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PHASE 1A STUDY REPORT

VOYAGER SPACECRAFT

VOLUME 6
OPERATIONAL SUPPORT EQUIPMENT

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TRW SYSTEMS GROUP

Redondo Beach, California

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INTRODUCTION

This volume contains functional descriptions of the operational support equipment (OSE) required to support the planned 1971 Voyager mission, employing the spacecraft defined in Volume 2. A description of the OSE required to support the 1969 Voyager test mission is presented in Volume 7.

Treatment of this material follows the suggested JPL format in the Phase IA Work Statement and is divided into these five sections:

I. OSE Objectives and Criteria

II. OSE Design Characteristics and Restraints

III. OSE System Functional Descriptions

IV. OSE Subsystem Functional Descriptions

V. OSE Implementation Plan

Material contained in the attached appendices provides data on possible alternate approaches and tradeoffs while establishing a basis for the selected configuration of operational support equipment. These appendices are entitled:

A. Voyager Automatic Data Handling System

B. Voyager Launch Complex Equipment

C. Voyager Mission Dependent Equipment

D. Voyager Mechanical Alignment Plan

E. Advantages of Voyager 1969 Test Mission to OSE Development for the Voyager 1971 Mission

F. Voyager OSE Abbreviations

G. Voyager OSE Design Documents

The discussion of operational support equipment utilizes the JPL nomenclature for OSE in which:

System Test Complex (STC) covers all support equipment used for powering, monitoring, and recording during test of the spacecraft and its subsystems to determine design performance and establish flight readiness. This grouping includes all bench checkout equipment (BCE), unit test sets (UTS), system test sets (STS), and the automatic data handling system (ADHS) (see Figure 1).

Launch Complex Equipment (LCE) covers all support equipment used for powering, monitoring, and recording during pre-flight tests of the spacecraft at the Eastern Test Range (ETR). This grouping includes all operational support equipment in the spacecraft assembly facility, the explosive safe facility, on the launch pad, and in the blockhouse (see Figure 2).

Mission Dependent Equipment (MDE) covers all in-line components of the telemetry ground station at the DSIF, consisting of telemetry detectors, computer buffers, and command encoders, plus associated test equipment, including command decoders, data format generators, error rate testers, etc. (see Figure 3).

Assembly, Handling, and Shipping Equipment (AHSE) covers all mechanical equipment for assembling, handling, and transporting the Voyager planetary vehicle, the spacecraft assembly, and all of its subassemblies. Included is such equipment as slings, hoists, dollies, transporters, and shipping containers (see Figure 4).

To facilitate discussions within the five prescribed sections, the system level components of the STC, i.e., the system test sets, are discussed in Section III, and the subsystem components of the STC, i.e., the unit test sets, are discussed in Section IV. Similarly, AHSE is discussed in Section III at the system level for those items of mechanical operational support equipment (MOSE) associated with the flight spacecraft, the planetary vehicle, and the planetary vehicle plus the nose fairing. It is also discussed in Section IV at the subsystem level for those items of MOSE associated with the various subsystems of the flight spacecraft. It should also be noted that the STS appears not only as part of the STC associated with the production and test cycle, but also as part of the LCE in the launch cycle.

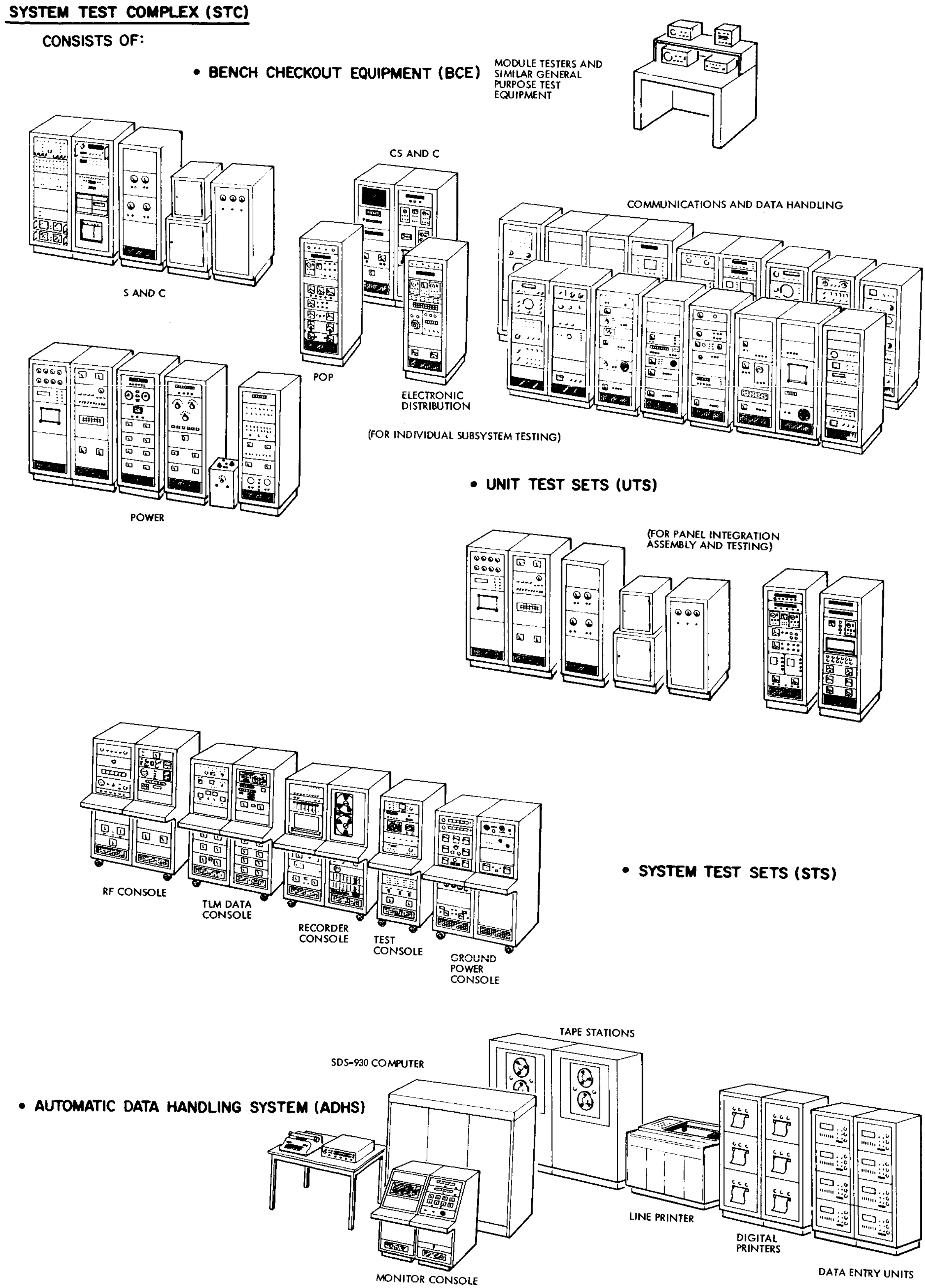


Figure 1. System Test Set Complex

LAUNCH COMPLEX EQUIPMENT (LCE)

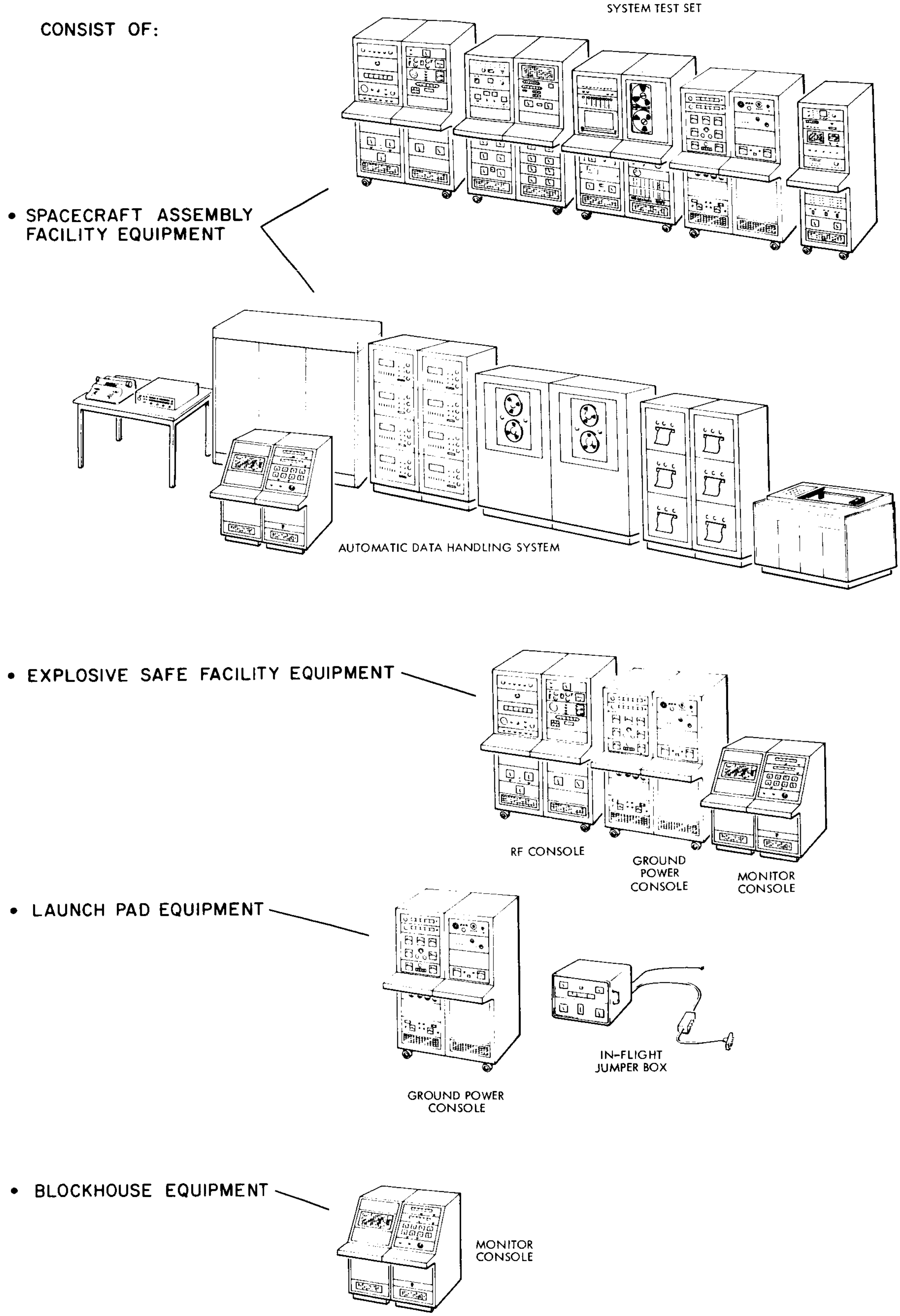


Figure 2. Launch Complex Equipment

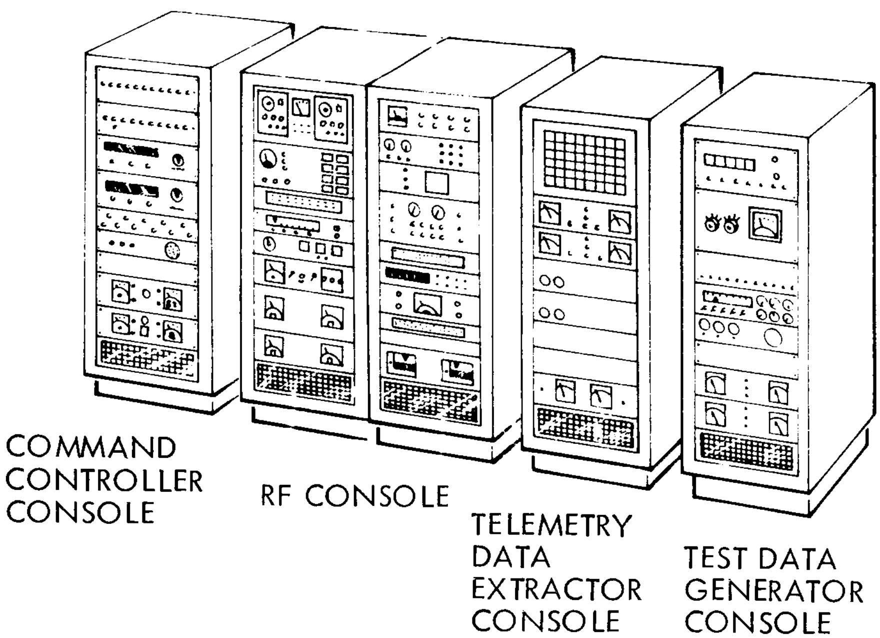


Figure 3. Mission Dependent Equipment

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT (AHSE)

CONSISTS OF:

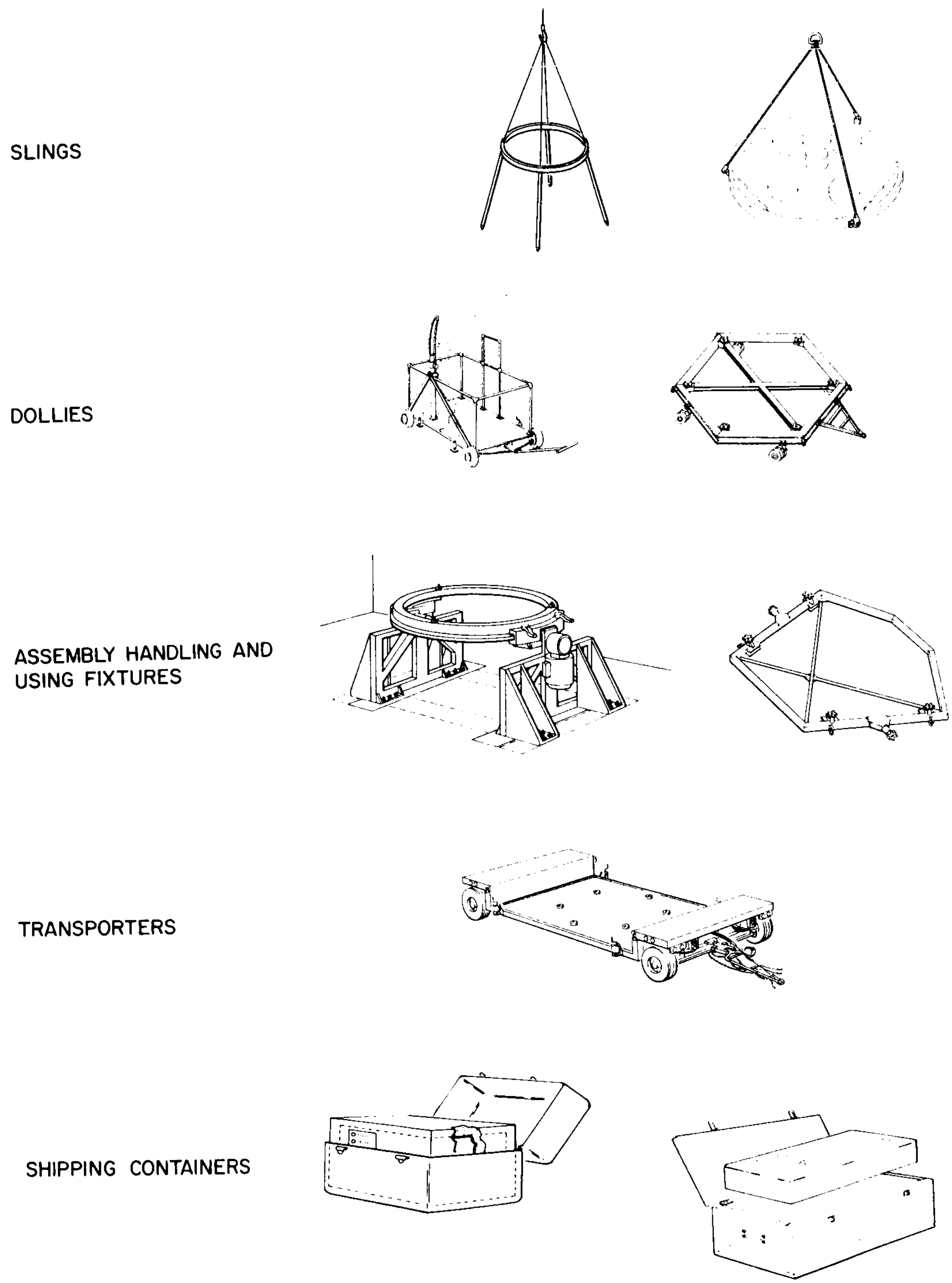


Figure 4. Assembly, Handling, and Shipping Equipment

To simplify discussions of implementation plans, design criteria, etc., OSE is also separated in this volume into electrical operational support equipment (EOSE) consisting of system test complex, launch complex equipment, and mission dependent equipment, and into mechanical operational support equipment (MOSE) comprising assembly, handling, and shipping equipment (AHSE).

Implementation of the testing philosophy is described here in which unit test sets are instrumented to support testing of specific subsystems, and system test sets are designed to provide end-to-end testing through many subsystems in series rather than duplicating the subsystem by subsystem testing capabilities of the unit test sets.

The level of semi-automatic test recording and automatic data handling selected for the Voyager OSE is described both in Appendix A, covering approaches and the tradeoff between manual and semi-automatic testing, and in OSE/VS-3-120. It is proposed to continue this investigation in Phase IB, with guidance from JPL as to a desired level of automation and sophistication of software.

A simplified flow chart of spacecraft development, test, and launch operations in provided in Figure 5, showing the relationship between spacecraft activities and development of OSE. A bar chart indicating major milestones in the OSE development is provided in Figure 18 in Section V of this volume.

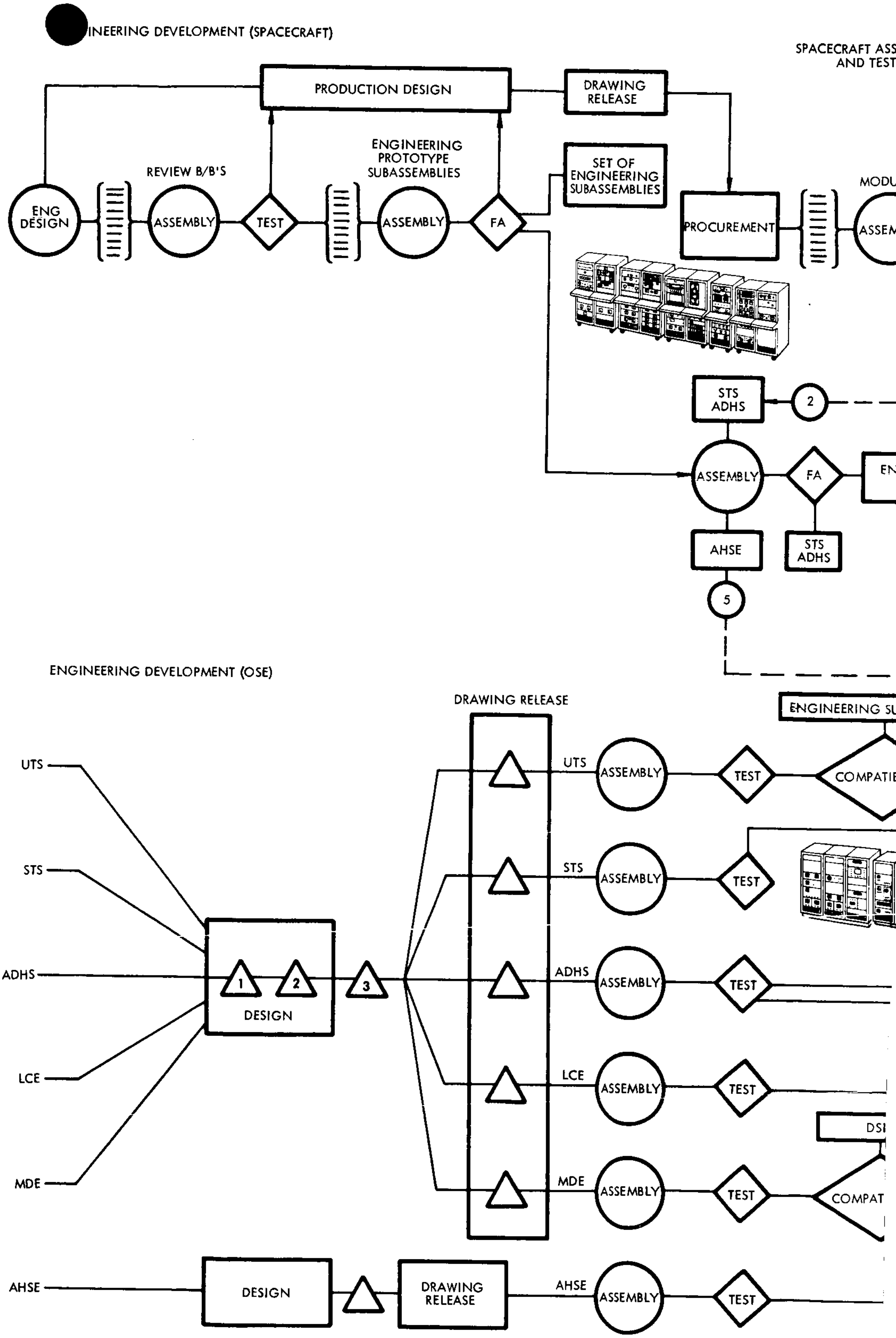
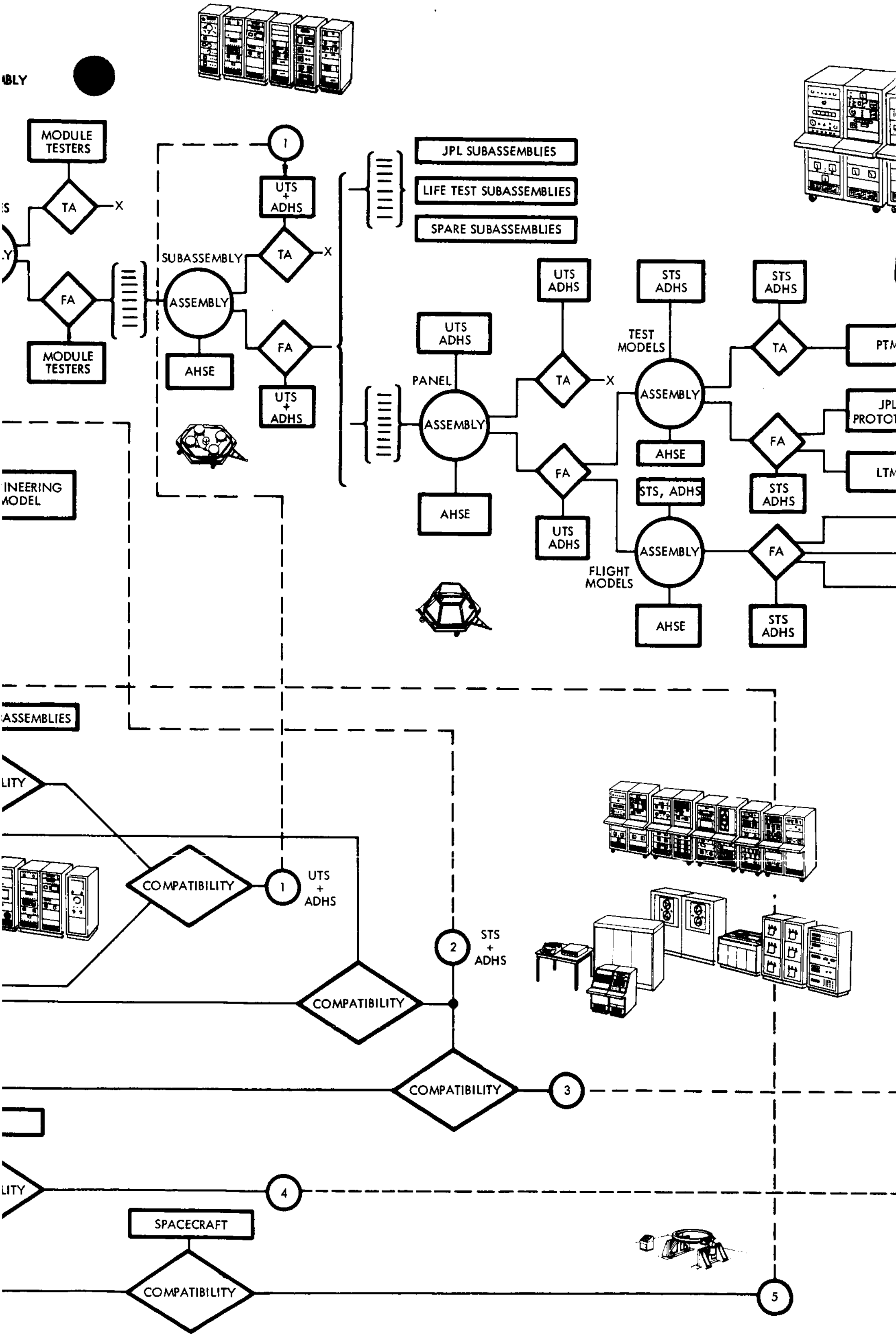


Figure 5. Flowchart of Spacecraft Development, Test, and Launch Operations

(1)



BLY
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MODULE TESTERS
SUBASSEMBLY
ASSEMBLY
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1
UTS + ADHS
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JPL SUBASSEMBLIES
LIFE TEST SUBASSEMBLIES
SPARE SUBASSEMBLIES
UTS ADHS
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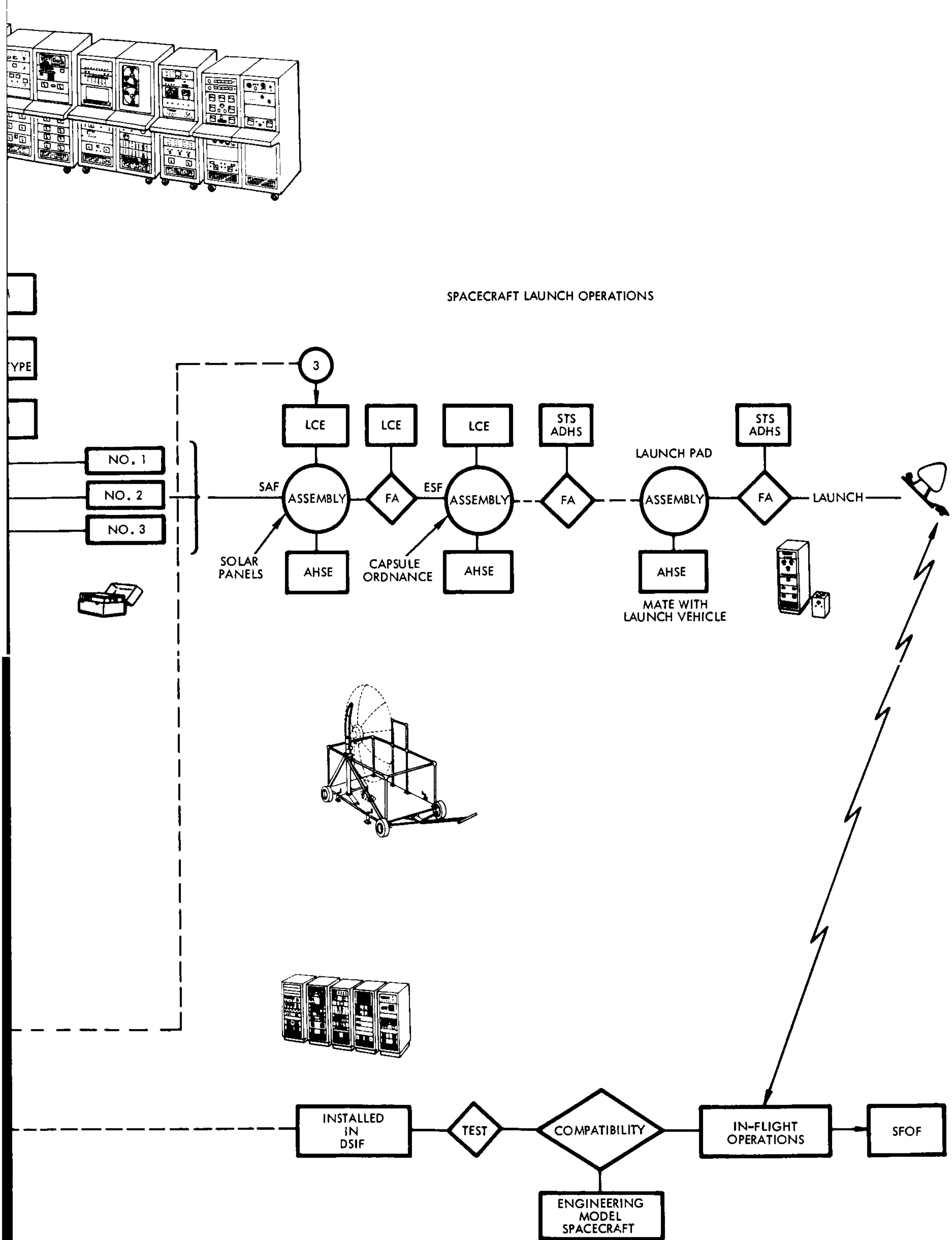
8 ②



SPACECRAFT LAUNCH OPERATIONS
TYPE
NO. 1
NO. 2
NO. 3
3
LCE
LCE
LCE
STS ADHS
STS ADHS
LAUNCH PAD
SAF
ASSEMBLY
FA
ESF
ASSEMBLY
FA
ASSEMBLY
FA
LAUNCH
SOLAR PANELS
AHSE
CAPSULE ORDNANCE
AHSE
AHSE
MATE WITH LAUNCH VEHICLE
INSTALLED IN DSIF
TEST
COMPATIBILITY
IN-FLIGHT OPERATIONS
SFOF
ENGINEERING MODEL SPACECRAFT

8 3

I. OSE OBJECTIVES AND CRITERIA

1. OBJECTIVES

This section provides the over-all ground rules, guidelines, and design philosophy for the operational support equipment required to support the Voyager program.

The basic objective of the OSE is to provide a tool to verify the design and the flight readiness of the Voyager spacecraft. The OSE will be mechanized to provide the capability of exercising the flight spacecraft through all of its standard and backup operating modes, and to provide mission dependent equipment capable of operating with DSIF equipment and verifying the compatibility of the DSIF equipment with the spacecraft.

2. GENERAL OSE

The following design criteria are applicable to the design of both the electrical and mechanical OSE.

The OSE provides for the highest practical probability that at the time of launch the Voyager mission will succeed in all of its objectives. In providing this, the OSE is capable of:

a) Verifying spacecraft design prior to the launch phase

b) Detecting spacecraft faults in time to prevent the launch and subsequent failure

c) Testing to a degree which does not prevent a good spacecraft from being launched

d) Providing for minimum launch turn-around time by using the most expeditious techniques to remove, repair or replace, retest, and remate on stand.

The OSE must provide safe conditions for the operating personnel as well as for the spacecraft. Precaution will be used in OSE design to prevent hazardous conditions, including explosive hazards and load margins, from existing. Physical protection of sensitive areas of the spacecraft will be given prime consideration during the OSE design.

Design of OSE is directed toward simplicity and conservatism, using proven techniques, procedures, materials, and vendor subassemblies or components.

Multiple use of specific OSE end items will be sought. Upgrading capabilities of end items to perform multiple functions is considered to be a standard design guideline except where costs become excessive, or simple and conservative design is compromised heavily.

All OSE is designed to provide complete interchangeability of end items of the same design. All OSE end items are designed so they may be functionally tested and actuated prior to acceptance for use with the spacecraft. Self-check capability is incorporated into design wherever possible. The design of OSE includes flexibility so that the same equipment, with minimum modification, can be used for both the 1969 and 1971 spacecraft.

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

The OSE must be designed to prevent the contamination of the capsule. This is accomplished by ensuring careful handling and testing of the capsule after it is mated with the spacecraft. In addition, the OSE is of such materials or will use such testing techniques that deleterious induced magnetic fields or RF interference will not be produced in the units under test, the spacecraft, or the capsule.

3. SPECIFIC VOYAGER OSE

3.1 Electrical OSE

3.1.1 Equipment Criteria

The EOSE consists of checkout equipment with the capability of performing electronic tests on the spacecraft units and systems which will confirm the ability of the spacecraft and its subsystems to operate within predetermined limits. The proof of the proper design of the various units and subsystems will be demonstrated after the spacecraft is completely assembled and integrated system tests completed in accordance with mission operational requirements. Electronic tests will be performed on the spacecraft during thermal-vacuum, vibration, and magnetic tests to verify proper operation during extreme operational environmental conditions. It is during this period of testing that confidence in the spacecraft design and in the capability of the OSE to demonstrate the proper spacecraft operation or to detect any faulty operation of the spacecraft is established. The OSE is capable of providing diagnostic assistance for fault isolation to the replaceable unit mounted on a spacecraft panel during spacecraft systems testing, and to the replaceable component level (i.e., resistor, capacitor, etc.) during unit testing.

Recorded history of the performance of each spacecraft will be provided and filed for comparison against the latest test data for each spacecraft throughout the test program. This will allow trend analysis to determine drift tendencies which could prove deleterious to spacecraft operation at some time after launch.

EOSE is designed to prevent damage or degradation to the flight spacecraft during test and checkout. EOSE incorporates fail-safe provisions to prevent flight spacecraft damage in the event of an emergency shutdown. For example, the EOSE is provided with a fuse for each hardline connection to the spacecraft and provides sufficient impedance or resistive isolation to the spacecraft test point to prevent damage to the spacecraft. Each unit test set, panel assembly test set, and launch complex equipment with all OSE required is completely validated prior to the mating with the operational spacecraft or its associated subsystems.

3.1.2 Test Operations

The EOSE supports the test operations to the following extent as a minimum:

No test or checkout operation will inadvertently overstress a spacecraft or its associated subsystems. During type approval qualification tests, parameters exceeding the design limits will be imposed on the unit under test in order to ascertain the proper design tolerances. Marginal conditions will be used in in-line tests and checkout, especially at the module level. This marginal testing will include subjecting the unit under test to input signals and power whose characteristics are the extreme limits of the design tolerances, but not exceeding these tolerances.

Spacecraft system test and checkout operations will utilize artificial stimuli only in those situations where it is not feasible to provide normal system inputs. Single-use devices such as pyrotechnics will not be ignored during checkout. Tests to verify status and interfaces to a maximum degree will be employed during appropriate in-line or acceptance tests.

3.1.3 Types of Tests

The spacecraft will undergo a series of tests with its associated OSE and with other elements at an early date relative to the launch period to detect design deficiencies in system interfaces early in the schedule. The following tests will be performed:

a) Compatibility of each unit test set with its unit will be established as early as possible

b) Compatibility of the OSE and the spacecraft subsystems will be established at the subsystem level prior to delivery to the system test area

c) Intersubsystem compatibility will be established during the engineering model phase and completed during the early part of the proof test model phase

d) Spacecraft-LCE compatibility will be demonstrated during the PTM phase

e) Spacecraft-capsule compatibility will be demonstrated during the PTM phase

f) Spacecraft-DSN-MDE compatibility will be established during the PTM test phase; spacecraft communication-DSIF compatibility will be demonstrated with spacecraft prototypes and during the PTM test phase

g) MDE software will be demonstrated with the DSIF equipment during the PTM phase.

Those quantitive tests (analog and digital data) that indicate system operational condition will predominate; tests which indicate GO/NO-GO parameters will be used only as a gross check.

Each module will be subjected to low-level vibration tests to uncover any gross problems incurred during the manufacturing process. These tests will not create systems degradation. Where feasible, all subsystems will be operated and monitored during thermal-vacuum chamber testing.

3.1.4 Test Data and Parameters

In the checkout program development, all data from development tests, acceptance tests, integrated system tests, and preflight tests form the basis for determination of spacecraft readiness. Data obtained during the checkout flow will be used to determine spacecraft readiness and to provide trend data where applicable.

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

Parameter limits for the acceptance tests will be determined by the results of development tests and mission performance requirements. During all test sequences, critical parameters will be evaluated in view of test specifications and reliability requirements.

3.1.5 Test Procedures

All tests will be conducted with adherence to a written test procedure which has been previously validated, reviewed, and approved. In instances where test content is the same or similar for different checkout phases, comparable procedures and equipment will be utilized regardless of location. Standard procedures will be established for all calibration and maintenance of OSE to assure consistency of testing and confidence in test results obtained at all test locations.

OSE will be calibrated in accordance with approved procedure, and adequate records will be maintained to provide the required confidence in test results.

3.1.6 Test Contingencies

No malfunction, however slight, is to remain unexplained. No subsystem or system encountering an unexplained malfunction will proceed beyond that test phase where the malfunction occurred until the event is explained to the satisfaction of the Test Board, and the subsystem or system has successfully demonstrated performance to test specifications.

In the event that troubleshooting is required, an approved procedure will be utilized, and all steps taken will be documented in detail. Field locations will have the capability to isolate system malfunctions to the replaceable spacecraft panel unit.

When it becomes necessary to utilize spares or to modify a spacecraft or a unit during the course of a test, those minimum tests must be repeated which are necessary to re-establish confidence in the unit or system that has been affected.

Electrical and mechanical connectors, once mated in the spacecraft, will not be disconnected for the purpose of checkout.

3.1.7 Test Time

The checkout programs assure that the time between the last test on a flight component and the completion of the mission does not exceed its shelf life. Equipment operating time for test purposes is minimized, particularly for life-limited components. An accurate record of the time of operation of each spacecraft and its components will be kept throughout the test program to the launch time.

3.2 Mechanical OSE

All AHSE is fabricated of materials which are either non-magnetic or capable of successful deperming so that the maximum external artificial environment presented to the spacecraft system does not exceed 80 Oersteds. In particular, those items of MOSE which will be in contact with the spacecraft system following manufacturing (e.g., during the various systems tests and during the Cape Kennedy operations) must be considered to be fabricated of non-magnetic materials.

All MOSE is capable of positive grounding and is fabricated in such a manner to insure complete electrical grounding continuity.

All AHSE and special test equipment is designed to provide shock attenuation and vibration damping necessary to reduce externally imposed loads (e.g., from transportation media) to acceptable limits for the spacecraft system.

All AHSE involved in the transportation of the spacecraft system between contractor facilities and the Cape Kennedy launch site is designed for acceptance aboard feasible transportation media. Transportation between contractor facilities or Southern California test facilities is considered accomplished by tractor-trailer truck. Mounting fixtures, support pallets, shipping containers, etc., attenuate loads imposed by these media as necessary. Transportation of the spacecraft system between Southern California and Cape Kennedy may be accomplished by air, road, or rail.

All MOSE is designed to provide complete interchangeability of end items of the same design.

All MOSE is designed to ultimate design loads obtained by applying alternate factors of safety upon the limit loads. Limit loads are obtained by applying design limit load factors to the rated loads to establish the maximum load that the structure may be expected to encounter at any time during service. Ultimate load and limit load factors are contained in OSE/VS-2-110.

MOSE, which performs tasks of handling, transferring, and shipping the spacecraft system, provides design features which adequately satisfy environmental control constraints imposed by the spacecraft system. Environmental control requirements imposed by the spacecraft include: heating, air conditioning, humidity, and cleanliness control during transfer handling, assembly, and transportation (although these controls are provided by the launch vehicle system while the spacecraft system is on the launch pad, the MOSE must provide these controls up until vehicle mating), and shock and vibration control and attenuation.

II. OSE DESIGN CHARACTERISTICS AND RESTRAINTS

1. GENERAL OSE

The intent of this section is to define the design characteristics and restraints which are applied to the operational support equipment (OSE) necessary to support the Voyager 1971 mission test and evaluation program. Included are those design requirements which are applied to the design, fabrication and checkout of the unit test sets (UTS), system test set (STS), launch complex equipment (LCE), mission dependent equipment (MDE), and assembly, handling and shipping equipment (AHSE).

2. VOYAGER OSE

2.1 Electrical OSE Requirements

In order to best achieve its design goal, the OSE must have certain basic capabilities. These goals must be accomplished within the requirements discussed in the following paragraphs:

It is a test requirement that subsystem functions, as well as selected subsystem inputs and outputs, be monitored for quantitative evaluation of spacecraft performance. To meet this requirement, direct access test connectors of the appropriate type and quantity are provided as part of the spacecraft configuration. Direct access points are used for monitoring and controlling the spacecraft during all test phases with the exception of dummy run and launch pad checkout tests.

Sufficient spacecraft monitor points are provided to verify that the spacecraft is in a satisfactory launch condition; however, there is no provision made for performing any dynamic tests once the spacecraft has been placed on the launch pad. During this time, the primary spacecraft evaluation is made through the telemetry link.

Sufficient isolation is incorporated at the spacecraft end of each monitoring circuit to prevent possible damage in the event an inadvertent short or abnormal voltage is applied somewhere in the cabling or test equipment. In addition, OSE isolation is provided so that the monitoring of any of the spacecraft functions does not affect the performance of the spacecraft.

The STS and LCE have the capability of providing power to the spacecraft which simulates the solar capabilities of the spacecraft. This simulated power source permits the operation of the spacecraft during system and subsystem tests when the normal solar sources and battery are not available. Facilities are also provided to indicate when spacecraft power is on.

The OSE must be capable of performing its normal operations when connected to the spacecraft by a cable whose length is equivalent to the spacecraft umbilical cable used at the launch pad.

2.1.1 Unit Test Set (UTS) Requirements

Unit test sets for the Voyager program are capable of providing three primary functions:

a) Testing and checking out the individual units comprising a subsystem

b) Testing and checking out an assembled subsystem

c) Operating in conjunction with other unit test sets in the test and evaluation of a spacecraft panel.

Unit test sets are used to perform unit acceptance tests, unit type approval qualification tests, and panel qualification tests and panel acceptance tests.

Each unit test set has the capability of providing power to its subsystem. The input power has the same characteristics as that supplied to the subsystem from the spacecraft power subsystem and is capable of being varied beyond the tolerance limits of the spacecraft power supply.

Each unit test set has the capability of monitoring the transformer/rectifier (T/R) output of its associated subsystem. Monitoring of T/R voltages is accomplished by the use of meters having an accuracy of two percent or better full scale.

Each unit test set has the capability of simulating the output loading and the inputs which the subsystem or the unit experiences during spacecraft operation. All output functions are monitored as a result of input stimuli corresponding to spacecraft input tolerances and the test set is capable of varying the input parameters beyond the tolerance requirements.

2.1.2 System Test Set (STS) Requirements

The STS is used primarily in the evaluation of proper operation of the integrated spacecraft. This evaluation is accomplished by performing tests on the various spacecraft subsystems in all operating modes. The STS performs the following basic functions:

a) Receive the information which may be either via radiated RF or hardline from the spacecraft, and decode and display the data in usable form

b) Command the spacecraft into various modes of operation by signals which may be via radiated RF or hardline

c) Provide external physical stimulation to spacecraft optical sensors and solar cell modules

d) Provide, control, and monitor spacecraft ground power and battery charging power

e) Provide a centralized means of controlling and sequencing various tests

f) Simulate optical sensor output, and the various experiment outputs

g) Evaluate received data via the automatic data handling system (ADHS).

A central oscillograph recorder is provided as part of the OSE and is used to provide a record of spacecraft performance during system test by recording some telemetry as well as subsystem direct access monitor data.

Standard time is supplied to the OSE.

The prime function of the telemetry ground display OSE is the presentation of spacecraft telemetry data in a form adequate to permit the evaluation of spacecraft performance. During system test and launch operations, telemetry data is displayed by appropriate means at the system test set console. OSE readouts and displays are in a form requiring a minimum of data reduction to convert test results into meaningful data.

A junction box is provided to act as a distribution center for umbilical functions between the OSE and the spacecraft at each spacecraft system test area.

The ADHS in no way compromises the STS or LCE capability to meet their design objectives. Detailed requirements for the ADHS are contained in Section VI, paragraph 4 of this volume.

2.1.3 Launch Complex Equipment (LCE) Requirements

The LCE includes that EOSE required at the ETR to support testing of the flight spacecraft at the spacecraft assembly facility and of the planetary vehicle at the explosive safe facility and at the launch pad.

a. Spacecraft Assembly Facility

At this facility, the LCE validates the stabilization and control subsystem and performs an integrated system test. Additionally, it performs experiment/telemetry calibration and continuity, capsule, and Centaur interface checks.

b. Explosive Safe Facility

At this facility, the LCE performs a planetary vehicle ordnance test and modified integrated system test. The integrated system test performed at this facility is identical to that performed at the spacecraft assembly area except for the possible limitation imposed by the planetary vehicle configuration and range safety requirements.

c. Launch Pad

At the launch pad, the LCE supports a DSIF compatibility test and perform a modified integrated system test. This integrated system test is identical to that performed in the spacecraft assembly facility except for the possible limitations imposed by the planetary vehicle/Centaur and range safety requirements. Additionally, the LCE commands the planetary vehicle into its launch mode and monitors its performance for proper operation until launch.

2.1.4 Mission Dependent Equipment (MDE) Requirements

The MDE is designed to provide the following functions at the Voyager DSIF sites.

a. Primary (In-Line) Functions

- Command generation
- Telemetry detection
- Computer buffering.

b. Secondary (Supplementary) Functions

- Command detection
- Spacecraft status display.

c. Tertiary (Test and Maintenance) Functions

- Telemetry detection testing
- Simulated telemetry data generation
- Spacecraft simulation
- Station simulation
- General purpose measurement and calibration.

d. Computer Programming

2.1.5 Electrical Requirements

All elements of the OSE operate from a 105-125 volt, 60 cps, 1 ∅, three-wire power source and/or 120/208 volt, 60 cps, 3 ∅, four-wire, power sources. Each rack or group of racks has an appropriate overload protective device. The AC power distribution lines are protected from noise and the transient effects of power switching normally performed by subsystem OSE. This protection takes the form of isolation introduced at the power input terminals of each subsystem. The use of a triple or quadruple shielded isolation transformer (such as the topaz transformer products series) is required.

Electrical connections to the spacecraft from the OSE for purposes of monitoring will not influence the operation of the spacecraft. The electrical interfaces between spacecraft and OSE therefore provide such isolation as is required to eliminate any interaction. All circuits in the OSE which interface with the spacecraft have circuit returns which are isolated from the OSE chassis ground.

Commercial instruments (meters, scopes, etc.) are calibrated prior to incorporation into the system test complex and calibration sticker showing the calibration period affixed to the face of the instrument. (It will be incumbent upon the user to submit the instrument for calibration prior to the expiration of the calibration period.)

All major assemblies of the EOSE having the same part number are physically and functionally interchangeable. Mechanical and electrical interchangeability exists between like assemblies, subassemblies, and their replaceable parts. Interchanging like equipment is accomplished without physical or electricalmodification of any part of the equipment or assemblies, including cabling, wiring, and mounting.

The EOSE is designed to require a minimum of special test equipment to maintain calibration, perform adjustments and accomplish fault identification. The equipment is designed to facilitate maintenance.

The specific electrical requirements for mission dependent equipment are detailed in OSE/VS-3-130.

2.1.6 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 labs, SAF, ESF, etc., the design temperature range will be 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 highly desirable. For equipment operating in the umbilical tower J-box the operating design temperature will be 50 to 160°F and 95 per cent relative humidity.

Storage temperature limits for this requirement are changed to 32 and 120°F. The indicated storage temperature range is to be considered a design goal, as such it need not be tested.

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

The spacecraft performance will be evaluated in an RF environment which simulates the Saturn IB telemetry and beacons, ground RF, radars, etc. This evaluation is part of the RF compatibility test. The RF simulators will not be a part of the system test set.

To minimize the effects of RFI and line noise, all EOSE, as required, provides that line filtering necessary to reduce this problem to its practical limits.

The specific electrical requirements for MDE are detailed in OSE/VS-3-130.

2.1.7 Racks and Accessories

The following requirements are applicable only to that new equipment purchased for the Voyager program. Components procured for the Mariner program are considered acceptable.

The standard rack for Voyager OSE is specified in JPL Specification . For convenience in handling, rack multiples greater than dual will not be used.

Paint type and colors are specified in JPL Specification 30600. All equipment in the Voyager OSE requiring painting will be in conformance with this specification.

All console and panel accessories such as knobs, meters, indicator lights, etc., conform with the following:

- Standard control knobs - Single turn, JPL Specification 30603.
- Meters - Meters on special purpose equipment purchased from commercial sources as well as those fabricated at TRW conform with JPL Specification
- Writing desk - Test console may contain a writing desk.
- Engraving and identification - Panel faces, consoles, racks and rack chassis and other equipment requiring identification are identified as per JPL Specification Identification and markings by means of engraving are in accordance with JPL Specification . 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 is provided in the racks to ensure longevity of the equipment. The louvers or grills are located in either the bottom front or back of the rack; however, they are permitted in the top only if necessary. Due to the proximity of adjacent racks, louvers and grills are not to be placed in the sides of the racks.

Fans may be used; however, they must be electrically noise free, as per JPL Specification 30236.

At least one 120 volt 60-cycle duplex convenience outlet is provided on the front of each test position.

The specific racks and accessory requirements for MDE are detailed in OSE/VS-3-130.

2.1.8 Wiring and Cables

External cables interconnect consoles and connect console to facility for the purposes of supplying power and routing signals. All external cables conform to JPL Specification , except that MDE conforms to JPL Specification . Molded cables are built in accordance with JPL Specification . Internal cables interconnect components within a cabinet for the purposes of supplying power and routing signals. All internal cables are laced into a harness.

Umbilical cables are those cables which run from the spacecraft's umbilical connector through the space simulator interface plate and into the STC umbilical junction-box. The J-box then acts as a distribution center, from which cables are run to subsystem test position.

When all pins on a cable connector are not utilized, spare leads are incorporated into the cable. All unshielded spare leads are connected to ground at one end only, except when the capacitive effect is detrimental.

The following techniques for ground will be adhered to, both in design and fabrication of system test equipment and associated cabling, in order to eliminate, as far as possible, all potential ground loops and polarity problems between test equipment and the corresponding subsystem.

All OSE consoles and the spacecraft are tied together at a common point with wire No. 2 AWG or larger and , in general, follow the same physical route as the direct access cables. A common tie point is at or near the spacecraft system test fixture. Figure 6 illustrates a typical grounding scheme.

2.1.9. Shielding

a) All shielded leads have insulated shielding

b) All shielded test equipment leads from any one of the spacecraft subsystems have their shields common at the spacecraft end only (and are connected to the spacecraft structure)

c) When using coaxial connector shells, care must be taken to prevent ground loops by the use of proper isolation techniques

d) Shielding is not grounded by connection to a J-box structure, chassis or console frame

e) Shield potential leads are carried through all cables and junction boxes so that the continuity is not lost; cables are grounded by connecting these shields to a common equi-potential lead at the cable end closest to the spacecraft

f) Test equipment must not tie their chassis to the spacecraft power or signal returns.

Miniature coax cable, such as Microdot Coax, is not considered to be rugged enough for this application and will not be used for system test complex cable.

2.1.10 Special Equipment

As a precautionary measure, dummy loads are used in place of the actual subsystem cases during the initial turn-on operations following spacecraft assembly. A dummy load box is required for each subsystem assembly and this should simulate the maximum load each case would normally draw from the spacecraft main harness. In addition, the dummy load box provides those grounding paths normally completed through the respective subsystem.

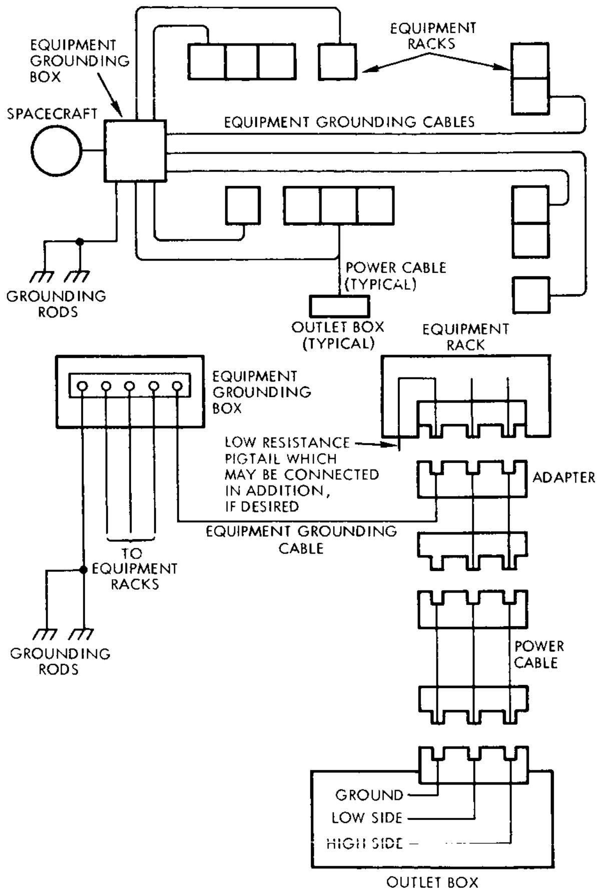


Figure 6. Typical Layout of System Test Complex

Break-in boxes or inline cable adapters which allow access to all leads at a connector interface are required for subsystem testing at SAF.

Line amplifiers may be used to counteract the interference and attenuating effect of long lines on low level signals. In the use of these amplifiers, consideration must be given to the following:

a) The line amplifiers used in the system test complex junction box must be electrically identical to those that will be used in the launch complex junction boxes

b) Direct access cable line amplifiers when used are placed as close to the spacecraft as feasible; however, they must be placed in a manner which will give the least interference or shadowing of the spacecraft with respect to the cold walls and light source in the space simulation chamber.

2.2 Mechanical OSE Requirements

2.2.1 General

Provision is made in the design of all mechanical operational support equipment (MOSE) to ensure that loads encountered during conditions of assembly, handling and shipping do not control the design of the spacecraft or any component to the extent that additional flight weight is required. It is a design goal that limit loads imposed on the spacecraft do not exceed the bending and axial load values specified in Figure 7. The design provides an appropriate combination of handling and transportation characteristics, considering safety of personnel, minimum hazard to the spacecraft, operational efficiency and ease of maintenance.

2.2.2 Strength and Rigidity Requirements

a. Limit Loads

Limit loads are the maximum loads that the structure may be expected to encounter at any time during service. For MOSE the limit load is normally the weight of the spacecraft or component and the associated item of MOSE multiplied by the limit load factor. The limit

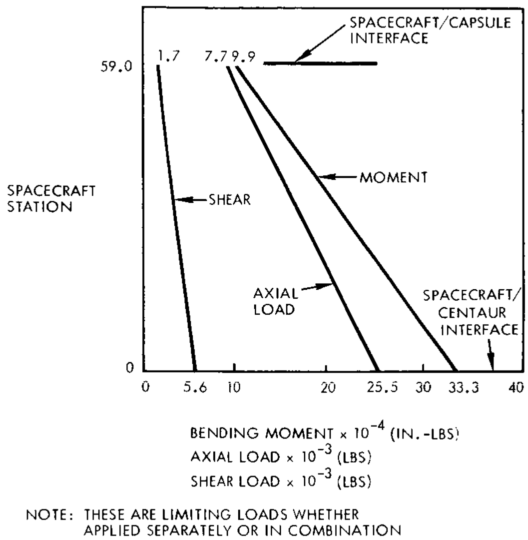


Figure 7. Maximum Allowable Load Envelope for Spacecraft During Transportation and Handling Modes

load acts at the combined center of gravity of the equipment and spacecraft or component.

The limit load factor is the ratio by which the weight of an item is multiplied in order to obtain the limit load acting on a structure or component. In some instances it may be the ratio by which an applied load is multiplied to obtain the limit load. Limit load factors to be used for design are specified in Table I.

The MOSE is designed to withstand the application of limit 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.

b. Ultimate loads

The ultimate design load is the limit load multiplied by an ultimate factor of safety.

The MOSE is designed to withstand design ultimate loads without failure. Failure is defined as inability to sustain ultimate loads.

c. Factors of Safety

The limit loads are multiplied by an ultimate factor of safety to obtain ultimate design loads. The ultimate factors of safety to be used for design are as follows:

Item	Ultimate Factor of Safety
All MOSE (except hoisting equipment)	2.0
Hoisting equipment (i.e., rotation or tilt fixtures, engine and propulsion handling fixtures, slings)	3.0

d. Material Properties and Allowable Stress Data

Material strengths and other physical properties are selected from reliable test results of recognized laboratories, reports from government agencies or manufacturer's guaranteed data. Strength allowables and other physical properties used are appropriate to the loading conditions, design environments stress state for each structural member.

Table I. Design Limit Load Factors

Equipment	Item No.	Direction	Limited Load Factors		Comment
Transporter, flight spacecraft	3-140	N_V $N_L = N_s$	+3.0 ±1.5	-2.0 ±1.5	
Assembly, handling and tilt fixture	3-140	N_V $N_L = N_s$	+3.0 ±1.5	-2.0 ±1.5	
Universal mounting ring	3-140	N_V $N_L = N_s$	+3.0 ±1.5	-2.0 ±1.5	
Adapter kit transporter of spacecraft and nose fairing	3-140	N_V $N_L = N_s$	+3.0 ±1.5	-2.0 ±1.5	Plus loads resulting from a 25 knot surface wind
Environmental cover (spacecraft)	3-140	N_V $N_L = N_s$	+3.0 ±1.5	-2.0 ±1.5	Depending on operational sequence, wind loads may also be included
Environmental covers (other than spacecraft)		N_V $N_L = N_s$	+3.0 ±1.5	-2.0 ±1.5	
Shipping containers		N_V $N_L = N_s$	+4.0 ±3.0	-3.0 ±3.0	
Handling dollies		N_V $N_L = N_s$	+2.0 ±1.0	-1.0 ±1.0	
Assembly, handling frames and fixtures		N_V $N_L = N_s$	+4.0 ±3.0	-3.0 ±3.0	Rigidity requirements must be examined
Protective covers		N_V $N_L = N_s$	+4.0 ±3.0	-3.0 ±3.0	
Weighing, c.g. and inertia fixtures		N_V $N_L = N_s$	+2.0 0	 0	All vertical forces shall be assumed to vary in direction from 0 to 1 from nominal rigging position

e. Analysis

The load factors are applied to the spacecraft and MOSE through their structural design centers of gravity and reacted statistically. The reactions are appropriate to the design condition and applied conservatively. Unbalanced forces and moments may be assumed reacted by spacecraft and MOSE inertia or reactions applied conservatively. Design of certain structural components may be directed by either stiffness or functional requirements; however, analyses are performed to verify that strength requirements are also satisfied.

f. Margins of Safety

The margins of safety are considered at both yield and ultimate load levels. All structures have a positive margin of safety which is computed in accordance with MIL-HDBK-5 procedures.

g. Structural Design Weights

The flight equipment weights to be used for structural design of MOSE are in accordance with TRW OSE Design Documents shown in Appendix G of this volume.

h. Design Limit Load Factors

The design of MOSE is based on the limit load factors specified in Table I and loads due to ground wind where applicable.

i. Dynamic Requirements

The acceleration input to the spacecraft or subassemblies during transportation and handling measured at the interface with the MOSE, is not greater than the equivalent acceleration input levels required for flight acceptance testing. Since flight acceptance vibration levels will not be defined until Phase IB, the vibration levels specified for Phase IA Section 4 of the Preliminary Voyager 1971 Mission Specification will be employed as the spacecraft input vibration level constraint. These dynamic test and design criteria will be updated during the development period to incorporate more accurate information on booster flight environment. Excessive localized resonances may be accounted for through the use of specialized supports or packaging equipment for the components under consideration.

j. Emergency Landing or Ditching

All MOSE that will be transported by air is designed to withstand accelerations due to emergency landings without any of the major components breaking loose and without external physical collapse. The spacecraft, components, and MOSE need not be in serviceable condition after the emergency landing. The ultimate load factors in accordance with MIL-A-8421 are shown in Table II.

Table II. Emergency Landing Load Factors

Direction		Condition	
N_v	4.5	0	0
N_s	0	±8.0	0
N_L	0	0	±8.0

k. Proof Tests

MOSE is tested to loads equivalent to those produced by the design limit load factors. Equipment used for hoisting is subjected to proof loads for a minimum period of five minutes. These tests will attempt to realistically simulate the actual loading at the points of attachment of the spacecraft or components to the MOSE. There will be no evidence of excessive deflection or permanent deformation after the proof loads have been removed.

l. Coordinate System

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

N_v = Vertical load, axis vertical relative to earth. Positive acting 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.

2.3 OSE General Requirements

The following paragraphs define the requirements which are applicable to both EOSE and MOSE.

All drawings are detailed to the extent necessary for reproduction of fabricated assemblies and ease of installation of OSE. Minimum documentation packages for the OSE include general assembly drawings, a functional description and schematics. This requirement is mandatory for all elements of the OSE. A complete set of schematics, test procedures, and operating instructions for each set of OSE will be filed in the areas where that particular OSE is used.

A Voyager interface control document will be generated and controlled by the Spacecraft contractor. This document and its control will ensure the interface compatibility of all OSE, OSE/ spacecraft and its components, OSE/ facilities, and OSE/ DSIF equipment.

Other documents which form a desirable adjunct to the minimum documentation package are operating instructions, wiring diagrams, maintenance and trouble shooting procedures, and spare parts lists.

Recognized principles of human engineering will be followed. These include, but are not limited to, the consideration of:

a) Intellectual, physical and psycho-motor capabilities of the intended user

b) Human space limitations for operation and maintenance

c) Visual and auditory perceptual requirements

d) Arrangement and readability of control and instrument panel displays

e) Safety factors minimizing potential human error in the operation of equipment

f) Sequence of operational requirements for the operator served equipment

Element and assembly identification will comply with the requirements as specified by JPL and each grouping of OSE will be identified with a nameplate.

3. OSE RELIABILITY REQUIREMENTS

Within the mission objectives for the Voyager 1971 flight there is a major emphasis placed upon downlink communications data from the spacecraft. Particular stress, therefore, must be given to the design for reliable system operations involving spacecraft communication and the OSE associated with the DSN. In this critical function, two equipments (the computer buffer and telemetry detector) identified as mission dependent equipment (MDE), play a big part in operational success. Figure 8 illustrates a single reliability assessment and logic diagram for these two equipments in their non-redundant usage configuration. Figure 9 provides a similar presentation of the assessed reliability for the command encoder, a third mission dependent equipment critical to uplink communications. (Reliability assessment details based upon part population estimates and a representative mission are provided for this study in Volume 4 Appendix B.)

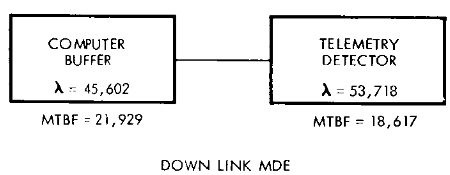


Figure 8. Single Reliability Assessment and Logic Diagram for Downlink MDE

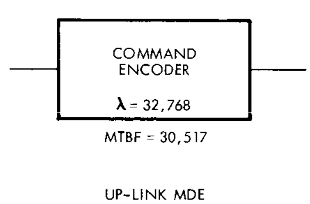


Figure 9. Reliability Assessment for Command Encoder

To evaluate the expected mission reliability for each of the MDE operational modes (downlink and uplink) it is useful to project their critical functions over the representative mission (8650 hours) used to assess the spacecraft success probabilities.

For downlink operations, although there will be three DSIF geographic loactions, it is assumed that two equipment sets (stations) can be made to function at any one time throughout the total 8650 hours. Furthermore, it is foreseen that any given station will implement an equipment switch over method equivalent to a sequential redundancy operational mode. This can be modelled conservatively as a simple parallel (both on) redundant configuration. Thus, with two stations each having redundant downlink communication capabilities, an operational mode exists wherein there will be a primary equipment set with three alternate (redundant) sets. Using the assessed reliability numerics of Figure 8, the total DSIF capability of downlink communications through the MDE (for no repairs) will be:

$$R_{DL} = 1 - (1 - R_T)^4$$

where $$R_T = \exp(-\Sigma\lambda t)$$

and $$\sum\lambda = (45,602) + (53,717) = 99,319(X10^{-9})$$

$$(\lambda = \text{failures per } 109 \text{ hours})$$

$$t = 8,650 \text{ hours}$$

Thus, $$R_{DE} = 1 - (0.576)^4 = 0.89$$

for the downlink success probability under a no-repair (worst case) condition. Under practical circumstances, however, each station can be expected to repair any one equipment set in a period of 48 hours during which time the remaining three redundant equipment sets can be expected to have reliability of:

$$R_{DL} = 1 - (1 - R_T)^3$$

where $$R_T = \exp(-\sum\lambda t')$$

$$\sum\lambda = 99,313 \times 10^{-9}$$

$$t' = 48$$

so that $$RDL = 1 - (4.77)^3 \times 10^{-9}$$

$$RDL \cong .999999$$

Thus, the MDE design objectives given as equipment MTBF (mean time between failure) values in Figure 8 are consistent with the downlink reliability objectives for the Voyager mission.

For the uplink MDE an estimated usage interval is 18 hours per transmission interval for an estimated 30 intervals. Thus the cumulative on-time is estimated at 540 hours over the total mission of 8650 hours. Assuming dependence upon a pre-selected DSIF station (and single redundancy of equipment) the operational success probability for uplink MDE functions will be:

$$RUL = 1 - (1 - R_T')^2$$

where $R_T1 = \exp(-\lambda' t')$

and $\lambda' = 32,768$

$t = 540$

Thus, $RUL = 1 - (.0176)^2 = 1 - 3.09 \times 10^{-4}$
$= .99969$

without repair. Under site repair conditions both uplink MDE equipments can be assumed in a state of operation prior to any one 18 hour transmission. Thus, the reliability for this interval can then be much greater than .99969 as a design objective and comparable with the equipment MTBF value given in Figure 9.

III. SYSTEM FUNCTIONAL DESCRIPTIONS

This section contains general descriptions of the Voyager systems operational support equipment. The section is subdivided into electrical system OSE (EOSE) and mechanical system OSE (MOSE).

1. SYSTEM EOSE

This section contains data on system level EOSE, i.e., EOSE used to support system level testing of the Voyager spacecraft and its associated supporting equipment. Included in these functional descriptions are:

1.1 System Test Set (STS)

This test set is used for integrated testing of the spacecraft during integration, assembly, and testing; testing in the environmental area; and testing at ETR in the spacecraft assembly facility (see Figure 10).

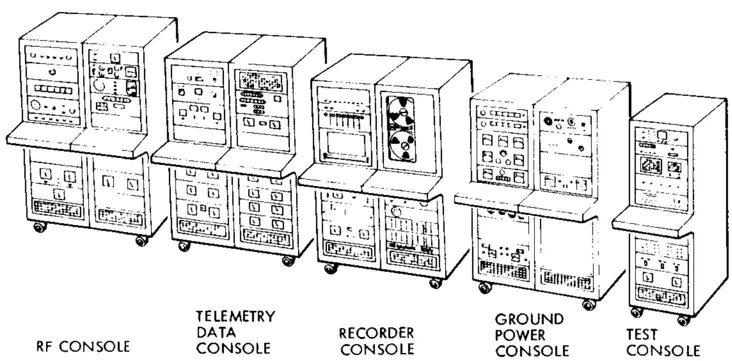


Figure 10. System Test Set

Tests at the propulsion test site (Capistrano), at the magnetic facility (Malibu), and in the environmental areas of TRW Systems will be supported by an STS in the Voyager assembly facility of TRW Systems, but with transfer of the RF console and the ground power consoles to the vicinity of the spacecraft. Similarly, the STS is used in the spacecraft assembly facility for tests in that facility and to support other tests in the ETR (such as tests in the explosive safe facility and on the launch pad) where, again, the RF consoles and the ground power consoles are grouped near the spacecraft under test while the system test set remains in the spacecraft assembly facility. In all cases, when the RF consoles and ground power consoles are used at remote locations, data flow between these units and the remainder of the system test sets is by wideband digital relay. See OSE Design Document OSE/VS-3-110 in Appendix G.

1.2 Automatic Data Handling System (ADHS)

ADHS is located with the STS to support system tests conducted by the STS during spacecraft tests in the TRW Systems integration assembly and test area, and at AFETR in the SAF, the ESF, and on the launch pad. The ADHS consists of a test director's console; an SDS-930 computer; manual input devices for transmitting data from the STS or associated equipment in the ESF, the blockhouse, or the launch pad; and computer peripheral equipment such as tape stations, line printers, character printers, paper tape punches, and readers. See OSE Design Document OSE/VS-3-120 in Appendix G and Figure 11.

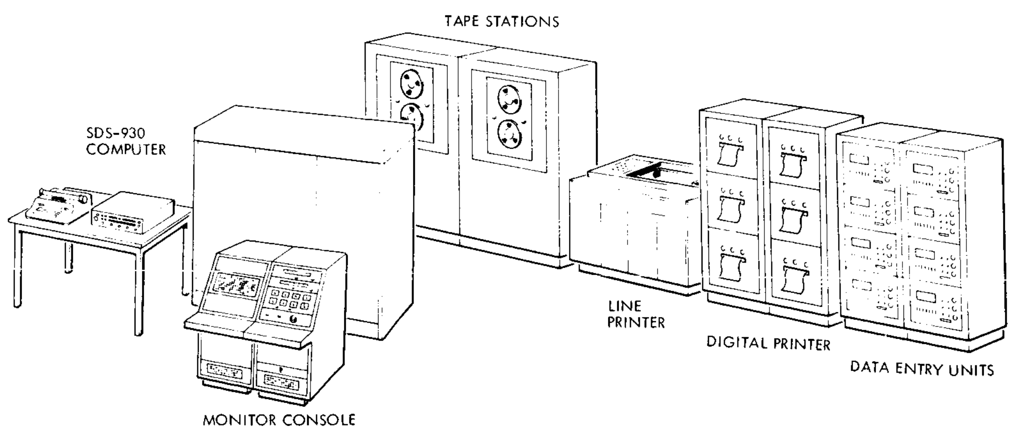


Figure 11. Automatic Data Handling System

1.3 Launch Complex Equipment (LCE)

LCE is used at ETR to support testing at the SAF, ESF, launch pad, and blockhouse. See OSE Design Document OSE/VS-2-120 in Appendix G.

1.3.1 Spacecraft Assembly Facility (SAF)

The system test set (STS) is used to conduct spacecraft tests at SAF and to support tests at remote locations in the ETR, such as the explosive safe facility and on the launch pad. Additionally, an automatic data handling system (ADHS) is used to support real time evaluation and recording of pertinent checkout data in conjunction with the STS (see Figure 12).

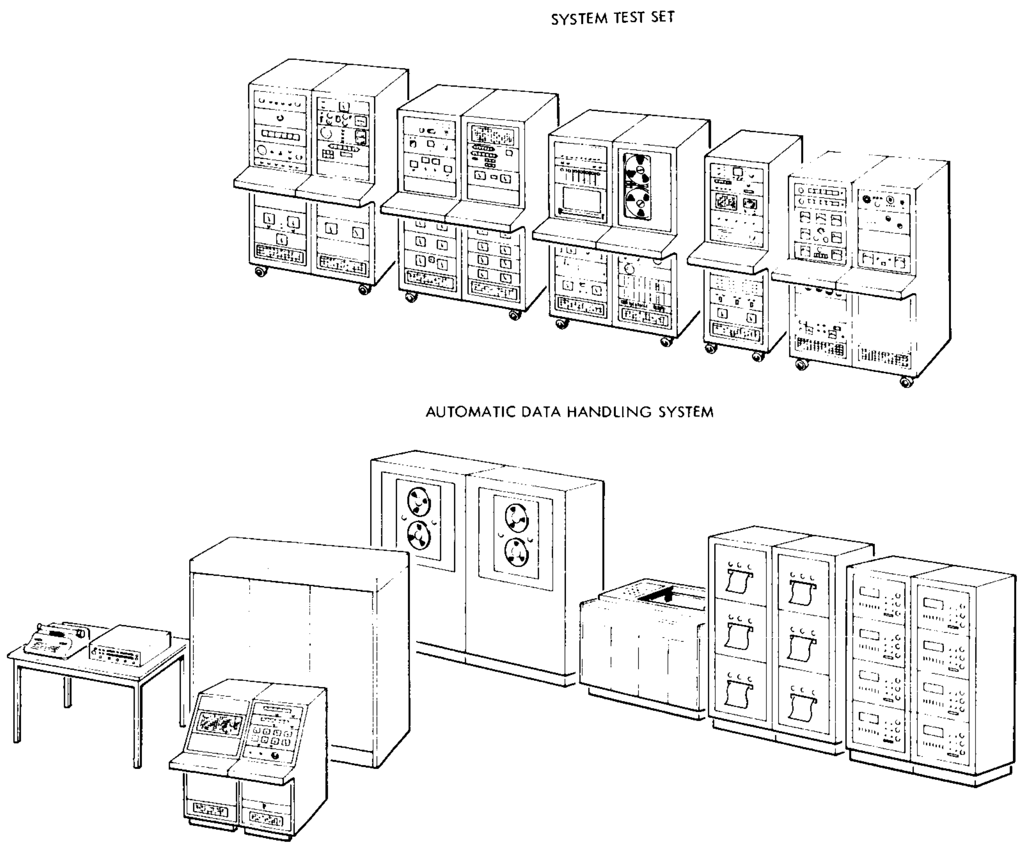


Figure 12. Spacecraft Assembly Facility

1.3.2 Explosive Safe Facility (ESF)

In the ESF, installation of the capsule and ordnance are checked out using the RF consoles and ground power consoles of the STS located in the SAF. To provide local display capability, a duplicate blockhouse monitor console is included in the ESF complement (see Figure 13).

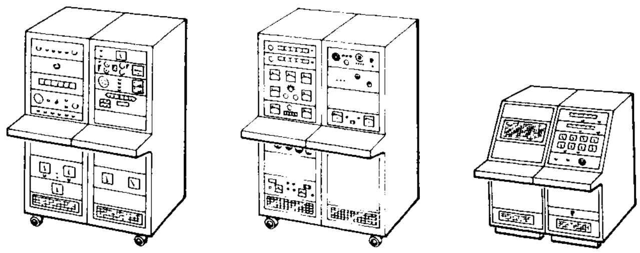


Figure 13. Explosive Safe Facility

1.3.3 Launch Pad Equipment

Launch pad equipment consists of the ground power consoles and the in-flight jumper control equipment used in connection with the STS in the SAF, and with the monitor console in the blockhouse (see Figure 14).

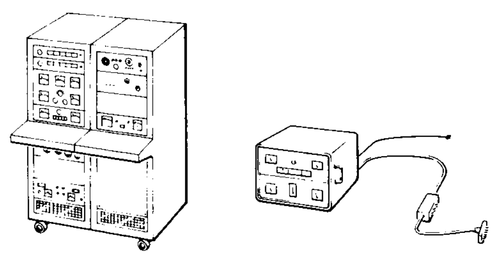


Figure 14. Launch Pad Equipment

1.3.4 Blockhouse Monitor Console

The blockhouse monitor console provides control and display of the power subsystem status, and display of spacecraft and telemetry status. Inputs for driving the blockhouse monitor console are derived from the ADHS in the SAF and from hardlines from the launch pad (see Figure 15).

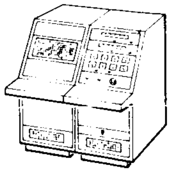


Figure 15. Blockhouse Monitor Console

1.4 Mission Dependent Equipment (MDE)

MDE used at the DSIF includes in-line equipment such as telemetry detectors and computer buffering, as well as command generation and supporting test equipment such as transponders, data format generators, error rate testers, command detectors, etc. See OSE Design Document OSE/VS-3-130 in Appendix G and Figure 16.

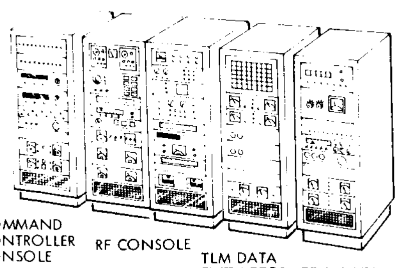


Figure 16. Mission Dependent Equipment

2. SYSTEM MOSE

The system MOSE is required for the assembly, checkout, and transport of the flight spacecraft, the planetary vehicle, and the planetary vehicle and nose-fairing combination.

2.1 Assembly,Handling and Shipping Equipment (AHSE)

The MOSE required to support the Voyager system assembly, checkout, and transportation program is the AHSE. A preliminary functional description of the overall AHSE is included in OSE/VS-3-140 in Appendix G of this volume, while descriptions of the units of AHSE are contained in the following:

a) Transporter, flight spacecraft - OSE/VS-3-140-1

b) Assembly, handling and tilt fixture OSE/VS-3-140-2

c) Transport recorder - OSE/VS-3-140-3

d) Weight, center of gravity, and moment of inertia fixture -OSE/VS-3-140-4

e) Shipping container, standard modules - OSE/VS-3-140-5

f) Work platforms, mobile - OSE/VS-3-140-6

g) Adapter kit, Centaur/shroud transporter - OSE/VS-3-140-7

h) Sling assembly, planetary vehicle and nose fairing - OSE/VS-3-140-8

i) Purge unit, freon/ethylene oxide - OSE/VS-3-140-9

j) Planetary vehicle/nose fairing mating and assembly fixture - OSE/VS-3-140-10

k) Sling, flight capsule - OSE/VS-3-140-11

l) Hoist beam and slings, flight spacecraft - OSE/VS-3-140-12

m) Tag lines - OSE/VS-3-140-13

n) Launch stand access platforms - OSE/VS-3-140-14

o) Universal mounting ring, flight spacecraft and planetary vehicle - OSE/VS-3-140-15

p) Environmental cover, flight spacecraft - OSE/VS-3-140-16

q) Hoist sling, environmental cover - OSE/VS-3-140-17

r) Platform, auxiliary access - OSE/VS-3-140-18

IV. OSE SUBSYSTEM FUNCTIONAL DESCRIPTIONS

This section contains general descriptions of the Voyager subsystem OSE. The section is subdivided into electrical subsystem OSE (EOSE) and mechanical subsystem OSE (MOSE).

1. SUBSYSTEM EOSE

This section contains data on subsystem level EOSE, i.e., EOSE used to support subsystem level testing of the Voyager spacecraft. The various unit test sets employed in production to assure acceptable performance levels of spacecraft subsystems are covered under their respective subsystems (see Figure 17).

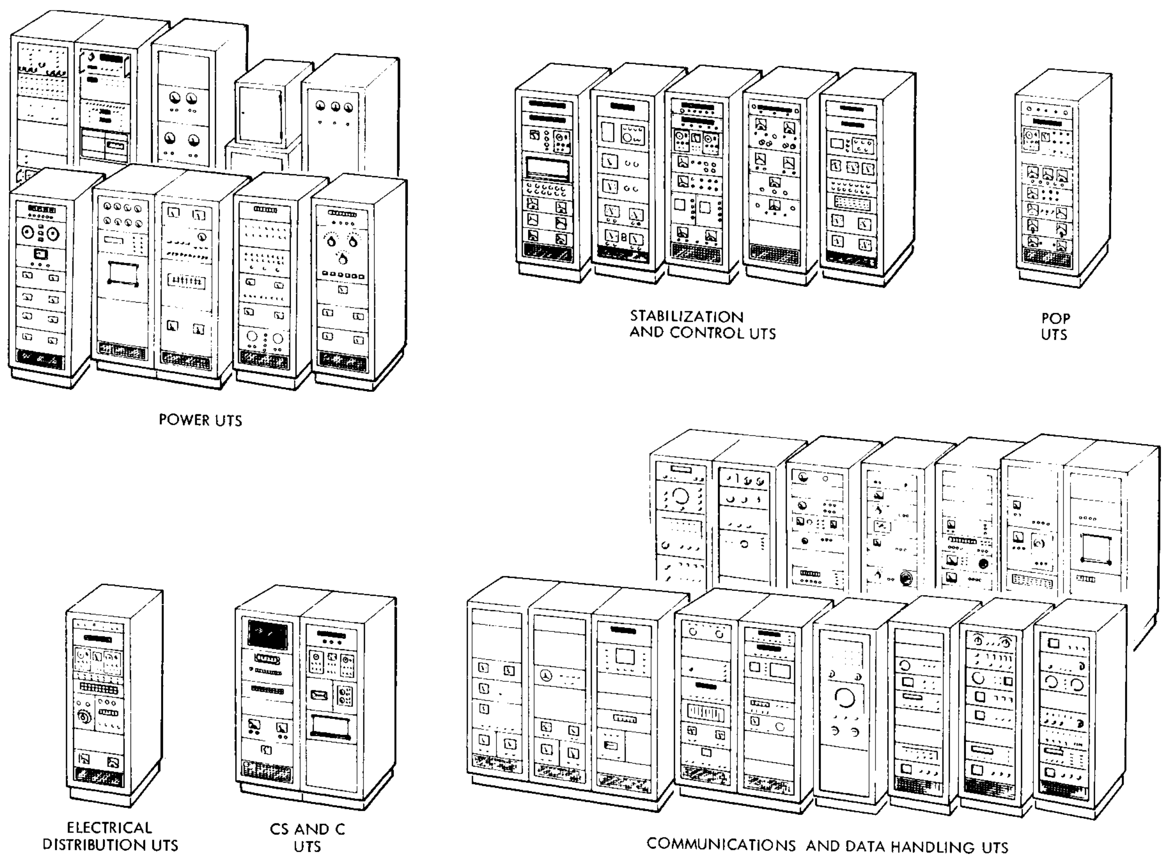


Figure 17. Electrical Operational Support Equipment

To obtain flexibility in scheduling, it is planned to perform integration assembly and testing on a panel level following qualification of units by unit test sets. This activity will be performed in an area apart from the spacecraft integration assembly and test area, and will be supported by a selection of unit test sets as appropriate rather than designing OSE specifically to duplicate the functions of the unit test set. Following is a discussion of the various unit test sets grouped under their respective spacecraft subsystems.

1.1 Communications and Data Handling Subsystem Unit Test Set

These unit test sets are required to test and evaluate the Voyager spacecraft communications and data handling subsystems. The test sets are used individually to test the associated flight units or collectively to test the integrated subsystem. The preliminary functional descriptions of the UTS are contained in the following documents.

a) S-band communications UTS, OSE/VS-4-311-1

b) VHF communications UTS, OSE/VS-4-311-2

c) Command detector UTS, OSE/VS-4-311-3

d) Data handling system UTS, OSE/VS-4-311-4

The antennas and coupling devices, both S-band and VHF, will be mated to the integrated spacecraft and evaluated utilizing the system test set.

1.2 Stabilization and Control Subsystem Unit Test Set

These unit test sets are required to test the Voyager stabilization and control subsystem units which consist of the rate gyro assembly, sun sensor, star sensors, control electronics assembly, and actuators. Each of these units is provided with its own associated unit test set and the use of these unit test sets in one area can be used to check out an integrated stabilization and control subsystem in the following modes: acquisition, cruise, re-orientation, mid-course velocity correction, de-boost engine burn, and orbital operations.

The preliminary functional descriptions of the UTS are contained in the following documents:

a) Rate gyro assembly UTS, OSE/VS-4-411-1

b) Sun sensor UTS, OSE/VS-4-411-2

c) Star sensors UTS, OSE/VS-4-411-3

d) Stabilization and control electronics assembly UTS, OSE/VS-4-411-4

e) Actuator UTS, OSE/VS-4-411-5

1.3 Central Sequencing and Command Subsystem Unit Test Set

This unit test set is required to test the Voyager central sequencing and command subsystem (CS and C). Since the units comprising the CS and C subsystem are packaged into one integral unit, this unit test set provides the capability of testing the CS and C subsystem either as a unit prior to or after being mounted on its spacecraft panel.

The preliminary functional description of this test set is contained in OSE/VS-4-451-1.

1.4 Power Subsystem Unit Test Set

The unit test sets required to test the Voyager power subsystem units which consist of the main AC power inverter unit, the 410 cycle single phase inverter unit, the 820 cycle two phase inverter unit, the battery control unit, the control electronics assembly and the battery unit. Because of the similarity of test requirements, capability of testing the three different inverter units is combined into the power inverter unit test set. Each of the other power subsystem units is tested by its own associated unit test set. The use of these unit test sets in one area can be used to check out an integrated power subsystem when mounted on its spacecraft panel.

The preliminary functional descriptions of the UTS are contained in the following documents:

a) Solar panel UTS, OSE/VS-4-461-1

b) Power inverter UTS, OSE/VS-4-461-2

c) Battery control UTS, OSE/VS-4-461-3

d) Power control electronic assembly UTS, OSE/VS-4-461-4

e) Battery UTS, OSE/VS-4-461-5

1.5 Electrical Distribution Subsystem Unit Test Set

This unit test set required to test the Voyager spacecraft planet oriented package (POP). The test sets required to test the other scientific experiments aboard the spacecraft will be provided as GFE along with the scientific experiment. The POP unit test set will provide the capability of testing the Mars sensor, gimbal drive and pickoff, and the gimbal electronics portions of POP in the alignment and servo modes.

The preliminary functional description of this UTS is contained in OSE/VS-4-581-1.

1.6 Propulsion Subsystem Unit Test Set

Analysis of the test requirements for the propulsion subsystem has disclosed that no electrical unit test sets are required. Functional operation and verification of the propulsion subsystem electrical components (valves, feedback pots, etc.) will be made during integrated system tests by the system test complex.

2. SUBSYSTEM MOSE

Subsystem MOSE is required to assembly, handle, transport, store, align, and protect the various Voyager subsystems during the Voyager factory and field operations.

2.1 Science Payload Subsystem MOSE

The Science Payload Subsystem MOSE is required for the transport, protection, storage and alignment of the body and boom mounted Science Payload Subsystem. The functional description of this subsystem MOSE is included in OSE/VS-4-210. Preliminary functional descriptions of the units of the CDHS are contained in the following documents:

a) Alignment set - OSE/VS-4-210-1

b) Shipping container, experiment booms - OSE/VS-4-210-2

2.2 Communications and Data Handling Subsystem (CDHS) MOSE

The CDHS MOSE is required for the assembly, handling, transport and storage of the CDH subsystem. The functional description of this subsystem MOSE is included in OSE/VS-4-310. Preliminary functional descriptions of the units of the CDHS are contained in the following documents:

a) Dolly 6' elliptical parabolic antenna - OSE/VS-4-310-1

b) Hoist beam 6' elliptical parabolic antenna - OSE/VS-4-310-4

c) Shipping container, 3' parabolic antenna - OSE/VS-4-310-3

d) Shipping container, 6' elliptical parabolic antenna - OSE/VS-4-310-4

e) Shipping container, lo-gain antenna - OSE/VS-4-310-5

f) Shipping container, flight capsule receiving antenna - OSE/VS-4-310-6

2.3 Stabilization and Control (S and C) Subsystem MOSE

The S and C subsystem MOSE is required for the protection, transport and storage of the Voyager S and C subsystem. The functional description of this MOSE is included in OSE/VS-4-410. Preliminary functional descriptionsof the units of S and C subsystem MOSE are contained in the following documents:

a) Alignment fixture, S and C nozzles - OSE/VS-4-410-1

b) Protective cover, S and C nozzles - OSE/VS-4-410-2

2.4 Power Subsystem MOSE

The power subsystem MOSE is required for the assembly, handling, protection, transport and storage of the Voyager power subsystem. The functional description of this MOSE is included in OSE/VS-4-460. Preliminary functional descriptions of the units of power subsystem MOSE are contained in the following documents:

a) Assembly and handling frame, solar panel segment - OSE/VS-4-460-1

b) Protective cover, solar panel segment - OSE/VS-4-460-2

c) Shipping container, solar panel segment - OSE/VS-4-460-3

d) Handling dolly, solar panel segment - OSE/VS-4-460-4

e) Sling, solar panel segment - OSE/VS-4-460-5

f) Shipping container, battery - OSE/VS-4-460-6

g) Shipping container, power amplifier - OSE/VS-4-460-7

2.5 Thermal Control Subsystem MOSE

The thermal control subsystem MOSE is required for the assembly, handling, protection, transport, and storage of the Voyager thermal control subsystem. The functional description of this MOSE is included in OSE/VS-4-510. Preliminary functional descriptions of the units of thermal control subsystem MOSE are contained in the following documents:

a) Assembly and handling fixture, spacecraft louvers - OSE/VS-4-510-1

b) Shipping container, spacecraft louvers - OSE/VS-4-510-2

c) Handling and shipping container, insulation - OSE/VS-4-510-3

2.6 Structural Subsystem MOSE

The structural subsystem MOSE is required for the assembly, handling, protection, transport, and storage of the Voyager structural subsystem. The functional description of this MOSE is included in OSE/VS-4-520. Preliminary functional descriptions of the units of structural subsystem MOSE are contained in the following documents:

a) Dolly, structural section - OSE/VS-4-520-1

b) Shipping container, miscellaneous structure - OSE/VS-4-520-2

c) Sling, propulsion/pneumatic structural section - OSE/VS-4-520-3

d) Interface match tool, spacecraft/flight capsule - OSE/VS-4-520-4

e) Interface match tool, spacecraft/Centaur adapter - OSE/VS-4-520-5

2.7 Pyrotechnic Subsystem MOSE

The pyrotechnic subsystem MOSE is required for the assembly, handling, protection, transport, and storage of the Voyager pyrotechnic subsystem. The functional description of this MOSE is included in OSE/VS-4-530. Preliminary functional descriptions of the units of pyrotechnic subsystem MOSE are contained in the following documents:

a) Shipping container, explosive train - OSE/VS-4-530-1

b) Handling and arming kit - OSE/VS-4-530-2

2.8 Planet Oriented Package (POP) Subsystem MOSE

The POP subsystem MOSE is required for the assembly, handling, protection, transport, and storage of the Voyager POP subsystem. The functional description of this MOSE is included in OSE/VS-4-580. Preliminary functional descriptions of the units of POP subsystem MOSE are contained in the following documents:

a) Assembly fixture and dolly, POP - OSE/VS-4-580-1

b) Shipping container, POP - OSE/VS-4-580-2

c) Hoist beam, POP - OSE/VS-4-580-3

2.9 Propulsion Subsystem MOSE

The propulsion subsystem MOSE is required for the assembly, handling, protection, transport, and storage of the Voyager propulsion subsystem. The functional description of this MOSE is included in OSE/VS-4-610. Preliminary functional descriptions of the units of propulsion subsystem MOSE are contained in the following documents:

a) Sling, retropropulsion motor - OSE/VS-4-610-1

b) Dolly, retropropulsion motor - OSE/VS-4-610-2

c) Alignment fixture, retropropulsion motor - OSE/VS-4-610-3

d) Alignment fixture, midcourse engine - OSE/VS-4-610-4

e) Shipping container, retropropulsion motor - OSE/VS-4-610-5

f) Shipping container, midcourse motor - OSE/VS-4-610-6

g) Pneumatic test set - OSE/VS-4-610-7

h) Pneumatic fill cart - OSE/VS-4-610-8

i) Propellant transfer and handling cart - OSE/VS-4-610-9

j) Alignment fixture midcourse motor/steering vanes - OSE/VS-4-610-10

k) Universal handling fixture, hydrazine-helium tank - OSE/VS-4-610-11

l) Sling, hydrazine/helium tank - OSE/VS-4-610-12

V. OSE IMPLEMENTATION PLAN

1. INTRODUCTION

1.1 Implementation Plan Purpose

The purpose of this implementation plan is to identify, plan, and organize the activities required to accomplish the development of the operational support equipment (OSE) used in support of the Voyager spacecraft operation. The objective of the plan is to specify all phases of the development work in order that due consideration of these tasks at the beginning will provide an efficient, well-organized approach to the development, with nothing unscheduled or unplanned. The result is the provision of OSE on schedule and within the cost budget.

1.2 Scope

This OSE implementation plan outlines the work to be accomplished from the time of Phase IB program go-ahead through the OSE design and fabrication, to the delivery and utilization of the equipment at the factory and in the field. The OSE for the Voyager program, as described in Sections I through IV of this volume, consists of the following categories of equipment:

a) Spacecraft unit test sets (UTS)

b) Bench checkout equipment (BCE)

c) Assembly, handling, and shipping equipment (AHSE)

d) System test set (STS)

e) Automatic data handling system (ADHS)

f) Launch complex equipment (LCE)

g) Mission dependent equipment (MDE).

Electrical operational support equipment (EOSE) is that equipment necessary to support the electrical test and evaluation, launch control, and mission operations of the Voyager spacecraft at all phases from laboratory development through mission completion.

Mechanical operational support equipment (MOSE) consists of specialized handling, protective, alignment, and transportation equipment necessary to support all vehicle activities from initial assembly operations through installation on the launch vehicle at the launch complex.

1.3 Development Phases

Specific activities are defined for the program of development and provision of OSE for the Voyager program. The development phases may be categorized as OSE analysis, design, manufacture, and evaluation. A summary of the tasks involved in each of these phases is given in the following paragraphs.

1.3.1 OSE Analysis

Following Phase IB contractual go-ahead, effort will be required in the completion of OSE requirements analysis, including gathering of additional OSE guidelines and background data as well as receipt of additional spacecraft test requirements and constraints where changes and additions may have occurred. The analysis effort will include the firming of OSE capability requirements, quantity determination, and completion of the OSE tradeoff tasks during completion of an OSE program plan in concurrence with generation of the final design/performance specifications for both hardware and software.

1.3.2 Design and Development

The preliminary system design for OSE will be completed, followed by detailed electrical and mechanical equipment design tasks. Breadboard circuit design and development testing, where required, will be accomplished during this phase, which will include three design reviews. The design phase will include human factors engineering and OSE parts analysis and preferred parts selection.

1.3.3 Manufacture

Following release of fabrication drawings OSE hardware will be built and electrical and mechanical assembly will be accomplished, with in-process and integration tests performed under surveillance of TRW Systems Quality Assurance.

1.3.4 Evaluation

The test activities will include initial fabrication checkout equipment integration, calibration, and OSE qualification tests, if required, as well as acceptance tests on each item of equipment. Following the functional evaluation and acceptance test phases, the OSE will be available for utilization in factory and field spacecraft support operations.

Certain of the subordinate tasks in these development phases have been accomplished during Phase IA and certain others have been accomplished in part. In general, the establishment of program ground rules and the basic requirements and constraints on the OSE have been preliminarily defined and functional descriptions of the OSE have been prepared during Phase IA. The updating of these items and the development work through the preparation of an OSE program plan, design/performance specifications, functional schematics, and OSE conceptual design documentation will be accomplished during Phase IB. During Phase II, the OSE detailed design, manufacture, evaluation, and sustaining engineering work is anticipated.

1.4 Critical Areas

Certain potentially critical areas exist in the development of OSE for support of the Voyager spacecraft program. The following areas will be given specific attention, and through the methods indicated, any possibility of these becoming pacing items in the Voyager program will be precluded.

1.4.1 Spacecraft Test Requirements

It is necessary that the acquisition of spacecraft functional test requirements for each of the types and levels of test be accomplished in a timely and accurate manner. Most of this has been accomplished during the Phase IA program. During the first part of Phase IB, the firming of additional or a changed test requirements data, will be accomplished prior to initiation of detailed OSE design in order that the design accomplished may be as accurate and complete as possible with minimum changes incurred.

Close and continuous liaison with spacecraft subsystem and system designers will be continued so that communication of test requirements data to OSE engineers will not be delayed.

1.4.2 Spacecraft-OSE Interfaces

Preliminary definition of the spacecraft/OSE electrical and mechanical interfaces has been accomplished during Phase IA. Detailed definition of these interfaces and of the man/machine interface will be continued and finalized during the Phase IB program. One of the major potential problem areas is that of interface with incurred problems in communication, in requirements definition, equipment design, and usage. Extensive control is, therefore, planned for the interfaces between checkout equipment and spacecraft, checkout equipment and experiments, and between the checkout equipment and personnel. Interface definition control and compatibility evaluation will be achieved through generation of interface criteria documentation which will:

a) Identify interfaces through block diagrams and lists

b) Identify interface types

c) Define interface format, content, and notation in a standardized minimum but complete manner

d) Identify the interface participants and those responsible for preparing, distributing, and maintaining the interface documentation

e) Define review and approval methods.

The potential interface problems will be minimized by a systematic procedure of identification, interface detail definition, definition of the impact of equipment design changes which are the mutual responsibility of a number of different organizations, and the maintenance of system compatibility through tightly controlled engineering data management.

1.4.3 Spacecraft Test Data Point Access

To provide for a maximum of evaluation information during the spacecraft test program, adequate test access to spacecraft subsystems and units must be available. This potentially critical area has been considered during the preliminary design work of Phase IA. The work will be continued during Phase IB, where the testing process will be considered during spacecraft design such that the proper tradeoff can be made between access to sufficient data points and the possible provision of an excess over that required for proper flight system status evaluation for all levels of test from unit through integrated system. Solution to this potential problem will be the participation of test and evaluation engineering personnel in the spacecraft design tradeoff studies.

1.4.4 Long-Lead Procurement

Some material or equipment included in the OSE involves an unavoidable long time procurement cycle. In these cases, advanced material releases are required prior to completion of the design, design review, and drawing release procedures. Upon relatively firm commitment on the use of such equipment, and with the approval of JPL, releases for its procurement will be accomplished. It is the experience of TRW Systems that such long lead item procurement has been successful, and essentially no losses have resulted from taking such action.

1.4.5 Configuration Control

One of the prime sources of problems in a complex system development is that of control of the end item configuration. Control implies the timely communication of information on the end item to affected parties relative to the item configuration as well as continual direction and monitor of each item's adherence to functional requirements and standardization, and compatibility in documentation. This will be achieved through adequate planning, culminating in the preparation and maintenance of the minimum configuration control documentation consistent with achieving this objective.

1.4.6 LCE and MDE Impact

Particular emphasis will be placed on the LCE and MDE reliability and configuration because of the in-line nature of most of the equipment involved.

2. OSE SCHEDULE

Information is provided on the OSE development schedule during Phase IB and II in the form of milestone schedules and quantity utilization charts.

2.1 Milestone Schedule

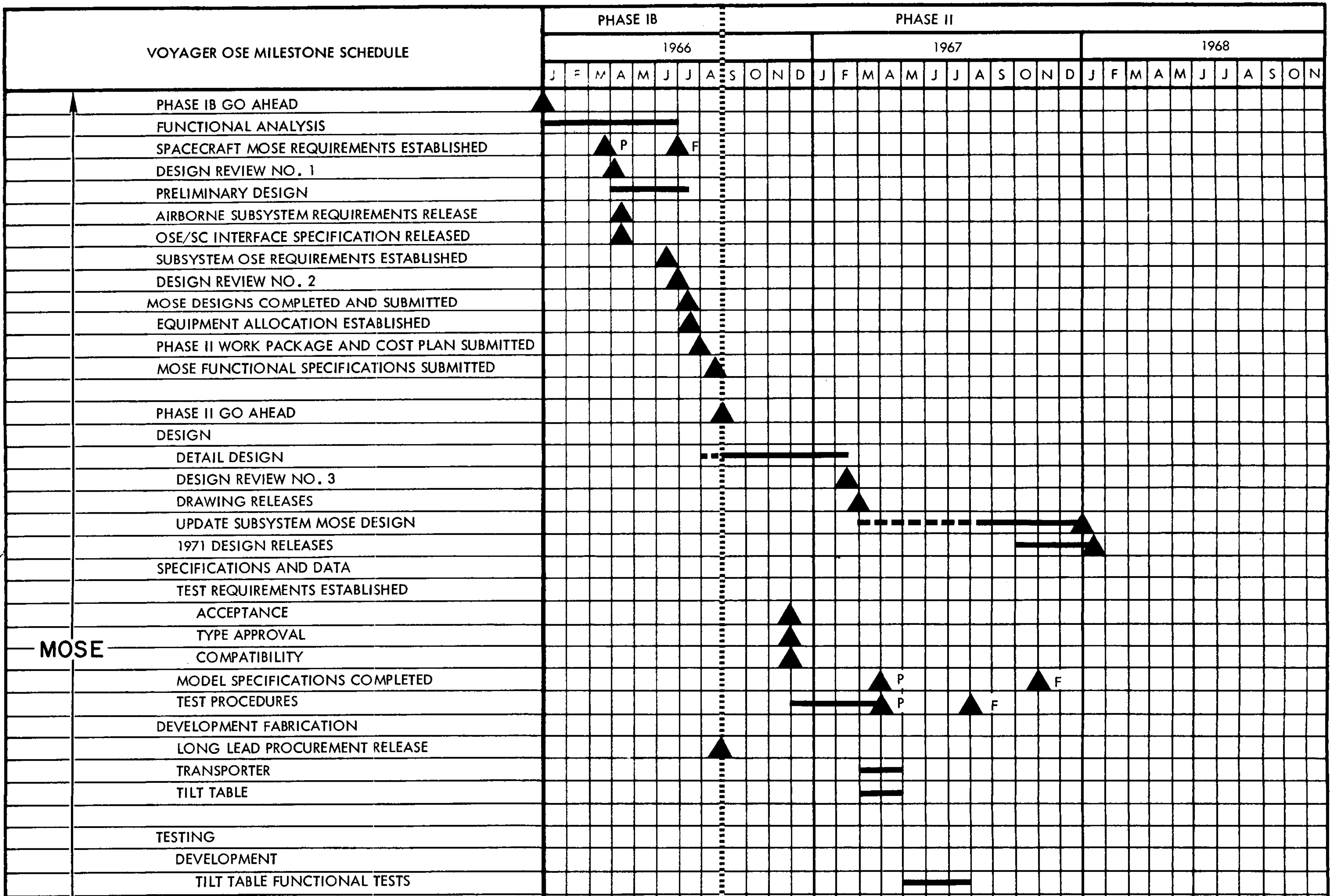
An OSE milestone schedule is given in Figure 18. This figure indicates the primary phases of OSE development as a function of time, following receipt of Phase IB contract go-ahead. Preliminary and final accomplishment dates are indicated where significant, as well as the types of equipment (assembly, handling, and shipping equipment, unit test sets, system test sets, automatic data handling system, launch complex equipment, and mission dependent equipment) through the phases of design, fabrication, test, and utilization. The sequential list of major milestone items indicates the estimated time span for each of the major tasks to be accomplished during OSE development.

2.2 Quantity and Utilization

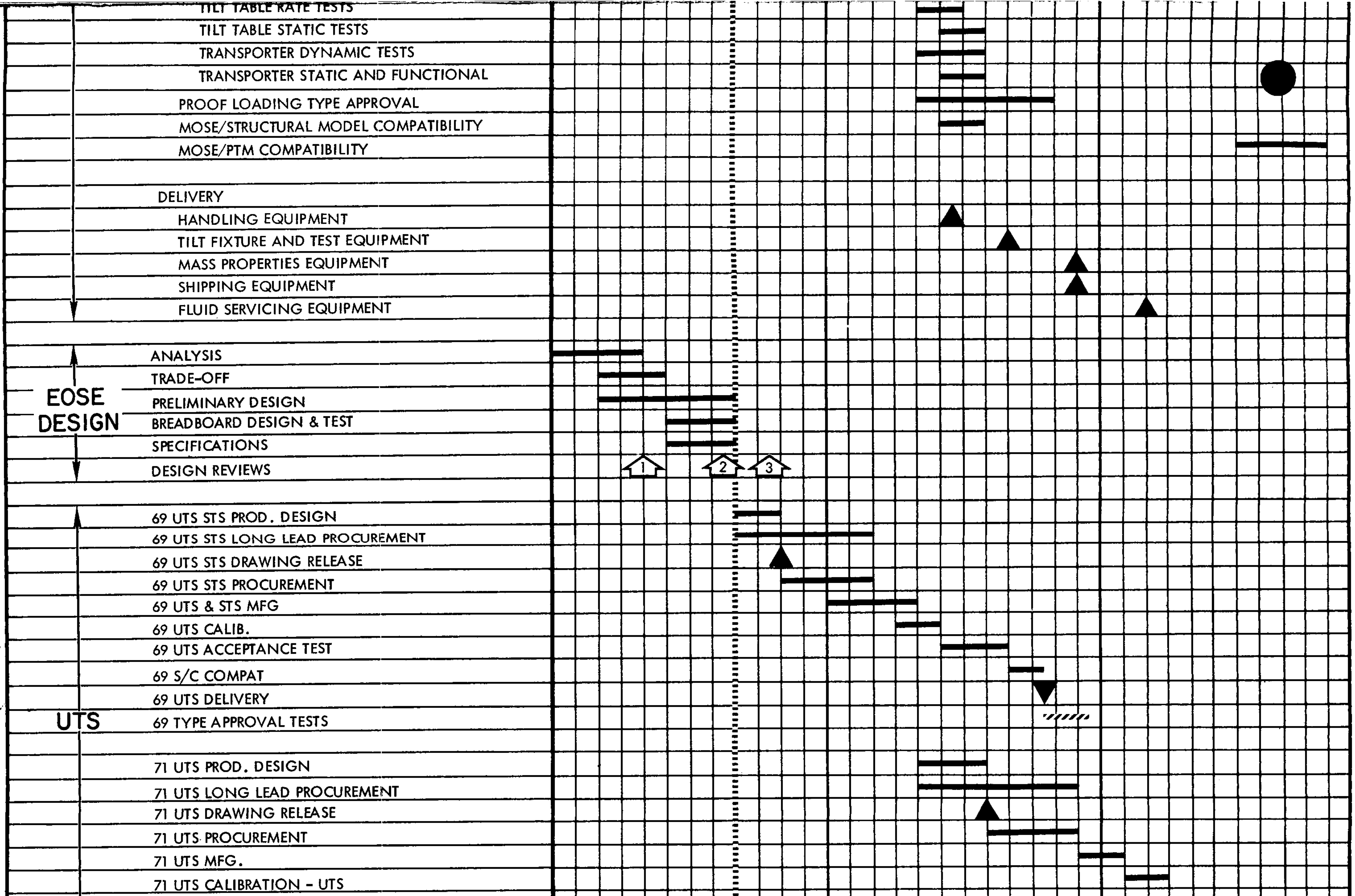
Table III is a chart showing a tabulation of EOSE quantities, categorized by type of equipment and where this equipment will be employed. Table IV contains similar data on anticipated utilization of MOSE.

2.3 Implementation Activities Summary

The activities of the EOSE development phases are shown on the development program network diagram of Figure 19, which indicates input requirements and tasks performed for each specific development event and shows the relationships between specific EOSE development tasks to be accomplished during this program. The diagram is at the first level of detail and each event shown may be subdivided into many subordinate events. During the preparation of the OSE program plan during Phase IB, the specific activities related to design, fabrication, and test of OSE will be additionally defined.


VOYAGER OSE MILESTONE SCHEDULE
PHASE IB
PHASE II
1966
1967
1968
J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N
MOSE
PHASE IB GO AHEAD
FUNCTIONAL ANALYSIS
SPACECRAFT MOSE REQUIREMENTS ESTABLISHED
DESIGN REVIEW NO. 1
PRELIMINARY DESIGN
AIRBORNE SUBSYSTEM REQUIREMENTS RELEASE
OSE/SC INTERFACE SPECIFICATION RELEASED
SUBSYSTEM OSE REQUIREMENTS ESTABLISHED
DESIGN REVIEW NO. 2
MOSE DESIGNS COMPLETED AND SUBMITTED
EQUIPMENT ALLOCATION ESTABLISHED
PHASE II WORK PACKAGE AND COST PLAN SUBMITTED
MOSE FUNCTIONAL SPECIFICATIONS SUBMITTED
PHASE II GO AHEAD
DESIGN
DETAIL DESIGN
DESIGN REVIEW NO. 3
DRAWING RELEASES
UPDATE SUBSYSTEM MOSE DESIGN
1971 DESIGN RELEASES
SPECIFICATIONS AND DATA
TEST REQUIREMENTS ESTABLISHED
ACCEPTANCE
TYPE APPROVAL
COMPATIBILITY
MODEL SPECIFICATIONS COMPLETED
TEST PROCEDURES
DEVELOPMENT FABRICATION
LONG LEAD PROCUREMENT RELEASE
TRANSPORTER
TILT TABLE
TESTING
DEVELOPMENT
TILT TABLE FUNCTIONAL TESTS
P
F
P
F
P
F

570


TILT TABLE RATE TESTS
TILT TABLE STATIC TESTS
TRANSPORTER DYNAMIC TESTS
TRANSPORTER STATIC AND FUNCTIONAL
PROOF LOADING TYPE APPROVAL
MOSE/STRUCTURAL MODEL COMPATIBILITY
MOSE/PTM COMPATIBILITY
DELIVERY
HANDLING EQUIPMENT
TILT FIXTURE AND TEST EQUIPMENT
MASS PROPERTIES EQUIPMENT
SHIPPING EQUIPMENT
FLUID SERVICING EQUIPMENT
EOSE DESIGN
ANALYSIS
TRADE-OFF
PRELIMINARY DESIGN
BREADBOARD DESIGN & TEST
SPECIFICATIONS
DESIGN REVIEWS
1
2
3
UTS
69 UTS STS PROD. DESIGN
69 UTS STS LONG LEAD PROCUREMENT
69 UTS STS DRAWING RELEASE
69 UTS STS PROCUREMENT
69 UTS & STS MFG
69 UTS CALIB.
69 UTS ACCEPTANCE TEST
69 S/C COMPAT
69 UTS DELIVERY
69 TYPE APPROVAL TESTS
71 UTS PROD. DESIGN
71 UTS LONG LEAD PROCUREMENT
71 UTS DRAWING RELEASE
71 UTS PROCUREMENT
71 UTS MFG.
71 UTS CALIBRATION - UTS

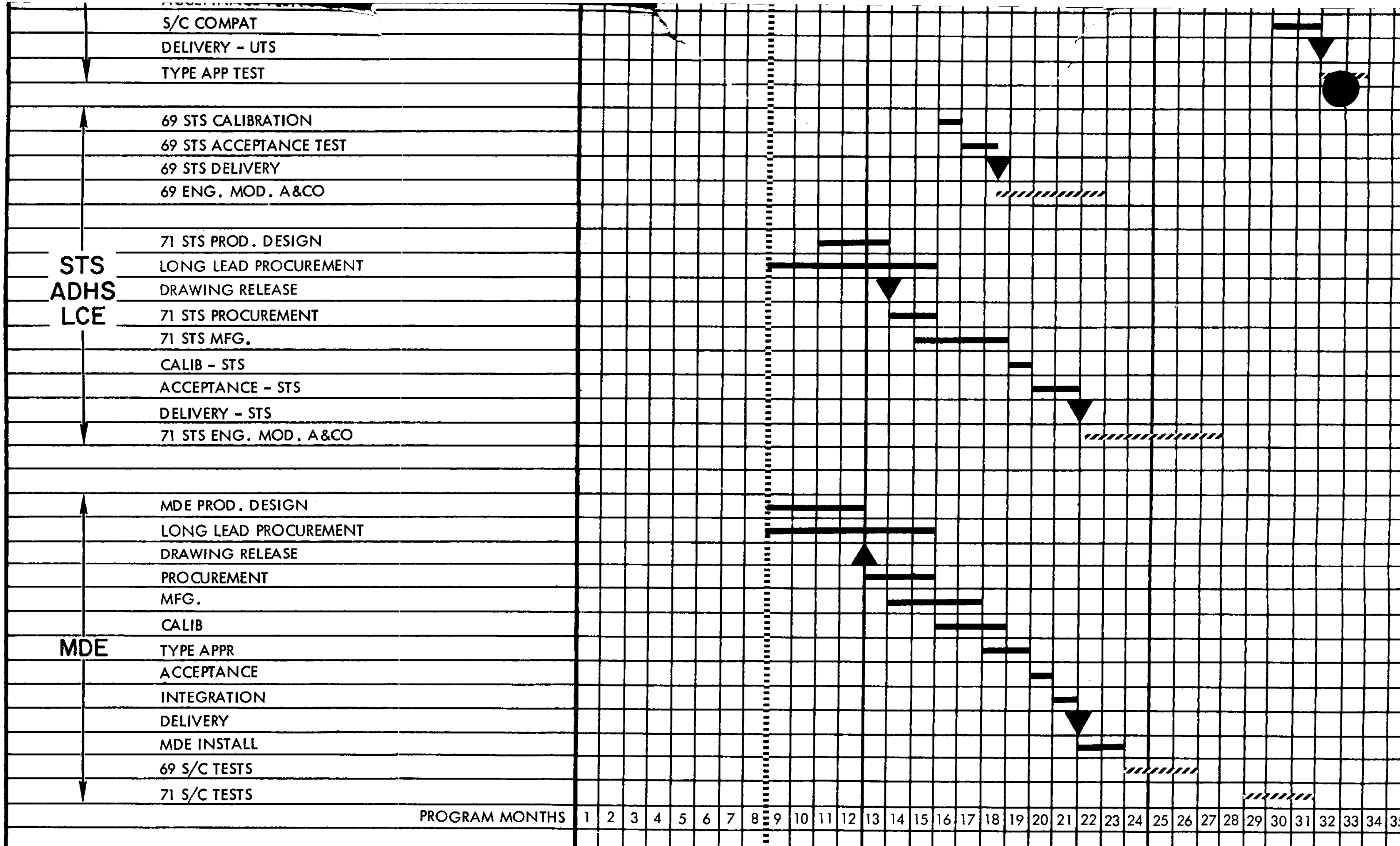


Figure 18. Voyager OSE Milestone Schedule

3

Table III. Estimated Quantities of EOSE

			UTS					STS	ADHS	LCE			MDE	GFE
		Telecommunications	Power	S and C	CS and C	Science	Electrical Distribution			ESF console	Blockhouse	Pad equipment		
Assembly area	1969	4	3	3	3	3	3	4	4	1	1	1	-	4[a]
	1971	-	-	-	-	-	-	4	4	-	-	-	-	4[a]
Panel assembly area	1969	2	2	2	2	2	2	-	-	-	-	-	-	-
	1971	-	-	-	-	-	-	-	-	-	-	-	-	-
Mag prop area (Malibu)		-	-	-	-	-	-	-	-	-	-	-	-	-
Static fire (Capistrano)		-	-	-	-	-	-	-	-	-	-	-	-	-
Launch area	1969	-	-	-	-	-	-	-	-	1	1	1	-	-
	1971	-	-	-	-	-	-	-	-	1	1	1	-	-
JPL test area	1969	-	-	-	-	-	-	-	-	-	-	-	-	-
	1971	1	1	1	1	1	1	1	1	1	1	1	-	1[a]
DSN	1969	-	-	-	-	-	-	-	-	-	-	-	4*	2[b]
	1971	-	-	-	-	-	-	-	-	-	-	-	-	-
TOTAL		7	6	6	6	6	6	9	9	4	4	4	4	9[a], 2[b]

*Johannesburg
Canberra
Goldstone
Cape 71
a = SDS-930
b = SDS-910

Figure IV. Estimated Quantities of MOSE

Item No.	Nomenclature	Use Location: Transportation	Subcontractor plant	TRW plant	Remote test sites	JPL	AFETR	Quantity Required: 1971	1969
	Assembly, handling and shipping equipment (Flight spacecraft and overall flight spacecraft) ()SE/VS-3-140)								
3-140-1	Transporter, flight spacecraft	X						4	3*
3-140-2	Assembly, handling and tilt fixture			X		X	X	7	5*
3-140-3	Transport Recorder	X						4	3*
3-140-4	Fixture, weight, cg and MOI			X			X	2	2*
3-140-5	Shipping container group, standard modules	X	X	X	X	X	X	50	50*
3-140-6	Work platforms, mobile		X	X			X	7	5*
3-140-7	Adapter kit, Centaur/shroud transporter						X	2	NR
3-140-8	Sling assembly, planetary vehicle and nose fairing						X	2	NR
3-140-9	Purge unit, freon/ethylene oxide						X	2	NR
3-140-10	Planetary vehicle, nose fairing mating and assembly fixture						X	2	NR
3-140-11	Sling, flight capsule			X	X	X	X	2	NR
3-140-12	Hoist beam and slings, flight spacecraft		X	X	X	X	X	4	4
3-140-13	Tag lines						X	2	2*
3-140-14	Platform, launch stand access						X	2	2*
3-140-15	Universal mounting ring, flight spacecraft and planetary vehicle	X	X	X	X	X	X		
3-140-16	Environmental cover, flight spacecraft	X				X		4	3*
3-140-17	Hoist sling, environmental cover		X	X	X	X	X	4	[illegible]*
3-140-18	Platform auxiliary access			X	X	X	X	6	6*
3-140-19	Transporter adapter cradle, 1969 Test Spacec Spacecraft							NR	3

Notes: * = 1969 uses 1971 equipment as is or with removable mod kits
NR = Not required

Figure IV. Estimated Quantities of MOSE (Continued)

Item No.	Nomenclature	Use Location: Transportation	Subcontractor plant	TRW Plant	Remote test sites	JPL	AFETR	Quantity Required: 1971	1969
	Science payload sybsystem (OSE/VS-4-210)								
4-210-1	Alignment Fixture, science payload			X		X	X	4	NR
4-210-2	Shipping container, experiment booms	X						5	NR
	Communications and data handling subsystems (OSE/VS-4-310)								
4-310-1	Dolly 6 ft parabolic antenna			X		X	X	4	3*
4-310-2	Hoist beam, 6 ft parabolic antenna			X		X	X	4	3
4-310-3	Shipping container, 3 ft parabolic antenna	X						5	NR
4-310-4	Shipping container, 6 ft dish antenna	X						1	1*
4-310-5	Shipping container, Low-gain antenna	X						5	4*
4-310-6	Shipping container, flight capsule receiving anntenna	X X						5 5	NR NR
	Stabilization and control subsystem (OSE/VS-4-410)								
4-410-1	Alignment fixture, stabilization and control nozzles			X		X	X	4	3*
4-410-2	Protective covers, stabilization and control nozzles	X	X	X	X	X	X	28	24
	Power Subsystem (OSE/VS-4-460)								
4-460-1	Assembly and handling frame, solar panel segment	X	X	X	X	X	X	30	12
4-460-2	Protective covers, solar panel segments	X	X	X		X	X	30	12

Notes: * = 1969 uses 1971 equipment as is or with removable mod kits
NR = No required

Figure IV. Estimated Quantities of MOSE (Continued)

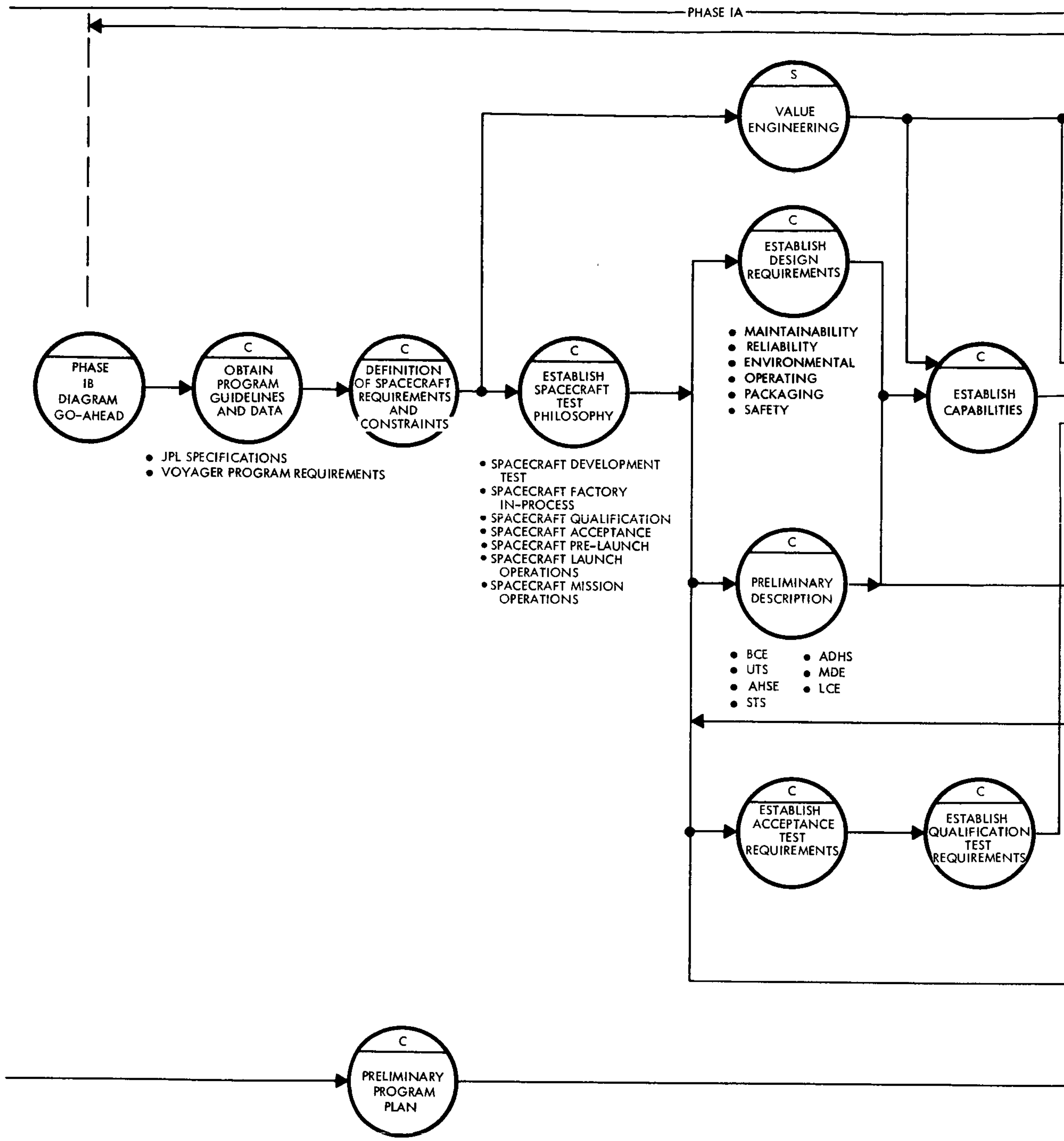
Item No.	Nomenclature	Use Location: Transportation	Subcontractor plant	TRW plant	Remote test sites	JPL	AFETR	Quantity Required: 1971	1969
	Power Subsystem (Continued)								
4-460-3	Shipping container, solar panel segments	X						15	6
4-460-4	Handling dolly, solar panel segments		X	X		X	X	18	8
4-460-5	Sling assembly, solar panel segment		X	X		X	X	6	5
4-460-6	Shipping container, battery	X						10	10*
4-460-7	Shipping container, power amplifier	X						2	2*
	Thermal control subsystem (OSE/VS-4-510)								
4-510-1	Assembly and handling fixture, spacecraft louvers		X	X		X	X	20	16*
4-510-2	Shipping containers, spacecraft louvers	X						5	4*
4-510-3	Handling and shipping container, insulation	X	X	X				4	3*
	Structural subsystem equipment (OSE/VS-4-520)								
4-520-1	Dolly, structural sections		X	X		X		4	3
4-520-2	Shipping containers, miscellaneous spacecraft structure	X						4	3
4-520-3	Sling, propulsion/pneumatic structural section		X	X	X		X	4	3
4-520-4	Interface match tool, spacecraft flight capsule		X	X				2	NR
4-520-5	Interface match tool, Spacecraft Centaur adapter		X	X				2	2
	Pyrotechnic subsystem (OSE/VS-4-530)								
4-530-1	Shipping container, explosive train	X						4	NR
4-530-2	Handling case, arming kit				X		X	2	2

Notes: * = 1969 uses 1971 equipment as is or with removable mod kits
NR = Not required

Figure IV. Estimated Quantities of MOSE (Continued)

Item	Nomenclature	Use Location: Transportation	Subcontractor plant	TRW plant	Remote test sites	JPL	AFETR	Quantity Required: 1971	1969
	Planet oriented package subsystem (OSE/VS-4-580)								
4-580-1	Assembly fixture and dolly, POP			X		X	X	4	NR
4-580-2	Shipping container, POP	X						2	NR
4-580-3	Hoist beam, POP			X		X	X	3	NR
	Propulsion subsystem (OSE/VS-4-610)								
4-610-1	Sling, retropropulsion motor		X	X		X	X	4	NR
4-610-2	Dolly, retropropulsion motor		X	X		X	X	4	NR
4-610-3	Alignment fixture, retropropulsion motor			X			X	4	NR
4-610-4	Alignment fixture, midcourse engine			X	X		X	4	4*
4-610-5	Shipping container, retropropulsion motor	X							NR
4-610-6	Shipping container, midcourse engine	X						2	2*
4-610-7	Pneumatics test set		X	X	X		X	3	2*
4-610-8	Pneumatic fill cart		X		X		X	3	2*
4-610-9	Propellant transfer and handling cart		X		X		X	3	2*
4-610-10	Alignment fixture, midcourse engine/steering vanes			X	X		X	4	4*
4-610-11	Universal handling fixture, hydrazine/helium tank		X	X			X	4	4*
4-610-12	Sling, hydrazine/helium tank		X	X			X	3	3*

Notes: * = 1969 uses 1971 equipment as is or with removable mod kits
NR = Not required



PHASE IA
S
VALUE ENGINEERING
C
ESTABLISH DESIGN REQUIREMENTS
• MAINTAINABILITY
• RELIABILITY
• ENVIRONMENTAL
• OPERATING
• PACKAGING
• SAFETY
C
ESTABLISH CAPABILITIES
PHASE IB DIAGRAM GO-AHEAD
C
OBTAIN PROGRAM GUIDELINES AND DATA
• JPL SPECIFICATIONS
• VOYAGER PROGRAM REQUIREMENTS
C
DEFINITION OF SPACECRAFT REQUIREMENTS AND CONSTRAINTS
C
ESTABLISH SPACECRAFT TEST PHILOSOPHY
• SPACECRAFT DEVELOPMENT TEST
• SPACECRAFT FACTORY IN-PROCESS
• SPACECRAFT QUALIFICATION
• SPACECRAFT ACCEPTANCE
• SPACECRAFT PRE-LAUNCH
• SPACECRAFT LAUNCH OPERATIONS
• SPACECRAFT MISSION OPERATIONS
C
PRELIMINARY DESCRIPTION
• BCE
• UTS
• AHSE
• STS
• ADHS
• MDE
• LCE
C
ESTABLISH ACCEPTANCE TEST REQUIREMENTS
C
ESTABLISH QUALIFICATION TEST REQUIREMENTS
C
PRELIMINARY PROGRAM PLAN

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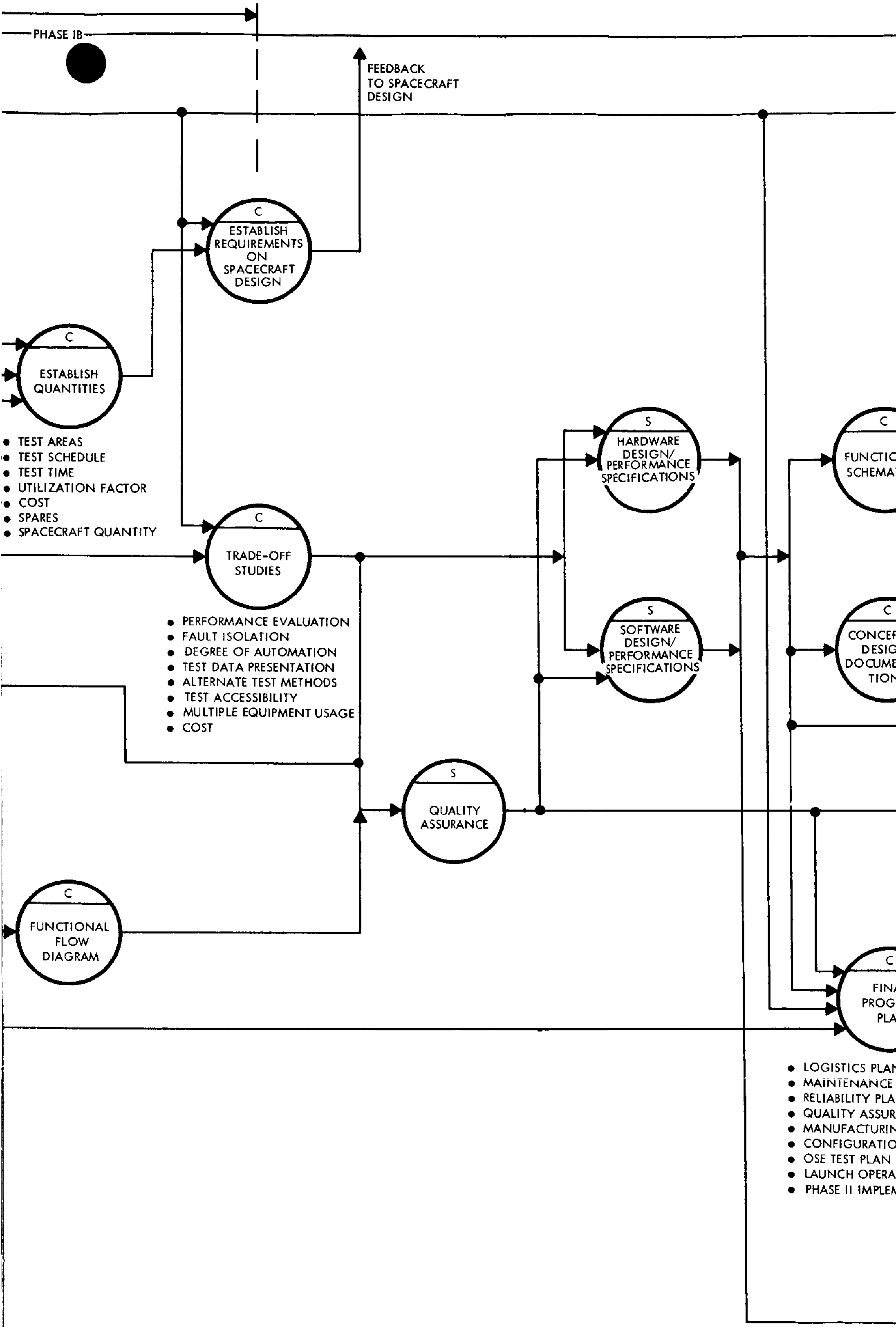

PHASE IB
FEEDBACK TO SPACECRAFT DESIGN
C
ESTABLISH REQUIREMENTS ON SPACECRAFT DESIGN
C
ESTABLISH QUANTITIES
• TEST AREAS
• TEST SCHEDULE
• TEST TIME
• UTILIZATION FACTOR
• COST
• SPARES
• SPACECRAFT QUANTITY
C
TRADE-OFF STUDIES
• PERFORMANCE EVALUATION
• FAULT ISOLATION
• DEGREE OF AUTOMATION
• TEST DATA PRESENTATION
• ALTERNATE TEST METHODS
• TEST ACCESSIBILITY
• MULTIPLE EQUIPMENT USAGE
• COST
S
HARDWARE DESIGN/ PERFORMANCE SPECIFICATIONS
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SOFTWARE DESIGN/ PERFORMANCE SPECIFICATIONS
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QUALITY ASSURANCE
C
FUNCTIONAL FLOW DIAGRAM
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• OSE TEST PLAN
• LAUNCH OPERA
• PHASE II IMPLEM

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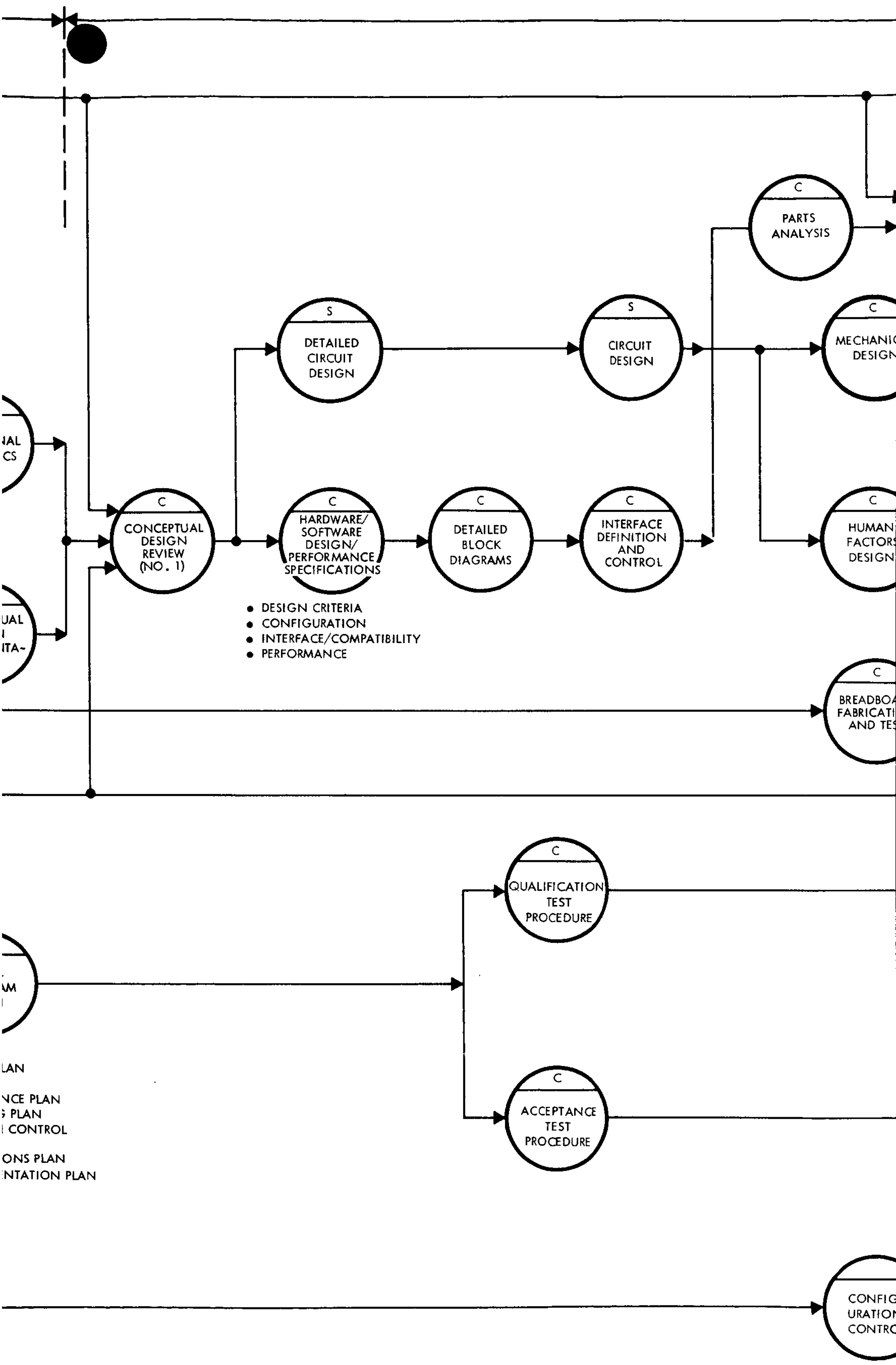
S
DETAILED CIRCUIT DESIGN
S
CIRCUIT DESIGN
C
PARTS ANALYSIS
C
CONCEPTUAL DESIGN REVIEW (NO. 1)
C
HARDWARE/ SOFTWARE DESIGN/ PERFORMANCE SPECIFICATIONS
C
DETAILED BLOCK DIAGRAMS
C
INTERFACE DEFINITION AND CONTROL
• DESIGN CRITERIA
• CONFIGURATION
• INTERFACE/COMPATIBILITY
• PERFORMANCE
C
QUALIFICATION TEST PROCEDURE
C
ACCEPTANCE TEST PROCEDURE

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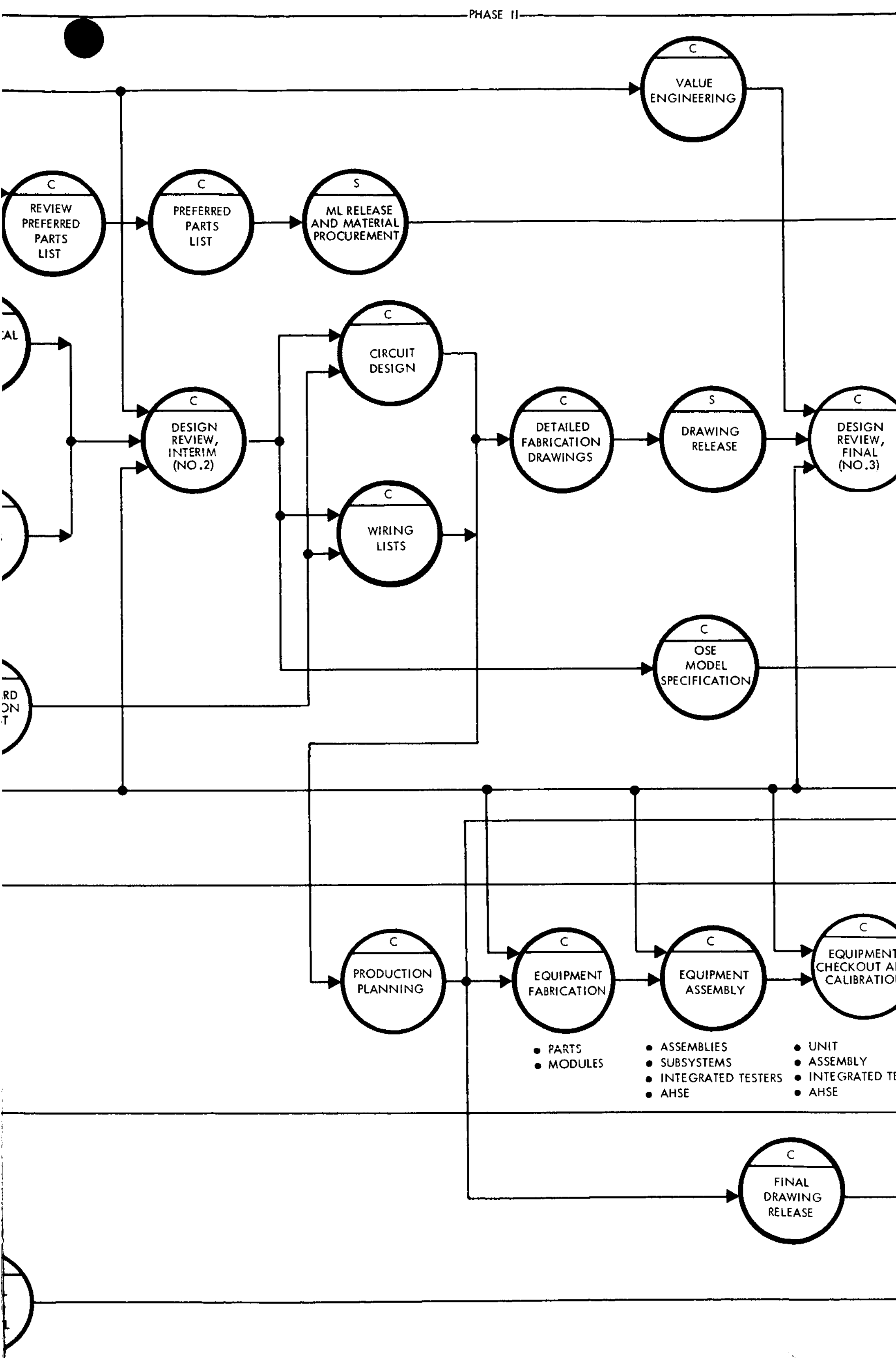
PHASE II
C
VALUE ENGINEERING
C
REVIEW PREFERRED PARTS LIST
C
PREFERRED PARTS LIST
S
ML RELEASE AND MATERIAL PROCUREMENT
C
CIRCUIT DESIGN
C
DESIGN REVIEW, INTERIM (NO.2)
C
DETAILED FABRICATION DRAWINGS
S
DRAWING RELEASE
C
DESIGN REVIEW, FINAL (NO.3)
C
WIRING LISTS
C
OSE MODEL SPECIFICATION
C
PRODUCTION PLANNING
C
EQUIPMENT FABRICATION
C
EQUIPMENT ASSEMBLY
C
EQUIPMENT CHECKOUT A
CALIBRATIO
• PARTS
• MODULES
• ASSEMBLIES
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C
FINAL DRAWING RELEASE

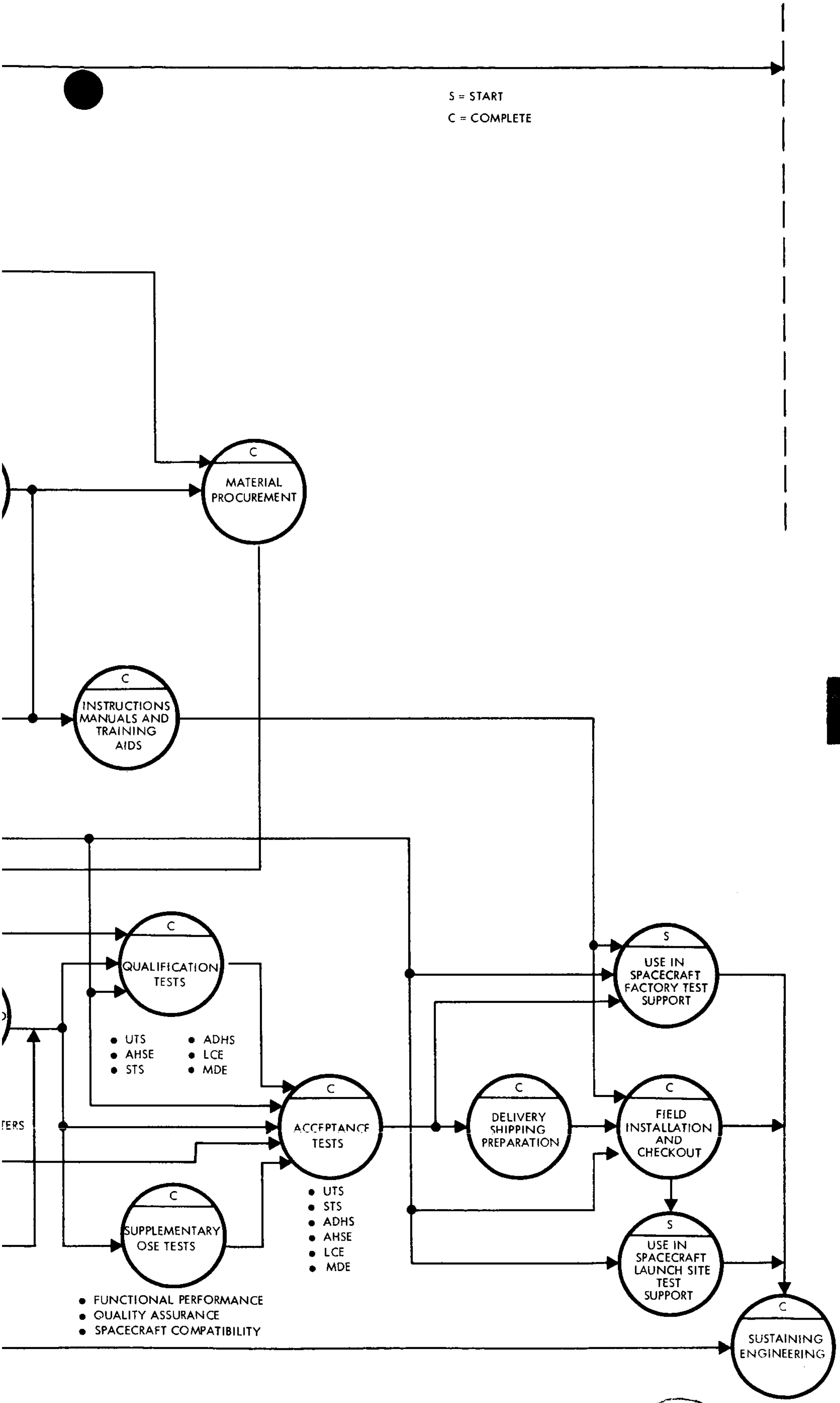

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Figure 19. Voyager Development Program Network (OSE)

3. IMPLEMENTATION PLAN, PHASES IB AND II

3.1 Analysis

3.1.1 Requirements and Spacecraft Test Philosophy

The initial OSE requirements analysis have been accomplished during the Phase IA effort of the Voyager program. Based upon the established program guidelines, gound rule data, specifications from JPL, and spacecraft preliminary design, a definition of the spacecraft requirements and constraints on the OSE and the OSE impact on the spacecraft system design has been partially completed, as described in the preceding sections of this volume. Final work is to be accomplished on this analysis during Phase IB and continued as an iterative process, certain groups of tasks being recycled through completion of spacecraft testing work of Phase II.

Predicated on these same program guidelines and spacecraft overall test requirements and constraints information, a spacecraft preliminary test philosophy has been established. This test philosophy specifies in general what data is to be acquired at each level and geographical location of spacecraft test, how the resulting information is to be handled, and also defines the test techniques to be applied. Final establishment of the test philosophy with required details will be accomplished in Phase IB.

For each item of the assembly, handling and shipping equipment (AHSE), unit test sets (UTS), system test sets (STS), automatic data handling system (ADHS), launch complex equipment (LCE), and mission dependent equipment (MDE) comprising the OSE, the identification and analysis of requirements placed upon this equipment will be completed at initiation of the program.

Input data required to develop the performance requirements, as well as the specific usage of this data, will be finalized for the following tests:

a) Spacecraft design/development tests - Tests performed to evaluate the adequacy of design of individual spacecraft components prior to their release for fabrication.

b) Spacecraft factory in-process and assembly tests - Tests performed during the fabrication process on subunits, units, and subassemblies (panels) as well as during the spacecraft assembly and integration phases. The spacecraft/OSE compatibility tests are included.

c) Spacecraft qualification tests - Tests which stress the spacecraft beyond its design limits performed to prove the margin of safety in the spacecraft design. Additional flexibility and range extension is provided in the OSE to support these tests.

d) Spacecraft acceptance tests - Functional tests performed on the spacecraft units and integrated system for the purpose of proving adequate functional performance under flight environmental conditions.

e) Spacecraft prelaunch field tests - Tests of an assembly and acceptance nature performed in the launch area at the spacecraft assembly building; the explosive safe facility and at the launch pad for the purpose of further evaluating functional integrity of the spacecraft following shipment handling.

f) Spacecraft launch operations - Tests performed during the countdown operations and actual launch control for the purpose of evluating immediate prelaunch spacecraft integrity and proper launch sequence.

g) Spacecraft mission operations - Tests imposed on the spacecraft during flight for the purpose of evaluation of operationsl status or command into troubleshooting or diagnostic modes if applicable.

The established program guidelines and spacecraft test require ments and constraints will provide the basis for completion of definition on the following general requirements placed on the OSE design.

a. Maintainability

The capability for rapid and easy adjustment, repair and replacement in the OSE will be attained through modular construction and standardization of panel design. Maintainability will be enhanced through self-test capability, multiple use of circuits, and through adequate human factors design. A minimum mean time to repair will be achieved through these methods.

b. Reliability

Maximum reliability within the constraints of cost will be achieved in the OSE through design simplicity in incorporating a minimum of components, minimum component complexity for a given function, proven and time-tested techniques, circuits, and components, component derating of power, temperature, voltage, humidity, etc., and through the use of redundancy of circuits and components when this is established to be required.

c. Environmental

The environmental conditions incurred in OSE transportation usually define the limiting requirements for checkout equipment design. Considerations will include also the temperature, humidity, pressure, shock, vibration, electromagnetic interference (EMI) and magnetic field interference at each of the operational locations, as well as those for transportation. For example, some of the OSE will be subjected to extended thermal/vacuum, vibration, etc., along with the spacecraft and must be suitably designed to withstand these environments.

d. Human Factors

In addition to the maintainability requirements which invoke human factors design, a prime factor is operability in order that maximum communication between man and equipment may be achieved for most satisfactory test control and monitor. The human factors design will provide equipment which is easy and convenient to use.

e. Packaging

Other than the factor of maintainability, primary packaging concepts will be determined by transportability and that which gives most satisfactory functional performance. Considerations of electrical noise, cross modulation, EMI and magnetic field interference are paramount in packaging decisions. The OSE must also be convenient in use and aesthetic in appearance.

f. Safety

As given in the OSE design criteria, Section I of this volume, the factor of safety in use and handling at all times is also paramount. Specific requirements for safety will be established early in the program.

The preliminary functional description of the OSE provided in Phase IA will be updated at this time with identification of inherent capabilies beyond the specific limits required, this factor having a direct bearing on the quantities of supporting equipment required on the program.

Functional flow diagrams to the next level of detail will be prepared for the OSE in Phase IB which will provide additional data on the following:

a) Identification of the OSE system functions and provision of an index for the complete set of system requirements documented

b) Emphasis on essential features of the system operation

c) Interrelationships between system functions

d) Indication of the sequential nature of relationships between system functions

e) Non-sequential but prerequisite relationships between functions

f) Identification of discrete events which may be required during the performance time of a given function.

The Voyager system factory-to-launch functional flow diagrams will be restudied and reworked in the light of the chosen spacecraft design. OSE tasks to be performed throughout the whole sequence of events will be defined. For each box in the functional flow chart, a requirements allocation form (RAF) will be written. On the RAF the functions occurring at that point will be described and the required EOSE and MOSE will be listed.

The requirements for the acceptance testing of the OSE will be established following refinement of program guidelines and firming of the spacecraft test philosophy. This will be followed by generation of OSE qualification test requirements.

Handling envelopes and constraints resulting from the configuration of the chosen spacecraft will be identified. These will range from the handling requirements of the smallest components to the handling and movement of the completely assembled spacecraft. Variations in the assembly and handling sequences and the resulting changes in the MOSE will be recorded for tradeoff study and analysis.

MOSE conceptual designs from the Phase 1A Study will be reviewed for changes resulting from the chosen spacecraft. Changes in handling envelopes, fixed-handling hardpoints, etc., will be incorporated in the conceptual descriptions and designs.

For each item of MOSE listed as required specific design requirements will be developed. These will include, for example, fabrication material requirements, weights and volumes to be accommodated, specific spacecraft hardpoints to be supported, mobility requirements, maximum g forces to be attenuated to specified vibration time history spacecraft criteria, electrical power, fluid requirements, pressures, and environmental considerations.

Environmental constraints on operation of the OSE or on the spacecraft will be identified and these factors considered in the OSE design requirements. For example, if an item of MOSE must support the spacecraft in a hard vacuum test environment, the design of the MOSE must consider outgassing of materials, need for contamination control, and the means for routing of MOSE utilities into the hard vacuum. At the launch area at ETR, the effects of a moist salt-laden atmosphere must be considered. Within transportation MOSE design, the heat, cold, and humidity of the transport mode and its loading areas must be evaluated for its deleterious effect on the contained spacecraft or components.

Facility restraints on OSE operation at the DAC manufacturing areas, the TRW assembly and checkout areas, and the ETR receipt, checkout and launch facilities will be evaluated and incorporated in the design requirements. These constraints will range in a broad spectrum from such items as available crane capacities, hook heights, and operating speeds to road surfaces, ramp grades, and turn radii. The normally available industrial electrical power, its regulation and grounding will be utilized in OSE design, unless the spacecraft design imposes a different or more specific electrical requirement. Facility areas of uncontrolled environment will require that the MOSE provide protection to the spacecraft. Each function performed within a facility area will be scrutinized to insure that the OSE design will fulfill its supporting function with no degradation to the spacecraft system.

3.1.2 Tradeoff Studies

In the interest of providing OSE which will efficiently accomplish the spacecraft ground test objectives within schedule and at the lowest possible cost, several tradeoff studies are required to be performed. These have been started in Phase 1A and will be continued and completed during the Phase 1B effort. It is important that the OSE be considered an integral portion of the over-all Voyager system in that OSE functional requirements and results of tradeoff studies will have some explicit effect on the spacecraft design, particularly to obtain maximum spacecraft evaluation information at lowest program cost. The following factors are included in the tradeoff decisions being made, as discussed further in Section VI.1.

a) 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, constituting a tradeoff between current and future performance confidence and spacecraft statistical reliability.

b) Level of fault isolation - Determination of specific level of fault isolation capability. This is a tradeoff between test data point availability, maintenance philosophy, and EOSE capability.

c) Degree of automation - Determination of the degree to which the test operations of program control, stimuli generation, and data handling. 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.

d) Method of test data presentation - Determination of the optimum method of test data presentation. This involves 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.

e) Alternate test methods - The various methods of acquiring data from the item under test such as single stimuli, transfer functional analysis, and continuous monitor, each under nominal or marginal testing conditions, are among the alternative test methods which may be employed.

f) Test accessibility of spacecraft subsystems and units - A tradeoff must be employed 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.

g) Multiple usage of equipment - A tradeoff study will be completed in evaluation of the quantity of OSE required as a function of the quantity, schedule, and location of the spacecraft test operations with the intent of minimizing the OSE equipment quantity through multiple usage of this equipment. This will include consideration of sequential time sharing, ADHS multiplexing, effect on equipment automation, and refurbishing equipment for use on subsequent spacecraft models. The impact of spacecraft schedule and, in particular, the launch window impact on OSE requirements, forms a major part of this tradeoff.

h) Test philosophy differences at system, subsystem, and unit level - Each of the tradeoff studies listed above must be considered specifically for testing at the integrated system, subsystem, and unit levels in that requirements at these levels may be substantially different.

i) OSE cost - A major overriding factor in the OSE tradeoffs to be completed during Phase IB is that of OSE 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 which have been discussed above. Value Engineering check points will be made at several stages during the development of the OSE, including the establishment of equipment capabilities, quantities, the OSE requirements on spacecraft design during the OSE final program plan generation, and during the design reviews discussed below.

3.1.3 Equipment Quantities

The next phase of the program involves the final determination of OSE quantity needs. Predicted on the final schedule and fabrication quantity of spacecraft as well as the test philosophy, the quantity of supporting equipment for each of the areas of test listed in paragraph 3.1.1 will be changed from that established during Phase IA if necessary. The following considerations are paramount in making OSE quantity decisions.

a. Test Area Equipment Requirements

Separate equipment is required to support each of the geographical test areas of factory assembly, environmental test, magnetic test site, launch complex assembly, and the launch site.

b. Test Area Schedule

The OSE quantity is a function of the schedule applicable to each of the geographical test areas and the time-sharing or sequential testing possibilities which may be applied to these areas.

c. Spacecraft Subsystem and Unit Quantities

The specific quantities of spacecraft equipment is a prime factor in determination of the OSE quantity and has a direct bearing on the test area schedule. Initial results of a Phase IA study indicates that, in some cases, more than one test set of a specific type is required to support spacecraft equipment checkout at a given geographical location at one time.

d. Spacecraft Subsystem and Unit Test Time

The test time spans required on individual spacecraft subsystem and unit equipment have a direct bearing on OSE quantities. The test time is a function of equipment set up, response time of the equipment under test, OSE degree of automation, and equipment tear down elapsed time.

e. Utilization Factor

OSE equipment quantities are dependent upon the percentage utilization factor of individual items. The higher this factor, the fewer the quantity of specific items. Through proper spacecraft equipment test scheduling, maximum OSE operability and sufficient flexibility and mobility of the OSE, the utilization factor may be increased.

f. Costs

One result of the cost tradeoff studies performed is the consideration that an additional test station is justified if a cost for such a station is less than that incurred by an increase in manpower or material costs through a delay in spacecraft scehdule testing operations.

g. Spares

Spare requirements are determined primarily through the maintenance philosophy and plan. The spare requirement is a function of the packaging techniques, and through module replacement capability, a reduction in equipment downtime is achieved. From the above analysis of the functional flow diagrams, OSE end item requirements lists will be developed. These lists will be broken down to generic groupings. The total allocation for each OSE item, its location, and whether it will be shared by two or more sites will be established. See Table III.

3.1.4 Support Planning

The OSE program plan will be finalized following the completion of requirements identification and concurrent with the generation of the equipment hardware and software specifications. This plan will identify all aspects of the OSE program and will be composed of the following separate subordinate plans:

a) Logistic Plan - to include a definition of the spares requirement and training needs for use of the OSE at each of the geographical locations of test.

b) Maintenance Plan - definition of the maintenance activities required to maximize the OSE up-time.

c) Reliability Plan - definition of the methods employed in achieving maximum OSE reliability through proper design, parts application, and equipment utilization and handling techniques.

d) Quality Assurance Plan - to cover the requirements for Quality Assurance activities at each major OSE development phase from specification through design review, fabrication, and testing of the OSE hardware.

e) Manufacturing Plan - indicating the manufacturing process for the OSE from fabrication planning through material procurement through the fabrication steps to equipment assembly and preparation for checkout and calibration.

f) Configuration Control - stating the techniques and methods for establishing and maintaining OSE configuration, use of configuration control documentation, integration engineering, and communication of this information to cognizant personnel utilizing computer tab run techniques.

g) OSE Test Program Plan - specification of the work to be accomplished in development testing, fabrication checkout, qualification test, and acceptance tests. The purpose of these tests is to establish that the OSE is operating properly and is capable of evaluating the spacecraft performance status, and in case of malfunction, the malfunction detection. The test program will also determine the OSE capabilities of spacecraft fault isolation to a replaceable module.

h) Launch Operations - to include the installation of OSE at the field site, the prelaunch evaluation tests, and the actual launch activity.

3.2 Design

The work to be accomplished under the OSE design includes the firming of performance/design specifications, the system design tasks and the detailed equipment design work culminating in design drawings and reports.

3.2.1 Performance Design Requirement Specifications and Preliminary Design.

Further identification of each type of OSE required and the generation of updated design/performance requirement specifications for both hardware and software at the system and detail equipment levels will be accomplished. The formality and depth of detail of the specifications will be of a minimum consistent with the program requirements and the need for efficient equipment design and procurement operations.

a. Design Criteria

The ground rules and other criteria to be used by the design engineer will be stated explicity for each item of equipment for use at each spacecraft level of test.

OSE end item design criteria will be developed from the OSE design requirement documents. The purpose of the design criteria is to provide definitive controllable documents for each OSE item, so that design reviews can be made and customer approval can be obtained. 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 OSE. The purpose and usage of the specific item will be covered. The fixed spacecraft or spacecraft components restraints and requirements placed on the MOSE will be listed or shown in dimensioned drawing form. Applicable sections from standard or military specifications pertaining thereto will be listed. Spacecraft power and utility requirements affecting the OSE design will be listed as to power, type, regulation, total load, reserve factors, and duration of usage. The document must be an entity to eliminate inaccuracies and time lost in inefficient cross-referencing.

The Design Criteria will be in concurrence with the decisions made during the requirements analysis tradeoff studies and as defined in the support plans for maintenance, reliability, Quality Assurance and test.

b. Configuration

The specification will define the preliminary end item configurations as a guide to the design engineer in the interest of equipment compatibility and standardization. The end item configuration will be consistent with the requirement established during the analysis and with those defined in the maintenance and configuration control planning documents.

c. Spacecraft - OSE Compatibility

The requirements for compatibility between spacecraft and the supporting equipment will be defined in detail in conjunction with recommendation of the configuration control plan.

A preliminary design for each individual OSE item will be prepared by specification on the Design Form format. Use of the Design Form will permit evaluation of the design at an early stage by the TRW Voyager team and JPL. Any changes required can be efficiently accomplished at this stage prior to initiation of detailed design drawings. The Design Form will provide a standard means for controlled approval of individual OSE items and for recording all changes required by all agencies involved. It will also provide a source of basic OSE design requirements information for use at First Article Inspections for compliance to specifications. Following approval for initiation of design by the TRW Voyager team and JPL, the Design Forms would be subject to Configuration Control Board action for any change request.

A dynamics analysis study will also be initiated at this point to estimate ground handling loads based on part experience with similar items, to calculate spacecraft and MOSE restrictive load factors at various operating frequencies, and to estimate equivalent static loads. During this phase, the dynamics personnel will work closely with the structural analysts to provide the MOSE designers with realistic structural-member sizing for the conceputal MOSE designs.

A preliminary strength analysis estimate of the interaction of the spacecraft structure and the MOSE structure will be prepared. This analysis will be based on the spacecraft being primarily designed for flight loads, not handling loads. Inputs from the spacecraft and MOSE dynamics analysis will be considered. Where required, allowable constraints and loads into the spacecraft attach points will be determined and specified. The spring rates of planned MOSE will be calculated. A transportation MOSE design criteria will be written covering maximum g levels, static and dynamic environments, etc.

3.2.2 Detailed Design

Design work in the subordinate OSE areas of UTS, STS, ADHS, LCE, and MDE will be accomplished following completion of detailed specifications. The following sequence of design tasks will be performed in the development of subsystem designs.

OSE design layout drawings will be prepared for each item of OSE that had been outlined in the design criteria and scoped in the Design Forms. These preliminary drawings accommodating all constraints and fixed spacecraft dimensions will be prepared and issued for TRW Voyager Team review, coordination, and preliminary approval. A major purpose of the preliminary drawings will be to act as a media for obtaining a "meeting of the minds" on each OSE item and to enable interface checking on inter acting OSE items. Following review, final detailed design layout drawings will be produced.

Standard GFE components adaptable to the MOSE design will be selected and incorporated where possible. Use of such items will increase the cost effectiveness of the total Voyager OSE procurement program.

Alternate design solutions to problem areas will be developed as a means to evaluate the optimum approach. Each solution will undergo detailed review and tradeoff analysis as to its fulfillment of the design criteria. In order to minimize the design effort time, judicious use of previously designed circuitry or complete units from previous TRW Systems, Douglas Aircraft Corp., and Radio Corporation of America programs will be included in the design to the maximum extent possible.

The use, where practicable, of general purpose type commercial equipment and the application of developed and proven techniques OSE will keep program costs at a minimum and assure compliance with delivery schedules.

The detailed activities will proceed under the direction of an assigned OSE Work Package Manager (WPM). He will review the requirement on the OSE with the Spacecraft subsystem engineers for compatibility during the design activity. It will be the WPM's responsibility to follow the design through completion and acceptance test.

The detailed electrical and mechanical design of the OSE will be accomplished to produce equipment capable of meeting the functional requirements established. This design will include the following specific activities:

a) Functional Drawings - OSE functional drawings will be prepared and supplemented by functional descriptions as required in order to document the conceptual design in preparation for Design Review No. 1 held on the OSE. Documentation will be sufficient to identify the equipment plan and outline the functional capabilities, and how these satisfy the established spacecraft test requirements. Based on the final layout drawings for each individual item of OSE, component specifications and request proposals will be prepared where required and issued to vendors. The vendor information to be received on cost, fabrication problems, delivery schedules, etc., will provide a realistic means to evaluate the OSE design as proposed. The vendor proposals will be evaluated. From this review specific components and vendors will be selected for use in the final OSE design and fabrication. Design specification drawings for each of the selected MOSE items will list in detail all design requirements for spacecraft envelope, weight, and maneuvering; dynamics and strength criteria; fabrication material, constraints (nonmagnetic, outgassing, etc.); total weight allowable; shock attenuation requirements; proof testing test program for acceptance testing; and other pertinent design criteria.

b) Development Testing - Development testing activities in the OSE program will have been started during Phase IB but will be limited to critical areas not heretofore evaluated by members of the TRW Voyager team. Development testing will be accomplished during the OSE design phase and will include performance, environmental, and reliability type tests.

c) Mechanical Design Mechanical design, including rack and panel layout, interconnecting cabling, and handling equipment will be accomplished, including human factors design to assure maximum man/machine communication and test control efficiency. Subsystem layouts of the individual approved OSE items will be completed and reviewed by the TRW Voyager team. Approval of the design shown will be obtained for each OSE item.

d) Model Specification Preparation - A final specification will be prepared which fully defines the end item configuration of the OSE to include interface definition and control, both internal and external, for compatibility with spacecraft and other interfacing equipment.

e) Test Procedures Generation - Detailed Test Procedures will be generated for the OSE Qualification Teat and Acceptance Tests to be run on equipment following fabrication. Completion of test to these procedures, written to satisfy the requirements stated in the Test Program Plan, will provide assurance that the OSE is in the proper status for connection to the spacecraft or its subassemblies with no deleterious effect to the spacecraft.

f) Design Reviews - The following three design reviews will be accomplished during the OSE program:

- Conceptual Design Review (No. 1), held following definition of OSE functional requirements and the generation of OSO functional block diagrams schematics are layout drawings to evaluate the support equipment concept in the fulfillment of spacecraft test and handling requirements.

- The Interim Design Review (No. 2) will be held following a major portion of the electrical and mechanical design and test activity.

- The Final Design Review (No. 3) will be held following completion of fabrication drawings and preliminary release for which the completion of OSE design and development testing are requirements.

Design documentation will be prepared in support of Interim Design Review No. 2 and the Final Design Review No. 3 as required to adequately describe the OSE design status at these phases. Requirements for changes resulting from design review activity will be reflected in OSE design drawings and documentation following the reviews.

g) Advanced Material Procurement - Soon after the start of OSE design, a parts analysis will proceed, involving the review of Preferred Parts Lists for application of these parts to the OSE. This work is done in consonance with value engineering techniques, and is accomplished prior to release of fabrication drawings and standard material procurement. Advanced material procurement is limited to that approved by JPL and includes only that equipment of low risk use and long lead procurement.

- Strength analysis calculations will be completed for each MOSE design to provide final member sizing and to verify that all design loads have been accommodated. A final analysis of the interaction of the spacecraft structure and the MOSE structure will be completed. The results of these analyses will be fed into the final detail design drawings.

- An updating of the dynamic analysis calculations for each MOSE design will be performed. In addition, actual handling and transportation tests would be run using the actual MOSE and a spacecraft model with the MOSE and spacecraft instrumented with accelerometers and deflection gauges as required.

3.3 OSE Manufacture

Electrical and mechanical manufacturing procedures will be accomplished following both preliminary and final release of design and vendor control drawings. The vendor control drawings will outline in detail the components required and will contain all applicable specifications controlling fabrication, manufacture, quality control, and documentation. The OSE manufacture will be accomplished in the fabrication facilities of the members of the Voyager plan in conjunction with standard Quality Assurance requirements. In-process tests will be accomplished where desirable for efficient evaluation of the fabrication process integrity. Test equipment required to support the 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 OSE manufacture.

a. Drawing Release

Preliminary and final drawing release phases will be accomplished prior to and following Design Review No. 3, respectively. Advanced release, as previously stated, is necessary on some equipment to preclude schedule delays in material procurements and fabrication. Drawing release for fabrication constitutes that point in the design where formal approval of the Voyager Program Office and JPL is required for any subsequent engineering changes.

b. Material Procurement

The final material procurement activities will follow design review No. 3 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.

c. Production Planning

Production planning will be initiated before the release of drawings and will be detailed in the Manufacturing Plan. Since it is planned that fabrication to accomplished under the cognizance of the engineers who are responsible for the design, fabrication requirements may be determined well ahead of the drawing release dates. A compression of fabrication schedule is imperative in that OSE is required by the time Voyager flight vehicle equipment is ready for test even though OSE design completion must await vehicle design firming. 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.

d. Detailed Part Fabrication

Individual subunits 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.

e. Subassembly Fabrication

The fabrication of units will be scheduled such that those units required for testing of other or subsequent units will be fabricated first. During integration of OSE units into subassemblies, each functional group will be examined for the presence of interference as evidenced by performance degradation. In-process tests and inspections will be employed as required to validate this integration, including electrical/mechanical interfaces and any new interface problems becoming evident at this time will be corrected.

f. Major Assembly Fabrication

Units and subassemblies will be integrated along with commercial general purpose equipment into specific OSE items with in-process tests and inspections employed as required. Integration and interface testing will be accomplished during this activity to verify proper functional operation of constituent assemblies and no overall performance degradation as before. Following fabrication of assemblies, the final debugging operations and proving of the fabrication integrity will be accomplished on completely assembled equipment. Any final manufacturing errors will be corrected and the equipment will be adjusted and calibrated in preparation for the qualification and acceptance tests to follow.

g. Configuration Control

The configuration control procedures as specified in the Configuration Control Plan will be employed to provide a continuum of up to date documentation and information dissemination.

h. OSE 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 OSE. The documentation will be the minimum required to efficiently utilize the OSE and to meet program specified requirements.

3.4 Testing

A complete acceptance performance test will be conducted on each OSE assembly before it is connected to a unit of the spacecraft to ensure that no damage to development or flight hardware can occur. The performance test will be conducted in accordance with a detailed test procedure 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 of operation such as high and low line voltage and ambient temperature. This test will be conducted on component (chassis) level as well as on equipments. The accuracy of self test loops will be demonstrated and interface impedances measured. The adequacy of protective fail-safe circuits will be verified.

Following the performance test, for those items which will be used in proximity to the test piece during magnetic field intensity measurements, a magnetic test will be performed. The test will determine the OSE magnetic field intensity at a prescribed distance to verify it is at an acceptable low level.

Environmental tests appropriate to the particular OSE 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.

The following is additional information on the types of tests to which the OSE will be subjected.

3.4.1 Type Approval Test

Certain OSE will be subjected to type approval tests in accordance with the test requirements established on the basis of established program guidelines. The tests will be accomplished through a JPL approved type approval test procedure and the equipment will be subjected to extended environmental stress during the test. The OSE operation and its capability to withstand the rigors of these environments will be evaluated prior to, during, and following the test activity through measurements of its functional performance.

All Voyager MOSE will be programmed through a type approval test series. The purpose of the type approval testing is to verify that the MOSE meets the design specifications. For certain portions of the tests planned, mass and c.g. models of the component or spacecraft will be required. Transport type equipment will be subjected to road tests and will be instrumented to record to the environment present during spacecraft transportation.

3.4.2 Functional Performance

Functional performance tests will be accomplished on each element of the OSE as required at various stages during the manufacturing, integration through acceptance and following adjustment calibration or modification to verify that the OSE is suitable for formal test or utilization with the spacecraft.

3.4.3 Spacecraft—OSE Compatibility

The compatibility tests are accomplished following fabrication of the OSE and following subsequent modification for the purpose of verifying electrical and mechanical compatibility with the spacecraft and interfacing equipment. This is done to verify interface connections and ability of the OSE to control and monitor the spacecraft, as well as to protect against possible degradation to the spacecraft. The first tests are accomplished with simulated spacecraft models and circuits, followed by connection to the spacecraft in planned steps.

3.4.4 Acceptance Test

Each of the fabricated OSE Plans will be subjected to acceptance tests, including those which have undergone qualification tests. Acceptance tests will be performed in accordance with the established test plan, as defined by the program guidelines, and will be run according to a JPL approved acceptance test procedures. Acceptance test will evaluate the final operating integrity of the OSE prior to use with the spacecraft and other interfacing equipment.

3.4.5 Quality Assurance

In-process tests will be accomplished on subunits of the OSE during manufacture and on assemblies of these items in concurrence with Quality Assurance requirements of the TRW Systems. Through these tests confidence is acquired that equipment has been fabricated to quality standards and that no manufacturing defects exists.

During manufacture, production tests on MOSE parts and components will include sheet material thickness checks, visual inspections, check of fabricated parts against the design drawing, and conduct of any testing, such as, proof pressure, leak, flow, or load, that might be listed on the design drawing. Magnetic cleanliness will also be checked.

3.4.6 Test Data

During progress of each of the tests imposed on the OSE, test data will be acquired and recorded for review necessary to the evaluation of OSE status. This test data will be retained for later comparative analysis and reliability evaluation. Test data will be taken by equipment integrated into the OSE and by supplementary TRW Systems capital equipment connected to and calibrated with the OSE.

3.4.7 Delivery, Storage and Shipping

Delivery of the OSE will be made to JPL at TRW Systems following formal acceptance tests. Packing of this equipment for storage or shipping will be in accordance with approved procedures of TRW Systems under the cognizance of Quality Assurance. The OSE will be prepared for the environmentals to which it will be subjected during storage and shipping. For example, heavy equipment will be removed from cabinet-racks and separately stowed. Shipping is planned via air-ride vans for maximum electronic equipment safeguard.

3.4.8 Configuration Control

As defined in the OSE configuration control plan, any incompatibilities or anomalies detected during any of the OSE test activities will be corrected in hardware and software, and the documentation changes will be followed up with adequate information dissemination.

3.5 Sustaining Engineering

3.5.1 OSE Utilization

a) Use in Spacecraft Testing, Factory - The OSE of the subsystem and system test levels will be utilized in support of factory spacecraft testing operations. The development testing, fabrication, in-process, integration, qualification, and acceptance testing operations on the spacecraft will be supported by this equipment.

b) Field Installation - The OSE required to support field test and launch operations of the spacecraft will be shipped to the field and installed in place sometime prior to the launch activity. Following installation, the equipment will be calibrated and retested in accordance with the established procedures to verify that no degrading situation has developed during shipping and handling of the launch complex equipment. The field spacecraft tests following final assembly, the ESF, and the on-stand flight readiness evaluation, and the launch control operations are included in this support.

3.5.2 Equipment Modification Support

As a direct consequence of changes to the spacecraft, modifications will be required to be made to the OSE at each of the levels of test. The modification process will follow this OSE implementation plan in similar sequence, but necessarily of abbreviated scope.

Following a definition of OSE modification requirements, an engineering change order will be written followed by detailed design effort on the change. A modification kit will be procured through procurement of parts and fabrication of the required equipment to quality assurance standards. The completed kit will be shipped to the OSE location for installation onto 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 procedure for checkout and acceptance of the revised equipment will be prepared.

Following the 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 OSE test records.

Sustaining engineering activities on the Voyager OSE will continue the cognizance of quality assurance, as have the activities prior to delivery of the equipment. Configuration Control will be maintained in accordance with the configuration control plan, in all activities involving the OSE following equipment delivery to the customer.

It is anticipated that the OSE flexibility and capability will be sufficient to preclude modifications required by the majority of the possible changes to the spacecraft design.

APPENDIX A

VOYAGER AUTOMATIC DATA HANDLING EQUIPMENT

1. INTRODUCTION

Telemetry data, containing both experiment and engineering information, will be processed by the OSE in order to present information in the manner desired. For Voyager OSE, a computer was selected to accomplish the processing task in lieu of a special purpose equipment configuration because it provides the following:

- Capability of accepting variable telemetry formats without a hardware change, since all format interpretation is determined by the computer program rather than specialized hardware.
- Capability for data analysis, such as limit checking, resulting in reduced spacecraft checkout time. (Since computers can perform repetitive operations at a much higher rate and with fewer errors than human operators, this increased accuracy is available through elimination of human errors.)
- Capability of automatic telemetry format recognition in lieu of patchboard approach used in special purpose configuration.
- Automatic visual presentation and printout of experiment test data and data processing results in real time. The printout may consist of random word groupings or words within a subcom.

Additional advantages of a computer include:

- Use of automatic test sequencing
- Verification of commands
- Control of generation of commands. (Computer-controlled Command Encoder allows for automatic closed loop testing and automatic generation of any sequence of command transmission per computer program.)

2. GENERAL COMPUTER FUNCTIONS

A computer will be utilized as the central core around which the Automatic Data Handling System (ADHS) will be developed. The functions of the computer will include, but are not necessarily limited to, those outlined in the following paragraphs.

2.1 Real Time Processing of Telemetry Data

a) Formatting engineering data so that a subsystem specialist can readily locate and analyze the data pertaining to his subsystem on a hard-copy printout

b) Converting telemetry data to engineering units for presentation on hard-copy and other types of display

c) Performing scaling and curve-fitting calibrations

d) Performing limit checks, flagging and fully identifying all data which exceeds the preset limits

e) Periodical updating various displays throughout the test area

f) Performing event monitoring and activating appropriate alarms when indications of dangerous conditions occur

g) Processing experiment data and providing hard-copy output in a form readily comprehensible to knowledgeable experiment personnel.

2.2 Test Sequencing

a) Automatic cycling through established and documented test sequences, being susceptible to easy alteration of the order in which sequences are performed

b) Selecting inputs to data display and recording equipment in accordance with documented test procedures

c) Identifying data by time, source, and test sequence number

d) Possibly controling command generation to the spacecraft in accordance with documented test procedures

e) Possibly performing command verification.

2.3 Off-Line Data Analysis

a) Performing data reduction and analysis, on an off-line basis, of magnetic tapes containing raw telemetry data as well as tapes containing partially pre-processed data

b) Formatting data for recording on magnetic tapes compatible with IBM 729 Model II tape units.

3. INPUT DATA

Input data to the computer will come from several sources including, but not necessarily limited to, the following:

3.1 Paper Tape Reader

A standard paper tape reader will be included with the computer for program generation and debugging, performance of diagnostic and periodic maintenance routines, and other general off-line uses.

3.2 Punched Card Reader

Although immediate inclusion of punched card equipment in the computer complex may not be required, eventual inclusion will be anticipated for such off-line uses as program tape generation, test procedure generation, and possibly limited data reduction and analysis.

3.3 Typewriter

A standard input/output (I/0) typewriter will be included in the computer complex as a direct means of communication between the computer and computer programmers and operators.

3.4 Digital Magnetic Tape

Several magnetic tape units will be included in the computer complex for uses such as:

a) Permanent storage for the complete program library

b) Automatic input to the computer of test programs as required in the orderly execution of test procedures

c) Automatic input to the computer of calibration data, limit check data etc., as required

d) Temporary storage for the accumulation of data which cannot be processed in real time

e) Storage of partially processed or partially reduced data in a form convenient for off-line or non-real time analysis.

3.5 Telemetry Decommutator

All telemetry data will be assembled into words and transferred in parallel into the computer through this device.

3.6 Special Input Devices

Between 10 and 20 special devices for input of selected data pertaining to the various spacecraft subsystems will be utilized.

3.7 Remote Interrupt Panels

Some automatic interrupt controls will be remotely located on a test director's console to give him control over the execution of certain critical or frequently used computer programs.

3.8 Manual Command Generation

An input device using a digiswitch, keyboard, or some other suitable mechanism will be used for the generation and verification of commands.

3.9 Single Line Inputs

Several hundred inputs of hardline data not available on telemetry will probably be used, especially in the earlier test phases.

4. OUTPUT DATA

4.1 Paper Tape Punch

A standard paper tape punch will be included for program generation, debugging, and other general off-line uses.

4.2 Line Printer

A medium speed line printer (300 lines/minutes, 100 to 120 characters/line) will be included in the computer complex for the following purposes:

a) Printout of processed (formatted and converted into meaningful units) spacecraft subsystem data derived from telemetry inputs

b) Printout of data derived from hardline inputs

c) Printout of data derived from special input devices

d) Printout of processed experiment data

e) Periodic printout of all commands sent to the spacecraft in the order sent, assuming all commands to the spacecraft originate in the computer (and this is one of the best justifications for such a procedural constraint).

g) Printout of test procedures on reproducible masters

h) Memory dumps and other diagnostic printouts.

4.3 Typewriter

The uses of an I/O typewriter were discussed in Section 3.3; however, as an output device the typewriter can also be used as an emergency backup for the line printer.

4.4 Magnetic Tape Units

The uses of magnetic tapes were discussed in Section 3.4.

4.5 Plotters

Two small plotters are included for graphic recording of some data, particularly experiment data such as pulse height data, energy distributions, etc.

4.6 Command Status Displays

With the computer controlling command generation to the spacecraft, it is only natural that it drive command status displays on a test director's console so that the state of the spacecraft subsystems and experiments can be readily ascertained.

4.7 Numeric Displays

Several (perhaps 12 or more) numeric displays are located in areas convenient for use by subsystem specialists and experimenters. These can be combined with the special input devices mentioned in Section 3.6 in such a way that a wide variety of data can be requested for display, using a keyboard, digiswitch, etc.

4.8 Remote Printers

It is planned to have a number of alphanumeric printout devices located in areas convenient for use by subsystem specialists and experimenters, in addition to the numeric displays just discussed. Such devices have the full complement of alphanumeric characters and are capable of printout at rates of 60 to 100 characters/second. This requirement is fulfilled by output typewriters, small printers, etc. These have the capability of requesting a wide variety of data from the computer (the computer can also output and automatically update data to these devices in accordance with established test procedures).

5.0 DATA RATES

5.1 Telemetry Data

The maximum bit rate for telemetry data is 4096 bits/second; however, since telemetry data will be assembled into 7-bit words which will then be entered into the computer in parallel, only the word transfer rate of 585 words/second will be considered. This leaves a total of 1709 μsec / words for all processing in the real time mode. The telemetry frame is 448 bits, or 64 words, in length; consequently, data identity repeats approximately 9 times/ second (9 frames/second).

5.2 Single Line Inputs, Special Input Devices, Display Requests, and Automatic Updating of Displays

All of these functions can be performed by a single program which periodically scans all lines for activity and which can be automatically initiated by a timer at appropriate intervals. For instance, if a scanning rate of once per second is adequate, an interrupt can be generated by the one-second markers from a real time clock. It is estimated that this program would require the execution of 500 to 1000 instructions.

5.3 Magnetic Tape Transfer Rates

Magnetic tape units capable of recording at two different densities, 200 bits/inch and 556 bits/inch, at a tape recording speed of 75 inches/ second, will be employed. This will permit the generation of tapes compatible with either the IBM 729 Mod II or Mod IV tape units. Using such tape units, data transfer rates between the tape units and computer memory will be 15 K characters/ second at the low density and 41.7 K characters/ second at the high density.

5.4 Other Devices

Devices such as the typewriter, remote printers, line printer, etc. operate at relatively slow rates and will be serviced on a programmed or interrupt basis and will not significantly detract from the time available for real-time processing.

6. COMPUTER WORD SIZE

6.1 A computer word size of 24 bits will be used for the following reasons:

a) It allows ample bits for instruction and address definition

b) Although the basic telemetry word length is only 7 bits, some measurements will require two or more words for proper parameter definition

c) A 24-bit word can define six decimal digits for printouts and numeric displays of time-tagged data

d) In those cases where considerable leeway is required, a shorter word length would require a larger amount of double precision arithmetic which is costly in both machine time and memory capacity.

7. COMPUTER SPEED

Since all telemetry data repeats once per frame, programs for processing this data must be repeated once per frame. At the high bit (4096 bps) rate, and assuming an average of 2.13 machine cycles/instructor, an 8 μsec machine cycle will permit the execution of 6400 instructions per frame.

For real-time processing, similar to that previously employed on OGO and PIONEER, the following is an estimate of the maximum number of instructions which will be executed in any one frame:

a) 4500 for processing of telemetry data

b) 1000 for processing and transfer of data to tape

c) 300 for processing hardline inputs, displays, etc.

d) 200 for other tasks.

This yields a total of 6000 instructions which may have to be performed in a single frame time-out of a maximum available of 6400, leaving very little room for error or growth. A machine cycle of 6 μsec will increase the number of instructions which can be performed in a single frame time to 8400, thereby increasing the margin based on the above estimates. Since machine cycle may be equated to memory cycle, the computer memory cycle time will be not more than 6 μsec.

8. COMPUTER CORE MEMORY SIZE

The following paragraphs give estimates of program lengths for various programs that will be used. These estimates are based on experience gained in programs such as OGO, PIONEER, etc.

8.1 Programs Required

8.1.1 Executive Control Program

This program is the master program for controlling selection of other programs, either in core or on magnetic tape. It provides the test director and computer operators with a great deal of flexibility in sequencing test operations. Primary man-program interface is via the I/O typewriter, but additional human control may be exercised through the activation of interrupts. The estimated core requirement is 750 locations.

8.1.2 Telemetry Input Program

This program enters all data from the telemetry decommutator into core, and gives indication when sync has been lost, marking all data in the last frame entered prior to loss of sync. The estimated core requirement is 900 locations.

8.1.3 Hardwire Input Program

This program scans all hardwire inputs recognizing any activity which requires further action, identifying the source of such activity and causing the appropriate actions to be taken. The estimated core requirement is 500 locations.

8.1.4 Display Processing Program

This program updates all displays and honors all requests for display data. The estimated core requirement is 350 locations.

8.1.5 Experiment Input Processing Programs

These programs perform all necessary processing or preprocessing of all experiment data entered into core by the telemetry input program. There are individual routines for each experiment, each controlled by an automatic interrupt so that none, one, or several, or possibly all experiment programs can be operating concurrently. The estimated core requirement for all experiments is 2000 locations.

8.1.6 Experiment Output Processing Programs

These programs perform functions for outputting the experiment data in meaningful form, including the final processing of input data stored initially on magnetic tape. The estimated core requirement for all experiments is 4000 locations.

8.1.7 Spacecraft Subsystem Data Processing

This program will process all subsystem data for engineering analysis including such subroutines as are required for limit checks, curve fitting calibrations, formatting, and conversion to engineering units. The estimated core requirement is 5000 locations.

8.1.8 Other Subroutines

a) Tape write: 1000 locations

b) Printer control: 250 locations

c) Standard mathematical functions such as binary to decimal conversion, trig functions, etc.: 750 locations

d) Various utility routines: 500 locations.

8.1.9 Test Sequence Programs

These programs are called into the memory as required under Executive Control, and vary in nature and length.

8.1.10 Command Generation Program

This program controls the generation of all commands to the spacecraft, and maintains a log of all commands sent, and the order in which sent. It may become necessary to verify commands. The estimated core requirement is 750 locations.

8.2 Core Size

These programs total more than 16,000 locations. Since not all programs will be in core simultaneously, it is felt that a core memory of 16,384 locations will be adequate; expansion to 32,768 locations will be a capability of the machine allowing for errors and/or growth.

APPENDIX B

VOYAGER LAUNCH COMPLEX EQUIPMENT

The purpose of this appendix is to discuss alternate configurations of the OSE electrical launch complex equipment (LCE) and to recommend a specific configuration, which is discussed in OSE/VS-2-120. In order to systematically select a specific configuration for the LCE, requirements for this equipment must be established; once this has been done, alternate approaches may be analyzed to implement these requirements. Finally, from the alternate approaches, a specific LCE configuration can be chosen.

It is recognized that the exact choice of electrical LCE configuration must be considered tentative at this time. During phase 1B, additional information will be available (e.g., the lander configuration) which will permit a more exact functional analysis of the electrical requirements for the LCE. Accordingly, this will produce a firm choice of electrical LCE.

1. ELECTRICAL LCE REQUIREMENTS

1.1 Introduction

The operations at ETR begin with the arrival of the spacecraft and terminate with the launch of the planetary vehicle. These operations consist of checkout and test of the spacecraft in the spacecraft assembly facility, of the spacecraft and lander in the explosive safe facility, and of the planetary vehicle on the launch pad with the Centaur vehicle.

The test objectives at the spacecraft assembly facility are:

a) Determination by inspection and test that the spacecraft has not been damaged during shipment

b) Verification by test that the spacecraft is ready for subsequent test operations.

The test objectives at the explosive safe facility are:

a) Determination by inspection and test that the integration of the spacecraft with the lander has been successfully accomplished

b) Verification prior to transporting the planetary vehicle (the integrated spacecraft and lander) to the launch pad that all functions that can be checked are in operating condition.

The test objectives at the launch pad are:

a) Verification by a series of checks and tests that after the planetary vehicle is mated to the Centaur no problems exist in the planetary vehicle.

b) Verification by a series of checks and tests that the planetary vehicle and Centaur vehicle have no incompatibilities and that they and all supporting agencies and equipment are ready for the actual countdown and launch

c) Verification by a series of checks and tests that the spacecraft is performing properly during the countdown to launch.

1.2 Tests Performed

The tests to be performed on the spacecraft or planetry vehicle which require the support of the electrical LCE are as follow:

1.2.1 Spacecraft Assembly Area

a) Stabilization and control subsystem validation

b) Integrated system test

c) Experiment/telemetry calibration

d) Continuity and Centaur interface checks.

1.2.2 Explosion Safe Facility

a) Planetary vehicle ordnance test

b) Modified integrated system test (the integrated system test performed at this facility is identical to that performed at the spacecraft assembly area except for the possible limitation imposed by the planetary vehicle configuration and range safety requirements).

1.2.3 Launch Pad

a) DSIF compatibility test

b) Modified integrated system test (this integrated system test is identical to that performed in the spacecraft assembly area except for the possible limitations imposed by the planetary vehicle/Centaur configuration and range safety requirements)

c) Planetary vehicle performance monitoring until launch (after it has been commanded into its launch mode).

1.3 Summary

The integrated system test to be performed at the launch complex is identical to that performed at the factory with the possible exceptions as discussed above. Similarly, the validation of the stabilization and control subsystem, experiment/telemetry calibration and continuity and interface checks are identical to those which are performed at the factory facility. The DSIF compatibility test is a test which has not been performed at the factory; however, this compatibility test requires the spacecraft to be commanded via the RF link through all of its proper operating modes and this performance is monitored through the telemetry link. This requirement is identical with requirements for the system test set as discussed in OSE/VS-3-110. Further, ground power control and monitor must be provided to the planetary vehicle during the compatibility test as it was done at the factory. Consequently, this test generates no additional requirements. In summary, it has been shown that the functional and design requirements for the LCE are identical to those for the system test set. This is true to the extent of replacing a faulty black box on the spacecraft in the event of a failure at the launch complex, since this operation is comparable to installing a black box on the spacecraft at the factory during spacecraft integration.

2. ALTERNATE APPROACHES

To meet the requirements stated above, it becomes readily obvious that if the same equipment which supported the spacecraft test activities from initial spacecraft electrical integration, through spacecraft integrated system test, prototype qualification test, and flight environmental acceptance test can be used for support of the spacecraft and planetary vehicle at the launch complex, major advantages can be obtained. These advantages are as follow:

a) Design cost savings to the Voyager program

b) Cost savings to the Voyager logistics program

c) Cost savings in terms of the man machine interface learning curve.

The proposed configuration for the LCE is to utilize the system test set automatic data handling system and specialized textbooks (see Figure 1, OSE/VS-2-120).

It is proposed to provide the same equipment in the blockhouse/ launch pad facility and the explosive safe facility and the blockhouse/pad facility will consist of equipment identical to that provided in the ground power console and rack of the system test set except for an additional spacecraft status display. This status display is identical to that in the telemetry data console of the STS and will be driven by the STS.

During launch operations and test operations at the explosive safe facility the system test set and data handling system are located in the spacecraft assembly area. Commands are transmitted to the spacecraft via the RF link or by providing a bypass of the RF equipment both in the spacecraft and in the system test set. This is accomplished by coupling the output of the STS command encoder directly to the spacecraft via hardline and the STS receives the spacecraft and/or lander DTU output signals directly via hardline. This feature is required in order to facilitate testing when RF silence is imposed by the range.

During checkout and test of the planetary vehicle at the pad, STS test console functions will be required. Since these functions must be performed in close proximity to the spacecraft (due to loading limitations of the spacecraft interface signals) the drawers in the test console will be readily removable from the rack and inserted into portable suitcases which may be carried onto the gantry. The suitcases will be heavy duty water-tight containers with slides installed so that an item of rack mounted equipment may be removed from the rack, inserted in a suitcase and used on the gantry.

During the terminal countdown with the STS and automatic data handling system located in the spacecraft assembly area, it is possible to assure proper planetary vehicle performance by interrogating the spacecraft, and to monitor its telemetered functions via the RF link using high gain antennas. By this method of operation the following advantages are achieved:

a) Minimum launch facility complex equipment interfaces are required

b) Minimum quantity of OSE is required

c) Minimum movement of the OSE and minimum amount of time are required to support launch operations.

It should be noted that each of the drawers expected to be taken onto the gantry will contain its own power supply and will require standard facility 110V AC power.

APPENDIX C

VOYAGER MISSION DEPENDENT EQUIPMENT

1. MAJOR CONSIDERATIONS

The major philosophical consideration in the design of the Voyager MDE is reliability of the MDE with particular emphasis on the primary in-line functions of command generation, telemetry detection and computer buffering. Also of major consideration are the means of test, fault isolation and maintenance of the MDE.

1.1 Reliability

In order to provide for the first of these considerations, the MDE is of conservative design and uses high reliability components and processes throughout. To provide an even higher order of reliability, the primary in-line units are supplied in redundant pairs which are mounted in the same equipment rack in close proximity to each other so that change-over can be initiated immediately upon detection of a malfunction. In order to minimize the loss of reliability of the primary signal paths due to switching mechanisms, signals are applied to the inputs of both units of the redundant pairs. The input circuits of the redundant units are designed in such a way that a failure in the input circuits of one of these units (except for a short at the signal branch point) will not prevent the like unit from operating properly.

Reliability of the primary command generation link is enhanced through a series of command verification steps, both manual and computer controlled, which insure the correctness of the data prior to transmission to the spacecraft.

Power supplied for the primary function units is also redundant and the power supply in use may be selected by application of its input power.

Outputs of these power supplies are isolated in such a way that failure of one supply will not affect the operation of the redundant unit, and damage will not result if both supplies of a redundant pair are energized.

MDE reliability is proven through the accumulation of operating time from installation through to mission usage. During this interval a history of parts performance is accrued and analyzed to establish failure rates and to confirm design. Where possible, MDE design equipment is included in the various other test complexes to perform like functions.

1.2 Test, Fault Isolation and Maintenance

The primary in-line MDE functions and the secondary MDE functions, which are command detection, command verification, and spacecraft status display, may be functionally tested and to a large degree fault isolated, maintained and calibrated by means of the MDE test equipment which consists of specially designed and commercial test equipment. In addition, self-test features are built into the individual units.

Patch panels and switches are provided to facilitate test functions.

1.2.1 Command Generation Testing

The output of the PN generator may be connected to the input of the command detector. This connection permits the computer command verification during actual command transmission, in the event that the station monitor receiver output should not be available for any reason, and in addition, provides for a functional test of the MDE command generation and command verification chain without operation (or connection to) the station transmitter modulator.

The output of the command encoder unit can also be connected into the command detector behind its PN code circuits providing for a similar test with the PN generator out of the loop. This connection (combined with the use of the emergency operation mode of command generation) can isolate faults to the PN generator, the command encoder or within the command detector.

The purpose of the station simulation equipment is to provide for checkout of all MDE equipment without requiring the use of the DSIF station multiple purpose communications equipment. By use of the transmitting portion of the station simulator, the PN coded commands can modulate a carrier and return this signal through the receiving portion of the test transponder to the command detector. All primary and secondary command functions can be checked out in this way, including the computer command permissibility and bit-by-bit command checks (via the MDE computer buffering circuits) if the station computer is available.

Station communications equipment interface tests can be made using the normal command generation units with the station test diplexer output connected to the test transponder and with the output of the station monitor receiver connected to the command detector. The test transponder output may alternately be connected to the command detector.

The functional tests discussed above can be used to check the operation of the command signal loops for factory integration and acceptance testing, on-site integration and acceptance testing, compatibility testing, station readiness testing, fault isolation tests and for general maintenance purposes.

1.2.3 Telemetry Channel Testing

The data format generator (DFG) can provide simulated telemetry inputs to the computer buffer which, in conjunction with the station computer, can be used to test the computer buffer and the spacecraft status display units, except for those functions related to command generation.

The error rate test (ERT) can be used to analyze the performance of the telemetry detector under noisy signal conditions and can provide both baseband (bi-phased NRZ data, ranging between 128 to 4096 cps) and IF (phase modulated 10 Mc) as inputs to the telemetry detector. Normally, the 10 Mc signal is applied to the input of the telemetry detector, but the baseband signal can also be used, bypassing the discriminator circuitry, during trouble shooting. The DFG output may also be used as input to the ERT to provide a telemetry signal to the telemetry detector with meaningful data which can provide for a functional test of the loop consisting of the DFG, the ERT, the telemetry detector, the computer and the spacecraft status displays. This loop may also be checked using recorded telemetry NRZ data signals played back to the ERT.

The ERT baseband output may also be used to modulate the test transponder transmitter section. The test transponder modulated RF output can be connected to the station simulator receiver section, demodulated and returned to the telemetry detector.

This test provides for functional test of all MDE equipment associated with telemetry functions in a single loop test without use of the station multiple purpose communications equipment.

Station communications equipment interfacing with the MDE related to telemetry detection can be tested with the DFG or tape recorder connected to the ERT, the ERT connected to the test transponder, the test transponder connected to the station diplexer and the station receiver output connected (in the normal fashion) to the telemetry detector.

This configuration can be used for station compatibility and functional tests for all of the MDE equipment associated with telemetry including the computer and the spacecraft status displays.

As in the case of the command loops, many of the configurations described above can be used to check the MDE equipment associated with telemetry detection during factory on-site integration and testing, compatibility testing, station readiness testing, fault isolation and general maintenance.

2. STATION COMPATIBILITY

The Voyager MDE equipment design is influenced by several factors which are concerned with general DSIF station compatibility. Most important of these from a functional viewpoint are electromagnetic interference and environmental considerations.

2.1 Electromagnetic Interference (EMI)

Based on JPL'S requirements and TRW Systems past experience, it is felt that the EMI needs of the DSIF station can best be met by designing the Voyager MDE to meet the requirements of military specification MIL-I-26600 for Class II equipment.

It is proposed that the design features incorporated in the Pioneer MDE also be used for Voyager MDE. On the Pioneer Program a very conservative approach was taken which resulted in passing of the tests of MIL-I-26600 with wide margins of safety without any redesign or retest necessary.

The measures employed on the Pioneer MDE and recommended for the Voyager MDE are as follows:

a) Individual shielding of all electronic units

b) RF gaskets or close spacing of screws and rivets of all unit covers and panels

c) Double RF shielding of RF units of MDE

d) Dip brazing of seams of the test transponder shields

e) Added RF gasketing on doors, sidepanels and around equipment panel mounting surfaces where possible

f) Shielding of all internal and external rack wiring that may be suspected of producing or being susceptible to RF radiation

g) Enclosing all interrack and station interface cabling with a second shield consisting of jacketed flexible conduit

h) Use of RF shielding on all front panel components, e.g. conductive glass over readouts, RF shields behind entry switches, etc.

i) RF filter screens over openings for cooling air

j) Selection of commercial equipment for minimum EMI

k) Filtering of power supply input lines

It is not suggested that commercial equipment incorporated in the Voyager MDE be required to meet MIL-I-26600.

2.2 Environment

It is proposed that the MDE be designed to meet the temperature range of 32 to 100°F. Although the station cooling air will be controlled to a much narrower range, this wider range provides for operation in the event of failure of the station air cooling equipment and also provides for test of the equipment at the factory over a reasonably wide range without need for special conditioned air.

The equipment is designed to operate with cooling air humidity up to 95 per cent RH.

It is proposed that the equipment not be subjected to shock and vibration testing (or other environmental tests), but that the mechanical design be adequate to meet all normal handling requirements. It is proposed that conservatively designed packing crates for shipment of the MDE be approved prior to fabrication as a practical means of avoiding the need for shock test of the equipment in the shipping crates which could result in unnecessary damage to deliverable equipment.

2.3 Test Transponder

The test transponder is rack mounted for use at the DSIF station to simulate the Voyager spacecraft for compatibility testing when connected to the station communications equipment via the station test diplexer.

It may also be used at a remote location such as a collimation tower site for system tests. For this type of operation the transponder has been designed for portability. In addition to the test transponder with its AC operated power unit, a rechargeable battery, capable of operation for at least 8 hours, is provided. The necessary battery charging circuits are also incorporated. For portable use, the transponder and its power supply are mounted in carrying cases.

In order to provide realistic spacecraft/station compatibility it is desirable for the test transponder performance to duplicate to the maximum extent practical the performance of the actual spacecraft. For this reason the design of the transponder incorporates actual spacecraft receiver and low level transmitter units.

2.4 Recording

The recorder outputs of the command encoder may be recorded on magnetic tape or strip chart recorders in order that they may be observed (or played back to the command detector) following a test or actual command transmission. A command transmission as received by the station monitor receiver may also be recorded for subsequent play back or observation.

Tape recordings of DSIF station receiver output telemetry signals will normally be made during all data transmission. In the event of any failure in the MDE telemetry extraction signal chain, the recorded signal then may be played back for later data extraction.

As a backup to this, the outputs of the telemetry detection unit (NRZ data and synchronization signals) are also tape recorded and may later be played back to the computer buffer.

For test purposes, tape recorded telemetry signals may be played back to the input of the ERT or to the transmitter portion of the test transponder. This signal may be generated by the spacecraft, or by some other simulation means, such as a data processing computer.

The proposed MDE has the necessary interfacing circuits and switching circuits necessary to provide for the recording playback functions discussed above.

3. GENERAL PURPOSE TEST EQUIPMENT

General purpose test equipment is incorporated in the MDE so that the proper equipment will be conveniently at hand and properly connected to provide accurate and rapid measurement, observation and calibration of the MDE. Patch panels and switches are employed to connect the test equipment to the desired MDE test points for those measurements, observations and calibrations which will be most frequently required.

Additional patching will be provided to allow use of both built-in and external equipment at additional test points which may require only infrequent attention.

The expense associated with procurement of general purpose test equipment meeting the MDE EMI requirements is not considered justified. However, radiated and conducted EMI and susceptibility to EMI should be considered in the selection of this test equipment.

APPENDIX D

VOYAGER MECHANICAL ALIGNMENT PLAN

1. SCOPE

The alignment plan presents the sequence of alignments for the total Voyager spacecraft and forms the basis for detailed equipment and facility requirements and procedural needs.

This alignment plan is based upon Configuration VS-3-110.

2. APPLICABLE DOCUMENTS

3. SPACECRAFT ALIGNMENT ACCURACY REQUIREMENTS

The accuracy of the individual alignments are presented in Table I.

4. ALIGNMENT SEQUENCE

4.1 General

The alignment technique outlined in this section is based upon existing commercially available equipment which is capable of accuracies an order of magnitude greater than any Voyager requirement.

Most of the critical alignment requirements on the Mars Voyager spacecraft are angular. The basic reference system which best meets such requirements is a spherical coordinate system determined by the fine sun sensor and the Canopus sensor as shown in Figure 1. Most items requiring alignment have one axis which must be aligned to the fine sun sensor axis, with the "cone angle" set to zero degrees plus or minus a tolerance; and another (orthogonal) axis set to some "clock angle," plus or minus a tolerance, relative to the Canopus sensor axis. Therefore, the fine sun sensor and the Canopus sensor are used as basic references for all final alignments. Except for some special cases (e.g., thrust axis offsets and mass properties determinations) radial and axial locations are unimportant and are not measured.

It is necessary for practical reasons to establish a reference on the spacecraft structure which is basic to all alignments. This reference is a plane described by three of the six hard points at the spacecraft separation plane. This plane is aligned to local gravity by a system of 10 arc second sight levels as a preliminary step to the installation of the sensors which define the basic reference system.

Table I. - Voyager Equipment Alignment Requirement

	Equipment	Mounting	F.O.V	Centerline Location Cone Angle	Centerline Location Clock Angle	Accuracy of Alignment	Primary* Reference	Remarks
1.	Fine Sun Sensor	Body, Base	± 0.5°	0°	-	± 0.1°	Spacecraft Structure	Cone angle is 0° by definition Sensor is a basic reference
2.	Coarse Sun Sensor a	Body Solar	2π Steradian	90°	α	± 0.5°	To Item 8.	Bench alignment
	b	Panel	2π Steradian	90°	α + 90°	± 0.5°	" " "	α = configuration dictated
	c	" "	2π Steradian	90°	α + 180°	± 0.5°	" " "	
	d	" "	2π Steradian	90°	α + 270°	± 0.5°	" " "	
3.	Canopus Sensor	Body, Equip Section	30° x 4°	70 - 110°	0	± 0.5°	Fine Sun Sensor and S/C structure	Clock angle is 0° by definition Sensor is a basic reference
4.	Near Earth Sensor	Body Mtd.	± 20°	90°	90°	± 0.5°	To Item 3	Bench alignment
5.	Control Gyros	Body, Internal	-	-	-	± 0.1° 3 Axis	To Item 3	Bench alignment
6.	ACS JETS, 8 pitch and yaw	Body Solar Panel	-	0°, 180°	α, α ± 90°, + .80°	2.50° 2.50°	To Basic Ref.	Offsets to spacecraft Z axis
	4 roll	" "		90°	±α, α ± 180°			
7.	Hi Gain Antenna	Body Gimballed	± 2.5°	40 to 140°	30 to 150°	± 0.25°	To Basic Ref.	
8.	Lo Gain Antenna	Boom Mtd. Exp. Boom No. 2	OMNI ± 115°			± 3°		
9.	Mid Course Engine	Body Mtd. and Jet Vanes	-	0°	0°	± 0.20° ± 0.25 in.	-Parallel to Z axis -offset from Z axis	
10.	Retro Propulsion Motor	Body Mtd., Fixed Fluid Injection	-	180°	0°	± 0.20°	Parallel to Z axis Offset from Z axis	

*units are referred to the sun sensor-Canopus sensor coordinate system.

Table I. - Voyager Equipment Alignment Requirement (Cont'd)

	Equipment		Mounting	F.O.V.	Centerline Location Cone Angle	Clock Angle	Accuracy of Alignment	Primary* Reference	Remarks
11.	Planet Oriented Pkg.		Body Mtd	-			± 0.15°	S/C Ref. Axis	
12.	Science Payload Scan Sensor (Mars Sensor)		POP				± 0.05° ± 0.25°	To Basic Ref.	
13.	TV Experiment a		POP	± 10°	-	-	± 0.05°	To Item 13.	Bench Alignment
14.	TV Experiment b		POP	0.1°	-	-	± 0.01°	To Item 14.	Bench Alignment
15.	UV Spectrometer		POP	0.1° x 2°	-	-	± 1°	To Item 14.	Bench Alignment
16.	Scan Radiometer		POP	10 mr	-	-	± 0.5°	To Item 14.	Bench Alignment
17.	Infrared Spectrometer		POP	0.2° x 5°	-	-	± 1°	To Item 14.	Bench Alignment
18.	Meteoroid Flash		POP	60°	-	-	± 5°	To Item 13.	Bench Alignment
19.	Body Mtd. Exper. Pkg.		Body Mtd. Plate	π Steradians	90°	0°	± 0.5°	To Basic Ref.	
20.	Meteoroid Impact	a	Body Mtd. Plate	± 10°	0°	-	± 1°	To Item 20.	Bench Alignment
		b	" " "	± 10°	180°	-	± 1°	" " "	
		c	" " "	± 10°	90°	270	± 1°	" " "	
		d	" " "	± 10°	90°	90	± 1°	" " "	
21.	Cosmic Ray	a	Body Mtd. Plate	± 45°	0°	-	± 5°	To Item 20.	Bench Alignment
		b	" " "	± 45°	0°	-	± 5°	" " "	
		c	" " "	± 45°	180°	-	± 5°	" " "	
		d	" " "	± 45°	90°	0°	± 5°	" " "	
22.	Plasma	a	Body Mtd. Plate	± 20°	0°	-	± 3°	To Item 21.	Bench Alignment
		b	" " "	± 20°	0°	-	± 1°	" " "	
23.	Trapped Radiation	a	Body Mtd. Plate	± 45°	0°	-	± 5°	To Item 21.	Bench Alignment
		b	" " "	± 45°	90°	Config. Dic.	± 5°	" " "	
		c	" " "	± 45°	180°	-	± 5°	" " "	
24.	Magnetometer		Exp. Boom No.1	π Steradian	0°	-	± 5°	To Basic Ref.	Bench Alignment
25.	Ionosphere		Whip Antenna	OMNI	180°	Config. Dic.	Req. for Clearance	-	

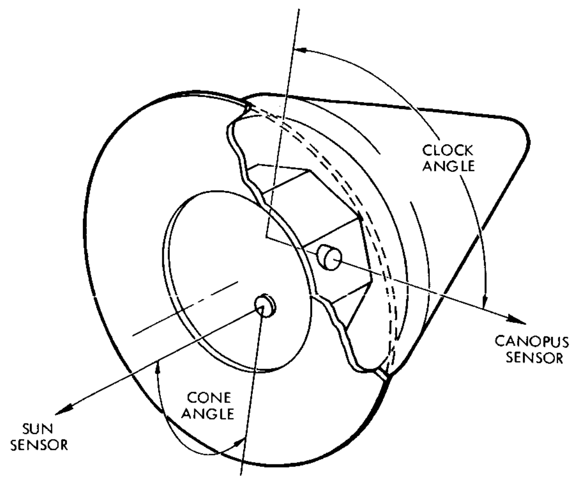


Figure 1. Basic Reference Coordinate System

4.1.1 The Sun Sensor Reference

The sun sensor axis, by definition, defines the zero reference for cone angle. A collimating mirror is installed on the sun sensor assembly and is aligned on the bench perpendicular to the sun sensor line of sight. This mirror then provides a reference surface for final alignments. The alignment of the sun sensor on the spacecraft is accomplished by leveling the structure to reference marks on the hard mounting points. The mirror on the sun sensor is then viewed with an auto-collimating plumb alignment assembly, as shown in Figure 2 and the sun sensor assembly is adjusted until it is plumb within 10 arc seconds (0.003 degrees). After the sun sensor has been aligned to local gravity, all alignments to the sun sensor axis can be made with standard leveling instruments, such as machinist's levels, sight levels and theodolites. Accuracies of 10 arc seconds are common with such instruments. The plumb alignment assembly is left in place during alignment operations and is periodically checked to assure that the sun sensor is still vertical. This check consists of turning on the collimator lamp and seeing that the returned reticle image is still superimposed on the instrument reticle image.

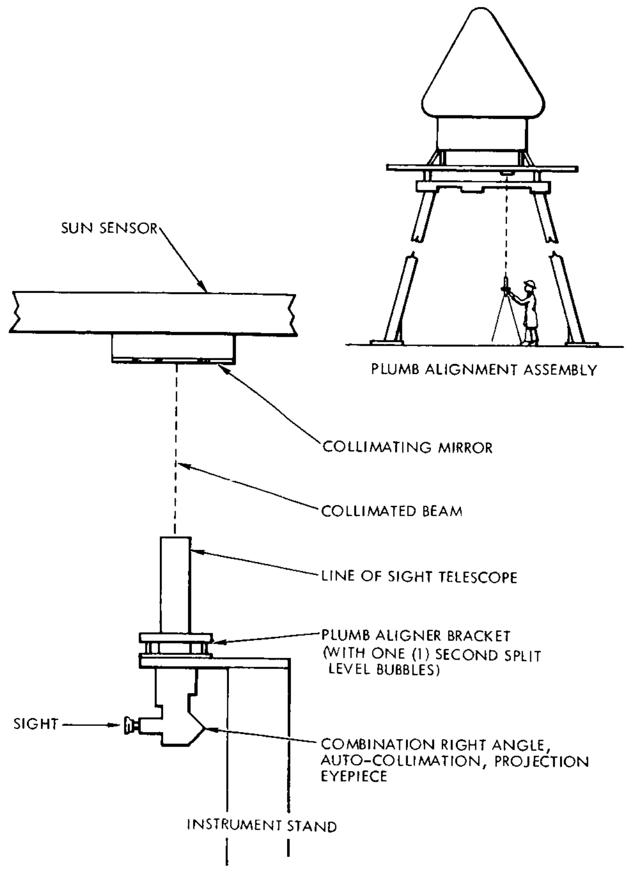


Figure 2. Sun Sensor Alignment

4.1.2 The Canopus Sensor Reference

The Canopus sensor axis defines the zero reference for clock angle. A collimating mirror is installed on the sensor assembly and aligned on the bench to the sensor line of sight. After installation of the sensor assembly on the spacecraft, the mirror is viewed with an auto-collimating theodolite. The vertical axis of the mirror is aligned to the prescribed angle to the sun sensor (presently planned at zero degrees) by adjusting the sensor assembly. The horizontal axis of the mirror defines the zero reference of the clock angle and does not have to be adjusted. The clock angles of other items to be aligned can be measured with one or two other theodolites as shown in Figure 3. Note that theodolites measure horizontal angles and are thus independent of the Canopus sensor to sun sensor angle (cone angle). This angle could be set to some value other than 90 degrees without affecting the alignment technique.

Also note that while a three-theodolite system is shown to obtain coverage on the opposite side of the spacecraft, this coverage could also

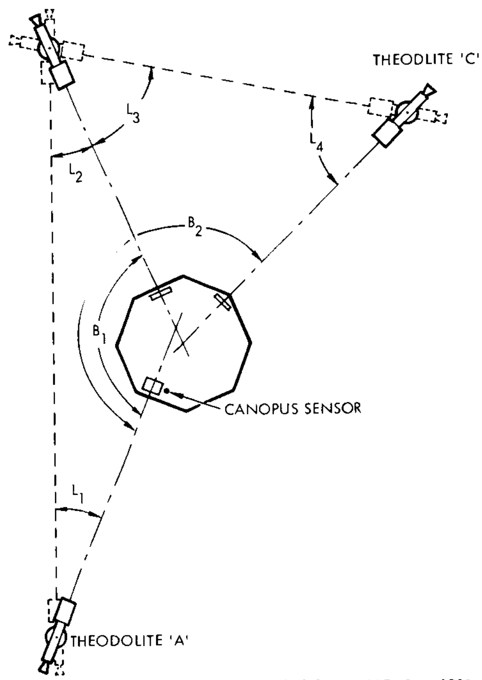


Figure 3. Direct Reading of Clock Angle with Theodolites

be obtained with right-angle pentaprisms or with an auto-reflection angle mirror. Standard theodolites, such as those used in second-order surveys, will provide angular measurement accuracies of better than 10 arc seconds.

4.1.3 Radial Linear Measurements

For the determination of mass properties, thrust-axis offsets, et al, axial and radial linear measurements are necessary. With the basic vehicle reference established (i.e., the sun sensor line of sight normal to local horizontal), these measurements are made with standard industrial optical equipment such as jig transits or theodolites and optical tooling scales in combination with special fixtures and alignment targets.

4.2 Typical Alignments

Requirements for alignments are shown in Table I. Typical alignments outlined in this section are intended to show proposed applications of the alignment concept presented under 4.1 above. Detailed procedures and equipment specifications will follow naturally as hardware designs develop. It is anticipated that many features of these alignments will change as shown in Figure 4.

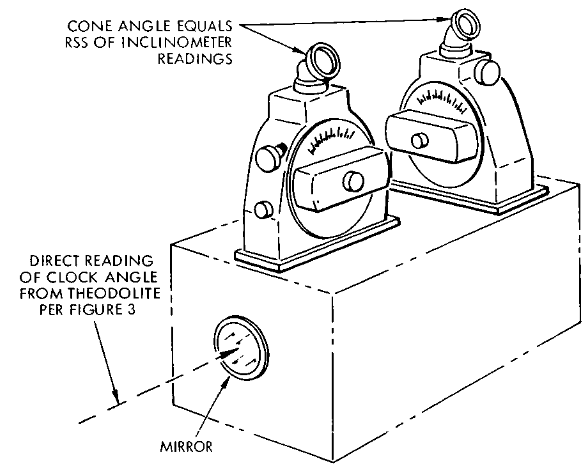


Figure 4. Typical Alignment of Spacecraft System Component

4.2.1 Canopus Sensor and Sun Sensor

These units constitute the primary alignment reference from which all other alignments are made. The spacecraft is set up in the vertical position and leveled by reference to the targets located on three of the six interface hardpoints with a system of 10 arc second sight levels. Installation and alignment of the sensors then proceed as outlined in Section 4.1 above.

4.2.2 Control Electronics Package

This package, consisting of the control gyros, is mounted as a unit in the same mounting location as the Canopus sensor. Their individual alignments upon their common mounting plate are accomplished during assembly by normal mechanical and electrical methods supported by Quality Assurance coverage. Such methods can easily meet the 0.1 degree requirements. The alignment of the Canopus sensor is accomplished by adjusting this entire package, thus the Canopus sensor alignment aligns all these components to the basic reference system.

4.2.3 Interplanetary Science Sensor Package (2)

These packages, containing micrometeoroid sensors, Cosmic Ray sensors, plasma probes, and trapped radiation sensors, are mounted as two units whose components are aligned to each other on the bench. Their alignment requirements are met exactly as are those of the control electronics package, i.e., the entire preadjusted package is aligned to the sun sensor with precision levels, and to the Canopus sensor with theodolites.

4.2.4 Planet Oriented Package

This package consists of a double-gimballed structure upon which are mounted the following seven experiments:

a) Science payload scan sensor (Mars sensor)

b) TV Experiment a

c) TV Experiment b

d) Ultraviolet spectrometer

e) Infrared spectrometer

f) Scan radiometer

g) Meteoroid flash experiment

The basic alignment characteristics of this package are determined by dimensional control and Quality Assurance coverage of the gimbal and structural components during fabrication. The completed package is mounted on the spacecraft and aligned to the basic sun sensor-Canopus sensor coordinate system to within ± 10 arc seconds with auto-collimating theodolites.

The pre-installation bench alignments of the seven critical experiments mounted on the P.O.P. (see Table 2) are accomplished in the electrical and optical laboratories by first locating the centerline of each experiment's field of view, then installing a collimating mirror (for TV experiments a and b, and the Mars sensor) or a mechanical fixture (for the remaining four experiments), and adjusting each instrument to bring its F.O.V. centerline to the required relationship with the package structure by optical or mechanical means.

4.2.5 Magnetometer Boom

This unit is checked for alignment in the deployed position. The boom is supported by a series of frictionless cradles such as the flotation fixtures used on the OGO spacecraft boom alignment. The magnetometer package has a mechanical layout which is viewed by a theodolite which is referenced to the sun sensor-Canopus sensor reference system. Axial and radial positions are determined with optical tooling scales and jig transits, which are set up by reference to the spacecraft structure.

4.2.6 Antenna Alignments

The alignment of the high gain antenna is similar to that for the P.O.P. due to the double-gimballed actuation. The bore sight line of the antenna is established in the laboratory and a fixture is installed. After the antenna is installed on the spacecraft, this fixture is viewed by theodolites which are referenced to the sun sensor-Canopus sensor reference system.

The alignments of all remaining antennas are non-critical and may be checked and adjusted with simple fixtures utilizing machinist's levels.

4.2.7 S and C Nozzles and Coarse Sun Sensors

The coarse sun sensors are mounted on the S and C nozzle blocks in the laboratory and are aligned to the nozzles when mounted, using simple mechanical methods. The nozzles are aligned on the spacecraft using plug fixtures which support machinist's levels. Axial and radial locations are checked by viewing the fixtures with optical tooling scales and jig transits which are set up by reference to the spacecraft structure.

4.2.8 Mid-Course and Retro Propulsion Engines

Both propulsion units are subject to the same alignment restraints, and both are handled in the same way for thrust vector alignment, i.e., a plug fixture carrying machinist's levels is inserted in the nozzles and the motors are shimmed until the levels center. The plug fixtures are also used to determine radial offset of the nozzle centerlines by viewing alignment targets on them with two jig transits set up to sweep the geometric axis of the spacecraft. The jet vanes of the mid-course engines are adjusted by observation of machinist's levels mounted on the vanes by means of special fixtures.

4.2.9 The Flight Capsule

The flight capsule must be aligned to the geometric axis of the flight spacecraft in order to hold the overall c.g. offset within the allowable envelope. This alignment is accomplished by the traditional method of sighting on capsule reference targets with two jig transits arranged to sweep orthogonal planes which intersect along the vehicle longitudinal axis. These transits are set up using the sun sensor-Canopus sensor system as a basic reference.

4.2.10 End-to-End Checks

All components of the spacecraft are automatically checked end-to-end to the basic reference when the techniques described above are used.

APPENDIX E

ADVANTAGES OF VOYAGER 1969 TEST MISSION TO OSE DEVELOPMENT FOR THE 1971 MISSION

The opportunity to privide OSE for a 1969 test spacecraft will greatly enhance the probability of mission success of the 1971 Voyager program. The factors leading to this conclusion are discussed below.

The 1969 test program would provide the opportunity to verify the compatibility of the OSE with the spacecraft during factory operations and launch complex operations. Additionally, the MDE could be verified with DSIF equipment prior to using the OSE for the 1971 program. This is particularly true because of the anticipated similarity of spacecraft subsystems and DSIF equipment requirements between the 1969 and 1971 programs. As a result of this spacecraft similarity, it is expected that the OSE will be similar; the EOSE in most cases will be identical.

In supplying OSE for the 1969 test program, the use of the OSE during all phases of the testing operations will provide the opportunity to uncover OSE design weaknesses, particularly in those areas where additional testing is required to provide information which could not be foreseen during the OSE design phase. Human engineering aspects of the OSE design will also be analyzed during the testing program so that the 1971 OSE design will be optimized with respect to the man/machine interface.

Because the 1971 OSE will be so similar to that used in 1969, there will be a guarantee that refurbished OSE will be provided on schedule or ahead of subsystem test schedule. This will allow additional time for test personnel to become familiar with the OSE prior to the actual test time for the 1971 spacecraft test program, and will ensure that the learning curve for the operators is at a peak early in the 1971 program. These factors are particularly important in ensuring the probability of meeting the launch window.

The opportunity to improve testing techniques will be provided as a result of the 1969 test program. Analysis of the types of spacecraft subsystem test points which should be provided, i.e., the use of quantitative data versus gross measurements and possibly GO/NO-GO criteria, and the tightening or relaxing of parameter tolerances will be accomplished which will benefit the 1971 OSE design.

The additional test ime provided by the 1969 test program will allow for complete debugging of hardware circuitry, analysis of critical circuit operation under actual test conditions, and improvement of design. Complete software debugging and analysis can be accomplished by running test programs and editing and improving the organization and sequencing of the computer test programs for factory, launch site, and field testing the spacecraft as well as for simulating and testing the DSIF software programs. Complete data management can be reviewed and analyzed to ensure the recording of the most critical parameters, and to provide close coordination and comparison of these parameters for trend analysis from test station to test station before or during the early phases of the 1971 program.

The test program will provide a better feel for logistics throughout the 1971 program because short cuts which become obvious during the 1969 program can be implemented during the 1971 program. This improved component and equipment handling and planning will result in higher reliability for the 1971 program and will expedite delivery to ensure meeting the 1971 launch window.

The functional analysis for the 1971 program will be greatly simplified as a result of requirements having been established for the test program. Need for additional tradeoffs for equipment designs and cost savings will be uncovered during the test program, which will result in more simple equipment and less expensive test approaches for the 1971 OSE.

The increased assurance of the 1971 flight success outweighs the cost of a 1969 test program, particularly since a 1971 program failure would include the total cost of providing an SIB launch vehicle. As shown in the implementation plan for the 1971 test program, the additional OSE requires amounts approximating 25 per cent, because the 1969 OSE can be refurbished and used for the 1971 program.

APPENDIX F

VOYAGER OSE ABBREVIATIONS

ADHS	Automatic data handling system
AFETR	Air Force Eastern Test Range
AHSE	Assembly, handling, and shipping equipment
BCE	Bench checkout equipment
BW	Bandwidth
c. g.	Center of gravity
CEA	Control electronics assembly
CS and C	Central sequencer and command
DAC	Douglas Aircraft Company
DAC/MSSD	DAC/Missile and Space Systems Division
DEU	Data entry unit
DFG	Data format generator
DSIF	Deep Space Information Facility
DTU	Digital telemetry unit
DVM	Digital voltmeter
EDS	Electrical distribution subsystem
EOSE	Electrical operational support equipment
ESF	Explosive safe facility
ERT	Error rate tester
ETM	Elapsed time meter
ETR	Eastern Test Range
EUT	Equipment under test
FSK	Frequency shift keying
GCE	Ground control equipment
IOVC	Input/output verification comparator
IST	Integrated systems test
J-Box	Junction box
JPL	Jet Propulsion Laboratory
LCE	Launch complex equipment
LO	Local oscillator

MDE	Mission dependent equipment
MOI	Moment of inertia
MOSE	Mechanical operational support equipment
NF	Noise factor
OSE	Operational support equipment
OV	Over voltage
PA	Power amplifier
PCEA	Power control electronic assembly
POP	Planet oriented package
PN	Pseudo noise
PS	Power supply
P/V	Planetary vehicle
RCA	Radio Corporation of America
RF	Radio frequency
RFI	Radio frequency interference
S/C	Spacecraft
S and C	Stabilization and control
SAF	Spacecraft assembly facility
SDS	Scientific data systems
SNR	Signal to noise ratio
STC	System test complex
STS	System test sets
TE	Test equipment
TLM	Telemetry
T/M	Telemetry
T/R	Transformer rectifier
TRW	Thompson Ramo Wooldridge Inc.
TWT	Travelling wave tube
UTS	Unit test sets
VCO	Voltage control oscillator
UV	Under voltage
VHF	Very high frequency
VSWR	Voltage standing wave ratio
VTM	Voltmeter

APPENDIX G

VOYAGER OSE DESIGN DOCUMENTS

LISTING

General

OSE/VS-1-110 OSE Objectives and Criteria
OSE/VS-2-110 OSE Design Characteristics and Restraints

System

OSE/VS-2-120 OSE Launch Complex Equipment
OSE/VS-3-110 System Test Set
OSE/VS-3-120 Automatic Data Handling System
OSE/VS-3-130 Mission Dependent Equipment
OSE/VS-3-140 AHSE
OSE/VS-3-140-1 Transporter, Flight Spacecraft
OSE/VS-3-140-2 Assembly, Handling and Tilt Fixture
OSE/VS-3-140-3 Transport Recorder
OSE/VS-3-140-4 Fixture—Weight, CG and MOI
OSE/VS-3-140-5 Shipping Container Group Standard Modules
OSE/VS-3-140-6 Work Platforms, Module
OSE/VS-3-140-7 Adapter Kit, Centaur/Shroud Transporter
OSE/VS-3-140-8 Sling Assembly, Planetary Vehicle and Nose Fairing
OSE/VS-3-140-9 Purge Unit, Freon/Ethylene Oxide
OSE/VS-3-140-10 Planetary Vehicle/Nose Fairing, Mating and Assembly Fixture
OSE/VS-3-140-11 Sling, Flight Capsule
OSE/VS-3-140-12 Hoist Beam and Slings, Flight Spacecraft
OSE/VS-3-140-13 Tag Lines
OSE/VS-3-140-14 Platform, Launch Stand Access
OSE/VS-3-140-15 Universal Mounting Ring, Flight Spacecraft/ Planetary Vehicle
OSE/VS-3-140-16 Environmental Cover, Flight Spacecraft
OSE/VS-3-140-17 Hoist Sling, Environmental Cover
OSE/VS-3-140-18 Platform, Auxiliary Access

Subsystem

OSE/VS-4-210 Science Payload Subsystem
- OSE/VS-4-210-1 Alignment Fixture, Science Payload
- OSE/VS-4-210-2 Shipping Container, Experimental Booms

OSE/VS-4-310 Communications and Data Handling Subsystem
- OSE/VS-4-310-1 Dolly, 6' Elliptical Parabolic Antenna
- OSE/VS-4-310-2 Hoist Beam, 6' Elliptical Parabolic Antenna
- OSE/VS-4-310-3 Shipping Container, 3' Parabolic Antenna
- OSE/VS-4-310-4 Shipping Container, 6' Elliptical Parabolic Antenna
- OSE/VS-4-310-5 Shipping Container, Low Gain Antenna
- OSE/VS-4-310-6 Shipping Container, Flight Capsule Receiving Antenna
- OSE/VS-4-311-1 S-Band Communications Unit Test Set
- OSE/VS-4-311-2 VHF Communications Unit Test Set

Subsystem (Continued)

OSE/VS-4-311-3 Command Detector Unit Test Set
OSE/VS-4-311-4 Data Handling Unit Test Set
OSE/VS-4-410 Stabilization and Control Subsystem
OSE/VS-4-410-1 Alignment Fixture, Stabilization and Control
OSE/VS-4-410-2 Protective Covers, Stabilization and Control Nozzle
OSE/VS-4-411-1 Rate Gyro Assembly Unit Test Set
OSE/VS-4-411-2 Sun Sensor and Near Earth Detector Unit Test Set
OSE/VS-4-411-3 Star Sensor Unit Test Set
OSE/VS-4-411-4 Control Electronics Assembly Unit Test Set
OSE/VS-4-411-5 Actuator Unit Test Set
Central Sequencer and Command Subsystem
OSE/VS-4-451-1 Central Sequencer and Command Unit Test Set
OSE/VS-4-460 Power Subsystem
OSE/VS-4-460-1 Assembly and Handling Frame, Solar Panel Segment
OSE/VS-4-460-2 Protective Cover, Solar Panel Segment
OSE/VS-4-460-3 Shipping Container, Solar Panel Segment
OSE/VS-4-460-4 Handling Dolly, Solar Panel Segment
OSE/VS-4-460-5 Sling Assembly, Solar Panel Segment
OSE/VS-4-460-6 Shipping Container, Battery
OSE/VS-4-460-7 Shipping Container, Power Amplifier
OSE/VS-4-461-1 Solar Panel Unit Test Set
OSE/VS-4-461-2 Power Inverter Unit Test Set
OSE/VS-4-461-3 Battery Control Unit Test Set
OSE/VS-4-461-4 Power Control Electronic Assembly Unit Test Set
OSE/VS-4-461-5 Battery Unit Test Set
Electrical Distribution Subsystem
OSE/VS-4-471-1 Electrical Distribution Unit Test Set
OSE/VS-4-510 Thermal Control Subsystem
OSE/VS-4-510-1 Assembly and Handling Fixture, Spacecraft Louvres
OSE/VS-4-510-2 Shipping Container, Spacecraft Louvres
OSE/VS-4-510-3 Handling and Shipping Container, Insulation
OSE/VS-4-520 Structural Subsystem
OSE/VS-4-520-1 Dolly, Structural Sections
OSE/VS-4-520-2 Shipping Container, Miscellaneous Spacecraft Structure
OSE/VS-4-520-3 Sling, Propulsion/Pneumatic Structural Section
OSE/VS-4-520-4 Interface Match Tool, Spacecraft/Flight Capsule
OSE/VS-4-520-5 Interface Match Tool, Spacecraft/Centaur Adaptor

Subsystem (Continued)

- OSE/VS-4-530 Pyrotechnic Subsystem
  - OSE/VS-4-530-1 Shipping Container, Explosive Train
  - OSE/VS-4-530-2 Handling Case, Arming Kit
- OSE/VS-4-580 Planet Oriented Package Subsystem
  - OSE/VS-4-580-1 Assembly Fixture and Dolly, POP
  - OSE/VS-4-580-2 Shipping Container, POP
  - OSE/VS-4-580-3 Hoist Beam, POP
  - OSE/VS-4-581-1 POP Unit Test Set
- OSE/VS-4-610 Propulsion Subsystem
  - OSE/VS-4-610-1 Sling, Retropropulsion Motor
  - OSE/VS-4-610-2 Dolly, Retropropulsion Motor
  - OSE/VS-4-610-3 Alignment Fixture, Retropropulsion Motor
  - OSE/VS-4-610-4 Alignment Fixture, Midcourse Engine
  - OSE/VS-4-610-5 Shipping Container, Retropropulsion Motor
  - OSE/VS-4-610-6 Shipping Container, Midcourse Engine
  - OSE/VS-4-610-7 Pneumatic Test Set
  - OSE/VS-4-610-8 Pneumatic Fill Cart
  - OSE/VS-4-610-9 Propellant Transfer and Handling Cart
  - OSE/VS-4-610-10 Alignment Fixture, Midcourse Engine/ Steering Vanes
  - OSE/VS-4-610-11 Universal Handling Fixture, Hydrazine/Helium Tank
  - OSE/VS-4-610-12 Sling, Hydrazine/Helium Tanks

OSE OBJECTIVES AND CRITERIA

OSE/VS-1-110

Data contained in Section I of Volume 6 will be placed in specification format and will become OSE/VS- 1-110 at a time mutually acceptable to JPL and the TRW Voyager Team.

OSE DESIGN CHARACTERISTICS AND RESTRAINTS
OSE/VS-2-110

Data contained in Section II of Volume 6 will be placed in specification format and will become OSE/VS-2-110 at a time mutually acceptable to JPL and the TRW Voyager team.

LAUNCH COMPLEX EQUIPMENT
OSE/VS-2-120

1. SCOPE

This document establishes the requirements for the Voyager OSE electrical launch complex equipment (LCE) used to evaluate pre-launch performance tests of the Voyager spacecraft at the launch complex.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-110	Voyager OSE System Test Set
OSE/VS-3-120	Voyager OSE Automatic Data Handling System

Military Standards

MIL-STD-129	"Marketing for Shipment and Storage"
MS - 33586	"Metals, Definition of Dissimilar"

3. FUNCTIONAL REQUIREMENTS

The LCE is used to support Voyager spacecraft test activities in three major facilities at the launch complex. These facilities are the spacecraft assembly facility (SAF), the explosive safe facility (ESF), and the blockhouse/pad terminal facility.

The launch complex equipment utilizes the same equipment drawer design from the system test set (STS), and automatic data handling system (ADHS), and ordnance unit test set wherever the functional and design requirements can be satisfied. New drawer level design to implement the functional and design requirements of the launch complex equipment are kept to a minimum.

3.1 Spacecraft Assembly Facility

The functional requirements for the LCE in support of spacecraft test activities at this facility are identical to those for the STS and ADHS except for the additional requirements listed below:

a) The LCE concurrently counts down two Voyager/ planetary vehicles at the launch pad

b) Concurrent with the countdowns, LCE performs tests (integration system test and detailed checkout tests) on a "standby" Voyager spacecraft located in the assembly facility.

3.2 Explosive Safe Facility

The functional requirements for the LCE in support of the Voyager integrated planetary vehicle test activities at this facility are identical to those for the STS and ADHS except for additional continuity measurements required on the integrated spacecraft.

3.3 Blockhouse/Pad Terminal Facility

The functional requirements for the LCE in support of spacecraft test activities at this facility are divided into two functional time periods, i.e., prelaunch checkout (spacecraft checkout prior to removal of the launch gantry) and terminal countdown.

a. Prelaunch

The functional requirements for the LCE in support of spacecraft prelaunch test activities are identical to those for the STS and the ADHS. However, all tests which require hardline signals to be transmitted and/or received via the flight spacecraft test connectors are accomplished with the test equipment located on a platform of the gantry adjacent to the planetary vehicle.

b. Terminal Countdown

The functional requirements for the LCE in support of spacecraft terminal countdown test activities are identical to those for the STS and ADHS except for the deletion of those requirements which are satisfied by the test console in the STS.

4. DESIGN REQUIREMENTS

The design requirements for the launch facility equipment are the same as those for the STS and the ADHS.

The blockhouse console consists of an STS ground power console plus a remote controlled (from the ADHS) spacecraft status display panel which is identical to the spacecraft status display in the telemetry console of the STS. The design requirements for these units are identical to those discussed in the STS functional description. The power rack located in the terminal room consists of the same equipment provided in the ground power rack of the STS. The units (drawers) contained in the test console of the STS are readily removable from the rack and inserted into portable containers which can be carried onto the gantry. The portable containers are heavy duty watertight containers with slides installed so that an item of rack mount equipment may be removed from the rack, mounted in a container and used on the gantry.

The portable containers are constructed so that they can be stacked one on the other. Both ends of each container are capable of being removed for access and air circulation.

5. FUNCTIONAL DESCRIPTION

The functional description of the LCE is identical with the STS and ADHS functional descriptions except for geographical location of certain equipment. Figure 1 is the conception of the launch equipment facility layout. Figure 2, 3, and 4 are artist's conceptions of equipment located in the blockhouse, pad terminal facility, gantry, and explosion safe facility. Most of the STS and all of the ADHS are located in the spacecraft assembly facility. A ground power console and ground power rack are located at each blockhouse/pad terminal and at the explosive safe facility. The operation of the LCE is identical to the operation of the STS in conjunction with the ADHS.

"Zero" type containers are used to house the drawers in the test console so that they can be carried onto the gantry close to the spacecraft.

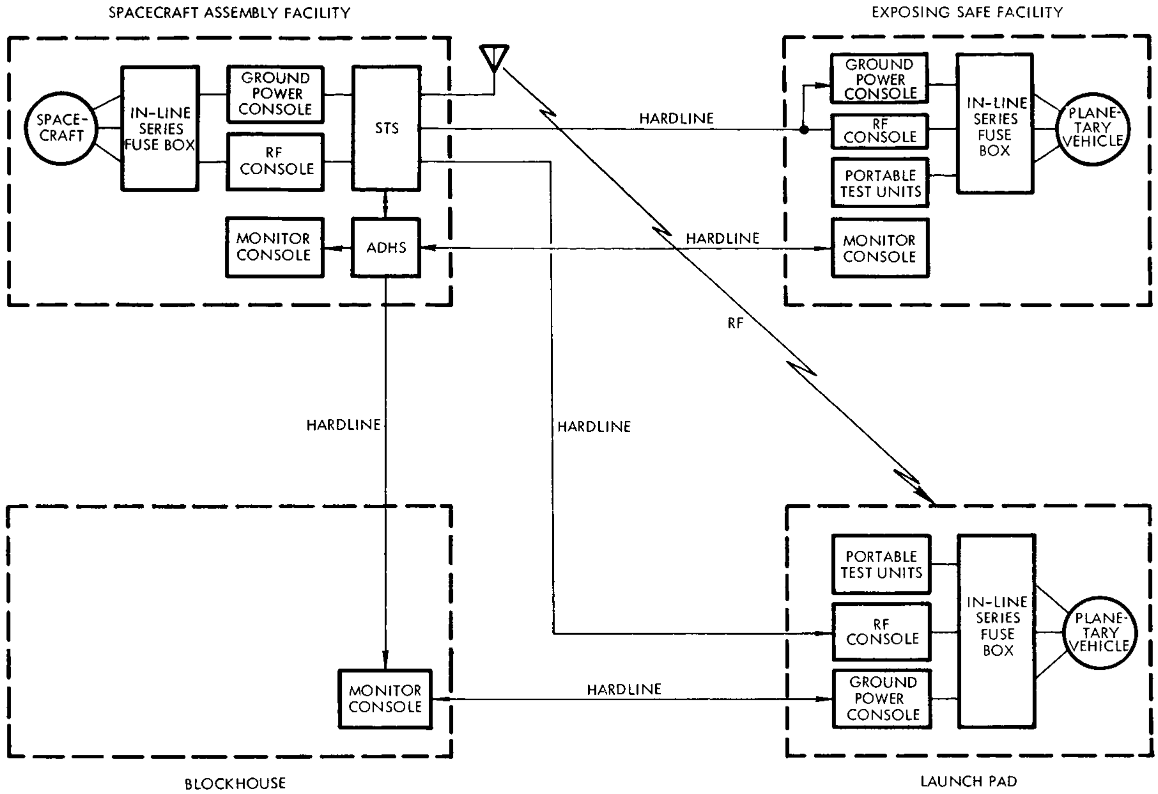


Figure 1. Launch Complex, Simplified Block Diagram

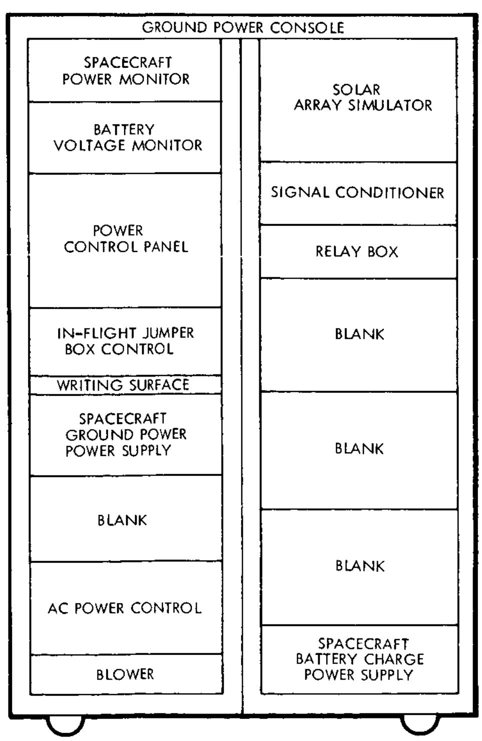


Figure 2. Battery Charge Rack

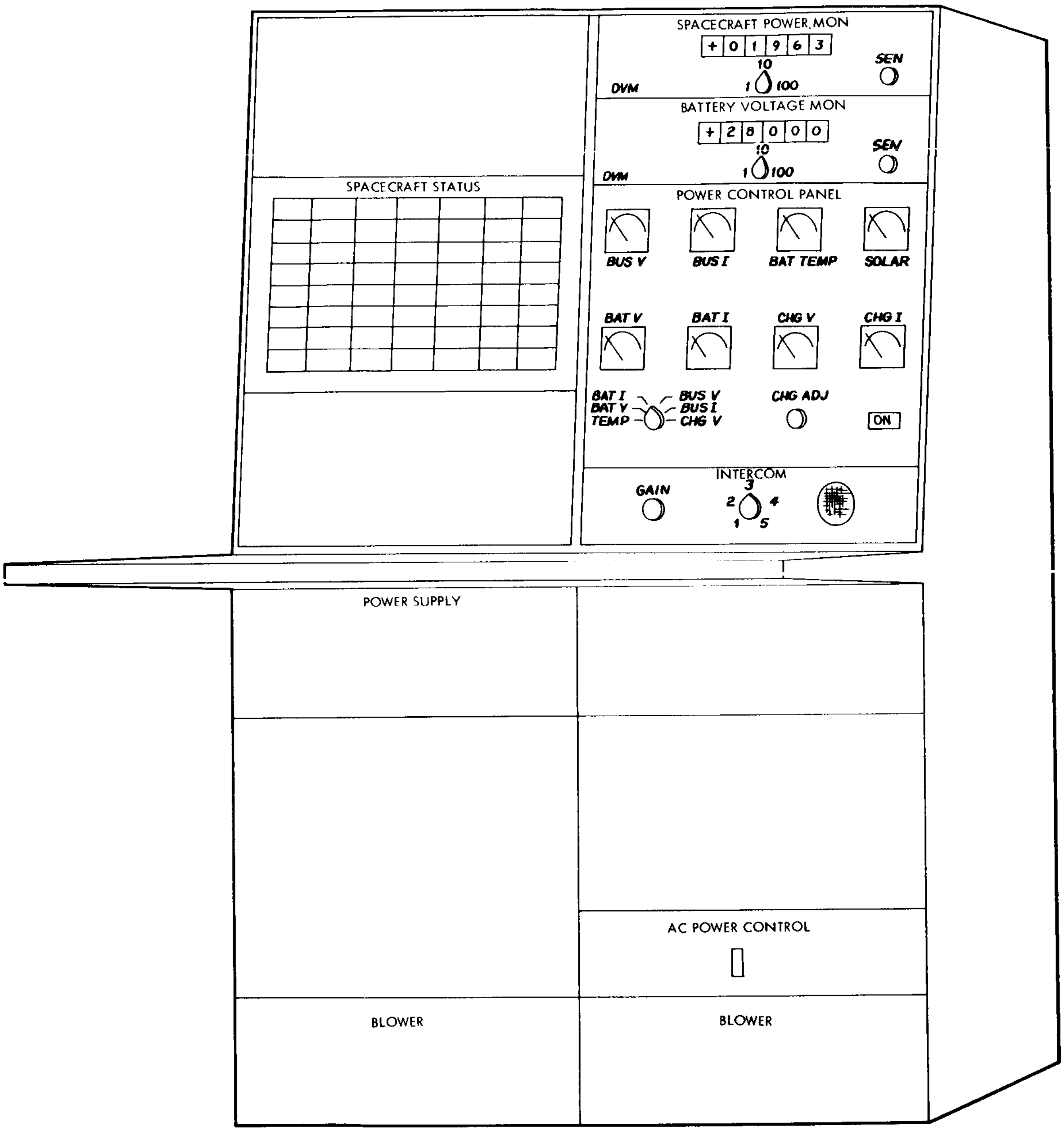


Figure 3. Launch Control and Monitor Rack

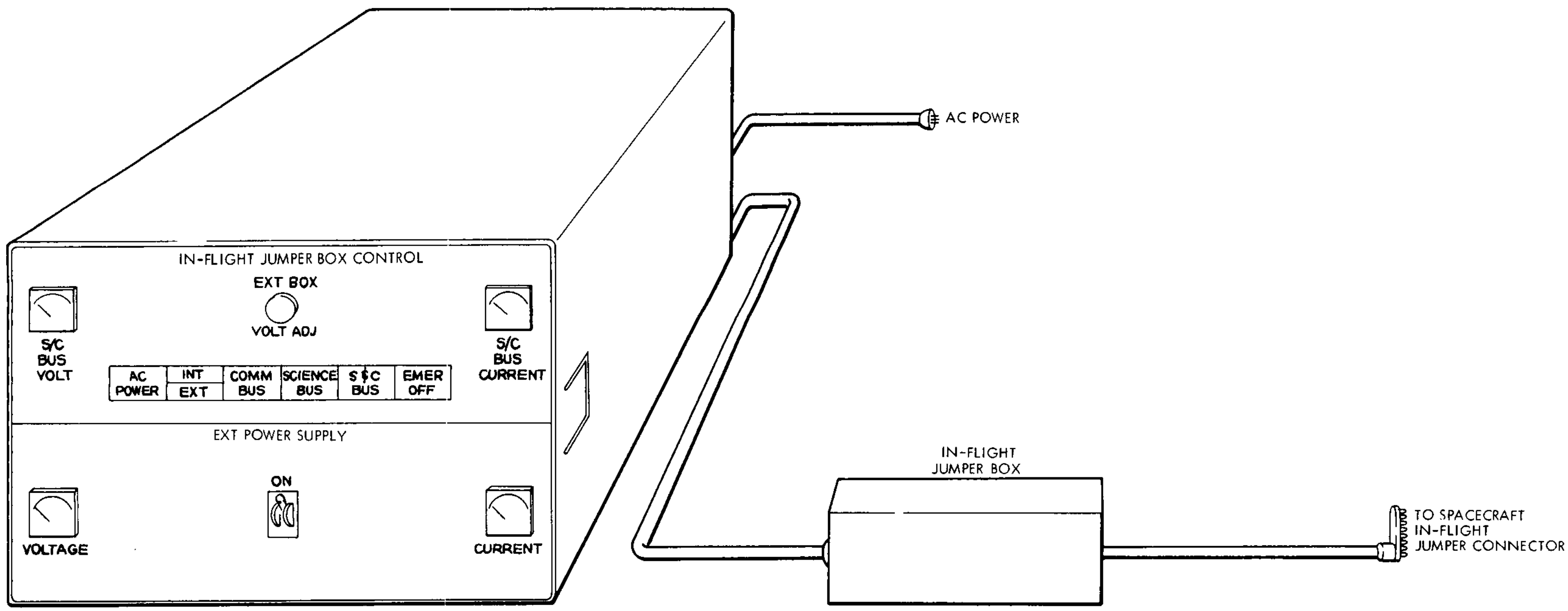


Figure 4. In-Flight Jumper Control

During the terminal countdown of the planetary vehicle the RF link will be utilized to command the planetary vehicle in its proper launch operating mode and monitor its performance.

5.1 Spacecraft Assembly Facility

Equipment at the SAF consists of:

a) Three RF consoles

b) Three telemetry data consoles

c) Three recorder consoles

d) One test console

e) One ground power console and rack

f) Two automatic data handling systems.

5.2 Blockhouse Facility

Equipment at the blockhouse facility consists of:

a) One ground power console

b) One spacecraft status display located in the ground power console.

5.3 Pad Terminal

Equipment at the pad terminal consists of:

a) One ground power rack

b) One set of test console drawers mounted in individual portable containers.

5.4 Explosion Safe Facility

Equipment at the ESF consists of:

a) One ground power console

b) One spacecraft status display located in the ground power console

c) One ground power rack

d) One set of test console drawers mounted in individual portable containers.

6. BOUNDARY DEFINITIONS

The individual console power is as follows:

Voltage	115 VAC ± 10 VAC
Frequency	60 ± 1 cps
Phase	Single (3-wire)
Current demodulator	Not to exceed amps

The integrated STS primary power is as follows:

Voltage	115 VAC ± 10 VAC
Frequency	60 ± 1 cps
Phase	Three (4-wire)
Current demodulator	Not to exceed amps

7. PARAMETERS

The parameters for the electrical LCE are identical to those discussed in the STS functional description.

8. CONSTRAINTS

The constraints for the electrical LCE are identical to those discussed in the STS functional description.

SYSTEM TEST SET
OSE/VS-3-110

1. SCOPE

This document establishes the requirements for the system test set (STS), a part of the electrical operational support equipment.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-110	Voyager OSE Automatic Data Handling System
OSE/VS-2-120	Voyager OSE Launch Complex Equipment

Military Standards

MIL-STD-129	"Marking for Shipment and Storage"
MS 33586	"Metals, Definition of Dissimilar"

3. FUNCTIONAL REQUIREMENTS

The STS in conjunction with the OSE automatic data handling system (ADHS) is used to support spacecraft test activities from initial spacecarft integration through spacecraft integrated system test, prototype qualification tests, flight environmental acceptance tests and preflight tests (at the launch complex) to and including the terminal launch countdown. The STS functional requirements are divided into the following categories:

- Commands to spacecraft
- Data acquisition from spacecraft
- Data processing and display (accomplish in conjunction with the OSE automatic data handling system)
- Stimulation
- Simulation
- Ground power
- Critical spacecraft monitoring
- STS self-test/fault isolation.

3.1 Commands to Flight Spacecraft

Commands are sent to the flight spacecraft to obtain all possible modes of spacecraft operation, i.e., the STS is capable of transmitting commands identical to those which would be transmitted to the flight spacecraft from the mission dependent equipment if the planetary vehicle were in space. In addition, commands are sent to individual units on the spacecraft to support subsystem integration and to facilitate fault isolation aboard the spacecraft.

3.1.1 Radiated and Hardline RF Signals

In orbit, the spacecraft receives commands via the radiated RF link. The STS has the capability of radiating commands to the spacecraft for short distances (less than 40 feet) in a checkout area (e.g., a factory or SAF) or for long distances (up to several miles) when the spacecraft is on the launch pad and the majority of the STS remains in the SAF. The capability of hardlining RF commands from the STS to the spacecraft is required during test activities which preclude use of radiated energy, for example, during simultaneous checkout of two spacecrafts or when RF silence is imposed due to an impending launch at the launch site.

3.1.2 Other Command Signals

Commands are provided by the STS which bypass the RF equipment in both the STS and the spacecraft. This hardline command signal is supplied directly to any one of the spacecraft decoder units.

3.2 Data Acquisition from Spacecraft

The STS is capable of acquiring data from the spacecraft. The principal communication path for spacecraft data acquisition is the telemetry link.

3.2.1 Radiated and Hardline RF Signals

In orbit, the spacecraft transmits telemetry data via the radiated RF link. The STS has the capability of receiving the radiated telemetry data from the spacecraft when the spacecraft is located several feet to several miles from the STS. In addition, hardlining of the telemetry data from the spacecraft to the STS is required during certain test activities for the same reasons as discussed above for the RF command hardline link.

The STS extracts the subcarrier from the incoming signal and supplies the extracted signal to the data processing portion of the OSE (STS and ADHS), to the STS magnetic tape recorder, or to both. The magnetic tape recorder provides the means of fault isolation and determining drift trends of spacecraft subsystems by permitting the recorded data to be played back into the data processing portion of the OSE (STS and ADHS) for evaluation at a later time.

3.2.2 Other Data Acquisition Signals

The STS is capable of receiving telemetry data which bypasses the RF equipment in both the spacecraft and the STS. This is accomplished by supplying the spacecraft digital telemetry unit (DTU) signals to the STS. The STS is also capable of recording the data on magnetic tape during this mode of operation.

3.3 Data Processing

The STS is capable of extracting the telemetry data from the incoming modulated signal supplied by the STS S-band receiver, by the spacecraft data handling subsystem, or as a playback from the STS magnetic tape recorder. The STS in conjunction with the OSE automatic data handling system (ADHS) processes the data by providing the following capabilities:

a) Selection of telemetry words from both the main frame and subcommutated frame

b) Conversion of telemetry words from binary to decimal numbers

c) Printout of all telemetry words or groups of telemetry words

d) Conversion of telemetry words to analog signals and display these on a strip chart (analog) recorder

e) Display of the contents of any selected telemetry word in decimal or octal

f) Display of the status of the spacecraft (e.g., battery on, battery off, ordnance armed, ordnance not armed, etc.) as indicated in the STS received telemetry data.

3.4 Stimulation

The STS provides stimulation to the spacecraft sun and star sensors, gyros, approach guidance sensor, Mars sensor, and solar cell modules. The purpose of stimulation is to make a gross check of the spacecraft stabilization and control subsystem, planet oriented package (POP), and power subsystem.

3.5 Simulation

The STS provides for simulation of the spacecraft sensors' activation (sun, earth, star, Mars, gyros, and approach guidance) to make possible a checkout of the stabilization and control subsystem. Simulation is also provided for the actuation of the spacecraft separation and deployment microswitches for tests of ordnance circuits and stabilization and control circuits. Simulation of the solar array output is provided to checkout the power subsystem under various solar array output conditions. The ranging code is simulated by the STS to provide a checkout of the telecommunications ranging circuitry. Finally, simulation of the interface signals between the spacecraft and capsule is provided to assure a high degree of confidence of detecting electrical interface problems prior to mating the spacecraft with the capsule.

3.6 Ground Power

The STS provides, controls and monitors ground power for the spacecraft. Provision for monitoring the spacecraft bus and spacecraft battery voltages is also included. The ground power voltage is variable (i.e., it simulates the output of the solar array) to enable simulation of different spacecraft positions in space. In addition, the STS provides the capability of charging and monitoring the performance of the spacecraft battery during spacecraft test activities.

3.7 Critical Spacecraft Monitoring

The STS provides for monitoring the three S-band and two VHF spacecraft receivers. The monitors required are signal strength, signal presence, loop phase detector, and loop stress. These monitors determine receiver characteristics in detail while integration tests are being performed.

3.7.2 Spacecraft Ordnance Monitoring

The STS provides for monitoring the spacecraft ordnance firing circuit outputs and simulating the spacecraft ordnance loads. Thus, an evaluation of the proper operation of the ordnance firing circuits can be obtained. In addition, a means of monitoring the ordnance circuits safe and arm condition is provided.

3.7.3 Spacecraft Stabilization and Control

The STS provides the capability of self-testing to a degree sufficient to determine its readiness prior to spacecraft testing and its GO/NO-GO status if a fault is detected during spacecraft tests.

4. DESIGN REQUIREMENTS

The STS is composed of an assembly of dolly-mounted equipment racks which are grouped functionally. Specifically, the functional groupings are:

- Radio Frequency Console
- Telemetry Data Console

- Recorder Console
- Ground Power Console
- Test Console.

Each of the above consoles operates independently of the other consoles of the STS or in conjunction with them.

The design requirements (implementing the functional requirements) of each of the consoles follow.

4.1 RF Console

The RF console provides the principal active link between the spacecraft and the STS. In performing this function it acts as a ground station to transmit commands to and receive data from the spacecraft. It also has the capability of making certain performance measurements that are required. In order to accomplish these functions it must be designed to the requirements discussed in the following paragraphs.

4.1.1 S-Band Command Carrier

The RF console generates an S-band command carrier to perform the following functions:

a) Testing spacecraft receiver sensitivity

b) Testing spacecraft receiver acquisition range

c) Testing spacecraft receiver frequency tracking rates

d) Carrying command and PN ranging signals modulated on the carrier of known modulation index

e) Providing an auxiliary RF power level sufficient to allow testing several miles from the spacecraft (e.g., at the launch complex).

4.1.2 S-Band Telemetry Carrier

The RF console receives an S-band telemetry and range code carrier and accomplishes the following functions:

a) Extracting the telemetry/PN subcarrier from the carrier

b) Measuring the carrier modulation index

c) Exercising turn-around ranging.

4.1.3 RF Power Measurements

The RF console makes RF power measurements as follows:

a) Spacecraft transmitter power output — all modes (low, medium, and high)

b) Spacecraft RF power amplifier power output

c) STS transmitter power output both at low and high RF levels.

4.1.4 RF Frequency Measurements

The RF console makes RF frequency measurements as follows:

a) Spacecraft transmitted carrier frequency

b) Spacecraft subcarrier frequencies

c) STS transmitter frequency

d) STS VHF capsule transmitter frequency.

4.1.5 VHF Capsule Data Carrier

The RF console generates a VHF capsule data carrier to perform the following functions:

a) Testing spacecraft VHF receiver sensitivity

b) Testing spacecraft VHF receiver frequency and tracking range

c) Carrying simulated capsule data as modulation on the carrier of known modulation percentage.

4.1.6 Command/PN Subcarrier

The RF console generates the appropriate command/PN subcarrier to modulate the up-link carrier for the following purposes:

a) Transmitting discrete commands to the spacecraft command decoders (selected by code scheme)

b) Testing the spacecraft response to commands

c) Notifying the computer of the commands sent.

4.1.7 Oscilloscope

The RF console provides an oscilloscope to observe and evaluate the following:

a) Spacecraft telemetry/PN subcarrier

b) Spacecraft command/PN subcarrier

c) Capsule data subcarrier

d) STS receiver loop test points.

4.1.8 Test Points

The RF console provides test points to monitor the various receiver and transmitter functions with test equipment external to the STS for STS calibration and spacecraft evaluation.

4.1.9 Antennas and Coaxial Cable

The RF console provides sufficient antennas and coaxial cable to permit simultaneous operation (transmit and receive) over one or more spacecraft antennas or antenna connections.

4.2 Telemetry Data Console

The telemetry data console in conjunction with the automatic data handling system (ADHS) collects, processes, displays and records data received from the spacecraft via the telemetry link. In addition, it has the capability of generating a command signal to the STS command encoder to provide the ability of sending commands to the spacecraft automatically. The computer (part of the ADHS) collects the telemetry data, extracts engineering and experiment words, converts binary data to binary coded decimal (BCD) equivalents and formats the information for display (STS) and recording (ADHS).

The majority of the equipment in the system is controlled by the computer, which receives its instructions from a program stored in the computer memory. The computer, under program control, performs the following functions:

a) Controlling test sequence

b) Generating commands

c) Establishing synchronization

d) Monitoring synchronization status on a continuous basis

e) Determining the telemetry format being received

f) Providing appropriate processing of the telemetry data

g) Acknowledging and checking commands transmitted to the spacecraft manually

h) Establishing loss of frame synchronization criteria.

By specifying the location of engineering and experiment words in the main frame and subcoms, all of the received data is processed, selected and displayed.

The telemetry data console is designed to perform the functions discussed below.

4.2.1 Demodulator Unit

The telemetry data console provides a demodulator unit to convert the telemetry subcarrier signal (as received from the STS receiver, STS magnetic tape recorder, hardline from the spacecraft digital telemetry unit, or from a telemetry data format generator which is utilized for STS self-test) into serial pulse code modulation (PCM) digital format.

The demodulator unit provides its output signals to a buffer unit and to a recorder console.

4.2.2 Buffer Unit

The telemetry data console provides a buffer unit for a central distribution point for all digital data between the digital computer (ADHS) and other components of the telemetry data console. The buffer accepts an instruction from the computer, remains in the ready state for the duration of the received input, and "addresses" the unit specified by the instruction code. When addressing is completed, data may be transferred to either direction between the computer and the selected unit.

The buffer unit accepts signals from the Demodulator unit, facility time code generator unit, data format generator (for self-test), command encoder, telemetry data displays, and digital computer. It provides output signals to the digital computer (ADHS), spacecraft status display, telemetry data display, and digital-to-analog converters.

4.2.3 Data Format Generator

The telemetry data console provides a data format generator to simulate normal STS telemetry receiver output information and nominal demodulator output signals. The signals are available for test purposes (STS self-test) to portions of the STS in the formats and modes of the Voyager system. The data format generator allows manual selection of mode, format, and bit rate transmission. In addition, it allows manual selection of contents of the telemetry words by addressing any one main frame word and fixing the contents of the word to any arrangement of bits, or selecting either a fixed pattern for all words or a distinctive word number for each word.

The data format generator provides output signals to the demodulator unit, command transmitter unit (RF console), and buffer unit.

4.2.4 Spacecraft Status Display Unit

The telemetry data console provides a spacecraft status display unit to display up to 100 bits of spacecraft status information. This information, extracted from the spacecraft telemetry data, is continuously held and updated. The display is in the form of indicator lamps.

The spacecraft status display unit receives its input signals from the buffer unit.

4.2.5 Telemetry Data Display Units

The telemetry data console provides telemetry data display units which permit manual selection and display of any telemetry or experimental words (up to six words) in decimal or octal notation. Identification of the displayed word consists of the main frame word or subcom word number. Mode, format, bit rate, sync status, and parity error status are also displayed. Should the computer lose telemetry synchronization, the telemetry data display units will retain the information received prior to synch loss. The telemetry data display units are capable of holding and updating all input information.

All input and output signals of the telemetry data display units are to/from the buffer unit.

4.3 Recorder Console

The recorder console provides the capability of recording various selected functions (from both the STS and spacecraft) for display, analysis and playback into the remaining portion of the STS. It consists of three major assemblies.

4.3.1 Instrumentation Patch Panel

An instrumentation patch panel is required to provide facilities to switch signals from specified equipment to select input channels of a magnetic tape recorder and a direct-write analog recorder, as well as to provide a means of obtaining the magnetic tape recorder output to any selected equipment input. The panel is equipped with 125 jacks to handle the signal inputs and outputs. In addition, the instrumentation patch panel supplies the necessary signal conditioning equipment to provide compatibility between the recording equipment and the source of recorded data.

4.3.2 Analog Recorder

a. Direct-Write Analog Recorder

- Outputs of the digital-to-analog converters from the ADHS (processed spacecraft telemetry data)

- Selected performance parameters from the spacecraft telemetry receivers

- Spacecraft stabilization and control input and output signals

- Output signal from the facility time code generator (for time correlation).

The recorder is an eight-channel, hot stylus type recorder with the following characteristics:

- Inputs: Eight direct input circuits for direct data recording; one circuit for a marker channel which has a capability of producing one-second timing marks or remote event marking

- Input Impedance: Channels 1 through 7 are balanced to ground, 5 megohms each side; channel 8 is 100 kilohms, single ended

- Input level: 0.5 to 200 mm per second

- Paper speed: 0.5 to 200 mm per second

- Frequency response: DC to 3 db down at 150 cps.

b. High Frequency Oscillograph Recorder

A high frequency oscillograph recorder is required to record the 511 cps Command/PN signal. This recorder is a portable galvo-mirror type utilizing ultra violet light and sensitized paper. The recorder is stowed in a spare drawer of the STS and patched into the Patch Panel when used.

4.3.3 Magnetic Tape Recorder

A magnetic tape recorder is required for permanent recording of selected signals received from the sapcecraft during integration and acceptance testing of the spacecraft system. The recorder console has the capability of recording the spacecraft data on the magnetic tape recorder concurrent with STS and ADHS processing of spacecraft data in real time. In addition, the magnetic tape recorder provides the capability of data playback into the data processing portion of the STS and ADHS for evaluation at a later time. Examples of the types of signals to be recorded are:

a) Video data output from the STS receiver (spacecraft telemetry data)

b) Digital binary data output from the demodulator unit

c) Spacecraft telemetry data obtained directly (via hard-line) from DTU

d) Output signals from the command encoder

e) Output signal from the facility time code generator

f) Voice (for annotation of recorder data)

g) Output signal from the data format generator (utilized during self-test of the STS.

The recorder contains seven channels (plus an additional channel for voice annotation) with the following characteristics:

a) Direct record/reproduce frequency response 400 cps to 2 Mc at 120 IPS

b) FSM record/reproduce 288 K bits per sec at 120 IPS

c) FM wideband record/reproduce 0-50 KC at 120 IPS.

4.4 Ground Power Console

The ground power console provides the following control functions:

a) Internal-external spacecraft power select

b) Power control for the spacecraft individual branch busses

c) Battery charging of spacecraft battery

d) Solar array simulation

e) Spacecraft timer reset.

Additionally, it provides the following:

a) Ground power current and voltage

b) Spacecraft battery charge current and voltage

c) Spacecraft battery discharge current

d) Spacecraft battery temperature

e) Spacecraft bus voltage

f) Spacecraft bus current

g) Spacecraft power status (external or internal power)

h) Spacecraft branch power bus status

i) Ordnance status (arm/safe)

j) Power on-time.

The ground power console contains ten major assemblies, as described in the following paragraphs.

4.4.1 Power Control Panel

The power control panel controls the following:

a) AC power to the spacecraft battery charge power supply

b) Application of the spacecraft battery charge power to the spacecraft

c) Voltage and current limits of the spacecraft battery charge supply when charging the spacecraft battery

d) Ground power mode of operation, i.e., solar array simulation mode or spacecraft battery charge mode

e) Functions to be observed on the spacecraft power monitor

f) Reset of the spacecraft timer.

In addition, it provides continuous analog monitoring of spacecraft battery charge current, spacecraft battery charge voltage, battery discharge current, battery temperature, bus voltage, bus current, and ordnance status.

4.4.2 Spacecraft Power Monitor

The spacecraft power monitor provides an accurate (±1 millivolt) 6-place digital display of spacecraft power functions as selected by the power control panel.

4.4.3 Battery Voltage Meter

The battery voltage monitor continuously provides an accurate (±1 millivolt) 6-place digital display of the spacecraft battery voltage.

4.4.4 In-flight Jumper Simulator Control

The in-flight jumper simulator control monitors ground power supply current and spacecraft bus voltage and current; controls and monitors ground power supply voltage and the application of power to the individual spacecraft branch busses; and controls internal-external spacecraft power. It also provides emergency control of power shutdown to the spacecraft.

4.4.5 Spacecraft Ground Power Supply

This supply provides 0 to 50 volts,   amps to the spacecraft.

4.4.6 In-flight Jumper Simulator

The in-flight simulator replaces the spacecraft in-flight jumper during spacecraft checkout. It provides the means to select by hardline signals the spacecraft power branches and spacecraft source of power.

4.4.7 Solar Array Simulator

The solar array simulator simulates the spacecraft power as received from the spacecraft solar arrays. The voltage (E) and current (I) outputs simulate the E/I curves of the solar array for various ranges from the sun. The simulated solar array outputs are also available as test points on the front panel for recording the signals transmitted to the spacecraft power bus. In addition, a preloading of the signal outputs is provided so that output levels can be established prior to application to the spacecraft.

4.4.8 Signal Conditioner

The signal conditioner provides complete isolation of the spacecraft and lander DTU output signals from the STS. It also contains level adjustments and test points for establishing compatible hardline communication with the remaining portion of the STS.

4.4.9 Relay Box

The relay box provides the means of remotely switching (via relay) the spacecraft power mode of operation, i.e., battery charge or solar array simulation. In addition, it acts as a function box between the spacecraft and power control panel.

4.4.10 Battery Charge Power Supply

The battery charger power supplies the ground power source for charging the spacecraft battery and the power source for solar array simulation.

4.5 Test Console

The test console is required for detail checkout and monitoring those critical spacecraft functions which cannot be tested or monitored by the RF and telemetry links.

In addition, the test console acts as a spacecraft/STS junction box in that all hardlines, except for ground power and RF connections, between the spacecraft and ground equipment, pass through the test console. The test point monitor and control panel located in the test console is utilized as the "single point" location of interface wiring.

This console tests the following:

a) Correct phasing and gain between the stabilization and control (S and C) sensors and the gas jet and the engine steering actuators

b) S and C mode control (e.g., launch mode; earth, sun, and star acquisition; cruise; midcourse direction and velocity correction; deboost; and orbit modes)

c) Gyro assembly output phasing through the S and C control electronics assembly to the gas jet and engine steering actuators

d) Spacecraft ordnance firing circuits

e) Correct phasing and gain between the S and C and the planet oriented platform

f) Correct phasing and gain between the S and C and the antenna servos.

It also receives, buffers and provides the biphase modulated signal from the spacecraft DTU to a test jack, monitors "signal present" and signal strength signals from each spacecraft receiver, and monitors loop detector signals for receiver figure of merit.

Additionally, the test console validates the spacecraft hardline operation of the STS, providing confidence to test personnel that the STS is operating properly.

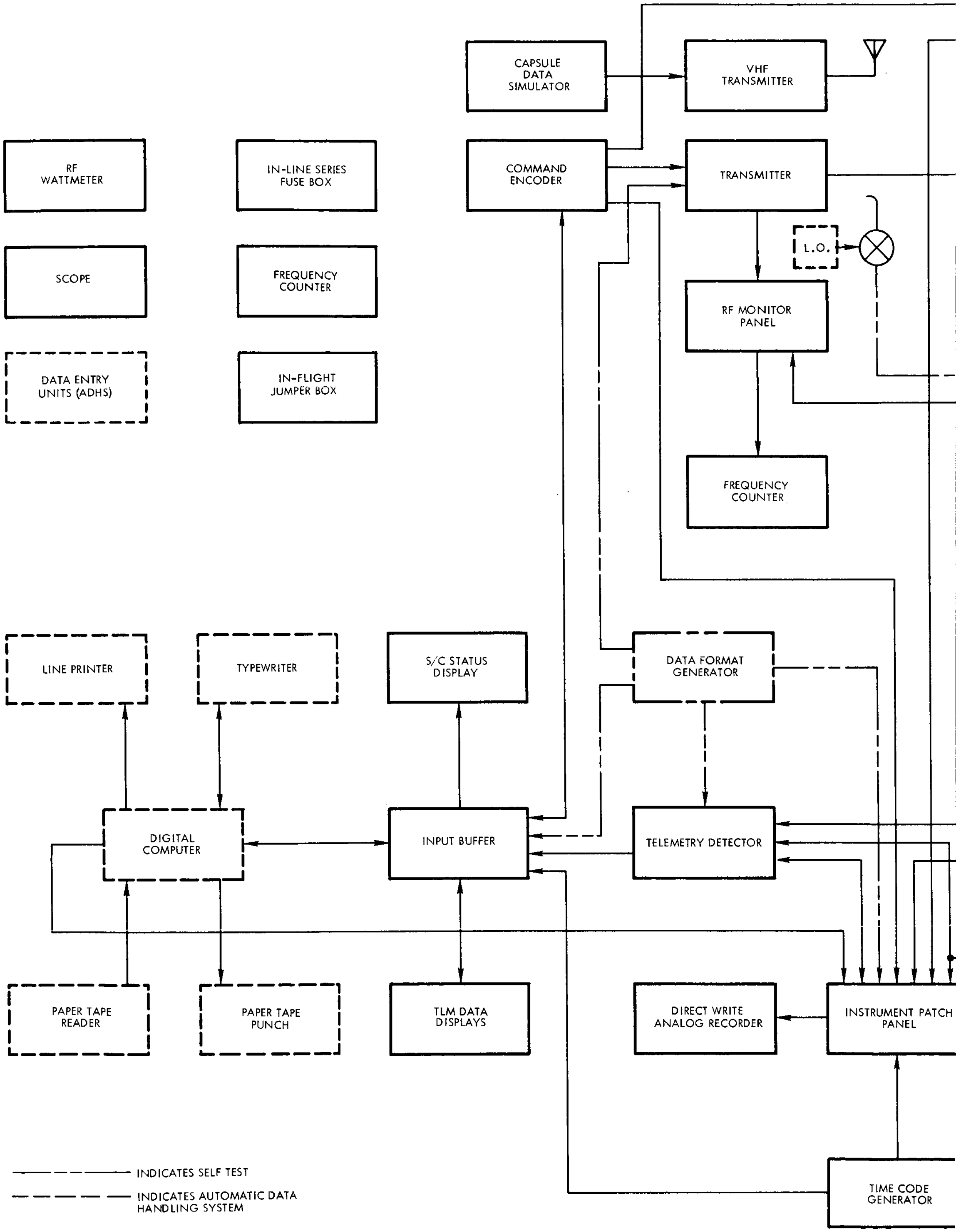
5. FUNCTIONAL DESCRIPTION

The STS serves as the central point for integrated systems tests. It contains, in conjunction with the ADHS, the command stimulus generators and the data acquisition, processing, measurement, display, and related equipment required for exercising and evaluating the operation of the Voyager spacecraft. A simplified block diagram is shown in Figure 1.

The prime communication path with the spacecraft is the RF link. A minimum number of hardline connections is required. This minimum is achieved by making maximum use of the telemetered functions for subsystem performance evaluation. The few hardlines provided are required for transmission of certain simulation and fault isolation signals (e.g., sun sensor simulation and command monitoring) which could not be accomplished over the RF link. Hardline interfaces with the spacecraft in no way degrade the reliability of the spacecraft subsystems nor do they provide an unrealistic test condition.

Self-test of the STS is by closed loop testing of the RF functions and by utilizing a data format generator as a source of simulated telemetry data. In addition, other functions, such as power supplies, are continuously monitored. Fault isolation to a replaceable unit is accomplished with general purpose test equipment.

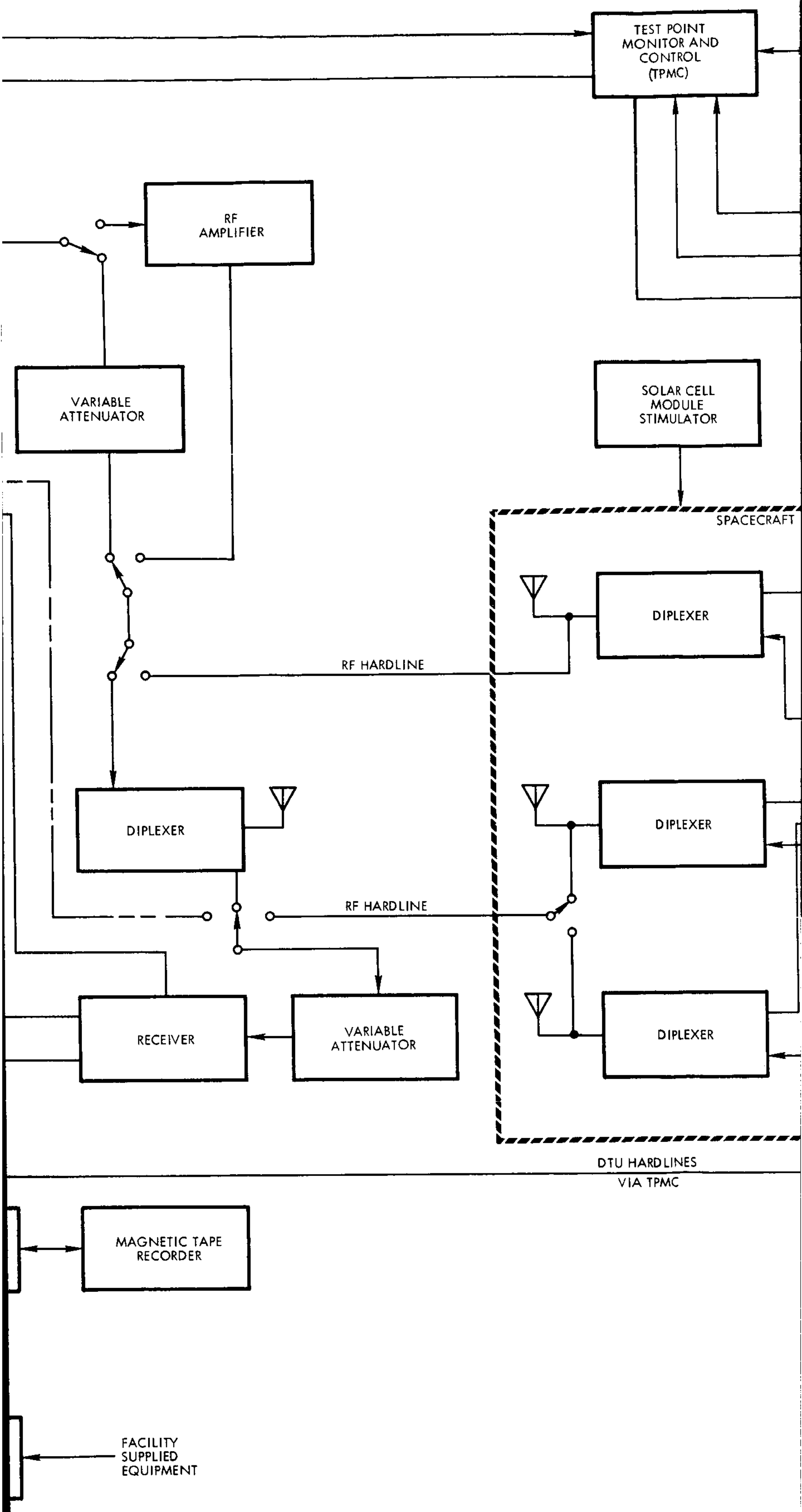
Commercial and specially designed equipment is assembled in rugged electronic equipment racks and carrying cases. The racks have recessed front panels for protection of meters and protection against inadvertent mid-adjustment. Certain individual drawers, which can be locked in place, are slide-mounted for maximum serviceability. Individual rack blower units are provided to circulate cooling air.



——— — — ——— INDICATES SELF TEST

— — — — — — INDICATES AUTOMATIC DATA HANDLING SYSTEM

310


TEST POINT MONITOR AND CONTROL (TPMC)
RF AMPLIFIER
VARIABLE ATTENUATOR
SOLAR CELL MODULE STIMULATOR
SPACECRAFT
DIPLEXER
RF HARDLINE
DIPLEXER
DIPLEXER
RF HARDLINE
RECEIVER
VARIABLE ATTENUATOR
DIPLEXER
DTU HARDLINES
VIA TPMC
MAGNETIC TAPE RECORDER
FACILITY SUPPLIED EQUIPMENT

3 1 (2)

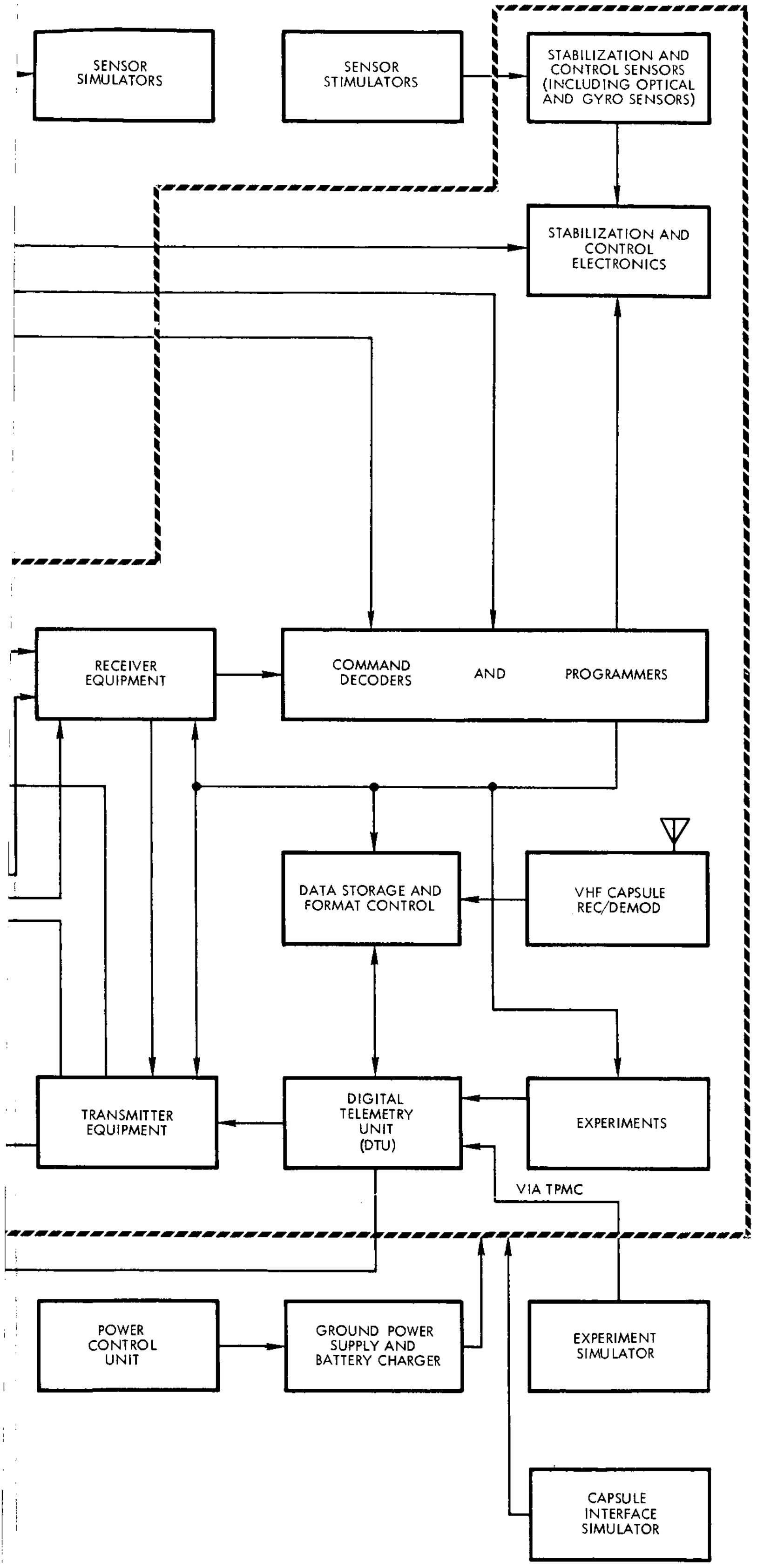


Figure 1. System Test Set, Block Diagram

3

Intra- and inter-rack cabling is located at the rear of the cabinets. To facilitate movement to the various test areas within TRW, the Malibu Magnetic Test Facility and the launch site, the racks are supported on caster dollies to allow limited mobility when the tie-downs are released.

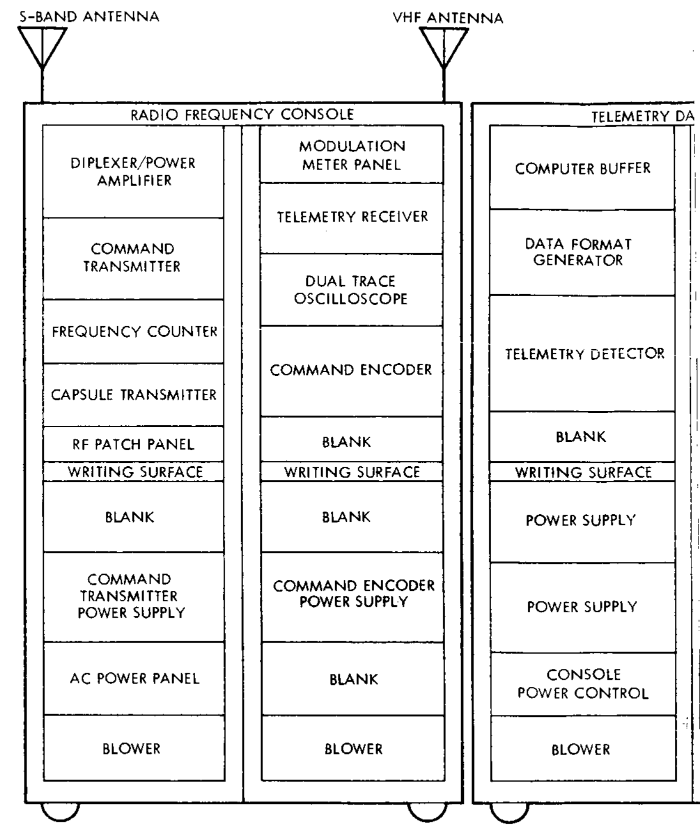
The STS consists of six consoles as shown in Figure 2. Each console can operate independently or in conjunction with other STS consoles. Descriptions of each of the consoles are given in the following paragraphs.

5.1 RF Console

The RF console operates in two primary modes which are called radiated RF lin and conducted RF link, and which differ in the medium used to communicate with the spacecraft. These modes may be combined to form a secondary or composite mode. In addition to these modes of spacecraft/test operation, there is a console self-test mode of operation. A console mode list is shown below:

- Radiated — low power, medium power, high power
- Conducted — low power, test (low power only)
- Composite — low power

Figure 3 depicts the console in the radiated low power mode. The command/PN signal format is generated in the command encoder and PN generator and is used to modulate the command transmitter. Monitors are provided at this point to establish the transmitter output power level and frequency. The modulated S-band up-link carrier is directed to the variable attenuator (0-160 db), the to the diplexer (transmitter port) and antenna. The downlink carrier is received at the antenna and passes through the diplexer (receiver port) and through the variable attenuator (0-60 db). The signal then arrives at the telemetry receiver where the subcarrier is extracted and sent to the demodulator and/or magnetic tape recorder.


S-BAND ANTENNA
VHF ANTENNA
RADIO FREQUENCY CONSOLE
DIPLEXER/POWER AMPLIFIER
COMMAND TRANSMITTER
FREQUENCY COUNTER
CAPSULE TRANSMITTER
RF PATCH PANEL
WRITING SURFACE
BLANK
COMMAND TRANSMITTER POWER SUPPLY
AC POWER PANEL
BLOWER
MODULATION METER PANEL
TELEMETRY RECEIVER
DUAL TRACE OSCILLOSCOPE
COMMAND ENCODER
BLANK
WRITING SURFACE
BLANK
COMMAND ENCODER POWER SUPPLY
BLANK
BLOWER
TELEMETRY DA
COMPUTER BUFFER
DATA FORMAT GENERATOR
TELEMETRY DETECTOR
BLANK
WRITING SURFACE
POWER SUPPLY
POWER SUPPLY
CONSOLE POWER CONTROL
BLOWER

33E ①

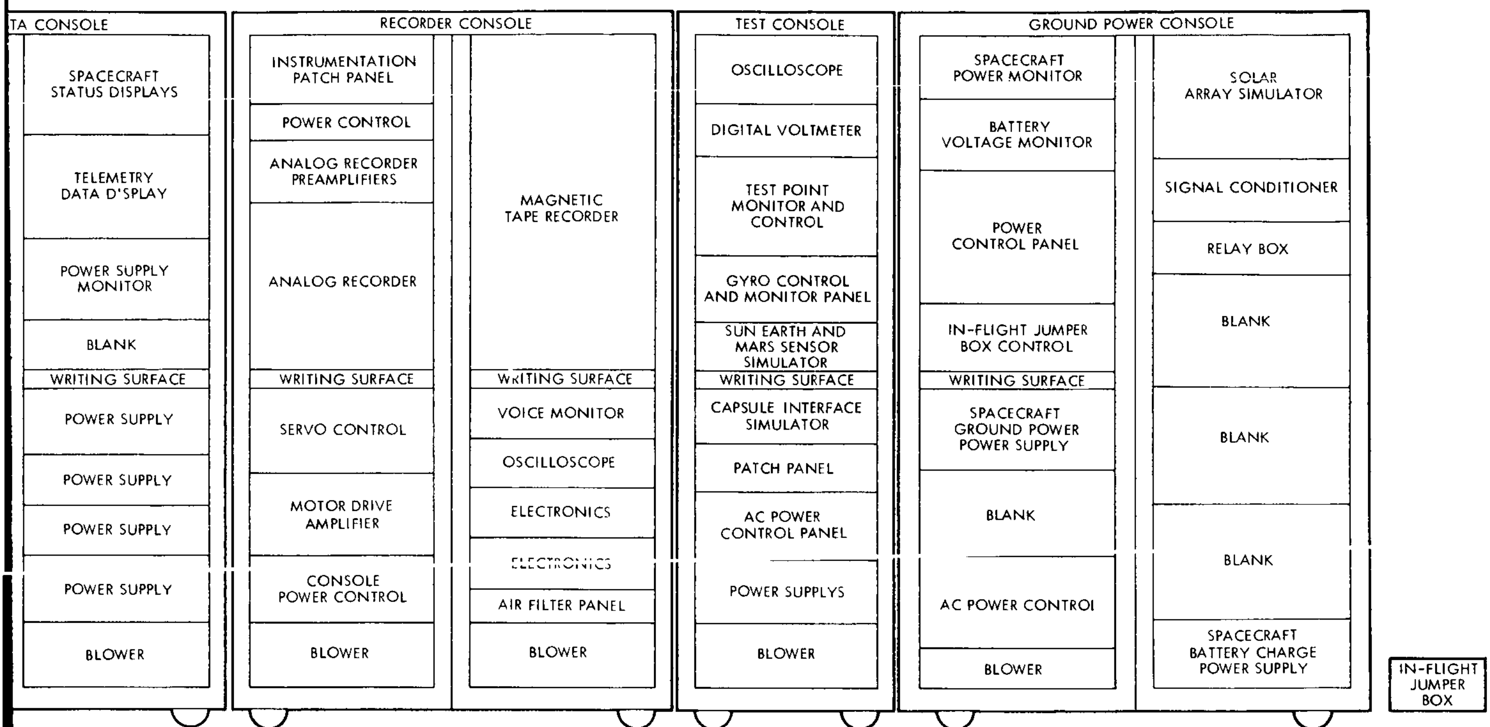


Figure 2. System Test Set, Rack Layout

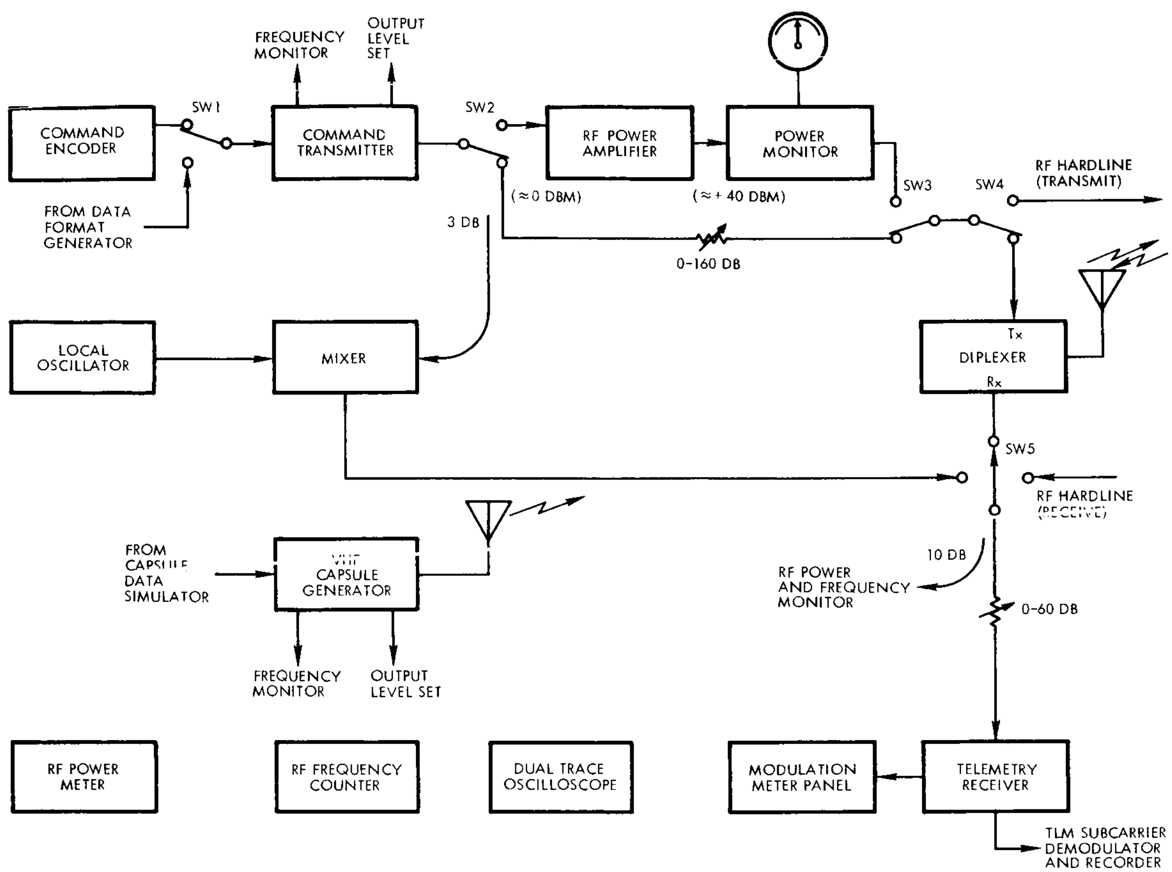


Figure 3. RF Console, Block Diagram

The operation of the radiated high power mode is identical to that of the radiated low power mode with the exception that switches SW2 and SW3 are positioned to direct the uplink carrier through the RF power amplifier in lieu of the variable attenuator. This increases the power level to enable transmitting commands over extended ranges.

The conducted low power mode differs from the radiated low power mode only in that switches SW4 and SW5 are positioned to direct the uplink and downlink carriers through the coaxial RF hardlines instead of through the diplexer. In this case, these hardlines are connected directly to the spacecraft.

The conducted test mode is to provide both a closed loop console self test of the downlink data system, and a verification of the uplink command system. In this mode the signal from the transmitter is mixed with a signal from the local oscillator to produce a frequency equal to the downlink carrier frequency. Switch SW5 is set to direct this mixed output signal to the telemetry receiver. The demodulator subcarrier is processed as simulated telemetry data (originating at the data format generator) or observed for correct ocmmand content as applicable.

The composite operation is identical to the radiated low power mode except that switches SW4 and SW5 are positioned such that a combination radiated/conducted situation occurs. One carrier, either uplink or downlink is carried over the coaxial RF hardline while the other is radiated.

The VHF capsule generator simulates the VHF carrier from the capsule and may be operated in conjunction with any of the modes mentioned previously.

The RF power meter, RF frequency counter, dual trace oscilloscope and modulation meter panel are used to make performance measurements on both the STS and the spacecraft.

5.1.1 Command Transmitter

The command transmitter is an S-band frequency source having the following characteristics:

Frequency range	2115 ± 5 Mc (crystal selected)
Power output	+6 dbm ± 3 db, calibrated to ±0.5 db
Frequency stability	±o.001 percent
Frequency ramp rate	5 cps/sec to 500 csp/sec, continuously
Modulation	PM, 0 to 1.5 radians, continuously variable
Output Z	50 ohms

5.1.2 <u>RF Power Amplifier</u>

When driven by the command transmitter the power amplifier exhibits these characteristics:

Saturated gain	40 ± 3 db
Power outputs	37 ± 3 db
Frequency bandwidth	2115 ± 5 Mc (3 db points)
Input/output impedance	50 ohms

5.1.3 <u>Diplexer</u>

The diplexer has these characteristics:

Frequency bandwidth	Rx to antenna, 2295 ± 5 Mc Tx to antenna, 2115 ± 5 Mc
Isolation	70 db, Rx to Tx
Insertion loss	1.0 db mac, both channels
Impedance	50 ohms
Power capability	20 watts

5.1.4 <u>Telemetry Receiver</u>

The telemetry receiver is a fully phased locked type with a true phase detector. It has the following characteristics:

Frequency range	2295 ± 5 Mc (crystal selected)
Acquisition and tracking range	±100 Kc
Noise figure	10 db
IF bandwidth	3.3 Mc
Loop noise bandwidth	Selectable 10, 25, 50 cps
Input impedance	50 ohms
Modulation index measurement	0.8 to 1.5 radians of accuracy of ±0.1 radian
Image rejection	Greater than 60 db
IF rejection	Greater than 80 db
PM video bandwidth	Low frequency limit, 2 times loop BW High frequency limit, 3.3 Mc
Video output level	High level, 4 V peak to peak into 600 ohms

5.1.5 Modulation Meter Panel

The modulation meter panel houses the meters associated with the receiver quadrature detector.

5.1.6 Local Oscillator

The local oscillator generates the difference frequency between the transmitter frequency and the receiver frequency. The other characteristics are as follows:

Output level	+6 dbm ±3 dbm
Output impedance	50 ohms
Modulation	None
Frequency stability	20 cps/^{o}C

5.1.7 Mixer

The mixer has these characteristics:

Input frequency	2115 ± 5 Mc (nominal)
L. O. frequency	180 Mc (nominal)
Output frequency	2295 ± 5 Mc (nominal)
Conversion loss	Not greater than 20 db

5.1.8 VHF Capsule Generator

The VHF capsule generator has the following characteristics:

Frequency	100 ± 10 Mc
Output level	0 dbm ± db max adjustable to -160 dbm ± 10 db
Output impedance	50 ohms
Modulation	FSK
Modulation bandwidth	10 bits/sec

5.1.9 Command Encoder/PN Generator

The command encoder/PN generator is a command link formating device having these characteristics:

Input	Command inputs are inserted manually or automatically from the computer. Manual inputs are accomplished by front panel switches.
Output	Combined command data and 511 bit pseudo-noise (PN) code
Selection	PN codes are selected at the front panel
PN code frequency	511 cps ± percent
PN code duty cycle	50 percent ± percent
Bit rate	1 BPS ± percent
Modulation	Bi-phase

5.1.10 RF Power Meter

The power meter is a commercial device with these specifications:

Frequency range	10 to 10,000 Mc
Power range	1 microwatt to 10 milliwatts
Accuracy	±3 percent of full scale

5.1.11 Frequency Counter

The frequency counter is a commercial device with these specifications:

Frequency range	0 to 3.0 GC
Accuracy	Less than 3 parts in 10^9 ± 1 count/day

5.1.12 Dual Trace Oscilloscope

The oscilloscope is a commercial device with these specifications:

Vertical amplitude passband	10 mv/div to 10 V/div
Vertical sensitivity	10 mv/div to 10 V/div
Sweep rate	0.5 microsec/div to 1 sec/div
Sweep delay	0.5 microsec to 10 sec

5.1.13 S-Band Antenna

This antenna is a commercial item with these specifications:

Frequency range	1 to 4 Kmc (minimum)
Gain	10 dbi
Beamwidth	50 to 60 degrees

5.1.14 VHF Antenna

This antenna is commercially available with these specifications:

Frequency range	136 to 138 Mc
Gain	Unity

5.1.15 Command Transmitter and Command Encoder Power Supplies

These supplies provide to DC voltages required for the command encoder, command transmitter and power amplifier.

5.1.16 RF Patch Panel

The RF patch panel is a monitor point for the communication system and contains these monitoring functions:

a) Receiver 1st L. O.

b) Receiver 2nd L. O.

c) Reference oscillator

d) Receiver video

e) Quadrature detector (modulation monitor)

f) Downlink power and frequency monitor

g) Receiver AGC.

5.2 Telemetry Data Console

The telemetry data console in conjunction with the ADHS forms the heart of the data handling portion of the STS. A block diagram is shown in Figure 4.

The majority of the equipment in the system is controlled by the computer, which receives its instructions from a program stored in the computer memory. The computer, under program control, performs the following functions:

a) Controls test sequence

b) Generates commands

c) Establishes synchronization

d) Monitors synchronization status on a continuous basis

e) Determines the telemetry format being received

g) Acknowledges and checks commands transmitted to the spacecraft manually

h) Establishes loss of frame synchronization criteria.

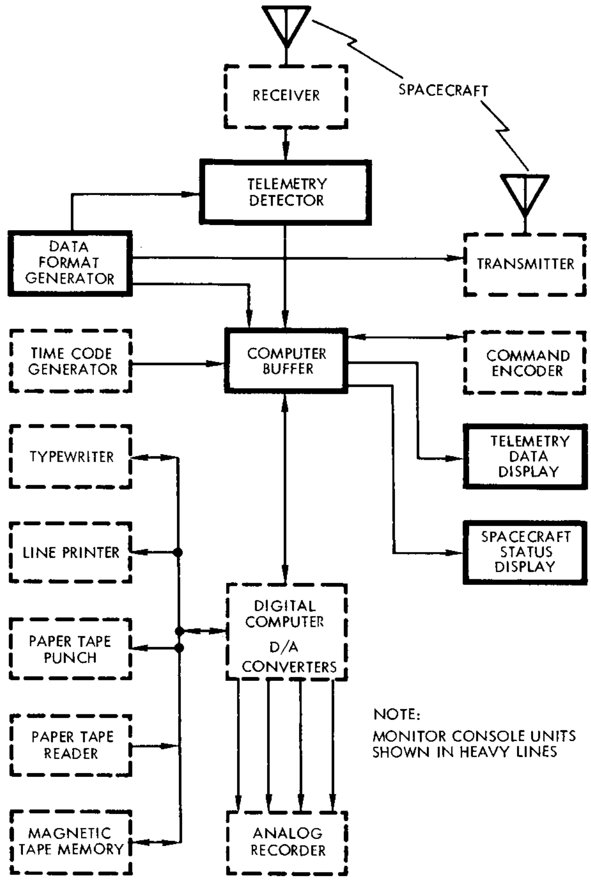


Figure 4. Telemetry Data Console, Block Diagram

5.2.1 Telemetry Detector

The telemetry detector receives the video (128 to 4096 cps, biphase) signal from the receiver and converts it to PCM plus sync for interpretation by the remaining portion of the data handling system. It receives inputs from any one of five different sources: direct from the telemetry receiver, via playback from the magnetic tape recorder, via hardline from the spacecraft and/or lander digital telemetry unit, or from the data format generator during STS self-test. The demodulated telemetry data output from the telemetry detector is routed to the buffer unit and the instrumentation patch panel (recorder console) for recording. The design for the telemetry detector is also utilized in the (MDE).

5.2.2 Buffer

The buffer accepts serial telemetry data from the demodulator and inserts parallel data into the computer. The buffer contains a 14-bit register used to store the serial telemetry data as it is received in real time. The register is instructed by computer command to put parallel telemetry data into the computer in either one-bit, one-word, or two-word groups. The capability of addressing the buffer and specifying the parallel data input pattern from the computer is provided to allow flexible control of the data input rates during and after the establishment of telemetry fram synchronization.

The buffer serves as a central point of distribution for inputs to the computer from the demodulator, the command encoder, the telemetry data display, the data format generator, and the time code generator, in addition to distributing output data from the computer to the telemetry data display and the spacecraft status display. Since all interfacing functions are not directly compatible with respect to one another, the buffer also possesses interface buffering and level conversion capability to insure interface compatibility.

5.2.3 Spacecraft Status Display

The spacecraft status display receives the spacecraft status telemetry information from the computer in its original telemetry word form.

The spacecraft status display displays the information on an indicator matrix containing an indicator lamp assembly for each spacecraft status item. The indicator lamp "on" condition signifies the "on" condition (or equivalent) of the spacecraft status item. The indicator "off" condition signifies the "off" condition (or equivalent) of the spacecraft status item. The display is updated each time new data is received from the spacecraft. Each of the 100 indicators are labeled in accordance with the spacecraft status item nomenclature.

5.2.4 Telemetry Data Display

The telemetry data display displays any word selected from the main frame or either subcom with appropriate identification. The display of the information is a function of computer conversion and formatting, hence it may be displayed in the decimal or octal number system. This unit displays the following information on a single request:

a) Data word content

b) Main frame word number (two-digit readout) if applicable

c) Subcom word number (two-digit readout) if applicable

d) Telemetry format

e) Telemetry mode

f) Telemetry bit rate

g) Parity status

h) Frame synchronization status.

The displayed data is updated each time it is received or placed on "hold" to prevent updating after display.

Requests for data display may be initiated from the computer typewriter or from digit switches on the telemetry data display. The telemetry data display selecting controls are implemented to provide word selection from the main frame or either subcom in either decimal or octal word number form.

The status display design will be utilized in the MDE.

5.2.5 Data Recording

Information may be permanently recorded on a line printer (ADHS) or a direct-write analog recorder. Four digital-to-analog converters, with 8-bit word storage each, are provided so that four channels of information can be simultaneously recorded. The over-all recording resolution is approximately ±2 percent. The words to be recorded are selected by typewriter input to the computer. The recorded information is updated each time it is received in the telemetry information. The digital-to-analog converters and their storage registers are supplied with the computer and are located inside the computer cabinet. The output of these converters is sent to the instrumentation patch panel (recorder console).

5.2.6 Self-Test

The data format generator is used to simulate the normal demodulator input (128-4096 cps biphase modulated signal) or the normal input buffer input to test the data handling system for proper operation. In addition, it can modulate the STS command transmitter for self-test of the RF loop as discussed in paragraph 5.4. Self-testing is accomplished by feeding simulated telemetry data into selected units and observing the resulting displays.

The data format generator can produce repeated 64-word real-time telemetry frames with all frame constants properly located within the main frame and subcoms. In one mode, the individual frame word slots (except those containing frame constants) contain their respective word number in binary. In this mode, any frame word may be addressed and a selected bit pattern inserted in the addressed word. Another mode will allow any selected bit pattern placed in all words.

This unit also provides telemetry format and bit rate selection simulating all spacecraft operating modes.

The data format generator design is used in the MDE.

5.3 Recorder Console

The recorder console provides the means for direct-write analog recording, magnetic tape recording, and magnetic tape playback.

5.3.1 Instrumentation Patch Panel

The patch panel is furnished with a trompeter type J-3, or equivalent, 50 ohm coaxial jacks and coaxial patches and plugs to insure minimum cross-talk and noise pickup. The jacks are arranged so that maximum use is made of looping-plugs during the most common test. Sufficiently long patch cords are furnished to provide complete flexibility in the use of the jacks. The panel is equipped with 125 jacks to handle the signal inputs and outputs of the recorder console.

Isolation transformers and L-pad attenuators are provided to assure compatibility between the recording equipment and the source of recorded data. The L-pad attenuators are adjustable by front panel control.

5.3.2 Analog Recorder

The analog recorder is a commercial device such as Sanborn Model 858-5461 or equivalent. DC preamplifiers, Sanborn Model 850-1000 or equivalent, and a frequency to voltage converter preamplifiers, Sanborn Model 850-2800 or equivalent, are also provided.

5.3.3 Magnetic Tape Recorder

The magnetic tape recorder is a commercial device such as Ampex Model FR-1600 or equivalent. The recorder has seven data channel record/reproduce electronics as well as speed lock and a scope monitor bay for monitoring the input and output signals. The recorder operates at tape speeds of 120, 60, 30, 15, 7-1/2, and 3-3/4 1 PS.

5.4 Ground Power Console

The ground power console consists of the ground power console, the power supply rack, and the in-flight jumper simulator.

The ground power console provides the central control and monitoring point for the spacecraft power control and battery charging and consists of five assemblies. During integration and test operations it is located in the checkout area convenient to the spacecraft. During launch operations, it is installed in the block-house for the control and monitoring of the spacecraft power.

The power supply rack contains the battery charger power supply, solar array simulator, signal conditioner, and relay box. The rack will be used in close proximity to the spacecraft during integration and test operations and in the terminal room during spacecraft countdown and launch.

Thein-flight jumper simulator replaces the in-flight jumper aboard the spacecraft during support of spacecraft activities. It providcs thc means of controlling spacecraft power distribution.

Figure 5 is a block diagram of the ground power console.

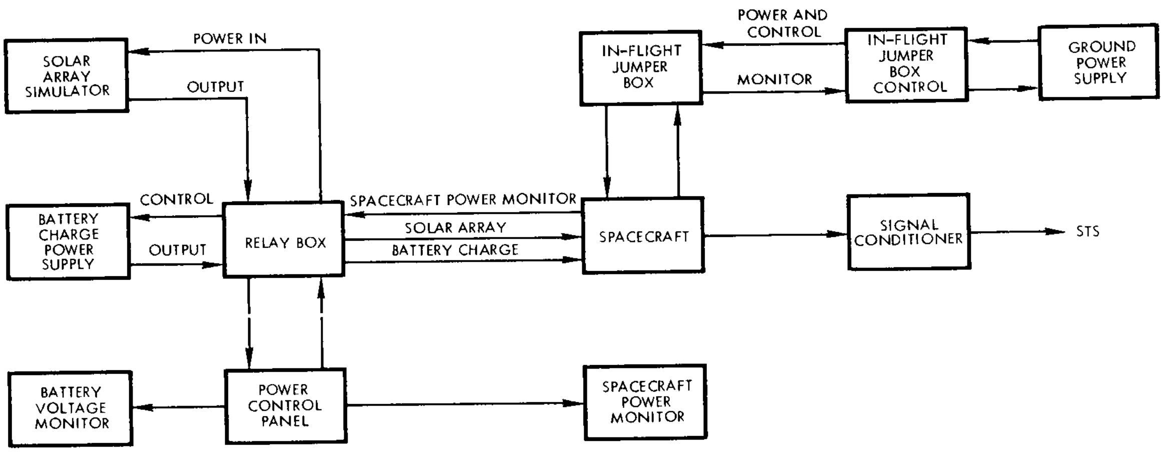


Figure 5. Ground Control of Spacecraft Power System, Block Diagram

5.4.1 Ground Power Control Console

The ground power control console consists of five major assemblies as discussed in the following paragraphs.

a. Spacecraft Power Monitor Unit

The spacecraft power monitor is a 6-place digital voltmeter Hewlett Packard Model 3460A or equivalent. Voltage measurement accuracy is ±2 millivolts on the 100 volt scale. The spacecraft voltages which may be measured by the power monitor unit are bust voltage, bus current, battery voltage, battery current, and battery temperature. The power monitor input is derived from a selector switch at the power control panel.

b. Battery Voltage Monitor

The battery voltage monitor is a digital voltmeter the same make and model as the spacecraft power monitor discussed above. Spacecraft battery voltage is derived from the power control panel and is continuously displayed.

c. Power Control Panel

This unit contains the controls discussed in the following paragraphs.

Battery Charger On/Off Switch. This switch provides power to a relay in the relay box that controls AC power to the battery charger.

Battery Charger Output Control Switch. This switch provides power to a relay in the relay box that applies the battery charger output to the spacecraft power system.

Battery Charger Voltage Control. This control is multi-turn potentiometer that remotely controls the battery charger voltage.

Battery Charger Current Control. This control is a multi-turn potentiometer that remotely controls the output current of the battery charger in the constant current mode of operation.

Power Monitor Input Selector. This selector is a multi-position rotary switch used to select the spacecraft signal to be monitored by the power monitor unit. Spacecraft signal inputs arrive at the selector switch via the relay box.

Battery Charger/Solar Array Mode Switch. This switch provides control to a relay in the relay box. The relay is used to control the spacecraft power operation mode. Interlocks are provided to prevent mode switch when the battery charger is turned on.

High Impedance Meters. The power control panel also contains high impedance (voltron or equal) meters to continuously monitor the following spacecraft functions obtained from the spacecraft via the relay box:

- Spacecraft battery temperature (monitors voltage output of a thermistor network located in the spacecraft batteries)

- Spacecraft voltage monitored from spacecraft battery output terminals
- Spacecraft battery current monitored from spacecraft transducer which provides the signal for the telemetry
- Battery charger voltage monitored from the battery charger power supply output terminals
- Battery charger current monitor from a shunt inside the relay box
- Spacecraft bus voltage monitored via the relay box from the spacecraft main power bus
- An indicator light is also provided to warn operator when the spacecraft ordnance is armed

d. <u>In-flight Jumper Simulator Control</u>

This unit contains a +28 volt modular power supply to provide voltage for the controls and indicators discussed in the following paragraphs.

<u>Spacecraft Internal/External Power Switch</u>. This switch provides power for control of a relay contained in the in-flight jumper simulator. The relay transfers the spacecraft load from the spacecraft batteries (internal) to the spacecraft ground power supply (external).

<u>Spacecraft Branch Bus Control Switches</u>. These three switches provide impulse commands to magnetic latching relays contained in the in-flight jumper simulator that controls the power to the science, communications, and stabilization branch busses.

<u>Spacecraft Ground Power Supply On/Off Switch</u>. This switch controls AC input power to the spacecraft ground power supply.

<u>Spacecraft Ground Power Supply Voltage Control</u>. This multi-turn potentiometer controls the ground power supply output voltage.

<u>Emergency Shut-down Switch</u>. This switch provides impulse commands to turn off the three spacecraft branch bus control relays in the in-flight jumper simulator.

Spacecraft Internal/External Power Indicator Lights. These lights are controlled by contancts of the external/internal relay contained in the in-flight jumper simulator.

Branch Bus On/Off Indicator Lights. These three lights are by contacts contained in the branch bus control relays located in the in-flight jumper simulator unit.

Spacecraft Ground Power On/Off Indicator Light. This light is controlled by contacts on the ground power On/Off switch.

Spacecraft Ground Power Supply Voltmeter. This meter is connected to the output lines of the spacecraft ground power supply.

Spacecraft Ground Power Current Voltmeter. This voltmeter has an ammeter scale, and derives current level information via a shunt in the in-flight jumper simulator.

e. Ground Power Supply

The ground power supply is a Harrison Model 6439A or equivalent power supply voltage output in adjustable 0-60 VDC. Current output is 0-15 amps. The output voltage is controlled by an over-voltage detection circuit to prevent spacecraft damage in case of power supply failure.

f. Running Time

A running time meter is provided which indicates the total time power has been applied to the spacecraft.

5.4.2 Ground Power Rack

The ground power rack consists of four major assemblies as described in the following paragraphs.

a. Relay Box

This unit contains relays for the remote control of the battery charge power supply AC power input, DC power output, and solar array simulator/battery charge mode from the spacecraft ground power console. It also contains an over-voltage protective circuit to remove power from the spacecraft to protect it in case of an over-voltage condition in the battery charge power supply. In addition, it contains a shunt for monitoring the spacecraft battery charger output current. This unit is also used as a junction and distribution box for signals going to and from the spacecraft and the ground power console.

b. Spacecraft Battery Charge Power Supply

Spacecraft battery charge power supply will be a voltage and current regulated, voltage and current variable, 0-60 VDC power supply such as the Harrison Model 6439A. The supply has a capacity of 0-15 amp and will be capable of charging the spacecraft battery while the spacecraft is being operated on internal power.

c. Solar Array Simulator

The solar array simulator simulates the outputs from the spacecraft solar array panels under varying flight conditions. This unit contains adjustable constant current regulators and voltage regulators which can be set to simulate current/voltage output characteristics as required. All output lines are diode protected to prevent spacecraft voltage feedback to solar array simulator circuitry. A switch is provided for control of each solar array output line. A switch is also provided on each output line to permit preloading of the output prior to application of power to the spacecraft. A front panel test point is provided for each output line to permit monitoring and recording of output data. A voltmeter and ammeter is provided for monitoring of the input and output voltages and output currents. Voltage and current selector switches are provied for the selection of input signals to the voltmeter and ammeter. Input power to the solar array simulator is derived from the spacecraft battery charger power supply via a relay in the relay box.

d. Signal Conditions

This unit contains the high input impedance isolation amplifiers for the DTU signals received from the spacecraft and launch. The amplifiers are adjustable to assure hardline compatibility with the STS.

5.4.3 In-flight Jumper Simulator

The in-flight simulator provides a means of connecting the ground power to the spacecraft. It also provides the capability of controlling the spacecraft external/internal power and the individual branch busses by means of magnetic latch relays. A current shunt is provided for monitoring the spacecraft bus current. The shunt is monitored by a meter in the in-flight jumper simulator control. The unit is designed for operation with the spacecraft in an environmental chamber and therefore contains a thermostat and heater to prevent freeze-up of relays.

5.5 Test Console

The test console comprises standard commercial test equipment and special equipment designed for the system testing of spacecraft functions not capable of being tested through the RF command and telemetry links. In order to perform these tests, the test console is capable of providing the spacecraft with the following stimulation test services:

Gyro	Pitch, yaw, roll torquing signals
Course sun sensor	Portable light source
Fine sun sensor	Portable light source
Near earth detector	Portable light source
Star sensor	Portable light source
Approach guidance sensor	Simulated star-field view
Mars sensor	Portable Mars simulator (heat source)

In addition, the test console provides simulation test services as follows:

Gyro	Pitch, yaw, and roll output signals
Course sun sensor	Output signals
Fine sun sensor	Output signals
Near earth detector	Output signals
Star sensor	Output signals
Mars sensor	Output signals
Boom	Deployment signals
Capsule to main spacecraft	Interface signals

The test console also provides monitor capability via hardline connection to spacecraft connectors as follows:

Gyro pitch, yaw, and roll output signals

Cold gas jet actuator signals

Mid-course engine TVC actuator signals

Deboost engine TVC actuator signals

Ordnance firing signals

Receivers signal present

Receivers signal strength

Receivers loop stress signal

Antenna drive signals

Boom deployed signals.

The control and stabilization system is operated in each of the flight modes with appropriate stimuli applied. Control and stabilization system performance are determined by monitoring the signals at test point, monitor, and control panel. Phasing and mode control tests are checked by inserting commands at the RF console and sequentially applying gross sensor stimulations or simulated sensor inputs and observing corresponding actuator operations and/or error signals. Actuator positioning is checked by applying appropriate input commands and sensor simulated inputs while monitoring the drive signal to the actuator as well as its feedback (position) signal.

A functional block diagram of the test console appears in Figure 6 and a panel-by-panel functional description is contained in the following paragraphs.

5.5.1 Oscilloscope

The oscilloscope is used in conjunction with the test point monitor and control panel to observe AC signals from the spacecraft and those supplied to the spacecraft during tests.

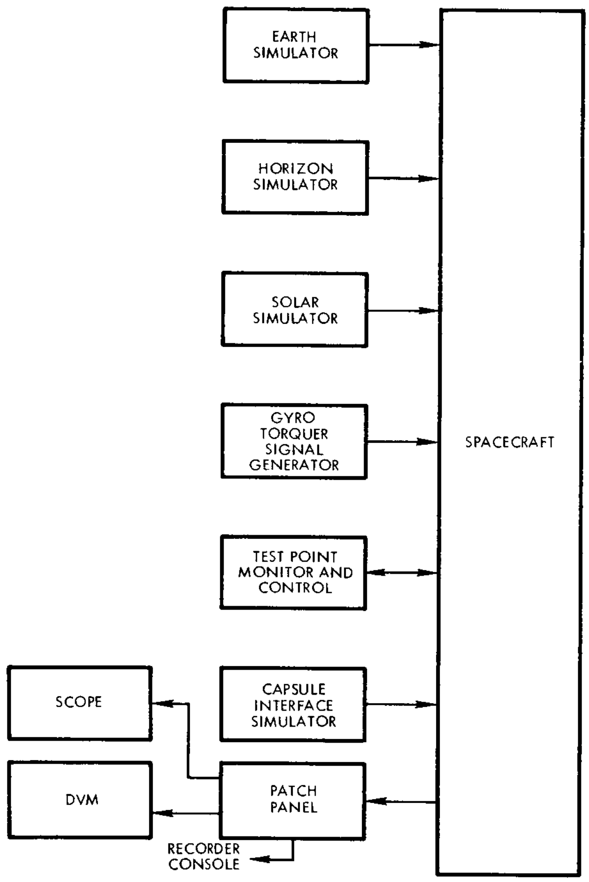


Figure 6. Test Console, Block Diagram

5.5.2 Digital Voltmeter

The digital voltmeter is used to measure the DC voltages appearing on the test point panel that are obtained as signals from the spacecraft during test and the DC signals supplied to the spacecraft.

5.5.3 Test Point Monitor and Control Panel

The test point monitor and control panel provides a terminal point for all of the hard wire test and control lines that are connected with the spacecraft during system test. Switches are provided to select signals for observation on the scope or measurement by the DVM. Located upon this panel are the ordnance firing circuit loads and indicators.

5.5.4 Gyro Control and Monitor Panel

This panel provides torquing signals to the gyro assembly to simulate spacecraft altitude change during tests.

5.5.5 Sun-Earth-Mars Simulator Panel

The sun-earth-Mars simulator provides electrical signals to the spacecraft stabilization and control electronics to simulate the presence of a signal from the sensor. These signals are not used when the sensor is stimulated.

5.5.6 Patch Panel

The patch panel provides a place to route signals to direct writer-type recorders.

5.5.7 Lander Simulator Panel

This panel provides simulation of all signals that appear at the lander/main spacecraft interface.

5.5.8 Validation Box

This portion of the test console provides a means of validating the overall operation of the STS. Functionally, it simulates the operation of the spacecraft in all critical modes which are required to checkout the STS. The validation box is functionally divided into the following sections:

a) AC Power Section — provides AC power to the validation box DC power supplies

b) Spacecraft Power Control Section — provides capability to checkout the STS power control of the spacecraft

c) Ordnance Monitor Section — simulates the spacecraft ordnance circuitry

d) Test Point Monitor and Control Section — checks out the test point monitor and control unit in the STS, including command line and reply line validiation

e) Start Reference Section — checks out the mass, sun, earth and Mars reference circuitry in the STS

f) Critical Circuit Validation Section — simulates those critical spacecraft monitoring points (e.g., redundant checkout operation).

6. BOUNDARY DEFINITION

6.1 Individual Consoles

The individual console primary power is as follows:

Voltage	115 VAC ± 10 VAC
Frequency	60 ± 1 cps
Phase	Single (3-wire)
Current demand	Not to exceed amps

6.2 System Test Sets

The STS primary power is as follows:

Voltage	115 VAC ± 10 VAC
Frequency	60 ± 1 cps
Phase	Three (5-wire)
Current demand	Not to exceed amps

7. PARAMETERS

The following parameters apply:

• Uplink frequency	2115 ± 5 Mc
• Downlink frequency	2295 ± 5 Mc
• Command bit rate	1 BPS (nominal)
• Telemetry subcarrier frequency	
• Capsule data carrier frequency	136 - 138 Mc
• Capsule data rate	10 BPS

- Cold gas jet actuating signals
- Mid-course engine TVC actuator signals
- Deboost engine TVC actuator signals
- Gyro unit pitch, yaw, roll output signal
- Antenna drive signals
- Spacecraft DC bust voltage sense
- Spacecraft battery voltage sense
- Ordnance safe/arm monitor
- Ordnance firing circuit current monitor
- Ordnance safe jumper
- Digital telemetry unit biphase modulation
- Receiver loop detector
- Receiver signal present
- Signal strength
- Boom deployed signal
- Separation signal
- Gyro torquing signals

- Simulated course sun sensor signal
- Simulated fine sun sensor signal
- Simulated earth sensor signal
- Simulated Mars sensor signal
- Simulated star sensor signal
- Course, fine sun, earth, star stimulation
- Ground power to spacecraft
- Ordnance firing circuit monitor returns.

8. CONSTRAINTS

8.1 Service Conditions

The STS is designed to operate within required limits in the following environment:

Temperature	$+50^{\circ}$ to $+100^{\circ}$F
Humidity	Not to exceed 60 percent
Altitude	Mean sea level to 5,000 feet
Salt atmosphere	Capable of operation in sheltered areas in coastal regions
Sand and dirt	Circulating air blowers contain washable filters
Shock and vibration	Equipment is designed to withstand normal handling for its intended usage and for shipping by electronic equipment commercial carriers when the usual packaging techniques for these carriers is employed.

8.2 Total Operating Life

With reasonable servicing and replacement of parts, the STS is designed for minimum expected operating life of 25,000 hours, an overall lifetime goal of 10 years.

8.3 Operating Time Record

An hourly operating time recording meter is provided on each console which records the time the main power switch is on. This record requires that primary power be connected.

9. ADDITIONAL INFORMATION

9.1 Electromagnetic Interference

To minimize the effect of electromagnetic interference, all subsystems have employed filtering and other safeguards to reduce this problem to practical limits.

9.2 Magnetic Field Interference

The STS does not interfere with any magnetic measurements which will be made on the spacecraft.

AUTOMATIC DATA HANDLING SYSTEM

OSE/ VS-3-120

1. SCOPE

This document defines the functional requirements for the Voyager OSE automatic data handling system (ADHS) which is used with the system test set (STS) to provide automatic data handling capability during testing of the Voyager spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL

OSE/ VS-1-110	OSE Objectives and Criteria
OSE/ VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

The Voyager ADHS is designed to provide remote data entry units and as a central complex for control of tests. It performs the following:

a) Primary processing for reduction and analysis of test data on a real time basis

b) All processing required for the updating of remote displays

c) Processing of incoming requests for data from remote test stations

d) Secondary, off-line data reduction and analysis.

It also controls the sequence in which pre-programmed tests are executed, subject to manual override by the test director; generates and modifies programs; and generates test procedures on reproducible masters.

4. DESIGN REQUIREMENTS

4.1 General Design Requirements

The ADHS is designed to perform the test control and data processing functions required to perform meaningful tests on a single Voyager spacecraft. The design of the ADHS permits growth in the required tasks, increased sophistication in the manner in which tests are performed, and upgrading of equipment parameters in future spacecraft. For example, the ADHS accommodates the present maximum telemetry data rate of 4096 bits/second; however, no equipment changes would be required if the telemetry rate were doubled, although some changes in computer programming would be required.

Efforts have been made to confine the equipment comprising the ADHS to proven, commercially available components.

The ADHS provides information to various test personnel (test director, subsystem specialists, experimenters, etc.) adequate enough in both quality and quantity to permit them to make accurate and current evaluations of spacecraft status and performance throughout the test period.

5. FUNCTIONAL DESCRIPTION

5.1 General

The ADHS consists of a test director's console, an SDS-930 computer, manual input devices for transmitting data from the STS or associated equipment in the ESF, the blockhouse, or the launch pad, and computer peripheral equipment such as tape stations, line printers, character printers, paper tape punches, and readers. Figure 1 illustrates a typical equipment configuration.

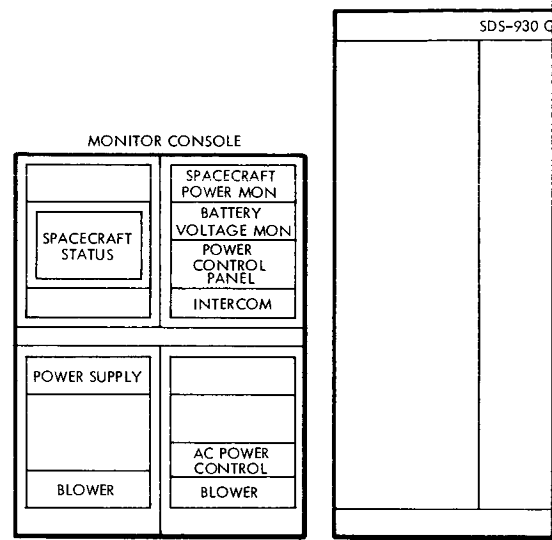
The functions of the ADHS include, but are not necessarily limited to, the following:

a) Real-time processing of spacecraft data from both telemetry and hardline sources

b) Test sequencing

c) Providing displays of various kinds (nixie tube readout, hard-copy printout, etc.) in formats meaningful to test personnel

d) Performing various off-line functions such as program generation, data reduction, etc.


MONITOR CONSOLE
SPACECRAFT STATUS
SPACECRAFT POWER MON
BATTERY VOLTAGE MON
POWER CONTROL PANEL
INTERCOM
POWER SUPPLY
BLOWER
AC POWER CONTROL
BLOWER
SDS-930 C

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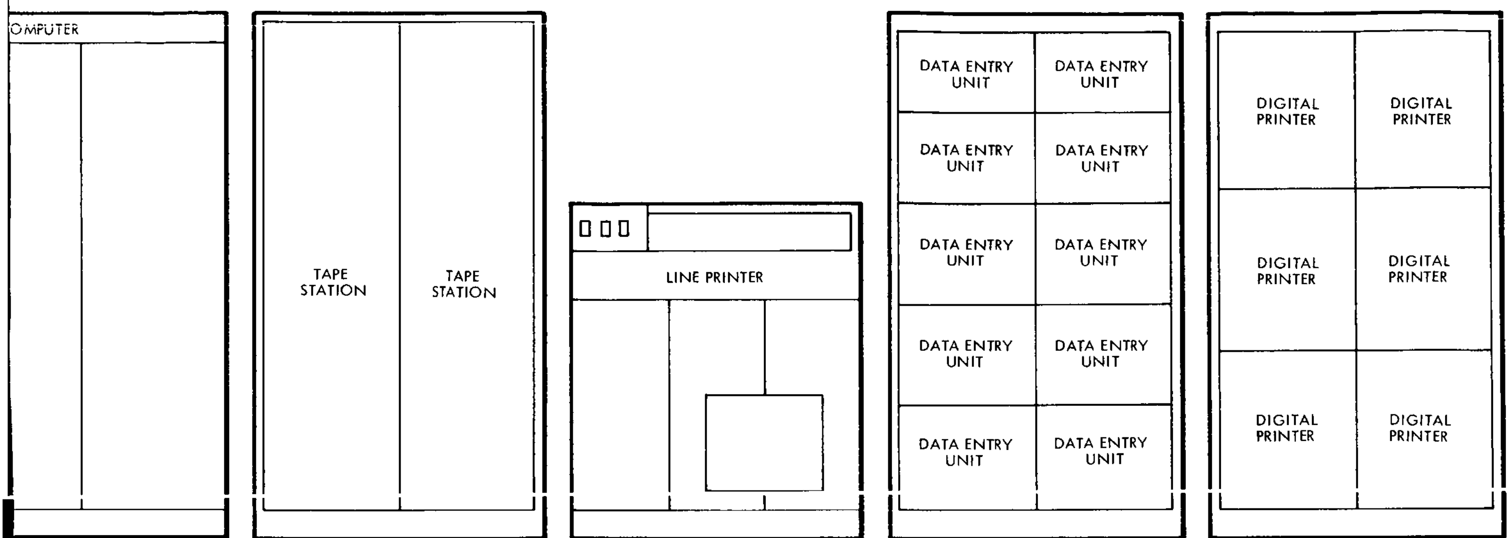
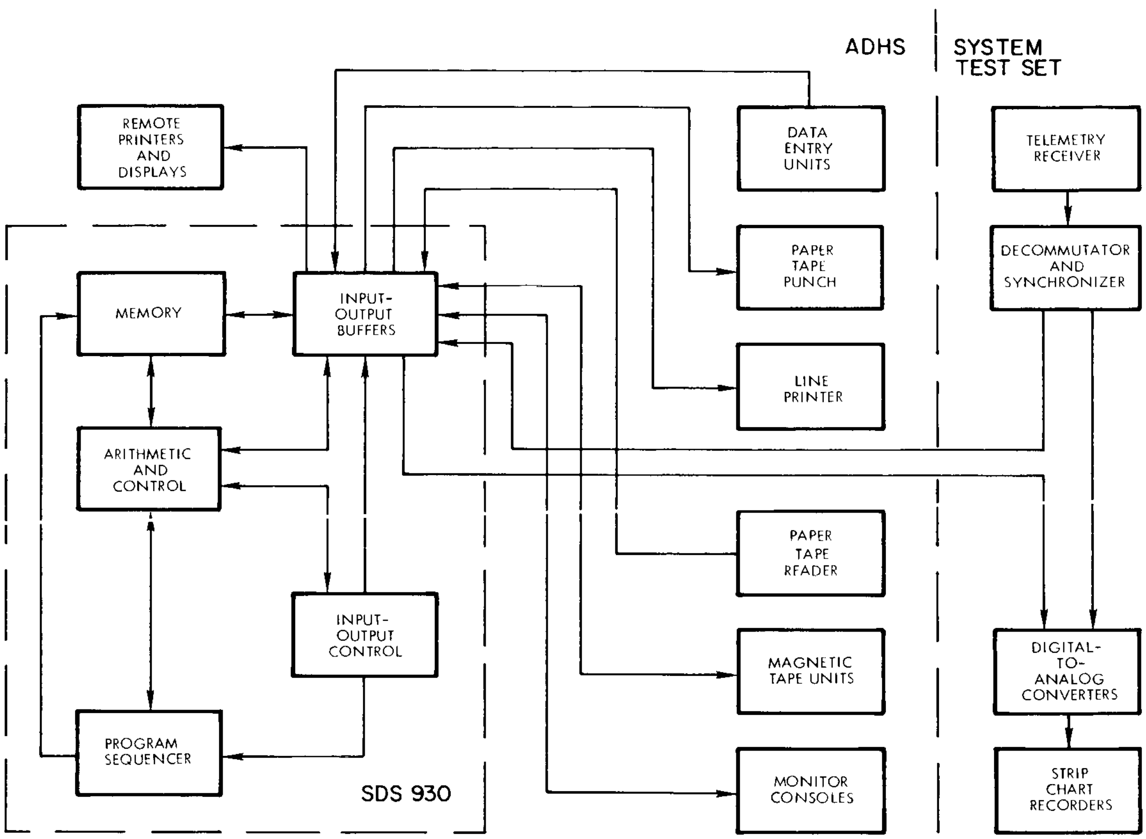


Figure 1. Typical Equipment Configuration

Figure 2 is a functional block diagram of the ADHS which indicates the functional interface with the STS, the SDS-930 internal functions, and the computer support equipment.



5.2 Unit Functional Descriptions

5.2.1 General Purpose Computer

The computer for the ADHS is the SDS-930, with the following characteristics:

- 24-bit word plus parity bit
- Binary arithmetic

- Single-address instructions with:

  Indexing (without timing penalty)

  Multi-level indirect addressing

  Programmed operators

- Basic core memory of 16,384 words, expandable to 32,768 words, all addressable with:

  0.7 microsecond access time

  1.925 microseconds cycle time

- 4,096, 8,192, and 16,384-word memory banks available

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

- Multi-precision programming facilities

- Programmed operators, permitting up to 64 special, user-specified, instruction codes that can vary from program to program

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

<u>Fixed-Point Operations</u>

ADD	3.85 microseconds
MULTIPLY	7.7 microseconds
DIVIDE	19.25 microseconds

<u>Floating-Point Operations</u>

(24-bit fraction plus 9-bit exponent)	
ADD	81 microseconds
MULTIPLY	59 microseconds
(39-bit fraction plus 9-bit exponent)	
ADD	91 microseconds
MULTIPLY	152 microseconds

- Program interchangeability with other SDS-900 Series computers

- Parity checking of all memory and input/output operations

- Two channels of priority interrupt standard, up to 1022 optional

- Memory non-volatile with power failure; power fail-safe feature (optional) permits saving contents of programmable registers

- Real-time programmable clock (optional)

- Up to four communication channels, time-multiplexed with computer operation, providing input/output rates of up to one word per 3.85 microseconds

- 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 1.925 microseconds

- 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

- Time-multiplexed input/output channels operate upon either words or characters. Character sizes of six or twelve bits can be obtained as desired. Direct access channels operate upon words and characters. The number of characters per word is under program control.

- Input/output with scatter-read and gather-write facility.

- Standard input/output equipment:

    Automatic input/output typewriter

    Control console

- Optional input/output equipment:

  Paper tape reader, paper tape punch

  Magnetic tape units (IBM compatible; binary and BCD)

  Punched card input and output equipment

  Line printers, graph plotters

  Auxiliary magnetic drum piles, disc files

  Communication equipment, teletype consoles, display oscilloscopes

  A/D converters, digital multiplexer equipment, and other special system equipment

- Facility to share memory between two or more computers
- FORTRAN II and symbolic assembler as part of complete software package
- All silicon semiconductors
- Operating temperature range: 10 to 40°C
- Dimensions: 124 inches x 25 1/2 inches x 65 inches
- Power: 2.5 KVA

5.3.2 Paper Tape Punch

The tape punch is the SDS Model 9132 with the following characteristics:

Speed	60 characters/sec
No. of channels	5, 6, or 7 (adjustable)
Width	19 inches
Height	10.5 inches
Depth	13 inches
Weight	35 pounds
Operating temperature range	50 to 120°F

Operating humidity range	20 to 85 percent
Power	Supplied from computer

5.3.3 Paper Tape Reader

The paper tape reader is the SDS Model 9330, with the following characteristics:

Speed	300 characters/sec
No. of channels	7
Width	19 inches
Height	7 inches
Depth	6.5 inches
Weight	50 pounds
Operating temperature range	50 to 104°F
Operating humidity range	10 to 90 percent
Power	Supplied from computer

5.3.4 Paper Tape Spooler

The spooler for use with the tape punch and tape reader is the SDS Model 9135, with the following characteristics:

Forward read speed	30 inches/sec
Manual forward speed	200 inches/sec
Rewind speed	200 inches/sec
Width	19 inches
Height	10.5 inches
Depth	10.5 inches

Weight	20 pounds
Operating temperature range	0 to 150°F
Operating humidity range	20 to 95 percent
Power	Supplied by computer

5.3.5 Magnetic Tape System

The tape system is made up of one SDS Model 9248 tape control unit and three SDS Model 9246 tape transports. It is possible to increase the number of tape transports to a total of eight without requiring an additional tape control unit. The magnetic tape system has the following characteristics:

Tape speed - read/write	75 inches/sec
Speed - rewind	150 inches/sec
Reels and hubs	10.5-inch take-up reel; 10.5-inch file reel with file-protect ring and IBM hub
Tape drive	Capstan pinch roller drive with reel control servos and vacuum buffer storage chamber
Recording method	NRZ1 (non-return-to-zero-change-on-ones)
Recording format	7-channel, 6 bits and parity, self-clocking, BCD or binary
Inter-record gap	3/4 inch
Recording density	200 or 556 characters/inch
Character read/write rate	15,000 or 41,667 character/sec
Tape	1/2-inch wide x 2400 feet long, 1.5 mil Mylar
End of tape sensing	Reflective marker photosensing
Head	7-channel, dual gap, IBM-compatible

Width	25 inches
Height	65 inches
Depth	27 inches
Weight	800 pounds
Temperature	50 to 90^{o}F
Humidity	40 to 70 percent
Power	15 amps at 115v ±10 per cent, 60 cycles
Heat dissipation	6500 BTU per hour.

5.3.6 Line Printer

The line printer is the SDS Model 9379 with a minimum printing rate of 628 lines/minute and a line length of 132 characters/line.

5.3.7 Remote Printout Devices

These devices are SDS Model 9137-B I/O typewriters. A total of eleven typewriters are supplied, one for operator-computer communications and ten for remote printout. Tye typewriters have the following characteristics:

Designation	IBM Selectric I/O typewriter
Typing mechanism	Type 908 "Golf Ball" element
Carriage width	11 inches
Input	Manual
Output	15 characters/sec
Width	18 inches
Height	9 inches
Depth	15 inches
Weight	25 pounds
Temperature	40 to 95^{o}F
Humidity	20 to 95 percent
Power	Supplied from computer

5.3.8 Test Director's Console (TDC)

A console is provided to facilitate the tasks of the test director in the control of tests. The TDC includes, but is not necessarily limited to, the items listed in the following paragraphs:

a. Status Displays

These displays provide means for making quick, complete, and current assessments of spacecraft operations.

b. Critical Function Controls

Means are provided on the TDC to enable the test director to exercise immediate and overriding control on a highest priority basis over critical functions (for example, initiation of power off commands to the spacecraft).

c. Selectable Displays

Means are provided on the TDC to enable the test director to select various spacecraft data.

d. Intercom System

Means are provided to enable the TDC to maintain bidirectional communication with all test stations.

e. Working Surface

The TDC is designed to provide a convenient working surface for the test director's use.

f. Storage Space

Drawer space is provided for storage of documentation and other material which the test director may require to perform his tasks.

Similar consoles will be provided in the ESF and the bench house to facilitate monitoring of spacecraft test operations.

5.3.9 Remote Displays

A total of twelve numeric displays are provided in the test area with means for selection of the data to be displayed. These are 6-character (digit) displays of a ''nixie'' or similar nature.

5.3.10 Special Input Devices

Means are provided for entering data into the computer from a standardized panel which is mounted conveniently on each unit test set, system test set, and item of launch complex equipment. This panel contains an input keyboard or a series of digiswitches, a readout device for verification of data entered, and a button for initiating entry of the data to the computer buffer.

6. CONSTRAINTS

6.1 Physical

The ADHS equipment is housed in standard Voyager racks as specified in Section 2 of this volume.

6.2 Environmental

Except as otherwise specified herein, the service environment is a laboratory type having a temperature of 60 to 90°F and a humidity of less than 50 per cent.

6.3 Electrical

The power subsystem test set operates from a single phase power source with a voltage of 115 ± 10 AC RMS and a frequency of 60 ± 1 cps.

6.4 Prime Power Connector

All prime power connections are via a cable with a standard "U" three-prong 115 volt AC cap.

7. INTERFACES

7.1 Equipment Interfaces

The only interface between the ADHS and other equipment is that between the telemetry decommutator in the STS and the SDS-930 computer. The STS assembles serial telemetry data into 7 bit words which are transferred to the SDS 930 in parallel. Word transfers are made under control of the priority interrupt system of the SDS 930.

7.2 Man-Machine 930

Man-machine interfaces have been designed to permit maximum flexibility with minimum confusion. These interfaces are as follows:

7.2.1 Human Input to the ADHS

Inputs from computer operators and other test personnel are of the following types:

a) Pushbutton--Well-marked and clearly defined pushbuttons which control the entry of data and requests for data and activation of interrupts

b) Digiswitches--Used for the selection of data to be displayed according to pre-defined tables

c) Typewriters--Used as an input to the computer, for more complicated communications between test personnel and the ADHS.

7.2.2 Outputs to Test Personnel

a) Numeric displays--Numeric values of various data measurements are presented on 6-digit displays in accordance with pre-selected requests

b) Typewriters--More extensive output requiring alphabetic as well as numeric descriptions are performed by typewriters controlled by the SDS-930

c) Line printer--For output too extensive for typewriters, a line printer is used to print out data under SDS-930 control.

DSIF MISSION DEPENDENT EQUIPMENT

OSE/VS-3-130

1. SCOPE

This document defines the functional requirements and configuration of the mission dependent equipment (MDE) of the Voyager electrical operational support equipment (EOSE). The MDE consists of the equipment, including computer solftware, needed at the DSIF to support the Voyager program.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

Environmental Specification, 8900

Preferred Parts Specification, 8905

Standard Modules Specification, 8906

DSIF General Requirements Specification, 8907

Console Rock Assemblies Drawing, J9157055D

NASA

Soldering of Electrical Connections, MSFC-PROC-158B

Military

Interference Control Requirements, MIL-I-26600

3. FUNCTIONAL REQUIREMENTS

The Voyager MDE is designed to provide the following general functions at the DSIF sites:

a) Primary (in-line) functions

- Command generation
- Telemetry detection
- Computer buffering

b) Secondary (supplementary) functions

- Command detection
- Spacecraft status display

c) Tertiary (test and maintenance) functions

- Telemetry detection testing
- Simulated telemetry data generation
- Spacecraft simulation
- Station simulation
- General purpose measurement and calibration

d) Computer programming

The primary functions of the MDE are those in-line functions which are essential to the DSIF link with the Voyager spacecraft.

The secondary functions of the MDE are those in-line functions which are not essential to the DSIF link with the Voyager spacecraft, but which are desirable operations monitoring functions.

The tertiary functions are those having to do with compatibility testing, station readiness testing, fault isolation, maintenance and calibration of the Voyager MDE.

4. DESIGN REQUIREMENTS

4.1 Over-all Design Requirements

The Voyager MDE provides for the functions described within the degrees of reliability and accuracies stated herein. This equipment is designed for mounting in standard JPL DSIF station equipment racks and for operation within the environmental constraints of the DSIF station. It interfaces with the multiple mission equipment present at the DSIF stations. A complete redundancy of the primary in-line functions is provided.

A critical design goal for the Voyager MDE is the attainment of a high degree of reliability to insure mission success. Initial design and selection of reliable parts is supplemented by:

a) Redundancy - The in-line components of the MDE are provided in a redundant configuration in which both equipments of a type are in-line; outputs are switchable. The design of the components is such that input isolation is provided to prevent failure in one unit from affecting the prime data path.

b) Accumulation of Operating Time - From installation at DSIF until launch operations, a prolonged period of calibration and operational readiness testing is scheduled. The equipment operation during this period will provide a measure of operational reliability.

4.2 Primary Requirements

4.2.1 Command Generation

A manually controlled command encoder unit is provided which is capable of generating each of the Voyager command words necessary for real time command of the spacecraft. The commands entered in this unit are automatically checked for permissibility before their transmission can be manually initiated. The command signal leaving this unit enters the pseudo noise (PN) generator where it bi-phase modulates a PN code which is repeated during each command bit period. The output of the PN generator is used to modulate the DSIF transmitter.

4.2.2 Telemetry Detection

The function of the telemetry detector is to extract the telemetry bit stream and synchronization signals from a subcarrier of the spacecraft to ground link.

4.2.3 Computer Buffering

The primary function of the computer buffer is to transform the telemetry data signal and its associated synchronization signals into a format that is acceptable to the station computer.

4.3 Secondary Requirements

4.3.1 Command Detection and Verification

The command detector receives the command PN code modulated with the command format from the station monitor receiver and extracts the command format from the signal. This signal is transmitted via the computer buffer to the computer where the command is checked bit-by-bit by comparing it to the command previously manually entered in the command encoder.

4.3.2 Spacecraft Status Display

Telemetry data signals regarding the spacecraft status are outputted from the computer through computer buffer circuits to the spacecraft status displays. These signsls are used to illuminate English language legends which indicate to the operator the operational status of the Voyager spacecraft.

4.4 Tertiary Requirements

4.4.1 Telemetry Detection Testing

The error rate tester (ERT) is used primarily to evaluate the performance of the telemetry detector. This equipment provides a simulated telemetry signal mixed with noise to the input of the detector. The ERT compares the output data signal of the detector with the input data signal, counts and displays the errors made by the detector at known signal-to-noise ratios.

4.4.2 Simulated Telemetry Data Generation

The data format generator (DFG) is used to provide simulated telemetry data for test of the MDE telemetry detector computer buffer computer combination. Selected codes may be entered in selected words of the data frame. Special words are automatically generated, and automatically correct or intentionally incorrect parity bits are produced. The various data format modes may be selected.

4.4.3 Spacecraft Simulation

A test transponder is provided with the MDE which is used to simulate the RF portions of the spacecraft for the purpose of compatibility and readiness testing of the in-line MDE. The normal input and output RF interface to the station is made via the station test diplexer. The transponder receiver output may be connected to the MDE command detector and the transponder transmitter may be modulated by simulated telemetry signals.

4.4.4 Station Simulation

Station simulation equipment is provided to thoroughly test the Voyager MDE without use of the station equipment. This equipment consists primarily of a transmitter, a receiver and diplexer which are low level analogs of the station equipment. When used in various combinations with the other MDE equipment provided, all other MDE (except for general purpose test equipment) may be thoroughly tested.

4.4.5 General Purpose Test Equipment

Various general purpose commercial test instruments are incorporated in the MDE to provide for observation, measurement and calibration of various signals throughout the MDE. The following equipment is required:

a) RF signal generator

b) Frequency counter

c) Spectrum analyzer

d) RF power meter

e) Digital voltmeter

f) Oscilloscope

g) Vacuum tube voltmeter

4.5 Computer Programming

Programming is provided that permits the station computer to decommutate the Voyager telemetry data, to output telemetry lines and to the MDE, to make MDE command checks and to accept station time signals. Telemetry data, command data and status data are also typed out on the computer typewriter.

5. FUNCTIONAL DESCRIPTION

The Voyager MDE is housed in five standard JPL DSIF equipment racks. These racks accommodate standard 19-inch wide front panel mounted units and each rack accommodates up to 77 inches total height of unit front panels. (See Figure 1.)

The command controller rack contains the unit (with redundancy) necessary for command generation, two command encloders, two PN generators, their power supplies, space for intercom equipment and a switch panel which is necessary to select which of the redundant units is to be operated.

RF racks Number 1 and Number 2 contain all MDE RF equipment, including the test transponder units, the RF commercial test equipment and the necessary power supply units. These racks also contain the additional commercial test equipment and patch panels associated with the RF and commercial test equipment. Since none of the equipment in the RF racks is primary in-line equipment, no redundancy is provided.

The telemetry extractor rack contains the spacecraft status display unit, two telemetry detectors, teo computer buffers, two power supply units and a switch panel used to select which of the redundant units is to be operated.

The test data generator rack contains the data format generator, the error rate test units and necessary power supplies. This equipment is provided for test purposes and is not redundant. The automatic data handling data entry unit and space for intercom equipment is also provided in this rack.

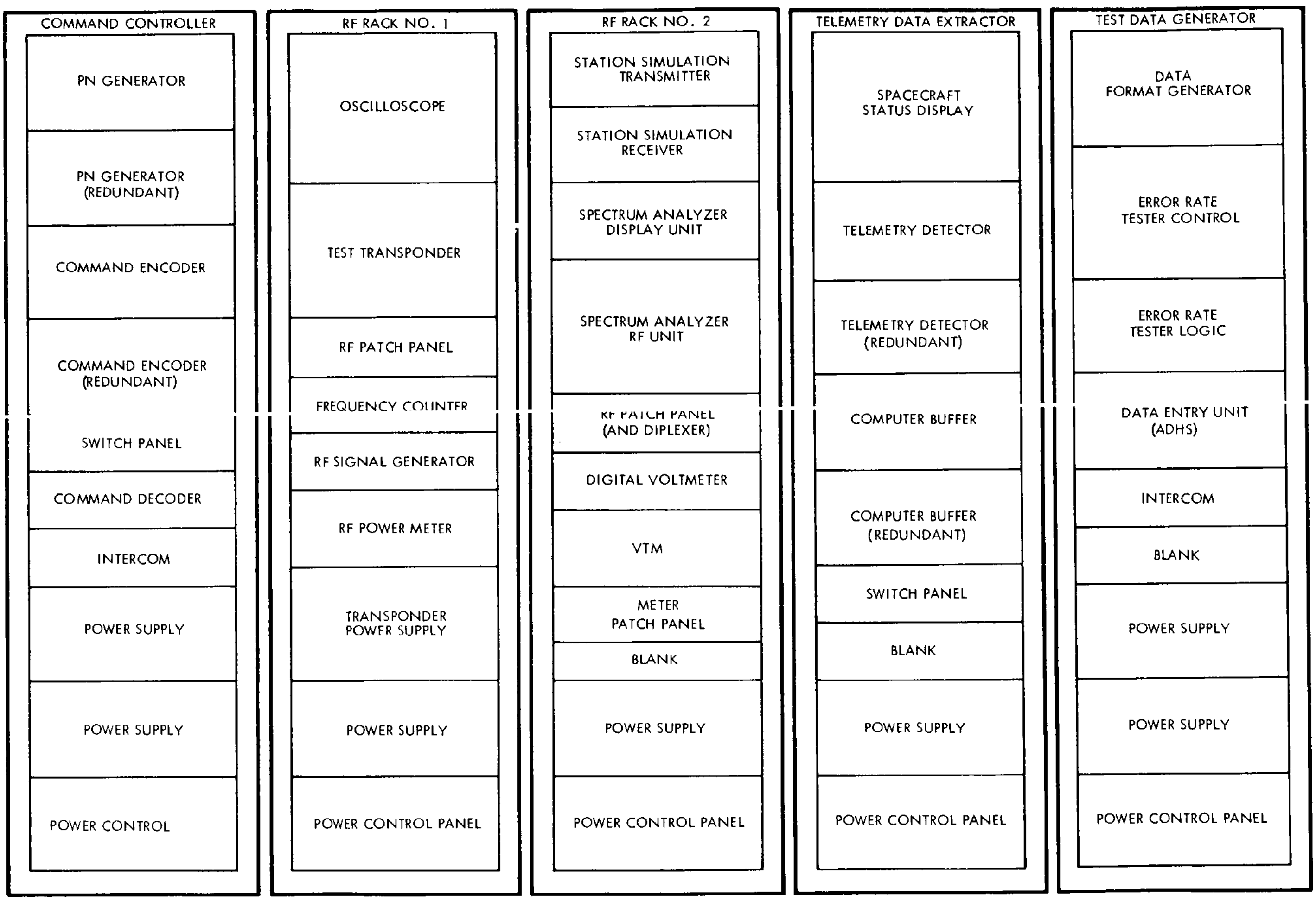


Figure 1. Mission Dependent Equipment, Rack Layout

The lowest space of each rack contains a power control unit which incorporates the necessary rack power circuit breakers, protective circuitry and a rack running time meter. The lower portion of each rack above the power control panel contains the necessary power supply units required for operation of the MDE.

Each of the MDE equipment racks contains a fan and motor to aid in distribution of cooling air throughout the rack. Cooling air is accepted through the bottom of the rack and forced through ducting up one side of the rack and directed at the various electronic assemblies through appropriately located outlets inboard of the ducting. The air flow exists through a grill at the top of the rack.

The racks are RF gasketed around all doors, side panels and around the electronic unit panel mounting surfaces (except for commercial units). The cooling air outlet grill is also designed to reduce RF radiation. All internal signal wiring which might generate high level RF radiation is shielded. All interrack and DSIF station interface wires (except power) are shielded individually and by flexible cable conduit.

All electronic units (except) for commerical test equipment) are RF shielded. Top and bottom covers are sealed by RF gasketing. Cooling air enters and leaves the units through small perforations in their side panels which are designed to limit RF leakage.

All front panel components are chosen for minimization of RF radiation. Where necessary, display devices are covered by conduction glass and other front panel components improved by added RF shielding behind the components.

A functional block diagram of the Voyager MDE is shown in Figure 2. Primary signal paths are indicated by heavy lines. MDE units in the primary signal paths are redundant, but redundant units are not indicated in the block diagram. Secondary signal patchs are indicated by narrow continuous lines and tertiary paths by dashed lines. General purpose test equipment, patch panels, switch panels and power circuits are not indicated in the block diagram.

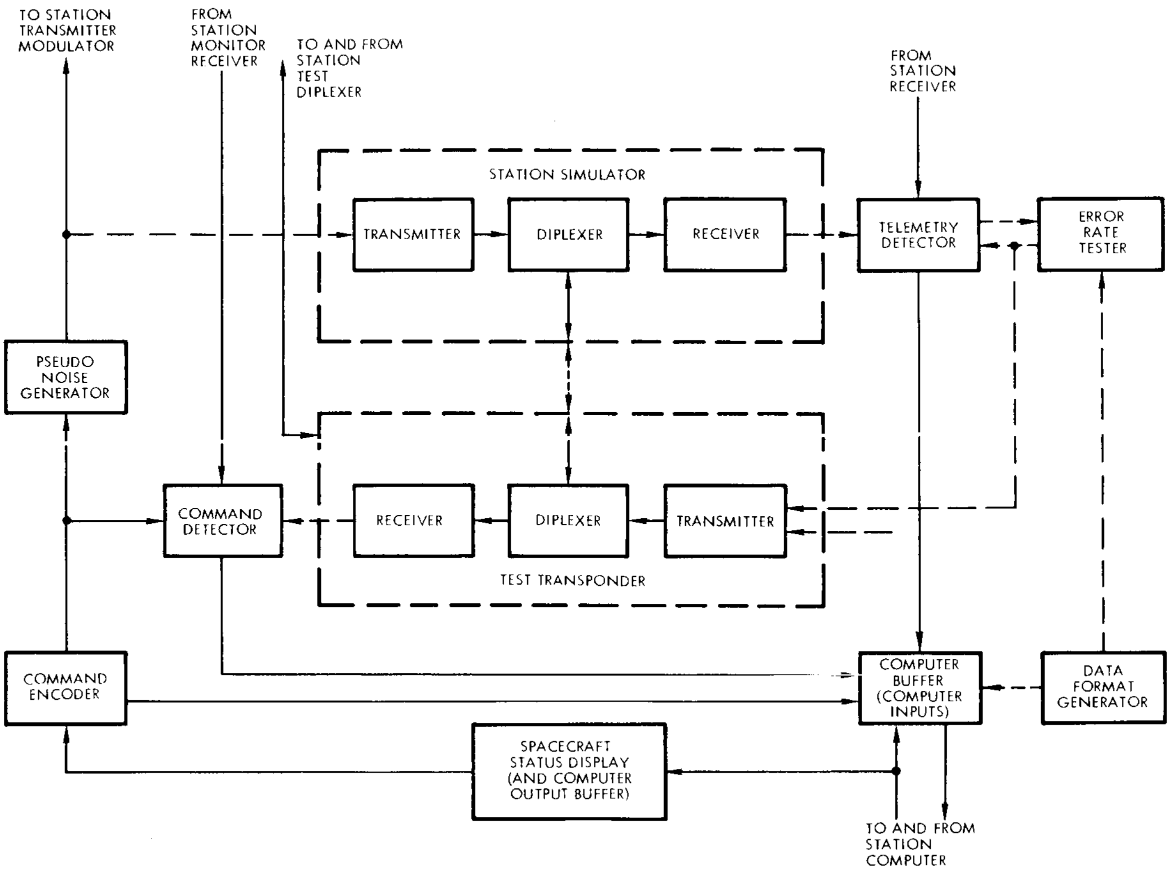


Figure 2. Mission Dependent Equipment, Block Diagram

Provisions are made for station recording of signals labelled with one. Signals labelled two may be recorded and later played back to the same equipment. The signal labelled three indicates that a test signal may be played back from a previously recorded tape.

5.1 Unit Functional Description

5.1.1 Command Encoder

The command encoder provides for manual entry of any commands which may be accepted by the Voyager spacecraft. Upon manual initiation the entered 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 may then be manually initiated. If the command is non-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). This effectively prevents transmission of erroneous commands. An emergency mode of operation is provided in which a command may be transmitted without the normal computer checks of the command.

5.1.2 PN Generator

The PN generator accepts the command encoder output signal consisting of an NRZ bit stream at a rate of one bit per second. This signal is utilized to bi-phase modulate 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 has the capability of generating both code formats required by the redundant spacecraft command encoders. PN format selection is made manually.

5.1.3 Telemetry Detector

The telemetry detector unit accepts the 10 Mc phase modulated telemetry signal from the DSIF station receiver. The bi-phase modulated PN signal is extracted from this carrier. The data bit stream, bit synchronization and word synchronization signals are then extracted from the PN signal.

Data extraction occurs with an error of less than one error in 1000 bits at a signal-to-noise ratio (S/N) within a few db of theoretical.

In order to initially acquire the telemetry data, the detector voltage controlled oscillator (VCO) 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 sweeps) of the VCO. When synchronization occurs the acquisition mode is automatically terminated.

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

5.1.4 Computer Buffer

The computer buffer accepts the serial telemetry and synchronization outputs of the telemetry detector and at proper intervals interrupts the DSIF station SDS 910 computer to enter the telemetry data and synchronization status signal in a format acceptable to the computer.

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) in a format acceptable to the computer. 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.

5.1.5 Command Detector

The command detector unit accepts the output of the DSIF station monitor receiver which is of the same general form as the output of the PN generator consisting of a 511 bit PN code which is repeated synchronously once each command bit period, and which is bi-phase modulated by the command bit stream.

The command detector recovers the command bit stream from the monitor receiver output signal and applies this signal 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.

5.1.6 Spacecraft Status Display

The spacecraft status display unit, in addition to having English legend 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, warning of improper system function as sensed by the computer.

5.1.7 Error Rate Tester (ERT)

The ERT generates a simulated spacecraft telemetry signal (having meaningless data words) which simulates the DSIF station receiver output. A noise signal may be added to this signal in a known ratio and the command signal applied to the telemetry detector input. The output of the telemetry detector is returned to ERT where the detected bit stream is compared bit-by-bit with the simulated telemetry bit stream.

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.

Three signal-to-noise ratios may be selected by means of direct switch calibration. One of these is at a S/N corresponding to theoretical performance of one error in 10^3 bits and the other two are within a few db of that level.

A true RMS voltmeter is provided to precisely adjust the signal and noise to standard levels prior tothe error rate test. Other ratios may be selected by adjusting the signal and noise levels to non-standard levels. Error sample bit counts of 10^3, 10^4, 10^5 and 10^6 may be selected. Bit rates corresponding to the various spacecraft telemetry bit ratio may also be selected.

The simulated telemetry output signal of the data format generator or playbacks of previously recorded telemetry data may also be selected to replace the ERT internal meaningless simulated telemetry data. The output of the ERT may also be applied to the MDE test transponder modulator input.

5.1.8 Data Format Generator

The DFG generates simulated telemetry data formats. Word selection switches are used to address any main frame word or any subcommutated word and, by means of a manually operated switch register, any possible binary data may be enetered in the selected word. Special words such as frame synchronization and subcommutation identification words are automatically generated. Words not selected for entry of the contents of the switch register (and which are not special words) will each contain a unique code or, alternately, all words (except special words) can be made to contain the contents of the switch register. Correct parity bits may be automatically generated for all words using parity, or incorrect parity bits may be selected.

The DFG may be operated at any of the bit rates of the Voyager telemetry data and any of the Voyager telemetry data formats may be selected.

The DFG output may be connected to the input of the computer buffer or it may be connected to the input of the ERT.

5.1.9 Test Transponder

The test transponder simulates the spacecraft communications link. In order that its performance will be as nearly identical to that of the spacecraft as possible, spacecraft RF units will be incorporated in its design.

The test transponder RF input-output is compatible with the DSIF station test diplexer, through which it receives a signal that, except for its low level, has the same characteristics as the signal transmitted to the spacecraft. The transponder in turn generates a low level coherent RF signal that is returned to the station receiver via the test diplexer, completing the simulated spacecraft RF loop.

The test transponder receiver output may be connected to the MDE command detector input. The test transponder transmitter may be modulated by the output of the ERT or by simulated or actual previously recorded telemetry. The transponder receiver input RF signal patch incorporates an attenuator of sufficient range to attenuate the test diplexer output signal to beyond the threshold of the transponder receiver. The transponder transmitter output RF signal patch incorporates an attenuator of sufficient range to attenuate the test diplexer input signal to beyond threshold of the station receiver.

The test transponder and its power unit are designed for either rack mounting or for mounting in portable carrying cases. In their portable configuration the transponder units may be powered by either an integral rechargeable battery or by commercial AC power. When in its portable configuration the transponder may be linked to the DSIF station by an external antenna.

5.1.10 Station Simulator

a. Transmitter

The DSIF station simulation transmitter may be tuned to transmit an RF signal at the command frequency of the Voyager mission. This signal may be modulated by the output of the MDE PN generator. Its RF output may be connected to the RF input of the test transponder. A commercially available signal generator will be utilized to simulate the DSIF transmitter.

b. Receiver

The DSIF station simulation receiver may be tuned to receive an RF signal at the telemetry frequency of the Voyager mission. Its input may be received from the RF output of the test transponder. The receiver output may be connected to the input of the MDE telemetry detector. A commercially available receiver will be utilized to simulate the DSIF station receiver.

c. Diplexer

The DSIF station simulation diplexer is used to connect the station simulation transmitter and receiver to the test transponder RF input-output path. This diplexer is mounted behind the RF rack Number 2 RF patch panel.

5.1.11 General Purpose Test Equipment

The following general purpose test equipment listed below is incorporated in the Voyager MDE in order to provide the measurement and calibration of the MDE:

a) RF signal generator - Hewlett Packard Model 8614A. Frequency range 800 to 4500 MC. Accuracy ±5 MC. Output level + 10 dbm to -127 dbm. 50 ohm output impedance.

b) Electronic counter - Hewlett Packard Model 5245L. Frequency range of 0 to 50 MC. Accuracy ± 1 count. With Frequency Converter Model 524A, frequency range 300 to 3000 MC, accuracy ± 10 counts.

c) Spectrum analyzer - RF section, Hewlett Packard Model 8551A, display section, Hewlett Packard Model 851A. Frequency range 10 MC to 10 gc. 10 spectrum widths between 100 kc and 2 gc.

d) RF power meter - Hewlett Packard Model 431B with Model 478A thermistor mount. Frequency range 10 MC to 10 gc. Accuracy ± 3 per cent of full scale. 7 ranges between 10 μw and 10 mw full scale.

e) Digital voltmeter - Hewlett Packard Model 3440A with Model 3445A AC/DC Range Unit. 4-place read-out with automatic polarity indication. DC accuracy ± .05 per cent ± 1 digit. AC accuracy from ± 2 digits and ± 0.1 per cent at 50 cps to ± 2 digital ± 0.3 per cent of full scale at 100 kc. Ranges 10V, 100V and 1000V full scale.

f) Oscilloscope - Tektronix Model RM45A. Frequency range DC to 30 MC. 18 sweep rates between 2 μ sec/cm and 1 sec/cm. With sweep delay variable between 2 μ sec and 10 sec. With Type C-A Dual Trace Plug-In.

g) Vacuum tube voltmeter - Hewlett Packard Model 400L rack mounted. 12 voltage ranges from 1.0 mv to 300 V full scale. Frequency range 10 cps to 4 MC. Accuracy from 2 per cent of reading at 50 cps to 5 per cent of reading at 4 MC. Input impedance 10 megohms shunted by 25 pt.

5.2 Computer Functions

The DSIF station computer is programmed to decommutate the spacecraft telemetry data. It is necessary that it establish and maintain word and frame synchronization. The current spacecraft telemetry data format and mode are also determined by the computer. Error checking, including parity and telemetry data limit checking, is also made.

Commands are checked for permissibility before their transmission is allowed and commands are bit-by-bit checked for errors during their transmission.

Data acquired by the computer is periodically typed out on the computer typewriter and also upon typewriter request. Telemetry data is transmitted via the DSIF station communication buffer over teletype lines to the space flight operations facility (SFOF).

Spacecraft status data is outputted to the spacecraft status display. DSIF station time is inputted to computer via the station communications buffer.

Computer programming and the list of permissible commands are entered into the computer by means of the computer punched tape reader.

6. BOUNDARY DEFINITION

The Voyager MDE provides command signals for modulation of the DSIF transmitting equipment, accepts the output of the DSIF receiving equipment and interfaces with the DSIF computing equipment. It also accepts the output of the DSIF monitor receiver and interfaces with the station test diplexer. Several points within the MDE may connect to the station recording equipment for magnetic tape and/or strip chart recording of MDE signals. The station tape recording equipment may be played back to several points within the MDE. Boundary signal details are discussed in Section 8.

7. CONSTRAINTS

7.1 Physical

The MDE equipment is designed to be housed in standard JPL DSIF equipment racks. The test transponder units may also be housed in portable carrying cases.

Electronic units do not weigh more than 100 pounds. Complete racks of MDE equipment do not weigh more than 1000 pounds.

Power dissipation in each MDE rack does not exceed 2000 watts.

7.2 Environmental

The MDE racks are designed to accept cooling air at the bottom of each rack. Normal cooling air temperature is to be $55 \pm 10^{\circ}$F. However, the MDE is designed to operate over an inlet air temperature range of 32 to 100°F.

The MDE will operate over a cooling air humidity range of 0 to 95 per cent RH.

7.3 Electrical

The Voyager MDE is designed to meet the requirements for conducted and radiated electromagnetic interference (EMI) and the requirements for susceptibility to EMI in accordance with military specification MIL-I-26600 for Class II equipment. This constraint does not apply to commercial equipment incorporated in the MDE.

8. INTERFACES

8.1 Command Signal

The command signal output of the MDE is defined as follows:

Command bit rate	One bit/second nominal
PN code bit rate	511 bits/second nominal
Signal amplitude	2 volts peak-to-peak nominal
Signal form	Sine wave
Modulation type	Bi-phase
Output impedance	50 ohms nominal

8.2 Telemetry Signal

The telemetry signal input to the MDE is defined as follows:

Subcarrier frequency	10 MC nominal
Modulation type	Bi-phase
Data bit rates	-----
Data bits per word	7
PN bits per word	63
Input signal amplitude	greater than 1 volt
Input impedance	50 ohms nominal

8.3 Test Transponder Interface

8.3.1 MDE Input RF

Frequency	2115 ± 5 MC nominal
Power level	----

8.3.2 MDE Output RF

Frequency	2295 ± 5 MC nominal
Ratio to input RF frequency	240/221
Power level	0 to -100 dbm

8.4 Computer Electrical Interface

The MDE computer programming is prepared on punched tape and entered in the SDS 910 computer by means of the punched tape reader. The punched tape has six data channels and one parity channel.

8.6 Recording Interfaces

8.6.1 Command Encoder Output

Bit rate	1 bit/second nominal
True voltage	+ 6 volts nominal
False voltage	0 volts nominal
Output impedance	50 ohms nominal

8.6.2 Telemetry Detector Output

a) Synchronization status signal - Discrete signal

b) Word synchronization signal - One pulse per word, one data bit in width corresponding to the bit of each data word, word rate 1/7 of bit rate.

c) Bit Synchronization signal - One pulse each data bit occurring at the of each bit: pulse width μ seconds.

d) Telemetry data - Data bit rate range to bits per second, NRZ signal format.

e) Signals (a) through (d) above have the following characteristics:

True voltage	+ 6 volts nominal
False voltage	0 volts nominal
Output impedance	50 ohms nominal

8.6.3 Test Transponder Input

The test transponder recorder input has the same characteristics as the telemetry signal described in paragraph 8.2.

8.6.4 Error Rate Tester Input

The ERT recorder input has the same characteristics as the data output signals of the telemetry detector described in paragraph 8.6.2.

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT
(Flight Spacecraft and Over-all Flight Spacecraft)
OSE/VS-3-140

1. SCOPE

This document defines the functional and design requirements and equipment description for all assembly, handling, and shipping equipment, (AHSE) required for the assembly, checkout, and transport of the flight spacecraft, the planetary vehicle and the planetary vehicle and nose fairing combinations used in the Voyager program.

The models covered by this document will conform to the requirements delineated herein and are identified as the VS-3-140 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL

V-MA-004-001-14-03	Project Document No. 45, Preliminary Voyager 1971 Mission Specification

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints
VS-3-110	Voyager Layout and Configuration

Government

MIL-D-3716A Amend. 2 14 May 1962	Desiccants, Activated for Dynamic Dehumidification
MIL-E-5556A Amend. 1 15 March 1963	Enamel, Camouflage, Quick Dry

MIL-M-008090D 21 February 1961	Mobility Requirements, Ground Support Equipment, General Specification For
MIL-C-13777D Amend. 1 5 November 1963	Cable, Special Purpose, Electrical, General Specification For
MSFC-SPEC-164	Cleanliness of Components For Use In Oxygen, Fuel, and Pneumatic Systems, Specification For

DAC/MSSD

Mechanical Support Equipment and Facilities Manual

3. REQUIREMENTS

The AHSE end items defined in the following paragraphs are designed to perform their specified functions with simplicity of design and operation, long service life, and low manufacturing costs as prime considerations. These end items are associated with the assembly, handling, positioning, testing, hoisting, protection, and storage of the assembled spacecraft structure (with or without any or all other spacecraft subsystems installed in place), as well as the planetary vehicle during assembly and as assembled and transported within the nose fairing at AFETR. The equipment defined below will accomplish all major mechanical handling and support functions.

3.1 Equipment Configurations

The equipment required to support the flight spacecraft and planetary vehicle consists of identifiable end items as shown in Table I. Separate functional descriptions of these end items form a part of this document and are separately treated.

Table 1. Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)

Item No.	Nomenclature
3-140-1	Transporter, flight spacecraft
3-140-2	Assembly, handling and tilt fixture
3-140-3	Transport recorder
3-140-4	Fixture, weight, c.g. and MOI
3-140-5	Shipping container group, standard modules
3-140-6	Work platforms, mobile
3-140-7	Adapter kit, Centaur/shroud transporter
3-140-8	Sling assembly, planetary vehicle and nose fairing
3-140-9	Purge unit, freon/ethylene oxide
3-140-10	Planetary vehicle/nose fairing mating and assembly fixture
3-140-11	Sling, flight capsule
3-140-12	Hoist beam and slings, flight spacecraft
3-140-13	Tag lines
3-140-14	Platform, launch stand access
3-140-15	Universal mounting
3-140-16	Environmental cover, flight spacecraft
3-140-17	Hoist sling, environmental cover
3-140-18	Platform, auxiliary access

3.2 Safety Requirements

3.2.1 Electrostatic Protection

The assembly and handling equipment incorporates safety features to eliminate the hazards of static electricity during maintenance and checkout of the spacecraft and its components. All end items coupled to these spacecraft components are operated at the same ground potential.

3.2.2 Magnetic Fields

The assembly and handling equipment is constructed of non-magnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oerstads at or around the spacecraft physical envelope.

3.2.3 Personnel and Equipment Safety

All assembly and handling equipment includes safety features to preclude damage to the spacecraft and its components or injury to the operating personnel during functional performance of the equipment.

3.3 Material

3.3.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.3.2 Fungi and Moisture Resistance

Those materials which resist the corrosive action of moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.4 Transportability and Storage

The assembly and handling equipment is designed for transportability by air or over land. This equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.6 Interchangeability

The design of the assembly and handling equipment requires tolerances or more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.7 Workmanship

All mechanical operating support equipment (MOSE) is designed, manufactured, and assembled using workmanship consistent with the interest of economy and quality production methods.

3.8 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology and at no time during use will there be degradation in the reliability of the spacecraft or spacecraft subassemblies.

3.9 Maintainability

The MOSE is designed so that repairs, adjustments, and overhaul can be readily accomplished by operating personnel using conventional, general purpose tools and equipment.

3.10 Identification and Marking

All MOSE carries adequate marking for identification, and lift points, rated loads, hazard warnings, and special instructions are on the equipment.

TRANSPORTER, FLIGHT SPACECRAFT
OSE/VS-3-140-1

1. SCOPE

This document defines the functional and design requirements and the equipment description for the flight spacecraft transporter required for the transportation and protection of the flight spacecraft.

2. APPLICABLE DOCUMENTS

TRW 1971 Voyager ASE Design Documents

OSE/VS-3-140	Voyager Operational Support Equipment, Assembly, Handling and Shipping Equipment, (Flight Spacecraft and Planetary Vehicle).
OSE/VS-3-140-3	Transport Recorder
OSE/VS-3-140-15	Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle
OSE/VS-3-140-16	Environmental Cover, Flight Spacecraft

3. FUNCTIONAL REQUIREMENTS

The flight spacecraft transporter provides support for the 1971 mission flight spacecraft (less solar panels and solar panel support structure) during transportation between:

- TRW Redondo Beach (using combinations of air and ground transport methods) to the assembly and check-out facility at AFETR
- TRW Redondo Beach to JPL Pasadena overland using existing surface roadways (prototype flight spacecraft less solar panels)
- TRW to Acoustic Test Facility and return overland using existing surface roadways (test facility located in Los Angeles area)
- TRW Redondo Beach to Magnetic Test Site (in Capistrano, California) and return over surface routes
- TRW Redondo Beach to Goldstone, California DSIF Station over surfacr routes and via air transport (engineering model for compatibility tests)

- TRW Redondo Beach to Sycamore Canyon, California using surface routes (engineering model for match mate tests)

- TRW Redondo Beach to White Sands, New Mexico using surface or air transportation (engineering model and PTM for acceleration tests)

- The subcontractor's plant to TRW Redondo Beach using surface routes (structural subsystem assemblies)

- The subcontractor's plant to TRW Redondo Beach and to the Static Firing Site at Capistrano, California using surface routes (midcourse propulsion/pneumatic module).

4. DESIGN REQUIREMENTS

4.1 Load Configuration

The spacecraft configuration is as shown in VS-3-110 except that solar panels and solar panel supports are removed to provide a minimum width load. This configuration presents the following envelope:

Width	106 inches
Length	222 inches
Height	59 inches
Weight	5500 pound maximum

4.2 Mobility

The transporter is designed in accordance with MIL-M-008090D, type III mobility.

4.3 Axle Loading

The total load per axle does not exceed 12,000 pound per axle.

4.4 Environment

4.4.1 Dynamic

The design of the transporter is predicated on isolating the spacecraft from the dynamic environment (shock and vibration) which would result from negotiating a one-inch washboard course at all speeds in the range of zero to fifty miles per hour. The washboard course is square edged 1 inch x 8 inch x 15 feet boards spaced at one-half the axle spacing for tandem axle vehicles or at two-foot spacing for single axle vehicles.

The length of the washboard course allows the entire prime mover-transporter combination to be on the test course for a minimum of ten bumps.

The design of the transporter and spacecraft suspension systems is such that during all transportation (land or air) the dynamic inputs to the spacecraft do not exceed the dynamic inputs (incremental g loads throughout the frequency spectrum) which will be experienced during a normal flight mission and in accordance with Voyager specification OSE/VS-2-110, Design Characteristics and Restraints.

4.4.2 Altitude

The transporter can successfully function at altitudes of from sea level to 8,000 feet.

4.5 Materials

Non-magnetic materials are employed wherever possible.

4.6 Tie Downs

Transport tie downs are provided and stored within the transporter.

4.7 Interfaces

4.7.1 Mechanical

- Universal mounting ring, flight spacecraft and planetary vehicle (OSE/VS-3-140-15)
- Environmental cover, flight spacecraft (OSE/VS-140-16).

4.7.2 Electrical

- Transport recorder (OSE/VS-3-140-3) power connection
- Aircraft power outlet.

5. EQUIPMENT DESCRIPTION

The flight spacecraft transporter consists of a "low boy" type trailer chassis with the following assemblies:

- Towbar and safety chain
- Front steerable undercarriage
- Aft fixed undercarriage
- Shock mounted spacecraft support platform
- Air brake system
- Mechanical emergency brake system
- Landing gear jack pads
- Electrical system (tail and running lights)
- Tiedowns
- Stowage compartment
- Spare tire storage
- Grounding straps

The chassis consists of an aluminum welded monocoque torque box chassis with suitable pads for mounting the subassemblies. The frame has a 14 inch minimum ground clearance and the wheel wells and wheels are designed in accordance with MIL-M-008090D.

The towbar is a purchased MIL-Spec towbar and safety chain assembly with a standard lunette eye.

The front steerable undercarriage is a self-contained unit including leaf, coil, or torsion spring suspension purchased from vendor MIL-Spec standard item inventory. The under carriage bolts with minimum adjustment and alignment to the towbar and transporter chassis frame.

The aft axle undercarriage mounts two standard truck tires per axle end on a nonsteerable spring axle assembly. The entire assembly bolts to the chassis with minimum alignment and adjustment of suspension linkage.

The spacecraft support platform forms the mating plane for the environmental cover and is fabricated from aluminum sheet and extrusions to provide environmental protection and support for the spacecraft. The support platform forms a drop well configuration to decrease the overall height of the transporter with its environmental cover attached. An integrated shock attenuation system isolates the spacecraft from all dynamic inputs. Extrusions with knife-edge seals provide an airtight, positive-pressure, leak-proof seal with the environmental cover.

The brake system is an air-fluid type with an emergency air reservoir stored on the transporter for automatic emergency braking, (i.e., if prime mover air supply lines are severed). Parking brakes are mechanically actuated with linkage located near the towbar attach point and can hold the vehicle on a 20 percent grade.

Landing gear jack pads are provided for frame stabilization and leveling during spacecraft loading operations. These jacks have self-contained gear boxes and ratchet actuator mechanisms.

The transporter electrical system consists of front and rear running lights on the environmental cover, and front and rear stop and clearance lights on the chassis frame placed in accordance with MIL-M-008090D requirements. Suitable electrical connections are provided for connection to the prime mover electrical system.

Shackles are attached to the transporter frame for aircraft transportation tiedown points.

A weather-proof storage box is attached to the chassis frame for tiedown and maintenance tools and miscellaneous equipment stowage. Spare tire stowage is provided in an accessible area on the rear end overhang of the transporter with the spare tire attached to a rack hinged or tracked to the chassis frame. Several grounding straps are connected to the transporter frame to provide grounding to roadbed during transportation. This design concept is illustrated in Figure 1.

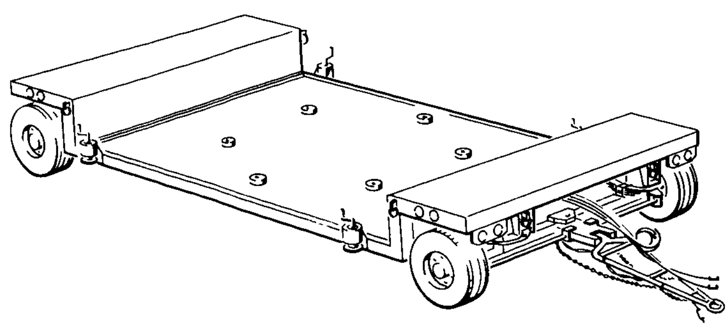


Figure 1. Transporter, Flight Spacecraft

ASSEMBLY, HANDLING AND TILT FIXTURE
OSE/VS-3-140-2

1. SCOPE

This document defines the functional and design requirements and the equipment description for the assembly, handling and tilt fixture required during spacecraft assembly and various subsystem tests and checkout procedures.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

<u>TRW</u> 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager Operational Support Equipment-Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-6	Work Platforms, Mobile
OSE/VS-3-140-15	Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle

3. FUNCTIONAL REQUIREMENTS

The fixture is designed to mount and hold the Voyager flight spacecraft or the planetary vehicle in the vertical position for assembly and test at TRW, Redondo Beach, and at AFETR assembly and test facilities. It also rotates the flight spacecraft or the planetary vehicle about its longitudinal centerline (Z axis) while in the vertical position at a controllable rate, and tilts the flight spacecraft or planetary vehicle in pitch and yaw at a controllable rate.

4. DESIGN REQUIREMENTS

4.1 <u>Load</u>

The fixture supports the total loads of the combined flight spacecraft and flight capsules for 1971 through 1977 mission opportunities. Flight spacecraft weights will approximate 5500 pound and capsule weights will vary from 2300 to 45 pound with corresponding changes in centers of mass.

4.2 Rotation

The fixture rotates the flight spacecraft or planetary vehicle about its longitudinal centerline (Z axis) through a minimum angle of ±15 degree. This rotation is power operated and controlled.

4.3 Pitch and Yaw

The fixture tilts the flight spacecraft or planetary vehicle through an angle of ±15 degree arc as measured from the vertical. The tilt is power operated with interlocks at both ends of travel.

4.4 Rotation Rate

The fixture provides the rotational movement at selectable rates of between $0.03 \pm {}^{00}_{-01}$ rpm to 0.2 rpm, with a maximum speed of 0.5 rpm.

4.5 Pitch and Yaw Rates

The fixture provides tilt movement at selectable rates of between $0.03 \pm {}^{00}_{-01}$ rpm to 0.15 rpm, with a maximum rate of 0.5 rpm.

4.6 Acceleration

With controls set for minimum speed, the instantaneous speed of rotation (Z axis) and pitch/yaw tilt does not exceed 0.03 rpm.

4.7 Console

The console mounts all controls and instruments required to operate the tilt fixture, and interfaces with required recording equipment.

4.8 Controls

The motor controls are reversible, equipped with "press to start" and "press to stop" buttons and a "jog" button for each motor.

4.8.1 Servo Provisions

The motors and controls are compatible with servo operation using externally generated signals.

4.8.2 Position Indication

Indicators and sensors indicate the position in all three orthogonal planes and are designed for connection to a computer.

4.8.3 Rate Indicators

Rate indicators and sensors indicate rate of change of position and have interconnections for computer use.

4.8.4 Safety

All motors and controls are capable of operating in an explosive hazard atmosphere.

4.9 Brakes

Self locking drivers as well as separate brakes on each drive provide positive position holding in the event of motor or gear failure.

4.10 Power

Motors are compatible with power available at the using sites.

4.11 Radio Frequency Interference

Radio frequency interference (RFI) is minimized by proper shielding and grounding. The values of RFI are significantly lower than those required to ignite propellants and explosive devices found in the explosive safe area, and do not interfere with communications of the spacecraft or its components.

4.12 Buzzer

A soft-tone buzzer sounds continuously while fixture movement is in progress.

4.13 Transportability

The fixture is transportable by truck or rail.

4.14 Hoisting

Lifting lugs are provided to permit pickup with an overhead hoist.

4.15 Interference with Spacecraft Sensors

An unobstructed access area is reserved for mounting calibration equipment, spacecraft sun sensors, and the midcourse propulsion module under the spacecraft/Centaur mating interface.

4.16 Height

The fixture supports the spacecraft in a vertical position with sufficient working head clearance under the spacecraft/Centaur interface plane. The overall height is minimized.

4.17 Materials

Non-magnetic materials are employed to the maximum extent possible.

4.8 Interface

The fixture mounts the flight spacecraft universal mounting ring, and operates compatibly during assembly of subsystem equipment with the work platforms, OSE/VS-3-140-6.

5. EQUIPMENT DESCRIPTION

The assembly, handling and tilt fixture consists of the following major subassemblies:

a) An aluminum "H"-beam support-structure assembly, and sub-frame assembly

b) Electric tilt motor, braking system, and gear box assembly

c) Trunnion-mounted caliper jaw support and drive assembly

d) Central control and monitoring console

e) Mounting ring and track.

The support structure assembly is fabricated of welded or bolted aliminum-rolled sections forming tow frame structure supports approximately 6 feet high, 7 feet wide, and 5 feet deep. These assemblies are bolt-mounted to a sub-frame assembly which is cast integrally with the concrete slab foundation, thus providing a closed load path. The support structure assemblies can be shimmed and aligned to required level and orientation by providing tolerance in the matching hole pattern to the subframe assembly.

The support structure assemblies mount the electric tilt motor, braking system, and gear box assemblies. Two identical electric tilt motor, braking system, and gear box assemblies are used and operate each of the two sets of caliper jaws. The caliper jaws are trunnion mounted to the support structure assemblies and driven through a worm gear set by the tilt motor. The braking system provides pre-selected 0 - 90^{o} hold points. During the 0 - 15^{o} tilt operations, a pre-selected mechanical override shall provide the specified tilt rate controlled by the central control monitoring console. The 90-degree tilt gear box includes a mechanical ratchet device capable of holding the spacecraft orientation at any pre-selected tilt angle position. The electric tilt motor is designed to supply armature power for clockwise and counterclockwise rotation and is capable of functioning as an electric brake, thereby controlling the tilt velocity of the system. The gear train is either planetary or a worm set or a combination of both to provide the required strength for horizontal tilt control. The magnetic flux of the tilt motor is varied to provide the required torque output. The armature, housing, and bearings are stressed to be loaded as an electric brake. Although tilt is required only through a ±15-degree range, the system provides for a full 90-degree tilt rotation with intermediate lock points to provide flexiblity in working attitudes for spacecraft subassembly installation. The 90-degree tilt is only used to provide additional orientation attitudes to the spacecraft bus structre, without solar array or support structure in place.

The caliper jaws grip the spacecraft mounting ring by preloaded V-notch rollers engaging the mounting ring track. A friction drive motor mounted on the caliper jaws provides the required torque for mounting ring rotation and mechanical clamps attached to the trunnion mounted caliper jaws lock the mounting ring in any desired position.

The mounting ring mounts and supports the spacecraft universal mounting ring (OSE/VS-3-140-20) and the spacecraft or the planetary vehicle. Leveling screws and pads attach the spacecraft to the mounting ring and permit horizontal plane leveling and alignment of the spacecraft or planetary vehicle. The circular configuration of the mounting ring and the spacecraft universal mounting ring, which attaches to the flight spacecraft at the hex point hardpoints on the spacecraft/Centaur interface plane, allows complete visual line of sight and access to the underside of the spacecraft. The spacecraft/Centaur interface plane is located about 7 feet from the floor level which allows access underneath the fixture for various sensor alignment checks, and other installation tasks. Small removable access platforms may be used under the fixture as required.

The central control console is electrically interlocked with both drive motors to provide remote automatic control. For Z-axis rotation, the fixture is locked in the horizontal plane position and the spacecraft may be rotated by the drive assembly to the required rate. During pitch or yaw tilt operations the support ring is locked by the caliper jaws to the appropriate azimuth (pitch or yaw) and the tilt motor operated. As the fixture tilts, the central controller actuates the electric braking system to provide the required speed control. The worm-gear set is engaged and driven by the tilt motor to return the spacecraft to vertical and the trunnion ratchet locks the spacecraft in any desired position.

The design concept is illustrated in Figure 1. A cutaway view of the tilt fixture drive, caliper jaws, and mounting ring is shown in Figure 2.

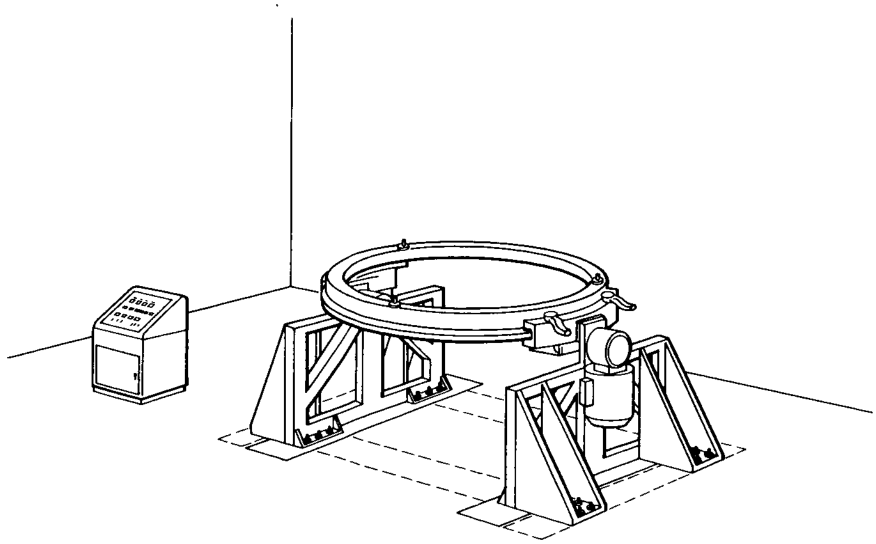


Figure 1. Assembly, Handling and Tilt Fixture

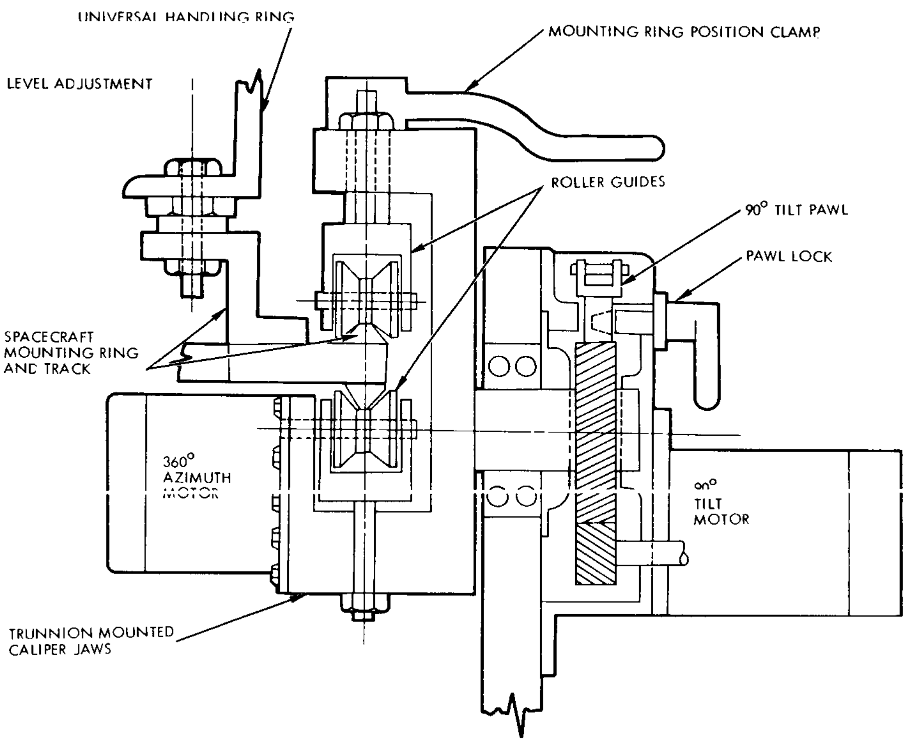


Figure 2. Tilt Fixture Drive, Caliper Jaws, and Mounting Ring (Cutaway View)

TRANSPORT RECORDER

OSE/VS-3-140-3

1. SCOPE

This document describes the functional and design requirements and the equipment design for the transport recorder.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager	OSE Design Documents
OSE/VS-3-140	Voyager OSE Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-1	Transporter, Flight Spacecraft
OSE/VS-3-140-16	Environmental Cover, Flight Spacecraft

3. FUNCTIONAL REQUIREMENTS

The transport recorder provides a record of the induced and natural environmental conditions to which the spacecraft on its transporter enclosed within the environmental cover has been exposed during transportation by land or air during the various interfacility shipments.

4. DESIGN REQUIREMENTS

4.1 Measurements

The transport recorder provides a permanent log of the measurement discussed in the following paragraphs. These measurements are recorded continuously during shipment.

4.1.1 Acceleration

The vibration and acceleration loads and frequency ranges are recorded in the lateral, longitudinal, and vertical directions and record of duration and integrated loading is also provided.

4.1.2 Temperature

The temperature within the environmental cover is recorded with the recorder functioning within the range of 0 to 150°F.

4.1.3 Humidity

The relative humidity within the environmental cover is recorded with the recorder functioning within the range of 0 to 100 percent RH.

4.2 Capabilities

The transport recorder is capable of continuous operation for at least 150 hours without reloading. The recorder is powered by self-contained batteries or by external 24 Vdc power sources provided by the land or air carrier.

4.3 Loads

The transport recorder is capable of withstanding the loads induced on the flight spacecraft transporter during transportation.

5. EQUIPMENT DESCRIPTION

5.1 General

The transport recorder consists of an aluminum console which contains a three-directional impact, acceleration, and gravity recorder and a temperature and humidity recorder. Both recorders contain styli for permanently recording the impact, humidity, and temperature readings on a chart or sensitive tape. The sensing elements of these recorders are attached to the flight spacecraft at critical areas or at the spacecraft/transporter mounting interface. The console contains removable windows for observing movement of the styli and for removal and replacement of the tapes.

5.2 System Interface

5.2.1 Mechanical Interface

The transport recorder is mounted within the environmental cover and the sensing elements are installed on the flight spacecraft.

5.2.2 Electrical Interface

The transport recorder is capable of being connected to an external power source when located within the environmental cover.

FIXTURE — WEIGHT, CENTER OF GRAVITY AND MOMENTS OF INERTIA OSE/VS-3-140-4

1. SCOPE

This document describes the functional and design requirements and the equipment description for the weight, c.g., and MOI fixture required to determine the mass properties of the flight spacecraft and planetary vehicle.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager OSE, Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-15	Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle

3. FUNCTIONAL REQUIREMENTS

The weight, c.g., and MOI fixture provides capabilities for mass properties measurements under the following assembly conditions:

a) Flight spacecraft only with propellants fully loaded

b) Flight spacecraft only with propellants fully depleted

c) Planetary vehicle fully loaded

Measurement of moments of inertia is limited to the proof test spacecraft. The purpose of MOI measurement is to verify principal MOI input to a computer program.

4. DESIGN REQUIREMENTS

4.1 Measurement Conditions

Weight, c.g., and MOI will be measured for the different configurations noted in Table I.

Table I. Mass-Properties Measurements Tests — Requirements

Configuration	Weight	Center of Gravity (c.g.)			Moments of Inertia (MOI)			Remarks
		Z-Axis	X-Axis	Y-Axis	I_Z	I_X	I_Y	
Flight spacecraft only								Proof-test model
1. Fully Loaded with propellants								
a. Antenna stowed, POP stowed	X	X	X	X	X	X	X	
b. Antenna deployed, POP stowed	X	X	X	X				
c. Antenna deployed, POP deployed								
2. Propellant depleted								
a. Antenna stowed, POP stowed	X	X	X	X	X	X	X	
b. Antenna deployed, POP deployed	X	X	X	X				
Planetary vehicle								Proof-test model
3. Fully loaded with propellant								
a. Antenna stowed, POP stowed	X	X	X	X	X	X	X	
b. Antenna deployed, POP stowed	X	X	X	X				
c. Antenna deployed, POP deployed	X	X	X	X				
Prototype spacecraft and actual flight spacecraft	X	X	X	X				Prototype and flight articles

4.2 Over-all Accuracy Requirement

4.2.1 Weight

a) Requirement 0.5 percent

b) Target o.25 percent

c) Capability 0.1 percent

4.2.2 Center of Gravity

a) Requirement 0.1 inch

b) Target 0.05 inch

c) Capability 0.03 inch

4.2.3 Moments of Inertia

a) Requirement 3 percent

b) Target 1 percent

4.3 Tare Weight

Optimum tare weight is 10 percent or less of the test item weight.

4.4 Instrumentation

4.4.1 Load Cell

The load cell and cables selected are comptible with measuring system accuracy and functional requirements.

4.4.2 Capacity

The load cell capacity is 6000 pound and capable of withstanding a minimum of 20 percent overload without permanent adverse effects to the system. The ultimate strength is 25 percent of the rated maximum load.

4.4.3 Measuring

The load cell measuring instrument has the following capabilities:

a) Confine instrument nonlinearity without load cell to within ±0.0005 percent of full scale

b) Confine zero instability to less than ±0.01 percent of full scale

c) Provide minimum of two load cell channels

d) Utilize temperature compensation for an eight wire, load cell cable system

e) Provide a zero-adjust control and a full-scale or span-adjust control for each load cell channel

f) Provide readout in percent of load cell capacity in 1 percent and 10 percent add-to-read switch settings. The minimum reading are equivalent to 0.10 pound when used with a 10,000 pound capacity load cell.

4.4.4 Load Simulator Box

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

4.4.5 Power Requirement

The load cell measuring equipment is designed to operate with 115 ± 12 V, 60-cycle, single-phase alternating current.

4.4.6 Environmental Requirements

All equipment furnished is in accordance with the requirements defined in previous paragraphs throughout a temperature range of 70 ± 30°F, and a relative humidity range of 20 percent to 90 percent.

5. EQUIPMENT DESCRIPTION

The weight, c.g., and MOI equipment consists of three major components as follows:

a) Forward support ring

b) Cradle

c) Weighing equipment.

The forward support ring is used to fulfill test requirements for the spacecraft fully loaded for all measurements except longitudinal center of gravity (Z-axis), and MOI about X and Y axis (I_X, I_Y). The cradle assembly provides the spacecraft measurements not obtained with the support ring, and all other measurements for the combined planetary vehicle (see Table I).

Weighing equipment consists of electronic load-cell force transducers used in tension, load-cell cables, linkages, and transducer readout instrumentation. A pictorial representation of this equipment is shown in Figure 1.

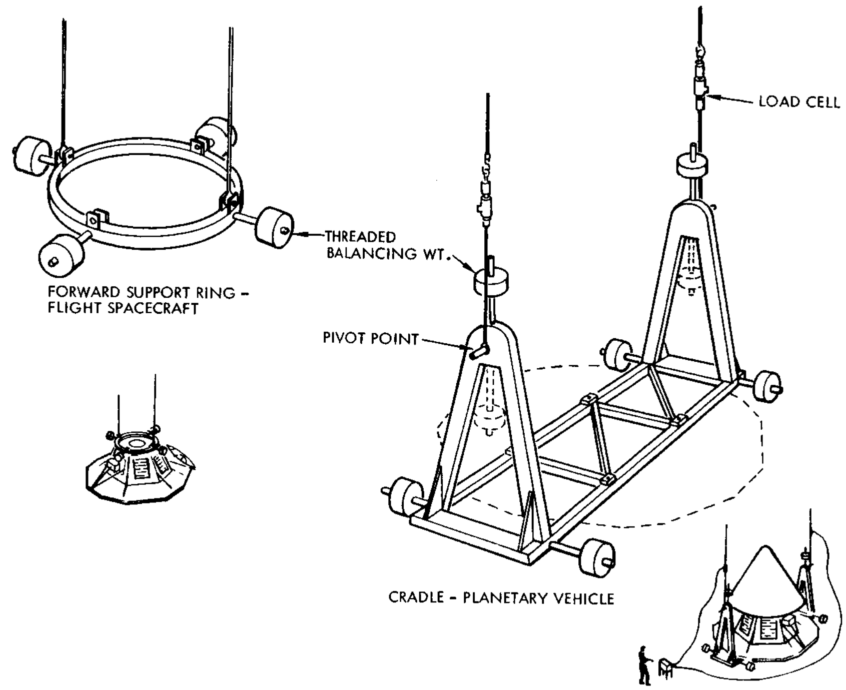


Figure 1. Fixture, Weight, c.g, and MOI

5.1 Forward Support Ring

The forward support ring is fabricated from high-strength aluminum alloy in the form of a torus which mates with the Voyager spacecraft/ flight capsule separation plane. This ring is fastened by three bolts at 120-degree intervals. Four support brackets are provided to attach either two wire cables for MOI measurements or a sling and load-cell linkage for weight measurement. Four arms are fastened about the periphery of the ring. These arms are essentially threaded rods which accept one threaded weight for each arm. The weights are used to determine the center of gravity and MOI (I_Z).

5.2 Cradle

The cradle provides a support for the flight spacecraft alone or when the combined measurements of the planetary vehicle are desired.

It consists of two vertical support members with supporting pivot rods, and is fabricated from I-beams and channels for the maximum rigidity and strength. The cradle doubles as a variable center of gravity pendulum.

5.3 Weighing Equipment

The weighing equipment consists of two load-cell linkages and a measuring indicator. Each linkage consists of a Miller type swivel, a 6,000-pound load cell, and wire-rope cable. One linkage is used when weight of the flight spacecraft alone is measured. Two linkages are required to support the cradle in a vertical position for the weight of the planetary vehicle. The readout equipment is a portable electronic weight kit.

5.4 Creep

Maximum load-cell creep and hysteresis do not exceed 0.5 percent of the applied load.

5.5 Modules and Temperature

Compensation is provided (from 25 to 125°F) to control the following:

a) Effect on sensitivity (output): +0.0013 percent of load/degree F

b) Effect of zero: +0.0025 percent of full scale/degree F

5.6 Linearity

Load cells are 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).

5.7 Load-Cell Cables

Cables have minimum moisture-absorption tendency, and all conductors are of compound F per MIL-C-13777. The cables are neoprene-jacketed, exterior-type highly flexible nonflammable, and are resistant to oils, solvents, chemicals, moisture and aging. They are 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.

5.8 Interface Definition

5.8.1 Mechanical Interface

The fixture is capable of supporting the Voyager flight spacecraft, the planetary vehicle, prototype spacecraft, and actual flight spacecraft.

The weighing equipment interfaces with hooks of overhead cranes and hoists at TRW, Inc. and at AFETR.

5.8.2 Electrical Interface

The load cell is capable of connecting to electrical power systems at the locations designated below.

SHIPPING CONTAINER, STANDARD MODULES
OSE/VS-3-140-5

1. SCOPE

This document defines the functional and design requirements and the equipment description for the standard module shipping container group required for the shipping and storage of standard sized spacecraft electronic and experimental modules.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager	OSE Design Documents
OSE/VS-2-110	Voyager OSE Design Characteristics and Restraints
OSE/VS-3-140	Voyager Operational Support Equipment, Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)

3. FUNCTIONAL REQUIREMENTS

The standard module shipping container group provides environmental protection to standard spacecraft electronic and experimental sub-system modules during transportation and storage.

4. DESIGN REQUIREMENTS

The standard module shipping container group design requirements conform to JPL Specification 20064A.

4.1 Physical Protection

The containers protect the standard modules from damage during transportation and storage.

4.2 Environment

4.2.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the standard electronic modules to less than that occurring during flight environments.

4.2.2 Humidity

The relative humidity in the closed container is less than 20 percent within a temperature range of 0 to 130°F.

4.2.3 Desiccation

Desiccants conforming to MIL-D-3464 are used to maintain the necessary environment.

4.2.4 Altitude

The shipping container group suffers no functional deterioration when subjected to altitudes experienced during air shipment.

4.3 Load Factors

The shipping containers are designed to load and handling factors in accordance with OSE/VS-2-110.

4.4 Venting

When required, venting provisions are made to accommodate altitude changes from sea level to 20,000 feet. Venting occurs through desiccants.

4.5 Transportability

The container is designed for transport by rail, truck or air.

4.6 Clearance

Clearance between the standard module and its inner (intermediate) container, and clearance between the inner (intermediate) and outer (shipping) container is at least two inches or 10 per cent of the maximum equipment dimension, whichever is greater.

4.7 Reusability

The intermediate and shipping containers are reusable.

4.8 Interface

The container group interfaces with all standard electronic and experimental modules which require shipment or storage when demounted from the spacecraft.

5. EQUIPMENT DESCRIPTION

The standard module shipping container group consists of an interior container (handling case), a shock mitigating system, an environmental cover (barrier material), and an exterior shipping container. The interior container is fabricated from a thermoplastic material conforming to acrylic butadiene styrene (ABS) material. The ABS container is of a rigid, ribbed, rectangular configuration capable of withstanding load factors specified in OSE/VS-2-110. The lid of the interior container is hinged with polypropylene material and the hasps are nickel plated. The standard handling container sizes are:

a) 10 x 12 x 11 inches I.D.

b) 10 x 10 x 20 inches I.D.

c) 10 x 10 x 40 inches I.D.

d) 17 x 8.5 x 19 inches I.D.

The shock mitigating inserts of the ABS container consist of three removable cushion pads, fabricated from 1.5 to 2.0 pound density polyurethane foam, ester base. The top and bottom pads are identical and are two inches thick. The center pad is die cut to accommodate the largest standard module of the group; the voids within the die-cut center pad are filled with 1.5 to 2.0 pound density polyurethane foam. A Two-inch minimum separation is maintained in all directions for standard modules of small dimensions or for multiple packing of small modules within the handling case. The standard modules to be handled and the applicable container sizes to be used are shown in Tables I through IV.

Standard modules classified as Type III equipment in JPL Specification 20064A are rigidly mounted to the ABS container. All loads are therefore transmitted to the modules through their normal flight attach points only. Shock isolation (i.e., rubber shear mounts attached to the ABS container or adequate cushioning within the shipping container) is provided to reduce imposed loads on the standard modules to less than those occurring during flight environments.

Table I. Standard Electronic Modules - Group A

Handling Case (ABS) 10x12x11" I.D.

Name	Dimensions of Module (inches)	Name	Dimensions of Module (inches)
J-box	6x6x6	Diplexer	6x3.4x2
Control electronic	6x6x6	Meteroid impact (4 sensors)	6x6x4
20 W power supply	6x6x6	Magnetometer	6x6x4.47
Pre-amplifier	6x3x1	Plasma (2 sensors)	6x6x4.47
VHF receiver	6x3x7	Ionosphere experiment	6x6x7.0
Command detector	6x3x2	Meteroid flash	6x6x5.34
Low power transmitter	6x6x3	I.R. spectrometer	6x6x4.0
Digital T/M unit and combined modulator	6x6x2	Scan radiometer	6x6x4
Mars scan sensors	6x6x2	ACS nozzle	3x3x5
Signal conditioner	6x6x1	Inverter 300 W 4.1 kc	6x6x2
Data storage unit	6x6x2.84	ACS nozzle	1.5x1.5x5
Capsule detector	6x3x1	Inverter 20W 820 cy	6x6x1
Modulator exciter (150mw)	6x3x5	Regulator	6x6x5.34
Receiver selector	6x3x1	Shunt element assembly	6x6x6
Sun sensors	3x3x3	Gyro	6x6x5
Transmitter selector	6x3x1		

Table II. Standard Electronic Modules - Group B
Handling Case (ABC) 10x10x20" I.D.

Name	Dimensions (inches)
Input recorder	6x6x10.56
Command decoder	6x6x10.56
Programmer	6x6x10.56
Power	6x6x10.56
Bulk storage unit	6x6x10.56
S-band receiver	6x3x9
Cosmic ray	6x6x14.25
Trapped radiation	6x6x14.25
UV spectrometer	6x6x16

Table III. Standard Electronic Modules - Group C
Handling Case (ABS) 10x10x40" I.D.

Name	Dimensions (inches)
Data automation equipment	6x6x36
Experiment	6x6x32

Table IV. Standard Electronic Modules - Group D
Handling Case (ABS) 17x8.5x19" I.D.

Name	Dimensions (inches)
Batteries	13x4.5x15
Canopus sensor	11x5x4

The handling case is environmentally controlled. Each standard module is placed in a reusable barrier material. The barrier material conforms to MIL-C-9959, Class II, Grade B, Amendment I, February 5, 1963, with a water vapor transmission value of 0.05 gms to 0.085 gms/100 sq. in./24 hours. The barrier material is made of one of the following materials: scrim foil, nylon reinforced polyvinyl chloride, fluorohalocarbon, or combinations thereof. The barrier material contains desiccant bags conforming to MIL-D-3464 with a humidity indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 20 per cent. The desiccant quantity required is calculated in accordance with MIL-P-116D, paragraph 3.5.6.

Prior to shipment, each standard module in its reusable barrier material is purged with dry nitrogen or dry air to a $0^{o}F$ dew point, desiccated, and evacuated. The encapsulated standard module is placed within the die-cut foam ABS container. The ABS container is placed in a reusable, cleated plywood container conforming to PPP-B-601. The void separating the interior container (ABS) from the exterior container (cleated plywood) is filled with 1.5 to 2.0 pounds density polyurethane foam, ester base. This design concept is illustrated in Figures 1 through 5.

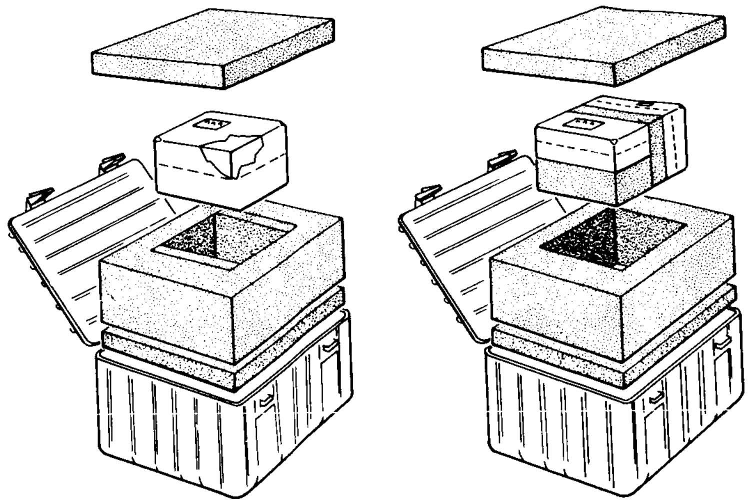


Figure 1. Shipping Container, Standard Modules (Packaging Standard Group A Modules)

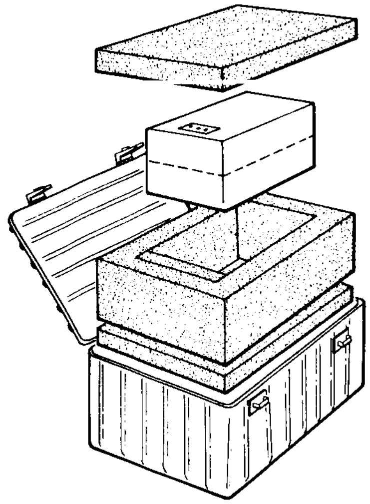


Figure 2. Shipping Container, Standard Modules (Packaging Standard Groups B, C, and D Modules)

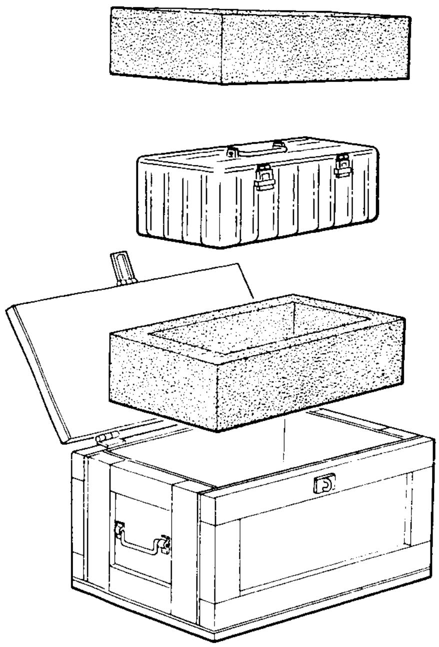


Figure 3. Shipping Container, Standard Modules
(Single Items in Groups A, B, C, D)

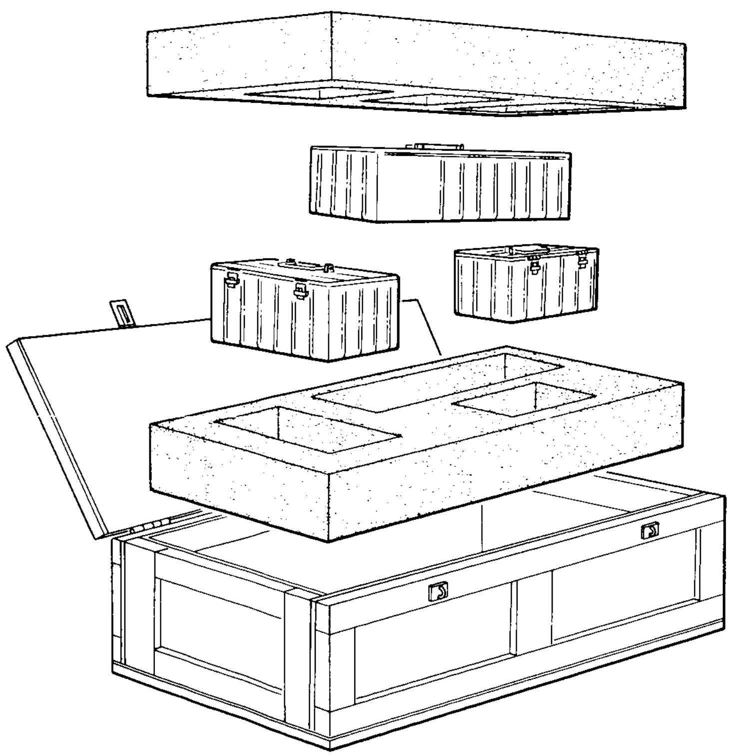


Figure 4. Shipping Container, Standard Modules
(Single Items in Groups A, B, C, D)

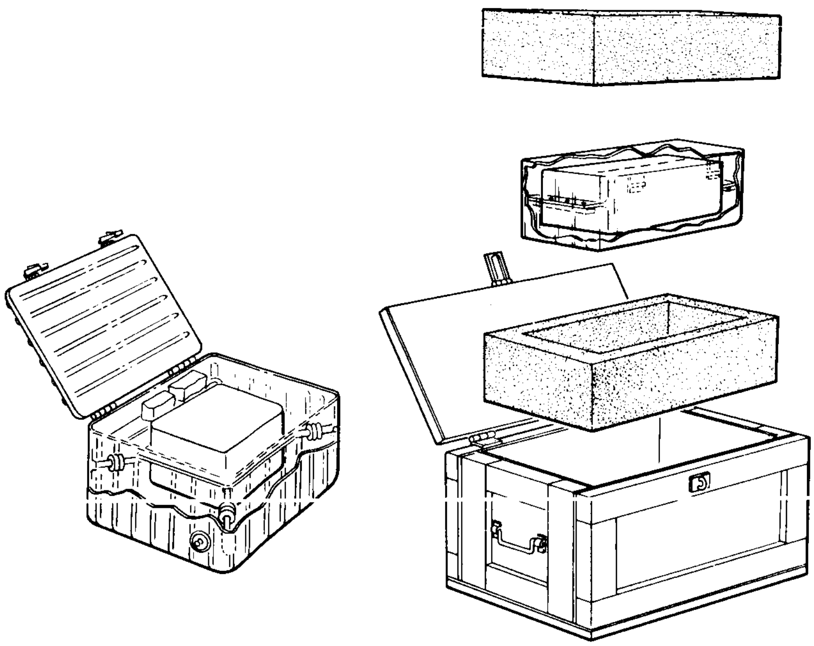


Figure 5. Shipping Container, Standard Modules (Shipping and Handling Container for Type III Modules)

WORK PLATFORMS, MOBILE
OSE/VS-3-140-6

1. SCOPE

This document defines the functional and design requirements and the equipment description for the mobile work platforms required for access to the flight spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting docuemtns are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager Operational Support Equipment, Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-2	Assembly, Handling, and Tilt Fixture

3. FUNCTIONAL REQUIREMENTS

The mobile work platforms provide working access and area for personnel around the periphery of the flight spacecraft when the spacecraft is mounted on the assembly, handling and tilt fixture and various test fixtures.

4. DESIGN REQUIREMENTS

4.1 Loads

The mobile work platforms are capable of supporting a load of 40 pound/ft^2 or 2000 pound on a 2.5 sq. ft. area. The safety rails are capable of withstanding a load of 40 pound per linear foot.

4.2 Mobility

The mobile work platform assembly conforms to the mobility requirements for Type I, Class 1, vehicles per MIL-M-008090D. The running gear consists of casters which are capable of full swivel and which contain parking brakes and swivel locks. No suspension or shock absorbing system is required.

4.3 Stability

Floor jacks are provided for stability when the platform is in place and the platform design provides stability when fully loaded.

4.4 Safety

The platform has safety rails and a kick plate to prevent loose tools or equipment from falling over the platform edge.

4.5 Height

The platforms provide a working level above the spacecraft solar cell plane when it is mounted on assembly and test fixtures.

4.6 Materials

Non-magnetic materials are employed wherever possible.

4.7 Interfaces

The platform is designed to work in conjunction with the assembly handling and tilt fixture (OSE/VS-3-140-2) and its central control and monitoring console. The platform is also used, as necessary, with other special test equipment which mounts the spacecraft.

5. EQUIPMENT DESCRIPTION

The mobile work platform consists essentially of two mirror image units capable of encompassing the total circumference of the flight spacecraft. Each platform consists of a rectangular floor area fabricated of aluminum grip strut or grating supported by a truss structure. The truss structure base provides flat plates to mount the rubber-faced casters. Safety rails, which may be inserted into the floor structure, are provided around the periphery of the platform; lightweight access ladder is attached to each unit. Several hand-operated floor jacks are provided to lift the platforms off the casters and provide stability to the positioned platforms. The platform design allows the working level to be positioned at an elevation coincident with the upper portion of the spacecraft structure when the spacecraft is mounted on the assembly handling and tilt fixture, and other special equipment. The platform work level is cantilevered so that the spacecraft solar panels and their support structure may be mounted to the spacecraft underneath the working level when the platform is in place. This design concept is illustrated in Figure 1.

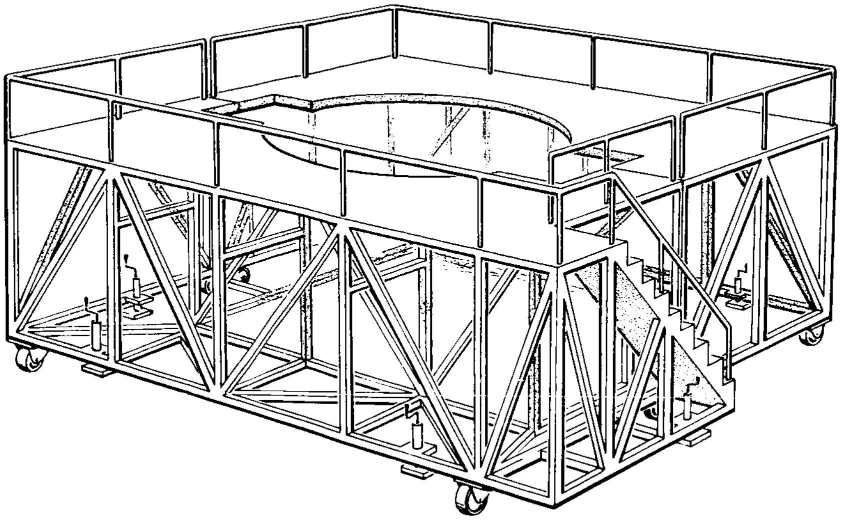


Figure 1. Work Platforms, Mobile

ADAPTER KIT, CENTAUR/SHROUD TRANSPORTER
OSE/VS-3-140-7

1. SCOPE

This document defines the functional and design requirements and the equipment description for the Centaur/shround transporter, adapter kit. This adapter kit permits Voyager program use of the transporter to be employed in the transport of the Centaur stage when mounted in its shroud.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager Mission OSE Documents

OSE/VS-3-140	Voyager OSE Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-10	Planetary Vehicle/Nose Fairing Mating and Assembly Fixture

3. FUNCTIONAL REQUIREMENTS

The planetary vehicle and nose fairing are transported from the explosive safe facility to the launch site. The planetary vehicle, encapsulated within the nose fairing and positioned, but not physically joined together, are oriented vertically (flight mode) on the OSE/VS-140-10 nose fairing mating and assembly fixture and installed on the Centaur/shroud transporter. The Centaur/shroud transporter deck mounts the adapter kit and accepts the combined load.

4. DESIGN REQUIREMENTS

4.1 Loads

The adapter kit accepts the load of the planetary vehicle, nose fairing, and the planetary vehicle/nose fairing mating and assembly fixture, approximately 13,000 pound.

4.2 Size

The nose fairing diameter is 260 inch and the planetary vehicle/Centaur interface diameter is 120 inch.

4.3 Hoisting

Provisions for hoisting the adapter kit are provided at a minimum of three locations for use with a stable hoist sling.

4.4 Load Factors

The load and handling factors are in accordance with OSE/VS-2-110.

4.5 Materials

Non-magnetic materials are used and plastic coatings are employed to prevent interface abrasions and scratches.

4.6 Fasteners

Fasteners are of the quick-release type and are attached to the adapter structure by a vinyl-coated cable or chain.

4.7 Shackles

A minimum of four MS-standard hoisting shackles are attached for adapter handling.

5. EQUIPMENT DESCRIPTION

5.1 General

The handling ring consists of a circular torque tube frame coincident with the 260-inch-diameter fairing field ring. A smaller 120-inch torque tube ring is attached by a truss frame to the 260-inch ring and the larger ring mounts pedestals that interface with the Centaur/shroud transporter deck. Pip-pin quick-release fasteners retain the fairing and planetary vehicle to the inner and outer rings. Flight hardware interfaces are protected by vinyl coatings. A pictorial representation of the equipment is shown in Figure 1.

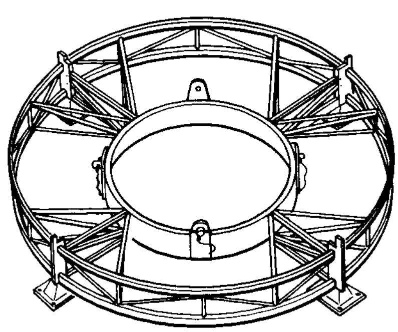


Figure 1. Adaptor Kit, Centaur/Shroud Transporter

5.2 Interface Definition

The equipment interfaces with Centaur/Shroud transporter deck and with 260- and 120-inch planetary vehicle/nose fairing mating and assembly fixture, OSE/VS-3-140-10.

SLING ASSEMBLY, PLANETARY VEHICLE AND NOSE FAIRING
OSE/VS-3-140-8

1. SCOPE

This document defines the functional and design requirements and the equipment description for the planetary vehicle and nose fairing sling assembly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager OSE Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-10	Planetary Vehicle/Nose Fairing Mating and Assembly Fixture

3. FUNCTIONAL REQUIREMENT

The planetary vehicle and nose fairing sling assembly supports the planetary vehicle and nose fairing combination assembled on the planetary vehicle/nose fairing mating and assembly fixture during installation or removal from the launch vehicle on the launch pad.

4. DESIGN REQUIREMENTS

4.1 Load Capabilities and Factors

The planetary vehicle and nose fairing sling assembly is capable of hoisting 11,000 pound plus the weight of the planetary vehicle/nose fairing mating and assembly fixture. It is designed to withstand load factors in accordance with OSE/VS-2-110. During use, no deflection at limit or applied loads which will be detrimental to the performance characteristics of the sling will be permitted.

4.2 Stability

The sling assembly provides stability during all hoisting and translational movement requirements of the planetary vehicle and nose fairing under the environmental conditions encountered at AFETR.

4.3 Ease of Installation and Removal

The sling assembly is easily installed or removed by use of positive-locking quick-disconnect fasteners.

5. EQUIPMENT DESCRIPTION

5.1 General

The planetary vehicle and nose fairing sling assembly consists of four stainless-steel wire-rope assemblies, an aluminum ring, and four aluminum struts. Each wire-rope assembly contains a clevis with a quick-disconnect fastener for attaching to the aluminum ring on one end and a fitting for attaching to a common hoist D-ring on the other. The aluminum ring is fabricated to conform to the outer diameter and conical shape of the nose fairing at a convenient station; it is required to provide stability to the nose fairing and planetary vehicle load combination during hoisting operations. The inner surface of the ring which contacts the nose fairing is lined with a cushioning material to prevent damage to the fairing. The aluminum ring is connected to hoist points on the outer ring of the planetary vehicle and nose fairing mating and assembly fixture by four aluminum struts for transmission of the vertical load. The assembly forms a cage around the planetary vehicle and nose fairing combined load to stablize the load during hoisting. A conceptual design of the sling assembly is shown in Figure 1.

5.2 System Interface

The sling assembly operates compatibly with overhead traveling cranes and gantry hoists at the explosive-safe facility and the launch complex. It attaches to the hoist points on the outer ring of the planetary vehicle/nose fairing mating and assembly fixture.

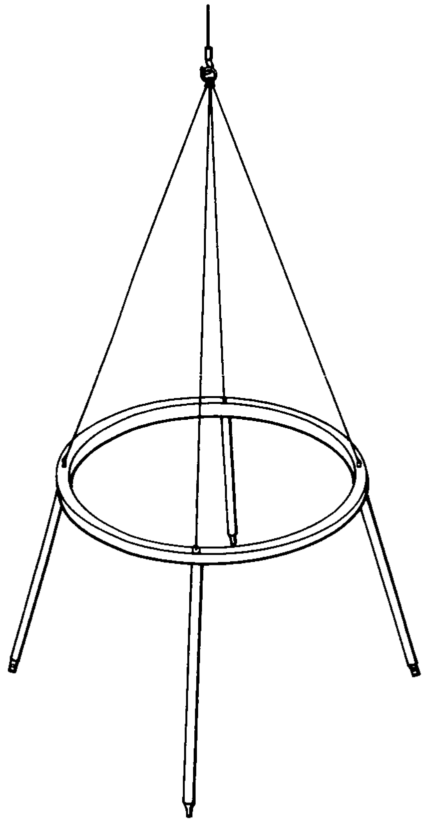


Figure 1. Sling Assembly, Planetary Vehicle and Nose Fairing

PURGE UNIT, FREON/ETHYLENE OXIDE
OSE/VS-3-140-9

1. SCOPE

This document defines the functional and design requirements and the equipment description of the purge unit. The purge unit provides external sterilization for the spacecraft in the explosive safe facility at AFETR.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager Assembly, Handling and Shipping Equipment

3. FUNCTIONAL REQUIREMENTS

The external surface of the flight spacecraft and the flight capsule is sterilized with 12-88 sterilizing gas mixture (12 percent ethylene oxide - 88 percent freon - 12) after they have been installed in the nose fairing. The sterilization is performed in the AFETR safe assembly facility, after which the entire planetary vehicle and nose fairing group are transported to the launch site without violating the aseptic barrier. The sterile package is mated to the Centaur adapter and shroud without disturbing the integrity of the aseptic barrier. External surface sterilization and dry nitrogen purge may also be required on the launch pad to retain the integrity of the spacecraft.

4. DESIGN REQUIREMENTS

4.1 Environmental Design Requirements

4.1.1 Temperature

The temperature inside the nose fairing is maintained at approximately 55°C during the sterilization cycle. Sufficient time is allowed prior to sterilization to permit all surface temperatures to equilibrate.

4.1.2 Humidity

The relative humidity inside the nose fairing is maintained at approximately 50 percent during sterilization.

4.2 Physical Design Requirements

4.2.1 Vacuum

A vacuum of 25 to 26 inch of Hga is drawn, which is required before pressurization with the sterilizing gas mixture, and again after pressurization to exhaust the gas and to vent the nose fairing.

4.2.2 Sterilization Gas Mixture

A 12 percent ethylene oxide - 88 percent sterilizing gas mixture is maintained at a pressure of 7 psig in the nose fairing during sterilization procedures.

5. EQUIPMENT DESCRIPTION

Equipment and supplies for the accomplishment of the sterilization cycle are shown in Figure 1 and are discussed below.

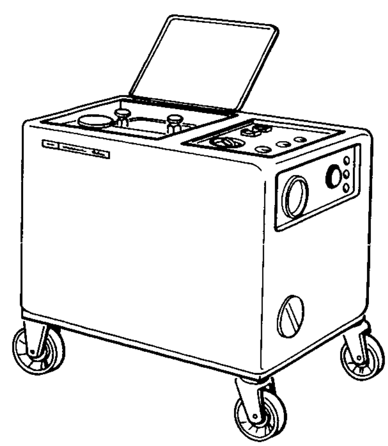


Figure 1. Purge Unit, Freon/Ethylene Oxide

5.1 Vacuum Pump

A vacuum pump is used to evacuate the nose fairing of 25 to 26 inch of Hga during sterilization.

5.2 Heating

Heating elements maintain the temperature in the nose fairing at approximately 55^{o}C during sterilization.

5.3 Humidity

Humidity controls establish and maintain the relative humidity inside the nose fairing at 50 percent during sterilization.

5.4 Sterilization Gas Mixture

A sufficient quantity of sterilizing gas mixture is used to fill the nose fairing to a pressure of 7 psig after it has been evacuated to 26 inch of Hga.

5.5 Timing

An interlocking timer automatically times and purges the nose fairing.

5.6 System Interface

The purge unit is physically connected to the planetary vehicle/nose fairing combination either through an access connection point in the nose fairing wall, or at the Centaur/flight spacecraft interface plane. A second unit is mounted on a gantry service tower so that, if required, sterilizing and/or purging can be continued after vehicle mating on the launch pad.

PLANETARY VEHICLE/NOSE FAIRING MATING AND ASSEMBLY FIXTURE OSE/VS-3-140-10

1. SCOPE

This document defines the functional and design requirements and the equipment for the nose fairing mating and assembly fixture required for mating operations at the launch pad.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

<u>TRW</u> 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager OSE Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-7	Adapter Kit, Centaur/Shroud Transporter
OSE/VS-3-140-8	Sling Assembly, Planetary Vehicle and Nose Fairing

3. FUNCTIONAL REQUIREMENTS

The planetary vehicle and nose fairing are encapsulated together, sealed, and purged while in the AFETR explosive safe facility prior to transporting of the assembly to the launch complex for installation on the launch vehicle. The encapsulation does not provide a mechanical load path between the nose fairing and the planetary vehicle and the entire assembly is installed on the Centaur/shroud transporter. A mechanical interface is required between the VS-3-140-7 adapter kit, the nose fairing, and the planetary vehicle at the 120 inch Centaur interface.

4. DESIGN REQUIREMENTS

4.1 <u>Mating</u>

The fixture is capable of hoisting and locating planetary vehicle and nose fairing in proper orientation over the Centaur adapter and shroud at Station 2048 of the launch vehicle during mating operations. The fixture also guides and positions the nose fairing over the planetary vehicle without interfering with the flight hardware.

4.2 Loads

The ring accommodates the static loads of the planetary vehicle and the nose fairing and provides rigidity and stability under the external environment (such as wind) loads experienced at AFETR during launch-complex mating operations.

4.3 Load Factors

The load and handling factors are in accordance with OSE/VS-2-110.

4.4 Materials

Non-magnetic materials are used and plastic coating is employed to prevent interface abrasions and scratches.

4.5 Fasteners

Two sets of fasteners are supplied with the fixture. One set attaches the fixture to the 120-inch -diameter Centaur flange and the other set to the VS-3-140-7 adapter kit. These fasteners are designed for quick release and rugged handling. Plastic-coated cables are used to attach the fasteners to the assembly fixture.

4.6 Shackles

A minimum of four MS-standard hoisting shackles is attached for ring handling.

4.7 Guide Rails

Removeable guide rails are designed to guide the nose fairing over the planetary vehicle during encapsulation at the explosive safe facility.

5. EQUIPMENT DESCRIPTION

5.1 General

The assembly fixture consists of a circular torque-tube frame concident with the 260 inch -diameter nose fairing field joint. A 120-inch torque tube ring is concident with the Centaur interface. The planetary vehicle/Centaur interface provides special pads for installing the 120-inch ring. Truss pylons connect the inner and outer rings together through small-diameter tension links that pass through access cutouts in the nose fairing and planetary vehicle during encapsulation. All interfacing surfaces are coated with vinyl to protect them from abrasion. A pictorial representation of the equipment is shown in Figure 1.

5.2 Interface Definition

The equipment interfaces with the 260-inch diameter nose fairing field joint, Centaur (120-in.)/planetary vehicle field joint and special mounting pads, and the VS-3-140-8 sling.

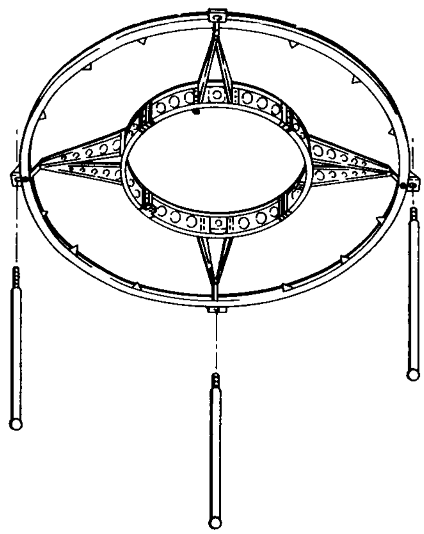


Figure 1. Planetary Vehicle/Nose Fairing Mating and Assembly Fixture

SLING, FLIGHT CAPSULE
OSE/VS-3-140-11

1. SCOPE

The functional description and preliminary design of the flight capsule sling is contained in this document.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL	Over-all Flight Capsule
TRW 1971 Voyager OSE Design Documents	
OSE/VS-3-140	Voyager Assembly, Handling and Shipping Equipment

3. FUNCTIONAL REQUIREMENTS

The capsule sling supports the flight capsule in the vertical position during removal from or installation into the flight capsule shipping container and provides hoisting and translational motion to the flight capsule during assembly and disassembly with the spacecraft interface.

4. DESIGN REQUIREMENTS

4.1 Loads

The sling is capable of hoisting a minimum load of 4,500 pound and is designed using the load factor in accordance with OSE/VS-2-110. No deflection at limit or applied loads detrimental to the performance characteristics of the sling will be permitted. Each sling is intially proof-tested to 80 percent of the yield and proof-tested periodically 1.5 times the rated working load.

4.2 Stability

The sling provides stable support for the flight capsule during hoisting and translational movement operations. To maintain stability, the hoist point is located directly above the c.g. of the flight capsule.

4.3 Ease of Installation

The sling is easily removed or installed through use of positive-locking quick-disconnect fasteners which attach to the spreader-bar assembly.

4.4 Length

The sling length is kept to a minimum to allow its use with cranes in buildings with restricted ceiling heights.

5. EQUIPMENT DESCRIPTION

5.1 General

The capsule sling consists of a spreader-bar assembly, a torus-shaped nylon- or dacron-fabric basket, and six wire ropes. The spreader-bar assembly contains six attach points with quick-disconnect fittings and a hoisting ring. The wire ropes are equipped with clevis-type swaged fittings for connecting the spreader bar to the webbed basket. The webbed basket conforms to the contour of the flight capsule canister outside of the flight spacecraft/flight capsule connection-point interface. It is split into two sections which lace together for ease of installation and supports the flight capsule over as much area as possible to allow the load on the canister to be minimized.

5.2 System Interface

The capsule sling operates compatibly with overhead traveling cranes, portable floor hoists, and other hoisting equipment.

HOIST BEAM AND SLINGS, FLIGHT SPACECRAFT
OSE/VS-3-140-12

1. SCOPE

This document defines the functional and design requirements and the equipment description of the Voyager flight spacecraft hoist beam and slings.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager Mission OSE Documents

OSE/VS-3-140	Voyager Assembly, Handling and Shipping Equipment

3. FUNCTIONAL REQUIREMENTS

The flight spacecraft hoist beam and slings are capable of hoisting the forward spacecraft bus structure and the assembled spacecraft bus during lifting and positioning requirements of the spacecraft or structure segments onto various test fixtures, assembly fixtures, and the spacecraft transporter. They also provide additional support and rigidity to the spacecraft structure during handling operations.

4. DESIGN REQUIREMENTS

4.1 Load Capabilities and Factors

The hoist beam and sling assembly is capable of lifting a minimum load of 5,500 pound. All other design load factors are in accordance with OSE/VS-2-110.

4.2 Stability

The hoist beam and slings provide stability during all hoisting and translational movement requirements of the specified structure.

5. EQUIPMENT DESCRIPTION

5.1 General

The hoist beam and slings consist of a triangular aluminum beam assembly with a captive bolt or similar attaching hardware at each corner, which attach to the hex points of the forward spacecraft bus structure. Three wire ropes are attached to the corners of the hoist beam assembly and are joined by a D-ring which serves as the assembly hoist point.

This concept is shown in Figure 1.

5.2 System Interface

The hoist beam and sling assembly is compatible with overhead traveling cranes, portable floor hoists, and other hoisting equipment at the manufacturing and assembly facilities, system test areas (both inplant and remote sites), and the AFETR facilities. The hoist beam attaches to the bolt-hole pattern in the upper structural plan spacecraft bus structure.

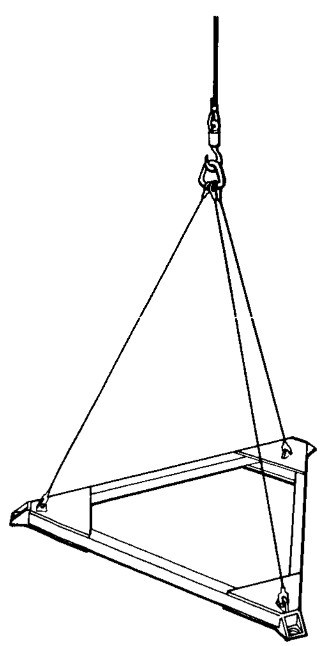


Figure 1. Hoist Beam and Sling, Flight Spacecraft

TAG LINES, VOYAGER HOISTING
OSE/VS-3-140-13

1. SCOPE

This document contains the functional description and preliminary design of the Voyager hoisting tag lines.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager Assembly, Handling and Shipping Equipment
OSE/VS-3-140-10	Planetary Vehicle Nose Fairing Mating and Assembly Fixture

3. FUNCTIONAL REQUIREMENTS

The tag lines are used to stabilize the planetary vehicle and nose fairing combination during hoisting and mating at the launch complex. Ambient wind loads will have a tendency to sway the structure as it is being hoisted into position. To reduce and control such swaying, several tag lines are attached to the planetary vehicle nose fairing mating and assembly fixture (VS-3-140-10).

4. DESIGN REQUIREMENTS

4.1 Loads

Working loads are not known at this time because no wind-load red-line erection criteria are available. However, it is estimated that the working loads will not exceed 200 pound. All other design load factors are in accordance with OSE/VS-2-110.

4.2 Ease of Installation and Removal

The tag lines contain clevis-type end fittings and positive-locking quick-disconnect attachments to facilitate installation and removal.

5. EQUIPMENT DESCRIPTION

5.1 General

The tag lines are made of braided nylon rope or strap with positive-locking quick-release fasteners at their connection ends. The fasteners require manual release features to preclude inadvertent release during use. A canister protects the tag lines from moisture and other corrosive elements during storage. A pictorial representation of the tag lines is shown in Figure 1.

5.2 System Interface

The tag lines attach to the planetary vehicle nose fairing mating and assembly fixture which in turn supports the planetary vehicle and nose fairing combination during launch-complex erection and mating. The tag lines are stored in their container when not in use.

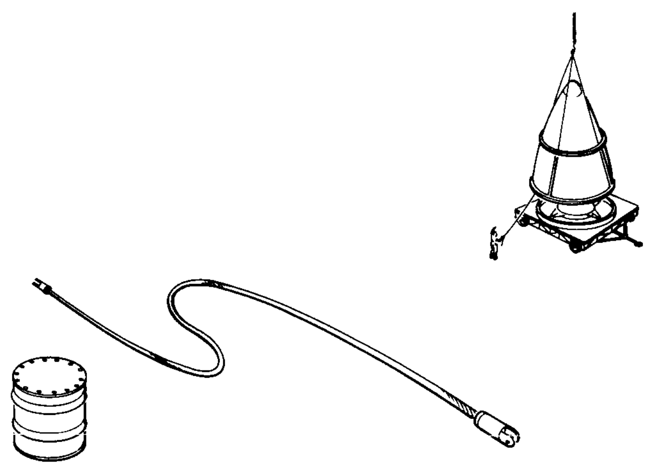


Figure 1. Tag Lines

PLATFORM, LAUNCH STAND ACCESS
OSE/VS-3-140-14

1. SCOPE

This document contains the functional description of the launch stand access platform.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager Assembly, Handling and Shipping Equipment

3. FUNCTIONAL REQUIREMENTS

The launch stand access platform provides working area for personnel at launch vehicle station 2048 (AFETR pads 34 and 37A or B) to allow activities such as installation or removal of the planetary vehicle and nose fairing combination with the launch vehicle, and to allow access for joining the planetary vehicle to the Centaur adapter section.

4. DESIGN REQUIREMENTS

4.1 Loads

The platform is capable of withstanding a live load of 40 lb/sq. ft. or 2,000 pound in an area of 2.5 sq. ft., and a load of 40 lb/linear ft. on the handrails in accordance with AFETR/NASA standard practice for KSC launch complexes.

4.2 Accessibility

The platform allows accessibility to the entire periphery of the launch vehicle at vehicle station 2048 for operating personnel and equipment used in mating procedures.

4.3 Removal

The platform is capable of being removed when not required. Provisions are made for temporarily locking the platform in position in the gantry when in use.

4.4 Safety

The access platform contains guardrails and kickplates around the platform for the safety of operating personnel. Guardrails are 42 inches high and kickplates are 6 inches high to prevent loose articles from falling overboard.

5. EQUIPMENT DESCRIPTION

5.1 General

This platform consists of a nonskid aluminum floor with kickplates and removable handrails around the floor. The platform consists of two or more separate sections with cutouts to provide access around the periphery of the launch vehicle. A ladder may be required to provide access from the launch complex gantry. Design is based on a knock-down modular concept such as is employed with standard removable access scaffolding or platforms no permanent physical attachment to the launch complex gantry is made. Setup and removal is manual, and the access platform contains no automatic drivers or motors for its positioning. (Note: No illustration of this design concept is provided because the configuration is dependent upon the specific launch complex gantry configuration. It is possible that existing gantry structure may provide adequate access, and that this MOSE end item may not be required.)

5.2 Interface Definition

The launch stand access platform is used in conjunction with the gantry structures at pads 34 and 37A or B.

UNIVERSAL MOUNTING RING, FLIGHT SPACECRAFT
OSE/VS-3-140-15

1. SCOPE

This document defines the functional and design requirements and the equipment of the flight spacecraft and planetary vehicle universal mounting ring.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager MOSE Assembly Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-1	Transporter, Flight Spacecraft
OSE/VS-3-140-2	Assembly, Handling and Tilt Fixture
OSE/VS-4-520-1	Dolly Structural Sections

3. FUNCTIONAL REQUIREMENTS

The flight spacecraft mates with various MOSE items and special test equipment during manufacturing, test, and prelaunch operations and connects through selected hard points on the flight spacecraft/Centaur interface plane. The spacecraft is oriented vertically in the (flight mode) during most of these operations. Installation and removal of the spacecraft from the various MOSE items and special test equipment are accomplished without damage to the critical flanges and mating holes of the spacecraft. Hoisting mounting, and spacecraft handling are independent of spacecraft surfaces, panels, and secondary load-path attach holes.

4. DESIGN REQUIREMENT

4.1 Loads

The ring accommodates the following static load conditions:

	Vertical	Horizontal
Longitudinal	7,800 pounds	--
Lateral	--	7,800 pounds
Bending	--	960,000 inch-pounds

4.2 Load Factors

The load and handling factors are in accordance with OSE/VS-2-110.

4.3 Materials

Non-magnetic materials are used and plastic coating is employed to prevent interface abrasions and scratches.

4.4 Fasteners

Two sets of fasteners are supplied with the handling ring. One set attaches the ring to the spacecraft interface bolt hole pattern (three or more) and is not loosened during the handling cycle. The other set attaches the ring to the interfacing MOSE and special test equipment and are designed for quick release and rugged handling. Plastic coated cables are used to attach fasteners to the handling ring.

4.5 Shackles

A minimum of four MS standard hoisting shackles is attached for ring handling.

5. EQUIPMENT DESCRIPTION

The handling ring consists of a segmented welded angle section approximately 10 feet in diameter. The spacecraft interfacing surface of the ring is coated with plastic to prevent scuffing and abrasion of the spacecraft/Centaur mating flange. Shackles are provided for ring handling at four points. The spacecraft fasteners are specially designed for attachment through the Centaur attach holes. The fasteners required for attachment to the MOSE are retained by plastic coated cables. A pictorial representation of the mounting ring is shown in Figure 1.

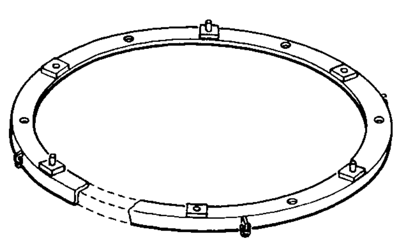


Figure 1. Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle

5.1 Interface Definition

The Centaur bolt-hole pattern is as specified in the Voyager mission specification.

The ring does not interfere with the aft sun sensors and experiment packages so that alignment and adjustment of this equipment can be easily effected when the spacecraft is mounted on the handling ring.

The equipment is compatible with the flight spacecraft transporter, the assembly, handling and tilt fixture, the structural sections dolly, and all special test equipment upon which the flight spacecraft may be mounted.

ENVIRONMENTAL COVER, FLIGHT SPACECRAFT
OSE/VS-3-140-16

1. SCOPE

This document defines the functional and design requirements and the equipment for the flight spacecraft environmental cover.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager OSE Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-1	Transporter, Flight Spacecraft
OSE/VS-3-140-3	Transport Recorder
OSE/VS-3-140-17	Hoist Sling, Environmental Cover

3. FUNCTIONAL REQUIREMENTS

The environmental cover provides environmental protection against the elements for the flight spacecraft during transportation and storage and provide environmental conditioning required by the spacecraft system. The environmental cover is used with the flight spacecraft transporter, transport recorder and environmental cover hoist sling. The transporter with the attached environmental cover is capable of safely transporting the spacecraft as defined in document OSE/VS-3-140-1. The transport recorder is mounted to the environmental cover in such a manner that recorded measurements (shock, vibration, temperature, and humidity) are readily visible. The environmental cover is mounted securely to the transporter by latching devices.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The cover protects the flight spacecraft from physical damage during transportation and shipment.

4.2 Environment

The cover protects the flight spacecraft from the following environment.

4.2.1 Temperature

The temperature within the closed container will be maintained within a range of 20 to 110°F.

4.2.2 Humidity

The relative humidity within the closed container will be less than 20 percent within a temperature range of 20 to 110°F.

4.2.3 Solar Load

The ambient solar intensity is considered to be 105 watts/sq ft of exposed surface area.

4.2.4 Condensation

No moisture condensation is permitted within the cover during transportation or storage.

4.2.5 Corrosion

No corrosive atmosphere is permitted within the cover during transportation or storage.

4.2.6 Dust

The envrionmental cover is equipped with filtering devices to preclude dust particle contamination in excess of 100 μ and which permits maximum air flow.

4.3 Altitude

The environmental cover will function satisfactorily at altitudes from sea level to 8000 feet (or greater if unpressurized cargo aircraft are used for transportation).

4.4 Venting

Venting provisions will be incorporated for air transport to withstand altitudes of 20,000 feet. Venting will occur only through desiccants to prevent moisture influx.

4.5 Hoisting

Hoist points are appropriately located for use in removing the cover from the transporter using the hoist sling (VS-3-140-17).

4.6 Attachments

The devices used to attach the environmental cover to the transporter are capable of withstanding normal road hazards and environment without degradation.

5. EQUIPMENT DESCRIPTION

5.1 General

The environmental cover is made of modular aluminum honeycomb panels capable of withstanding altitudes from sea level to 40,000 feet. The environmental cover is painted white; the paint conforms to MIL-E-5556 and has an emissivity of 0.8. An appropriate undercoating is also required. The environmental cover is cleaned in accordance with MSFC-SPEC-164 prior to being mounted on the transporter. The environmental cover is secured to the transporter and is provided with a relief valve to allow for pressure venting and dry nitrogen oxygen purging. A static desiccant system containing micron filters is mounted to the environmental cover to ensure a relative humidity of less than 20 percent and a particle size influx of less than 100 μ. The micron filter and desiccant canister allows maximum air flow. The transport recorder indicator is mounted to the environmental cover in such a manner that the measurements (shock, vibration, temperature, and humidity) are readily visible. The environmental cover is purged with dry, clean gaseous nitrogen or air to a dew point of $+10^{\circ}F$ prior to shipment by pressurized airplane. If an unpressurized aircraft is used for transportation, the environmental cover is purged to a dew point of $-32^{\circ}F$. For transportation in a C-133A, the environmental cover may require a tapered cross section to fit the cargo loading restrictions. In addition, a thermostat and heater system are placed within the environmental cover to maintain the internal temperature range between 20 and $110^{\circ}F$. The modular honeycomb panels, including the specified mountings (e.g., desiccant canister, thermostat, relief valve, and recorder indicator) are hermetically sealed. A sealant may be required to ensure an environmental seal among the aluminum modular panels, desiccant canister, relief valve, recorder indicator and thermostat. Indicating desiccant conforming to MIL-D-3716 is used in the desiccant canisters. As required, caution signs indicating nitrogen atmosphere are posted on the environmental cover. This design concept is shown in Figure 1.

5.2 Interface Definition

The flight spacecraft environmental cover has physical interface connection with the flight spacecraft transporter, the transport recorder, and the environmental cover hoist sling. In addition, the environmental cover may have electrical interface with ground power or aircraft/road carrier supplied power.

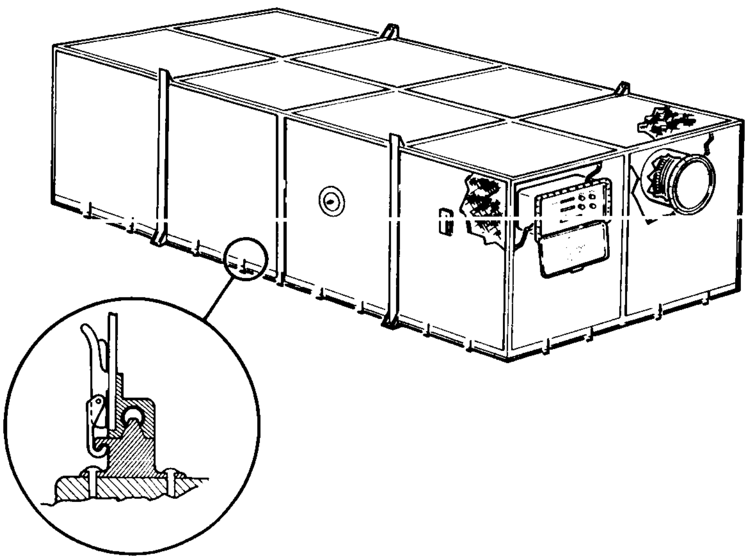


Figure 1. Environmental Cover, Flight Spacecraft

HOIST SLING, ENVIRONMENTAL COVER
OSE/VS-3-140-17

1. SCOPE

This document defines the functional and design requirements and the equipment description for the environmental cover hoist sling.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager MOSE Assembly, Handling and Shipping Equipment (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-16	Environmental Cover, Flight Spacecraft

3. FUNCTIONAL REQUIREMENTS

The environmental cover hoist sling supports the flight spacecraft environmental cover (VS-3-140-16) during installation or removal from the flight spacecraft transporter (VS-3-140-1).

4. DESIGN REQUIREMENTS

4.1 Loads

The environmental cover hoist sling is designed to load factors in accordance with OSE/VS-2-110.

4.2 Stability

The environmental cover hoist sling provides stability to the environmental cover during all hoisting and translational movement requirements and while positioning the cover on the transporter.

5. EQUIPMENT DESCRIPTION

5.1 General

The sling consists of four individual stainless steel wire ropes and a lifting eye assembly. One end of each wire rope is connected by a shackle to the lifting eye plate; the other end contains a selflocking safety hook which attaches to the hoist points on the environmental cover. Each cable is fabricated of proper length to assure that when assembled to the lifting eye assembly and the environmental cover, the lifting eye will be located directly above the environmental cover c.g. The cable connection points are appropriately coded for proper installation. The environmental cover hoist sling concept is illustrated in Figure 1.

5.2 Interface Definition

The safety hooks on the environmental cover hoist sling cables attach to the four hoist points on the flight spacecraft environmental cover (VS-3-140-16). The sling operates compatibly with overhead traveling cranes or mobile hoists.

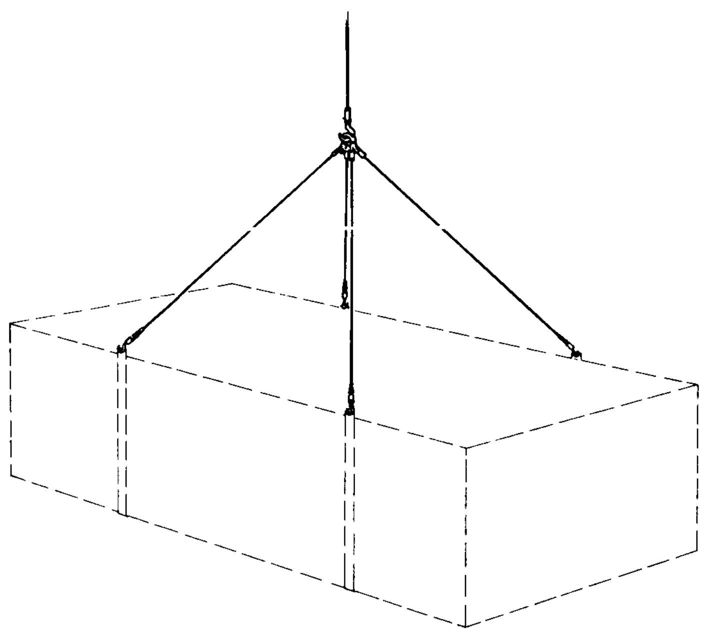


Figure 1. Hoist Sling, Environmental Cover

PLATFORM, AUXILIARY ACCESS
OSE/VS-3-140-18

1. SCOPE

This document defines the functional and design requirements and the equipment description for the Auxiliary Access Platforms required for access around the spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-3-140	Voyager Operational Support Equipment, Assembly, Handling and Shipping Equipment, (Flight Spacecraft and Planetary Vehicle)
OSE/VS-3-140-2	Assembly Handling and Tilt Fixture

3. FUNCTIONAL REQUIREMENT

An auxiliary access platform is required for use during assembly and checkout operations to provide working area around the periphery of the flight spacecraft when mounted on the assembly, handling and tilt fixture (VS-3-140-2) and other special test equipment. It provides personnel and equipment temporary access for purposes of installation and test of the various subsystem elements when the mobile work platforms (OSE/VS-3-140-6) are not installed. The platform also provides access to the flight spacecraft at various elevation levels.

4. DESIGN REQUIREMENTS

4.1 Loads

The auxiliary access platform is capable of withstanding a live platform load of 40 lb/sq ft or a 2,000-pound load on an area 2.5 ft square, and a load of 40 lb/linear ft on the handrails.

4.2 Safety

The platform contains guardrails and kickplates around the periphery of the platform for the safety of operating personnel. The platform guardrails are 42 inches high, and the kickplates are 6 inches high to prevent loose articles from falling overboard. The floor and ladder step plates are constructed of non-skid plate.

4.3 Accessibility

The platform provides adjustable working levels for operating personnel. Levels are required for operational functions to be performed on the flight spacecraft and flight capsule interface and the Centaur interface as well as other levels for installation and removal of various subsystem equipment and test connections.

4.4 Mobility

The platform conforms to the mobility requirements of Type I, Class 1 mobility of MIL-M-008090D. The running gear consists of four swivel casters with swivel locks, phenolic or similar wheels, and parking brakes.

5. EQUIPMENT DESCRIPTION

5.1 General

The platform consists of a rectangular aluminum base on casters, a nonskid aluminum floor and an access ladder. The working level is connected to the floor level by a ladder structure. A handrail is required on the ladder and around the periphery of the working level. A manually operated worm-gear box allows the platform to be positioned at any desired elevation and one or more such platforms may be used as required. Many standard GFE inventories will satisfy these functional requirements.

An alternate method of satisfying the functional requirements consists of providing a knock-down-type scaffolding platform without level adjustability. The preferred design concept is shown in Figure 1.

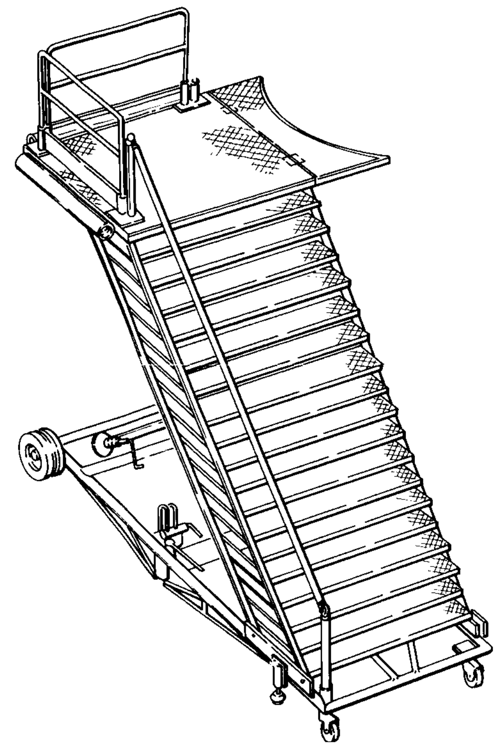


Figure 1. Platform, Auxilliary Access

5.2 Interface Definition

The auxiliary access platform is used in conjunction with the assembly handling and tilt fixture (VS-3-140-2) and other assembly and special test equipment and fixtures.

SCIENCE PAYLOAD SUBSYSTEM
OSE/VS-4-210

1. SCOPE

This document defines the general requirements equipment list and applicable documents for science payload subsystem MOSE required for the shipment, storage protection, and alignment of the body and boom mounted science payload subsystem used in the Voyager program.

The models covered by this document will conform to the requirements delineated herein and are identified as the VS-4-210 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

PPP-B-621A Amend. 2 12 April 1963	Box, Wood, Nailed and Lock-Corner
MIL-D-3464B 31 October 1955	Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification
MIL-C-9959 Amend. 1 5 February 1963	Container, Flexible, Reusable, Water-Vaporproof

DAC/MSSD

Mechanical Support Equipment and Facilities Manual.

3. REQUIREMENTS

The science payload subsystem MOSE defined in the following paragraphs, is designed to perform its specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations. The end items defined within this document are associated with the shipping, protection, and storage of the science payload subsystem experiment booms and alignment of the body mounted science experiments.

Table I. Science Payload Subsystem

Item No.	Nomenclature
4-210-1	Alignment fixture, science payload
4-210-2	Shipping container, experiment booms

The equipment is functionally described by each end item attached to this document and consists of items shown in Table I.

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The science payload subsystem MOSE incorporates safety features to eliminate the hazards of static electricity, when the equipment is used to support the science payload subsystem experiment booms.

3.1.2 Magnetic Fields

The equipment is fabricated of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the physical envelope of the experiment booms.

3.1.3 Personnel and Equipment Safety

The equipment includes safety features to preclude damage to the experiment booms and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact with each other, which may result in corrosion by electrolytic action, is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or over land and can perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured, and assembled using workmanship consistent with the interest of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of the equipment. Designs are based upon proven methods and technology, and at no time during use will there be degradation in the reliability of the science payload subsystem equipment.

3.7 Maintainability

The MOSE is constructed so that repairs, adjustment, and overhaul can be readily accomplished by operating personnel using conventional general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification, with lift points, rated loads, hazard warnings, and special instructions noted.

ALIGNMENT FIXTURE, SCIENCE PAYLOAD
OSE/VS-4-210-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the science payload alignment fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-210	Voyager OSE Science Payload Subsystem

3. FUNCTIONAL REQUIREMENT

The alignment fixture supports two orthogonally oriented level vials in a plane parallel to the mounting surface of the science payload.

4. DESIGN REQUIREMENTS

4.1 Accuracy

The accuracy of the fixture is at least one order of magnitude greater than required by the alignment specifications of the science payload.

4.2 Installation

The fixture must be installed without interference with any science payload components.

4.3 Fabrication

The fixture is rigid and light weight.

5. EQUIPMENT DESCRIPTION

5.1 General

The fixture consists of an L-shaped platform with three legs dimensioned to rest on three machined areas of the science payload structure. The top surface of the fixture is parallel to the mounting surface of the science payload structure within ±0.005 degrees when the fixture is installed. A ten arc-second coincidence level is mounted along each leg of the "L" on the top surface. These levels are calibrated to show zero deflection with the fixture standing on a surface plate which is level within 0.005 inches/foot, as shown in Figure 1.

5.2 Interface Definition

The fixture has mechanical interface with the science payload structure. No other interface exists.

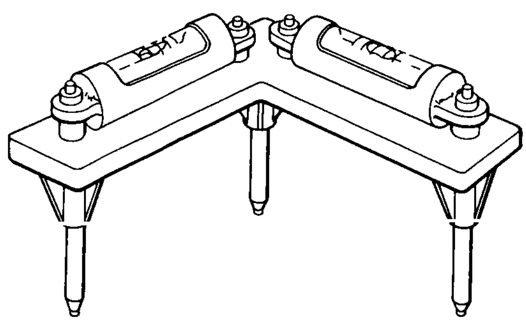


Figure 1. Alignment Fixture, Science Payload

SHIPPING CONTAINER, EXPERIMENT BOOMS
OSE/VS-4-210-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the experiment boom shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-210	Voyager OSE, Science Payload Subsystem

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the DeHaviland boom and magnetometer during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the experiment boom and magnetometer from physical damage during surface and air transportation and during periods of storage.

4.2 Weight and Size

The weight and size are minimum, within the constraints of providing the desired protection. The experiment boom envelope is 57 inches x 5 inches x 5 inches.

4.3 Environment

4.3.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the experiment boom and magnetometer to less than those defined in OSE/VS-2-110, Voyager Design Characteristics and Constraints.

4.3.2 Load Factors

The shipping container is designed to load and handling factors in accordance with OSE/VS-2-110.

4.3.3 Humidity and Temperature

The relative humidity is less than 20 percent, within a temperature range of 0 to 130°F.

4.4 Desiccation

Desiccants conforming to MIL-D-3464B are used to maintain the necessary environment.

4.5 Altitude

The shipping container functions as intended at altitudes consistent with commercial and military air transportation.

4.6 Purge

The environment around the experiment boom and magnetometer is purged with dry nitrogen or dry air prior to shipment.

4.7 Transportability

The container is capable of being transported by rail, truck, or air.

4.8 Reusability

The shipping container is reusable.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping container equipment consists of a shock mitigating system, an environmental cover (barrier material), and an exterior shipping container. The experiment boom and magnetometer is completely encapsulated in polyurethane foam or rubberized hair. The foam or hair nests the experiment boom and magnetometer in a manner which distributes the load equally. The encapsulated boom and magnetometer is enclosed in a barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963, with a water-vapor transmission value of 0.05 to 0.085 g/100 in.^{2}/24 hours. The barrier material is made of one of the following materials: scrim foil, nylon reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. This material contains desiccant bags conforming to MIL-D-3464B with a humidity indicator window capable of being easily inspected. The desiccant will be changed when the indicator shows a relative humidity of more than 20 percent. Prior to shipment, the barrier material is purged with dry nitrogen or dry air to a 0°F dew point, desiccated, and evacuated. The experiment boom and magnetometer, encapsulated in foam and enclosed in its barrier, is placed in a reusable wooden shipping container, conforming to PPP-B-621A. The wooden container provides necessary dunnage to prevent damage to the environmental cover. This design concept is shown in Figure 1.

5.2 Interface Definition

The shipping container is used to store and transport the experiment boom and magnetometer, but has no physical or electrical interface with other operating support equipment.

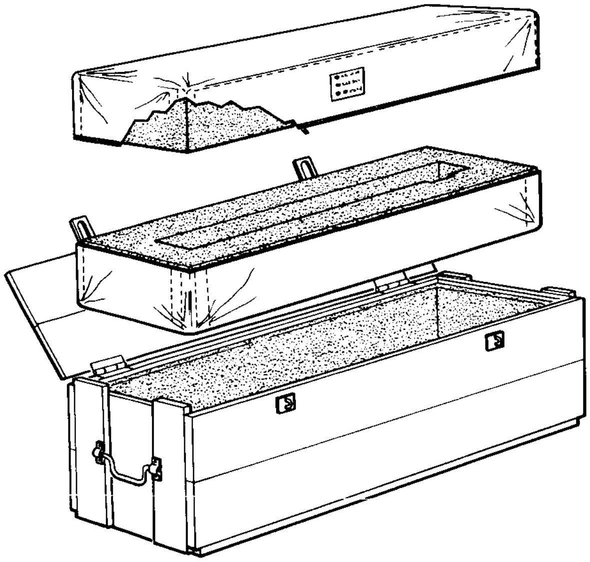


Figure 1. Shipping Container, Experiment Booms

COMMUNICATIONS AND DATA HANDLING SUBSYSTEMS
OSE/VS-4-310

1. SCOPE

This document defines the general requirements, equipment list and applicable documents for all communications and data handling subsystems MOSE required for the assembly, handling, transport, storage, and shipment of the Communications and data handling subsystems equipment used in the Voyager program.

The models covered by this document will conform to the requirements delineated herein and are identified as the VS-4-310 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

MIL-D-3464B 31 October 1955	Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification
MIL-C-9959 Amend. 1 5 February 1963	Container, Flexible Reusable, Water - Vaporproof
MIL-B-26195A 25 May 1962	Boxes, Wood-Cleated, Skidded, Load Bearing Base
PPP-B-601A Amend. 2 16 August 1963	Boxes, Wood, Cleated - Plywood
MIL-P-116D	Preservation, Methods of
MIL-M-008090D	

DAC/MSSD

Mechanical Support Equipment and Facilities Manual

3. REQUIREMENTS

The communications and data handling subsystems MOSE items defined in the following paragraphs are designed to perform its specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

The end items defined within this documentation group are associated with the assembly, handling, hoisting, shipping, protection, storage and alignment of the communications and data handling subsystems equipment. The equipment defined herein accomplishes these major mechanical handling and support functions.

The MOSE required to support the communications and data handling subsystems consists of the items listed in Table I. The functional descriptions are separately described.

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The communications and data handling subsystems MOSE incorporates safety features to eliminate the hazards of static electricity when the equipment is used to support the communications and data handling subsystems components. All MOSE coupled to these components is operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is fabricated of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem component physical envelope.

Table I. Communications and Data Handling Subsystem

Item No.	Nomenclature
4-310-1	Dolly, 6' Parabolic Antenna
4-310-2	Hoist Beam, 6' Parabolic Antenna
4-310-3	Shipping Container, 3' Parabolic Antenna
4-310-4	Shipping Container, 6' Parabolic Antenna
4-310-5	Shipping Container, Low-gain Antenna
4-310-6	Shipping Container, Flight Capsule Receiving Antenna

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the communications and data handling subsystems components and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or over land. It is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use will there be degradation in the reliability of the communications and data handling subsystems equipment.

3.7 Maintainability

The MOSE is constructed so that repairs, adjustments and overhaul can be readily accomplished by operating personnel using conventional general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification, with lift points, rated loads, hazard warnings, and special instructions noted.

DOLLY, ELLIPTICAL PARABOLIC ANTENNA
OSE/VS-4-310-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the dolly for the 6 foot parabolic dish antenna.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-310	Voyager OSE Communications and Data Handling Subsystems

3. FUNCTIONAL REQUIREMENTS

The high-gain antenna with the waveguide tube support, hyperboloid focuser, and microwave horn is transported as an assembly from the bonded stores area to an assembly area for installation on the flight spacecraft. The clean surface and relatively fragile antenna dish structure are protected during transit along standard industrial plant access alleys. Speed is limited to three mph or less while in plant, and movement is accomplished manually or by a shop mule or electric tractor. The installation or removal of the antenna assembly is facilitated by quick-release fasteners.

4. DESIGN REQUIREMENTS

A limited mobility dolly is designed in compliance with the functional requirements specified in Paragraph 3. The design requirements listed below apply.

4.1 Minimum Dimensions

80 inches x 80 inches x 12 inches.

4.2 Mobility

Type I, Class I, per MIL-M-008090D.

4.3 Prime Mover

Standard in-plant mule or electric tractor.

4.4 Axle Loading

Total load is 175 pounds plus the weight of the dolly.

4.5 Emergency Brake

A manual brake capable of retaining the loaded dolly on a 10 percent grade is required.

4.6 Steerable Undercarriage and Towbar

A standard MIL-M-008090D (20-inch pintle height) towbar is connected to a front steerable undercarriage to provide a turn radius of 10 to 15 feet.

4.7 Mechanical Connections

All mechanical connectors attaching the antenna to the dolly are of the positive locking, quick release type.

4.8 Jack Pads

Landing gear jack pads are provided for stabilizing the fully loaded dolly.

4.9 Shock Attenuation System

The total shock attenuation system includes spring-loaded rubber-coated caster wheels and foam stabilizing pads to limit transportation g loads to those discussed in OSE/VS-2-110.

4.10 Equipment Stowage

Provisions are made for stowage of antenna support assemblies and tiedown fittings.

5. EQUIPMENT DESCRIPTION

5.1 General

The VS-4-310-1 dolly consists of a standard dolly chassis with a sheet-metal deck. The following subassemblies are bolted to the chassis: towbar, front steerable undercarriage, aft fixed axle, antenna support pylon, stabilizing foam pads, emergency brake and stabilizing jacks. The chassis consists of a welded tube or torque box frame with suitable pads for attaching the subassemblies. The towbar is a purchased MIL towbar with a standard lunette eye. The front steerable undercarriage is purchased from commercial stock and bolts with minimum adjustment and alignment to the towbar and dolly-chassis frame. The aft fixed axle is a standard purchased part bolted to the chassis. The antenna support pylon supports the VS-4-310-2 hoist-beam assembly, and a vinyl bag provides contamination protection for the antenna dish. Stabilizing foam pads cushion the antenna dish and prevent it from vibrating freely. The emergency brake is purchased or included as an integral part of the caster assemblies and its actuator is located for foot operation. The jack pads are manually operated purchased parts used for stabilizing the dolly during inspection and checkout operation. The dish antenna dolly concept is shown in Figure 1.

5.2 Interface Definition

The dolly is used in direct conjunction with the antenna hoist beam assembly (VS-4-310-2) which provides the mechanical means for installing and mounting the antenna on the dolly. Clearances are sufficient to permit physical inspection and subsystem testing.

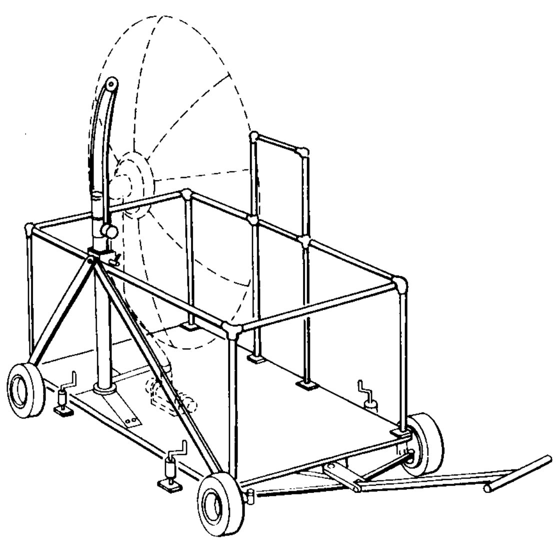


Figure 1. Dolly, Elliptical Parabolic Antenna

HOIST BEAM 6 FOOT PARABOLIC ANTENNA
OSE/VS-4-310-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the hoist beam for the 6 foot parabolic antenna.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-310	Voyager OSE Communications and Data Handling Subsystems
OSE/VS-4-310-1	Dolly, 6 Foot Parabolic Antenna
OSE/VS-4-310-4	Shipping Container, 6 Foot Parabolic Antenna

3. FUNCTIONAL REQUIREMENTS

The parabolic antenna hoist-beam assembly is capable of hoisting and supporting the antenna during its installation on or removal from the parabolic antenna shipping container (VS-4-310-4), the parabolic antenna dolly (VS-4-310-1), or the flight spacecraft.

4. DESIGN REQUIREMENTS

4.1 Loads

The hoist beam assembly is capable of lifting approximately 100 pounds and is designed in accordance with the load factors stated in OSE/VS-2-110.

4.2 Safety

The hoist beam assembly has the necessary safety features to prevent damage to the antenna and injury to personnel. Safety features include padding on antenna interfaces and positive connections to prevent inadvertent disconnection of the hoist beam assembly.

4.3 Rotation

The hoist beam assembly is capable of rotating the antenna package from horizontal to vertical for installation on the spacecraft. Allowance is made for 3 degrees of movement.

4.4 Stability

The hoist beam assembly provides stability to the 6 foot parabolic antenna during the hoisting and rotating operation.

5. EQUIPMENT DESCRIPTION

5.1 General

The assembly is fabricated from standard aluminum tubing. One end has a clamp fitting for attachment to the mounting-plate end of the antenna waveguide tube; the other end has a connection plate to permit attachment to the antenna hard point near the centerline of the antenna. A movable hoisting arm is provided which will be lock controllable so that the antenna can be rotated from a horizontal to a vertical position. The positive locking device allows the antenna to be retained in any required position. The hoist beam, when attached to the VS-4-310-1 dolly, assumes the supporting loads from the dolly frame. The parabolic Antenna hoist beam assembly concept is shown in Figure 1.

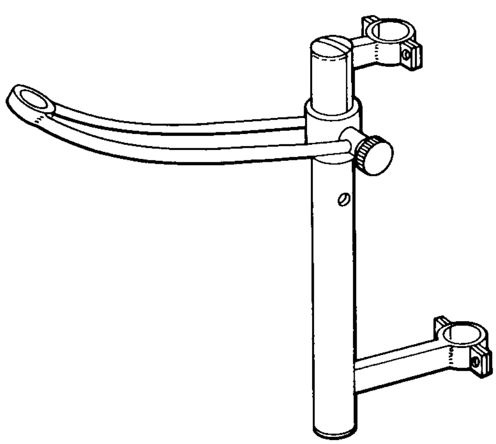


Figure 1. Hoist Beam 6 Foot Parabolic Antenna

5.2 Interface Definition

The hoist beam attaches to the hard points of the parabolic antenna and waveguide tube and mechanically interfaces with the parabolic antenna dolly (VS-4-310-1). The movable hoisting arm is compatible with hooks of standard hydra sets, overhead traveling cranes, and portable floor hoists.

SHIPPING CONTAINER, 3 FOOT PARABOLIC ANTENNA
OSE/VS-4-310-3

1. SCOPE

This document defines the functional and design requirements and the equipment description for the three-foot parabolic antenna shipping container required for the shipping and storage of this antenna.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-310	Voyager OSE Communications and Data Handling Subsystems

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the three-foot parabolic antenna during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the parabolic antenna from physical damage during surface and air transportation and while in storage.

4.2 Weight and Size

The weight and size are minimum within the constraints of providing the desired protection.

4.3 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the antenna to less than those occurring during flight environment.

4.4 Environment

4.4.1 Humidity

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.4.2 Condensation

No moisture condensation is permitted within the container.

4.4.3 Altitude

The shipping container functions as intended at altitudes consistent with commercial and military air transportation.

4.5 Load Factors

The container is designed to load and handling factors in accordance with OSE/VS-2-110.

4.6 Venting

When required, venting provisions are incorporated for air transport operating modes to withstand pressure differentials from sea level to 20,000 feet. Venting occurs through desiccants.

4.7 Transportability

The container is capable of being transported by rail, truck or air.

4.8 Reusability

The shipping container is reusable.

5. EQUIPMENT DESCRIPTION

The shipping equipment consists of a shock-mitigating system, an environmental cover (barrier material) and an exterior shipping container. The antenna is completely encapsulated in 1.8 to 2.0 pound density polyurethane foam. The foam nests the antenna in a manner which distributes the load equally. The encapsulated antenna is enclosed in a reusable barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963, with a water vapor transmission value of 0.05 to 0.085 gms/100 in^2/24 hours.

The barrier material is made of one of the following materials: scrim foil, nylon-reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. These materials contain desiccant bags conforming to MIL-D-3464B with a humidity-indicator window capable of being easily easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 20 percent. The required desiccant quantity is calculated in accordance with MIL-P-116D, paragraph 3.5.6. Prior to shipment, the barrier material is purged with dry nitrogen or dry air to a 0°F dew point, desiccated, and evacuated. The antenna, encapsulated in foam and enclosed in its barrier, is placed in a reusable, cleated plywood shipping container conforming to PPP-B-601. The wooden container provides necessary dunnage to prevent damage to the environmental cover. This design concept is shown in Figure 1.

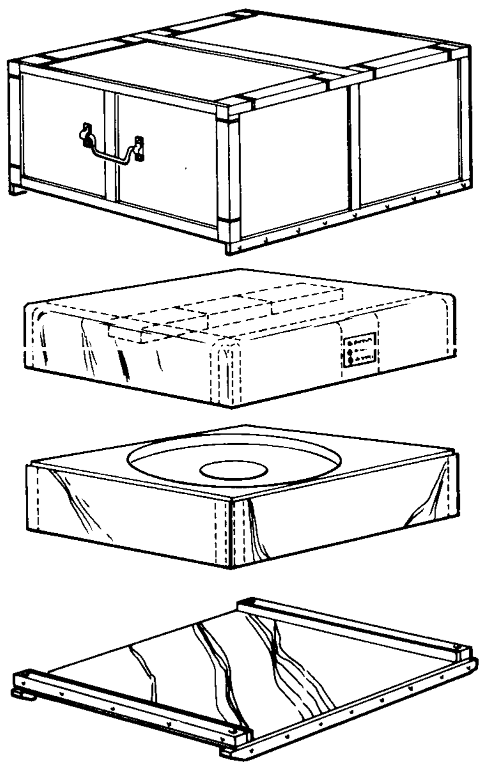


Figure 1. Shipping Container, Three-Foot Parabolic Antenna

SHIPPING CONTAINER, ELLIPTICAL PARABOLIC ANTENNA
OSE/VS-4-310-4

1. SCOPE

This document defines the functional and design requirements and equipment description for the shipping container for the elliptical parabolic antenna.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-310	Voyager MOSE Communications and Data Handling Subsystems

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the parabolic antenna and actuators during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the parabolic antenna from physical damage during surface and air transportation and while in storage.

4.2 Weight and Size

The weight and size are minimum within the constraints of providing the desired protection.

4.3 Environment

4.3.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the antenna to less than that occurring during flight environments.

4.3.2 Desiccation

Desiccants conforming to MIL-D-3464B are used to maintain the necessary environment.

4.3.3 Humidity and Temperature

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.3.4 Load Factors

The shipping container is designed to load and handle factors in accordance with OSE/VS-2-110.

4.3.5 Condensation

No moisture condensation is permitted within the container.

4.3.6. Altitude

The shipping container function as intended at altitudes consistent with commercial and military air transportation.

4.4 Venting

When required, venting provisions are incorporated for air transport operating modes to withstand altitudes from sea level to 20,000 feet. Venting occurs through desiccants.

4.5 Size

The container accommodates an antenna with a major diameter of six feet and a height of 31.25 inches.

4.6 Transportability

The container is capable of being transported by rail, truck or air.

4.7 Reusability

The shipping container is reusable.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping equipment consists of a shock mitigating system, an environmental cover (barrier material) and an exterior shipping container.

The antenna is completely encapsulated in polyurethane or polyethylene foam which nests the antenna in such a manner that the load is distributed equally. The encapsulated antenna is enclosed in a barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963, with a water-vapor transmission value of 0.05 to 0.085 gms/100 in^2/24 hours.

The barrier material is made of one of the following materials: scrim foil, nylon-reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. This material contains desiccant bags conforming to MIL-D-3464B with a humidity indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 20 per cent. The desiccant quantity required is calculated in accordance with MIL-P-116D, paragraph 3.5.6. Prior to shipment, the barrier material is purged with dry nitrogen or dry air to a 0°F dew point, desiccated, and evacuated. The antenna, encapsulated in foam and enclosed in its barrier, is placed in a reusable wooden shipping container conforming to MIL-B-26195. The wooden container provides necessary dunnage to prevent damage to the environmental cover. This design concept is shown in Figure 1.

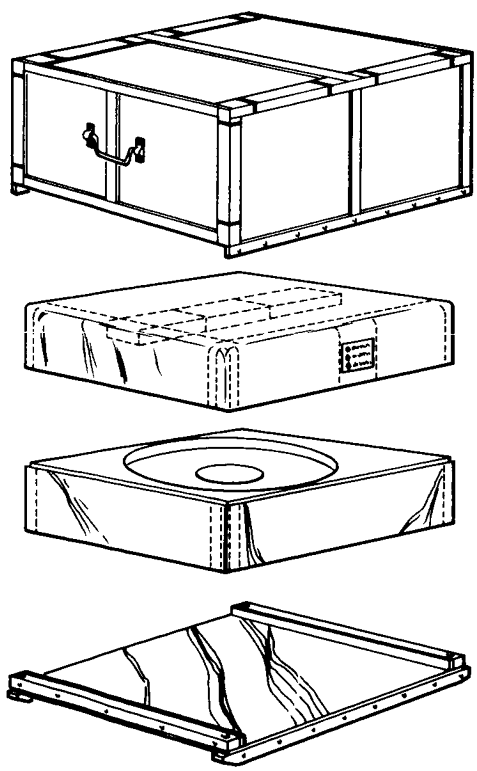


Figure 1. Shipping Container, Elliptical Parabolic Antenna

5.2 System Interface

The shipping container is used to store and transport the 6 foot parabolic antenna but has no physical or electrical interface with other operating support equipment.

SHIPPING CONTAINER, LOW GAIN ANTENNA
OSE/VS-4-310-5

1. SCOPE

This document defines the functional and design requirements and equipment description for the low gain antenna shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-310	Voyager OSE Communications and Data Handling Subsystems

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the low gain antenna and boom during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the low gain antenna from physical damage during surface and air transportation and during periods of storage.

4.2 Weight and Size

The weight and size are minimum within the constraints of providing the desired protection.

4.3 Environment

4.3.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the antenna to less than that occurring during flight environments.

4.3.2 Desiccation

Desiccants conforming to MIL-D-3464B are used to maintain the necessary environment.

4.3.3 Humidity

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.3.4 Load Factors

The shipping container is designed to load and handle factors in accordance with OSE/VS-2-110.

4.3.5 Condensation

No moisture condensation is permitted within the container.

4.3.6 Altitude

The shipping container function as intended at altitudes consistent with commercial and military air transportation.

4.4 Venting

When required, venting provisions are incorporated for air transport operating modes to withstand altitudes from sea level to 20,000 feet. Venting occurs through desiccants.

4.5 Transportability

The container is capable of being transported by rail, truck or air.

4.6 Reusability

The shipping container is reusable.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping equipment consists of a shock-mitigating system, an environmental cover (barrier material) and an exterior shipping container.

The antenna and boom are completely encapsulated in polyurethane or polyethylene foam which nests the antenna in such a manner that the load is distributed equally. The encapsulated antenna is enclosed in a barrier material conforming to MIL-C-9959, Class-II, Grade B, Amendment I, 5 February 1963, with a water vabor transmission value of 0.05 to 0.085 g/100 in^2/24 hr. The barrier material is made of one of the following materials: scrim foil, nylon-reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. This material contains desiccant bags conforming to MIL-D-3464 with a humidity-indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity or more than 20 per cent. The required desiccant quantity is calculated in accordance with MIL-P-116D, Paragraph 3.5.6. Prior to shipment, the barrier material is purged with dry nitrogen or dry air to a 0^{o}F dew point, desiccated, and evacuated. The antenna encapsulated in foam and enclosed in its barrier is placed in a reusable wooden shipping container conforming to MIL-B-26195 or PPP-B-601, depending on dimension and weight limitation. The wooden container provides necessary dunnage to prevent damage to the environmental cover. This design concept is shown in Figure 1.

5.2 System Interface

The shipping container is used to store or transport the low gain antenna but has no physical or electrical interface with other operating support equipment.

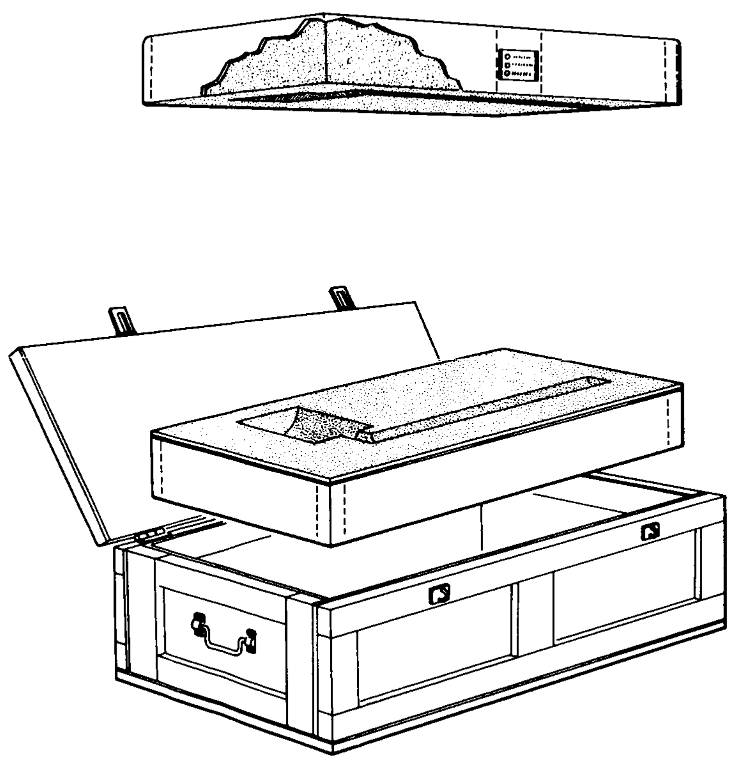


Figure 1. Shipping Container, Low Gain Antenna

SHIPPING CONTAINER, FLIGHT CAPSULE RECEIVING ANTENNA
OSE/VS-4-310-6

1. SCOPE

This document defines the functional and design requirements and the equipment description for the flight capsule receiving antenna shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-310	Voyager OSE Communication and Data Handling Subsystems

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the flight capsule receiving antenna during surface and air transportation and while in storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the antenna from physical damage during storage and transportation.

4.2 Weight and Size

The weight and size are minimum within the constraints of providing the desired protection.

4.3 Environment

4.3.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the antenna to less than those occurring during flight environment.

4.3.2 Humidity

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.3.3 Condensation

No moisture condensation is permitted within the container.

4.3.4 Altitude

The shipping container suffers no functional deterioration when subjected to altitudes experienced during air shipment.

4.4 Load Factors

The shipping container is designed to load and handling factors in accordance with OSE/VS-2-110.

4.5 Venting

When required, venting provisions are made to accommodate altitude changes from sea level to 20,000 feet. Venting occurs through desiccants.

4.6 Transportability

The container is designed for transport by rail, truck or air.

4.7 Reusability

The shipping container is reusable.

48. Interface

The shipping container interfaces with the flight capsule receiving antenna which requires shipment and storage.

5. EQUIPMENT DESCRIPTION

The shipping equipment consists of a shock mitigating system (foam chocks), an environmental cover (barrier material) and an exterior shipping container.

The shock mitigating system for the flight capsule receiving antenna container consists of several foam chocks, fabricated from 1.5 to 2.4 pound density polyurethane foam. The foam chocks support and stabilize the antenna and are of the thickness necessary to reduce the imposed loads on the antenna to less than that occurring during flight environment. The foam chocks are bonded to a fiberboard sheet and placed within the barrier material for rigidity.

The antenna is enclosed in a barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963. The barrier material contains desiccant bags conforming to MIL-D-3464B with a humidity indicator window capable of being easily inspected. The required desiccant quantity is calculated in accordance with MIL-P-116D, paragraph 3.5.6. The barrier material and the foam chocks separate at the same area to allow for ease of antenna removal.

Prior to shipment, the barrier material is purged with dry nitrogen or dry air to a 0°F dew point, desiccated and evacuated. The antenna in its foam chocks and enclosed in its barrier is placed in a reusable, cleated plywood container conforming to PPP-B-601 with fork lift capability. This design concept is illustrated in Figure 1.

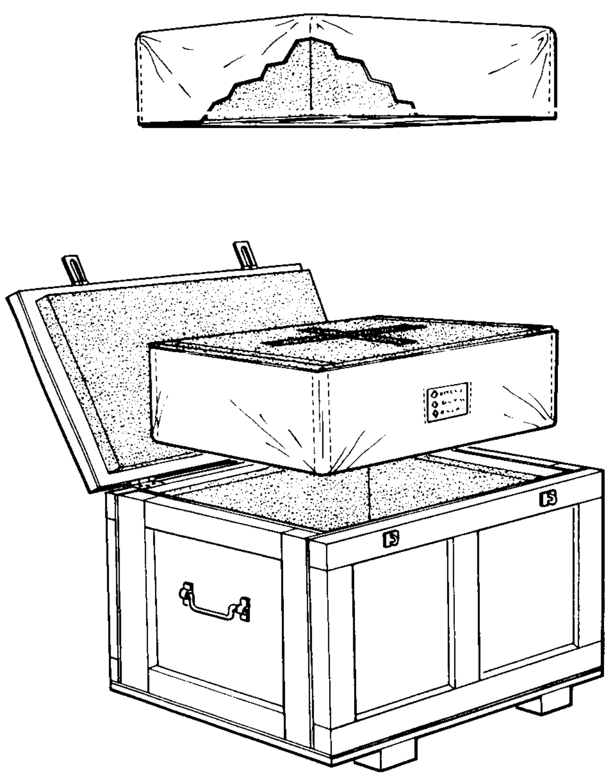


Figure 1. Shipping Container, Flight Capsule Receiving Antenna

S-BAND COMMUNICATIONS SUBSYSTEM
OSE/VS-4-311-1

1. SCOPE

This document defines the functional requirements for the unit test set (UTS) used to test and evaluate the Voyager S-band communications subsystem.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints
OSE/VS-4-310	Voyager Communications and Data Handling Subsystem
OSE/VS-3-120	OSE Automatic Data Handling System

3. FUNCTIONAL REQUIREMENTS

The unit test set is configured to support testing through the "black box" level of the various subassemblies comprising the subsystem. A functional breakdown of the S-band communications subsystem in the Voyager complex includes:

a) S-band receivers

b) S-band transmitters

c) Power amplifiers

d) S-band peripheral equipment.

To perform testing to the level desired, a building block test philosophy is incorporated into the design of the S-band communications UTS. By a repetitive test and add process using simulated output loading, the subassemblies and then the subsystem are integrated and tested.

This utilized test procedure lends itself to fault isolation at the black box level. By supplementing the test program with telemetry analysis and test point access within a black box and a test harness, sufficient information down to the internal characteristics of a black box is afforded the tester for fault analysis. Early stages of testing will verify telemetry validity over the complete operating range.

The UTS incorporates sufficient equipment to perform a complete OSE self-check cycle prior to flight hardware mating in any test mode. Self-contained analysis equipment provides a check of the stimulus data paths and test equipment to be utilized in a given test. In the self-check configuration, access points, T/M data (selected OSE parts) and patch panel connections provide OSE fault isolation to a specific equipment level. Initial design considerations include ease of access for maintenance purposes.

The UTS includes the necessary hardware and software to insure safe operation with the flight equipment. Limiting devices and fail-safe procedures prevent damage to equipment under test due to OSE failure or erroneous matings.

3.1 Required Functions

3.1.1 Overall Requirements

The S-band communications subsystem provides the following:

a) Input power to equipment under test (EUT)

b) Input power measurement equipment

c) Stimulus, data paths and analizing equipment for all levels of testing

d) Monitoring of T/M data and direct access points and permanent records of parameters being tested

e) Necessary interface equipment and test harnesses

f) Protection for EUT

g) UTS self-check capabilities

h) Adaptability with automatic data handling system (ADHS).

Additionally, this subsystem exercises system components over a complete range of operating conditions and maintains and records operating times where applicable.

3.1.2 S-Band Transmitter-Receiver

The S-band transmitter-receiver provides the following:

a) Stable phase-locked S-band transmitter for evaluating flight and UTS receiver characteristics

b) S-band receiver for evaluating flight and UTS transmitter

c) Controls necessary to exercise all test modes

d) Ranging data display and permanent record for waveform analysis.

In addition, the S-band transmitter-receiver supplies required test equipment to perform applicable subsystem and assembly measurements and generates PN data for ranging loop output analysis.

3.2 Test Functions

The S-band communications UTS tests the following portions of the communications subsystem:

a) Spacecraft S-band receivers

b) S-band receiver selector

c) Modulator exciters

d) Low power transmitter

e) Power amplifiers

f) Four-port coupler

g) Transmitter power detector

h) Transmitter selector.

4. DESIGN REQUIREMENTS

4.1 General

The S-band communications UTS provides the necessary stimuli, data paths, access points and analysis equipment to perform the specific testing described herein to the degree of accuracy given.

4.2 Spacecraft Receivers

The input stimuli for the S-band receivers is obtained individually for sensitivity checks and simultaneously for system checks. A patch panel arrangement for allowing this operation is shown in Figure 1.

4.2.1 Receiver Sensitivity

The receiver input signal during operation ranges between -141 to -50 dbm. The test set performs this measurement between -40 and -160 dbm. The accuracy of measurement over the performance range will be ±0.75 db and from -140 to -160 dbm ±1.0 dbm.

4.2.2 Noise Figure

The receiver noise figure is specified as 10 db maximum. The test noise generator is calibrated to ±0.1 db and overall noise figure measurements are accurate to ±0.25 db over the range of noise figures from 7 to 13 db.

4.2.3 Received Frequency

The test set is capable of measuring frequency from DC to 3 GC. The received frequency is measured with an accuracy of ±1 KC.

4.2.4 In-Lock Signal

The signal strength which causes the VCO to come into lock is measured with an accuracy of ±0.2 db.

4.2.5 Received Signal Strength

The received signal strength is calibrated from the I/Q detector output. The voltage is measured within ±2 percent.

4.2.6 Range Code Output

The amplitude of the range code output signal versus carrier signal level is measured within ±5 percent accuracy.

4.2.7 Command Subcarrier Output

The amplitude of the command subcarrier output signal versus carrier signal level is measured with ±5 percent accuracy.

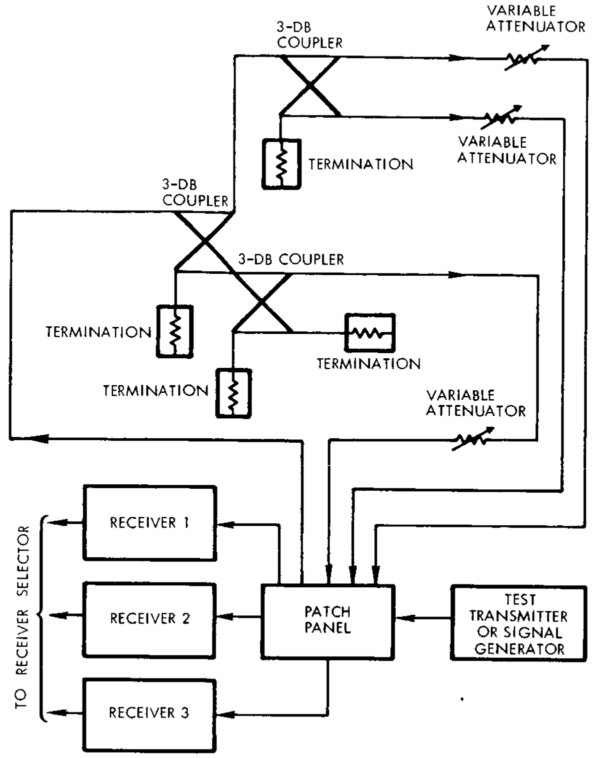


Figure 1. Receiver Excitation, Block Diagram

4.2.8 Reference (VCO) Signal Output When In Lock

The reference signal frequency when the signal level is sufficient to cause VCO lock is measured to an accuracy of 1 cps. The VCO output is specified as 0 ±3 dbm when in lock and below -30 dbm when out of lock. The in lock power and out of lock power levels are measured with ±3 percent accuracy.

4.2.9 Auxiliary Oscillator Frequency Stability

The short and long term stability is measured with an accuracy of 10 cps.

4.2.10 Special Test Equipment

The above tests may be accomplished with commercial test equipment. The UTS also contains custom designed equipment to perform the following specific tests on the receiver:

a) Static VCO frequency lock in range
b) Dynamic VCO frequency lock in range
c) Noise bandwidth with signal level
d) Phase modulation response
e) Static phase error
f) Dynamic phase error
g) Video response
h) Output limiting
i) Command output linearity.

Limits and measurement accuracy to be specified

4.3 Receiver Selector

The input stimuli to the receiver selector are obtained from a specifically designed test set and mounted in the UTS. The output signals from the selector are evaluated utilizing commercial test equipment. A block diagram of the test configuration for this article is shown in Figure 2.

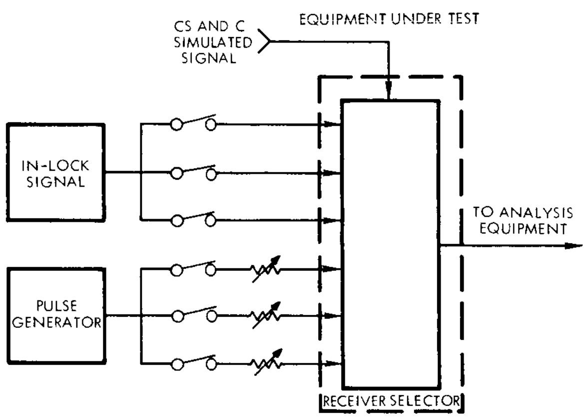


Figure 2. Receiver-Selector Checkout Configuration

The following tests may be performed on the receiver selector:

a) Selection modes test

b) CS & C command response

c) Immunity to line noise in CS & C channel

d) Input and output impedance of all channels.

4.4 Modulator-Exciter

To provide a complete test of modulator-exciter performance, the UTS performs the tests specified in the following paragraphs.

4.4.1 Modulation Index

The modulation index is specified as 0 to ±1.5 radians. The test set performs this measurement from 0 to 4 radians. The accuracy of measurement over the specified range is 5 percent or better.

4.4.2 Modulation Bandwidth

The modulation bandwidth is specified as 10 cps to 1.5 Mc or more with an accuracy of 5 percent or better.

4.4.3 Output Frequency

The nominal output frequency is specified as 2295 ± 5 Mc. The test set is capable of measuring this frequency with an accuracy of ±1 KC.

4.4.4 Output Bandwidth

The output bandwidth is specified as 3.5 Mc to the 1.0 db points. The test set is capable of measuring this quantity over a minimum of 5 Mc and to an accuracy of 5 percent or better.

4.4.5 Output Power

The output power is specified as 100 mv minimum into a 50 ohm load (VSWR $\geq$ 1.6). The test is capable of measuring power from 10 μw to 40 watts; within the performance range the accuracy is ±0.25 db.

4.4.6 Spurious Outputs

The test set is capable of measuring harmonically and non-harmonically related spurious signals to at least the fifth harmonic of the output frequencies over a dynamic range of 60 db or better. (Design specifications are 50 db below output signal.)

4.4.7 Special Test Equipment

The tests in sections 4.4.1 through 4.4.6 may be performed with commercial test equipment, employing the special test receiver and transmitter which is designed specifically for the UTS. The following tests may be performed with the use of the special test equipment:

a) Phase control

b) Phase stability

c) Linearity

d) Modulator sensitivity

e) Modulation stability.

4.5 Low Power Transmitter

The low power transmitter is similar to the modulator-exciter chassis, with the exception of an output power of one watt. The types of tests for the unit are identical to those shown in section 4.4.

4.6 Power Amplifiers

The communications and data handling system transmitter has two amplifiers, both units having an output power of 20 watts.

The UTS contains a complete set of test equipment to test the power amplifiers.

4.6.1 Saturated Gain

The saturated gain and 20 watt units are specified as 33 db ±1.5 db and 30 db ±1.5 db. The UTS is capable of performing this test over a gain range of 0 to 60 db with an accuracy of ±0.5 db.

4.6.2 Power Output

The output powers are specified as ±0.5 db of the nominal values previously stated. The test set measures output power from 0 to 50 watts with an accuracy of ±0.25.

4.6.3 Power Consumption Tests

To provide power consumption and tube power supply regulation data the UTS is capable of monitoring the special power supply of the power-output device. The following elements will be checked:

a) Helix voltage

b) Helix current

c) Collector current

d) Collector temperature.

Measurements (a) through (c) are made by employing very high precision high impedance resistor dividers and a six-digit digital voltmeter. The overall accuracy of DC power parameters is ±0.1 percent or better.

4.6.4 Noise Figure

The PA noise figure is specified as 30 db. The test noise generator is calibrated to ±0.1 db. and noise figure measurement is accurate to ±0.25 db over a noise figure range of 7-13 db.

4.6.5 Spurious Output

The PA design specification limits harmonic and non-harmonic spurious outputs to 45 db below nominal output power. The UTS measures over a dynamic range of 60 db.

4.7 Transmitter Selector

4.7.1 To provide a complete test of selector performance, the UTS performs the following types of tests (performance values and tolerances are not given):

a) Response to amplitude of CS & C signals

b) Amplitude of output signals

c) Input and output impedance

d) Logic response to power monitor input signals

e) Immunity to line bounce noise signals.

4.8 Four-Port Circulator

To provide a complete test of circulator performance the UTS performs the following types of tests:

a) VSWR

b) Insertion loss

c) Isolation or directivity

d) Power split ratio.

5. FUNCTIONAL DESCRIPTION

The UTS contains standard commercial test equipment and special test equipment specifically designed for testing the Voyager communications subsystem. The equipment is configured to provide hardline access for end-to-end test modes and a central patch/control panel for all interconnections at tests lower than subsystem levels.

Functionally, the UTS is an integral combination of rack-mounted special and commercial test equipments, test harnesses, and equipment necessary to complete the subsystem loops.

A functional block diagram of the S-band communications UTS is included as Figure 3. Only special test equipments and major data paths are shown; the patch panel configuration represents functional data paths for levels of testing now shown. A block diagram of the special test transponder test equipment is shown in Figure 4.

The physical configuration of the rack-mounted equipments contained in the UTS is shown in Figure 5. Consisting of seven racks, the UTS is functionally arranged to operate on an integrated level and, supplemented by a minimum of standard test equipment, provides detached units for parallel test programs.

The rack layout is composed of one double unit and five single units mounted on casters, with full length rear door access for maintainability purposes. Peripheral equipment includes a bench-mounted junction box.

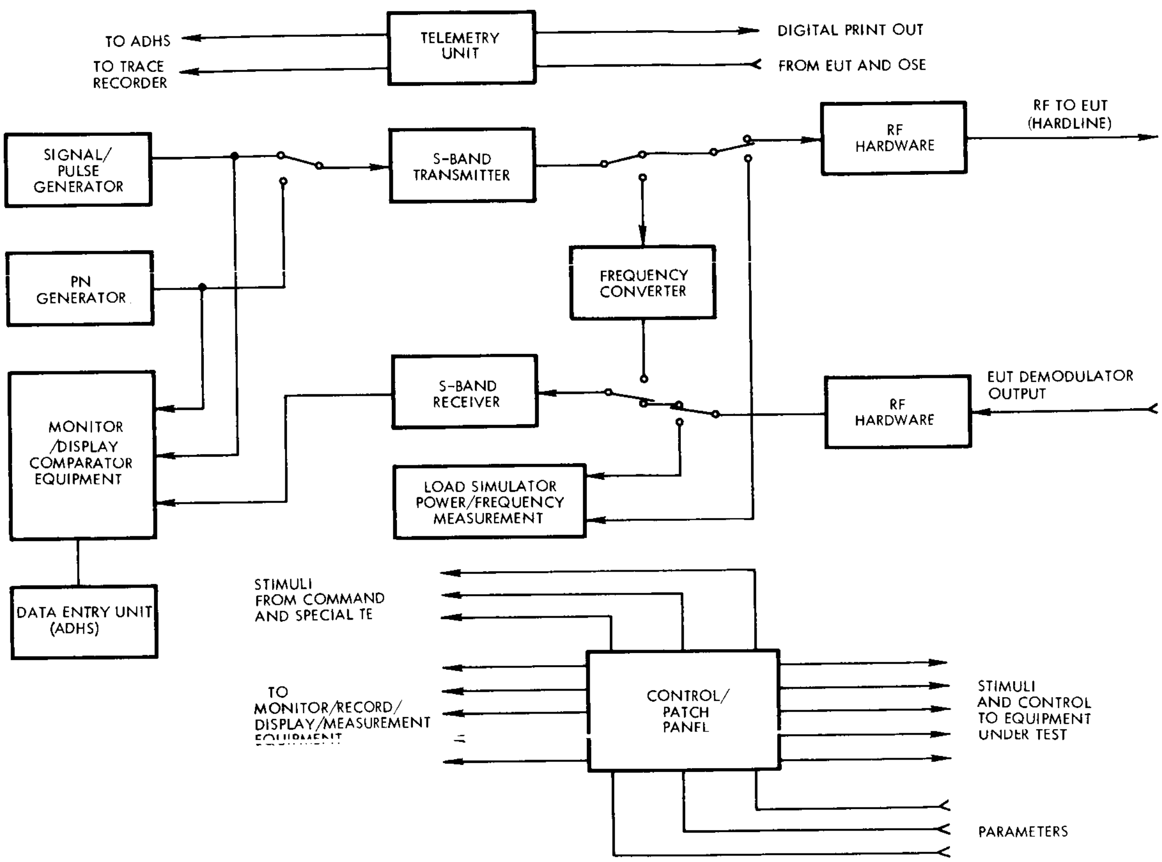


Figure 3. S-Band Communications Unit Test Set, Block Diagram

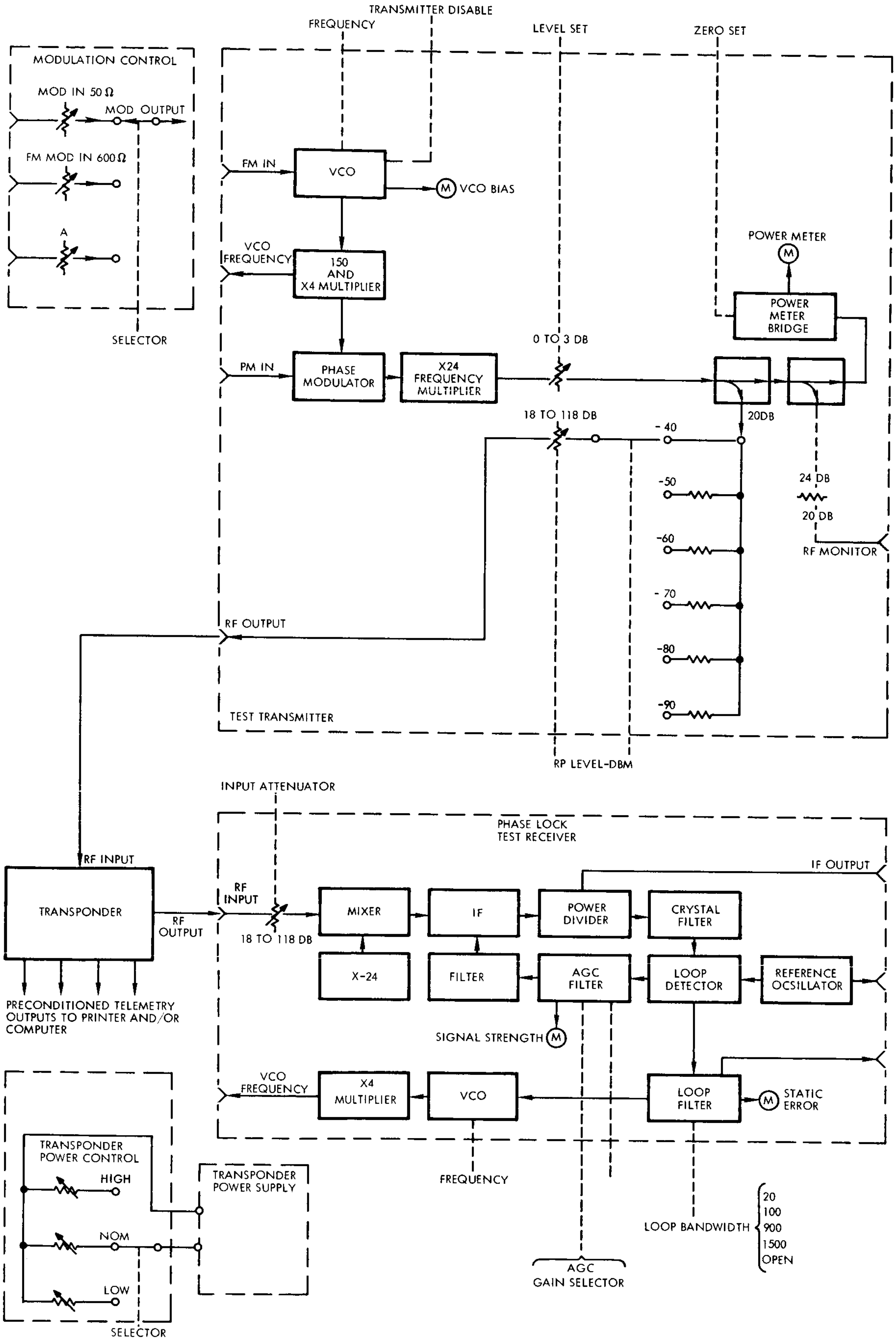


Figure 4. Special Test Transponder Test Equipment

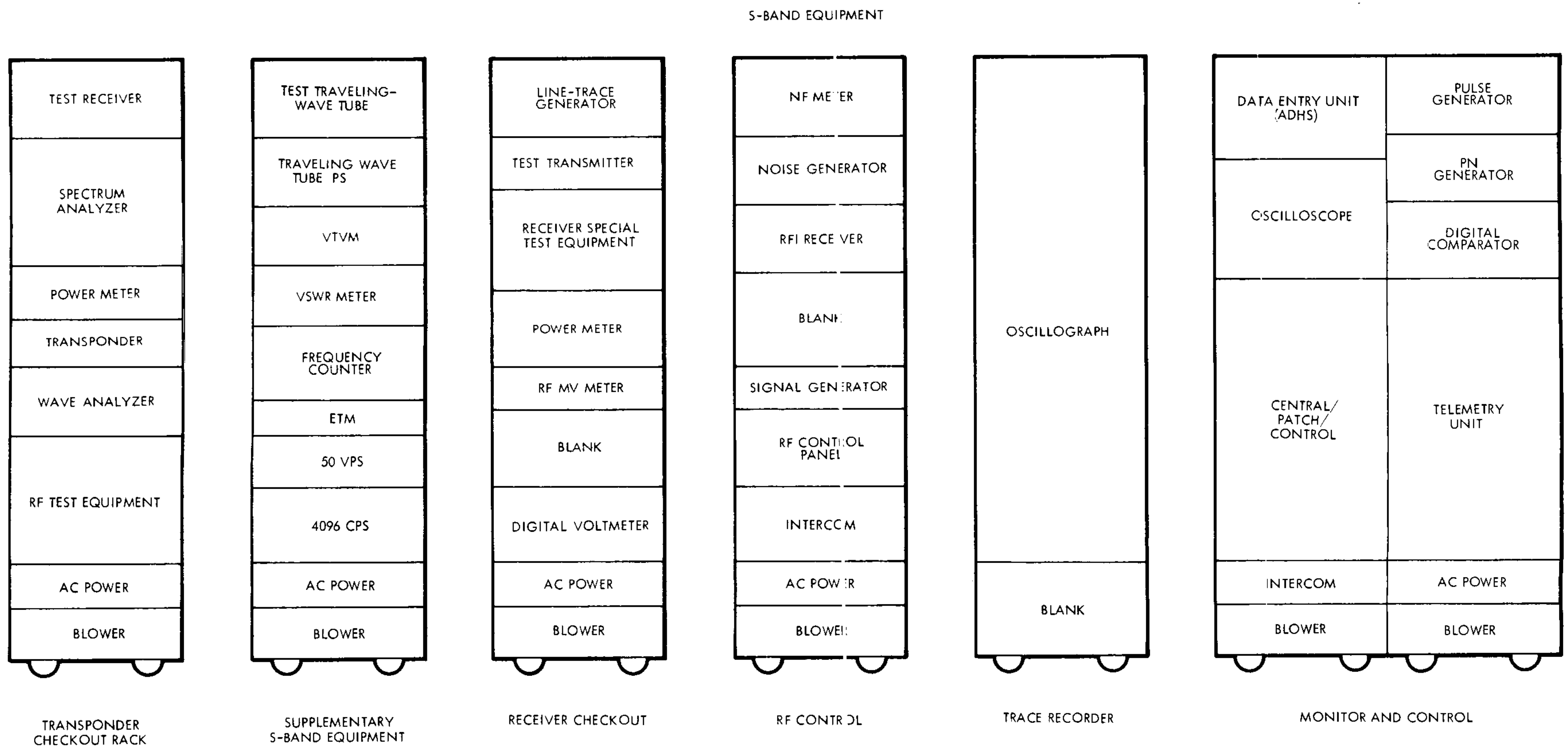


Figure 5. S-Band Communications Unit Test Set, Rack Layout

5.1 Functional Breakdown

5.1.1 Interface Equipment

The interface equipment of the VHF communications UTS consist of the junction box, simulated flight harnesses, and test cables and connectors. This equipment is detailed in Section 5.2 of this document.

5.1.2 Rack-Mounted Equipment

a. Transmitter Checkout Rack

The equipment contained in this cabinet is that special and standard commercial equipments used to perform basic tests on the spacecraft transmitter section. The major components are as follows.

Test Receiver. This unit provides the tool for evaluating the transmitter output (spacecraft and UTS) in the end-to-end loop analysis and, as such, is designed to minimize distortion of the transmitted signal characteristics.

Spectrum Analyzer, Wave Analyzer. Used in conjunction with other RF characteristic measurement equipments, this equipment provides analysis of the S-band signal for spurious and harmonic content.

Transponder Test Equipment. This is special test equipment as shown in Figure 4.

RF Hardware. The miscellaneous coupling devices, attenuators, and dummy loads, considered a permanent part of the test configuration, are mounted in this cabinet space.

b. Receiver Checkout Rack

This equipment is designed to provide a basic analysis of the spacecraft receiver section. Used in conjunction with the equipment contained in the supplementary S-band equipment rack and RF control rack, this equipment is capable of all receiver testing as specified in Section 4.2 of this document. The main components in the receiver checkout rack are as follows.

Test Transmitter. This transmitter is a specially designed piece of test equipment capable of transmitting a signal of minimum distortion to the spacecraft receiver and, using a frequency converter, to the UTS receiver. The test transmitter is used for all end-to-end loop evaluations.

Receiver Special Test Equipment. This equipment is illustrated in Figures 1 and 2 and is used to provide the proper receiver excitation and selection modes.

Power Meters. The power meter and RF milliwattmeter are used to measure UTS power output to the spacecraft receiver in addition to measuring spacecraft transmitter signal levels.

RF Hardware. This hardware consists of the necessary coupling devices, attenuators, dummy loads and power limiting devices considered as permanent connections in a test configuration.

c. RF Control Rack

This cabinet contains the required input stimuli, other than simulated signals, used to test the spacecraft communications equipment. RF patching for a given test mode is a function of this rack. The major components in the RF control rack are:

Noise Generator, Noise Figure Meter. These items are used to make accurate NF measurements of the applicable S-band equipment.

RF Control Panel. This patch panel configuration is utilized to make RF signal connections applicable to a given test mode. The rack space contains the facility for connections to hardware (couplers, dummy loads, etc.) as contained in the RF hardware portion of the cabinet.

d. Supplementary S-Band Equipment

The special and commercial test equipment included in this cabinet complement the transmitter and receiver checkout equipment to perform all levels of test described in section 4. The main equipment is as follows.

Test Traveling Wave Tube (TWT). This standard piece of test equipment provides the necessary UTS output power to measure the saturated characteristics of the flight TWTA. The test TWT power supply is included.

Flight Equipment Power Supplies. A 4096 cps power supply and a 50 VDC power supply (for the TWTA) are supplied along with accurate power metering. Elapsed time meters (ETM) record operating times based on input power application. The power supplies are appropriately interlocked to provide spacecraft protection.

Standard Test Equipment. The remainder of the cabinet includes the necessary test equipments to support testing of receiver and transmitter performance.

e. Trace Recorder

A multi-channel trace recorder provides permanent recordings and pictorial representations of the various waveforms and monitor points required in a given test mode. Having a 0-5 KC bandwidth, the recorder provides real time display for operator analysis.

f. Monitor and Control

This unit provides the central switching and control functions for signal routing, test initiation and conduction, and parameter monitoring. The following are included as major components.

Control/Patch Panel. This segment of the M&C unit contains the complete control switching and a patch panel configuration for signal routing and parameter monitoring. Terminations and coupling connections are made with equipment located in the space behind the front panels.

Monitor Oscilloscope. This standard commercial oscilloscope (RM45A or equivalent) serves as a signal monitor for analysis and fault isolation purposes. Selectable inputs are obtained from the monitor outputs of the patch panel.

T/M Unit. The T/M unit provides the scanning and digital display (print-out) of the hardline telemetry and direct access monitoring signals emanating in the spacecraft and OSE.

ADHS Buffering. A pre-programmed sampling of test stimuli, results, and operational parameters will be converted to ADHS format and interim storage provided for in the buffering unit. Transfer to computer input device will be manually initiated.

Pulse Generator, PN Generator, Digital Comparator. These devices provide the spacecraft signal simulation for system input-output comparison and error rate detection.

5.1.3 Major Commercial Test Equipments

Type	Typical Model
Signal generator	HP 8614A
Counter	HP 5245L
Counter plug-ins	HP 5253B, HP 5254A, HP 5262A
3db Couplers	Nardu 3033
Oscilloscope	HP 175A
Scope plug-ins	HP 1754A, HP 1752A
Spectrum analyzer	HP 851A plus 8551A RF Head
RF Millivoltmeter	HP 411A
Power meter	HP 431B
Calorimetric power meter	HP 434A
Slotted line	Norder 231N
Slotted line probe	Narda 229
Directional coupler	Narda 3003-30
Signal sampler	Midrolab HM-30N
VTVM	HP 400LR
Sampling voltmeter	HP 3406A
2 Watt test TWT and power supply	Alfred Model 561
Noise figure meter	HP 340B
S-band noise source	HP 349A
VSWR meter	Narda 441C
RFI receiver	Polarad Model CFI
Wave form generator	HP 3300A (3301A Plug-In)

Analyzer	HP 310A
Digital voltmeter	HP 3460A
Dummy Loads:	
Microlab (2)	TD5FN 900-12GC 100 watt
Microlab (3) DC-10GC 200 watt	TA5FB
Variable electronic filter	SKL 302
	Telonic SM-2000 L-6 Plug-In
	Telonic Series TRB
T/M Readout:	
Input scanner programmer	Dymec 2901A
Slave scanner	Dymec 2902A
Digital recorder	HP 562AR
Recorder jackfield panel	Trim 1024040
Oscillograph	Sanborn Series 4500
Attenuator transfer standard	HP S302B
Attenuators	HP 355C and 393A
Elapsed time meters	

5.1.4 Special Test Equipment

P. N. generator

S-band test transmitter

S-band test receiver

4096 cps power supply

Line transient generator

Test cables

Junction box

Digital comparator

50 VDC power supply

5.2 Test Configuration

The relationship of types of UTS equipment, test facilities, and equipment under test (EUT) is shown in Figure 6. Types of UTS equipment involved are as follows.

5.2.1 Simulated Flight Equipment

Test harnesses, dummy loads, etc., are provided to test integrated subsystems or subassemblies. Test harnesses for this purpose are designed to closely simulate the actual flight harness; breakout points are included in the test harness with sufficient isolation to prevent influencing system operation. All equipment in this category are qualified to flight specifications.

5.2.2 Junction Box and Test Cables

The junction box is designed to be included in all phases of testing. Comprising the junction box are terminations, impedance matching devices, line drivers, and signal paths required to test and evaluate the EUT. In this capacity the junction box is required to operate in close proximity to the EUT. Test cables route the signals from the "black box" T/M points and direct access monitor points to the junction box. This equipment is qualified to flight specifications.

5.2.3 External Cabling

The UTS supplies all external cabling to the EUT. Interfacing with the junction box at a test connector, this cabling is designed to operate over the OSE operating range.

5.2.4 Rack-Mounted Equipment

This equipment is described in section 5.1.

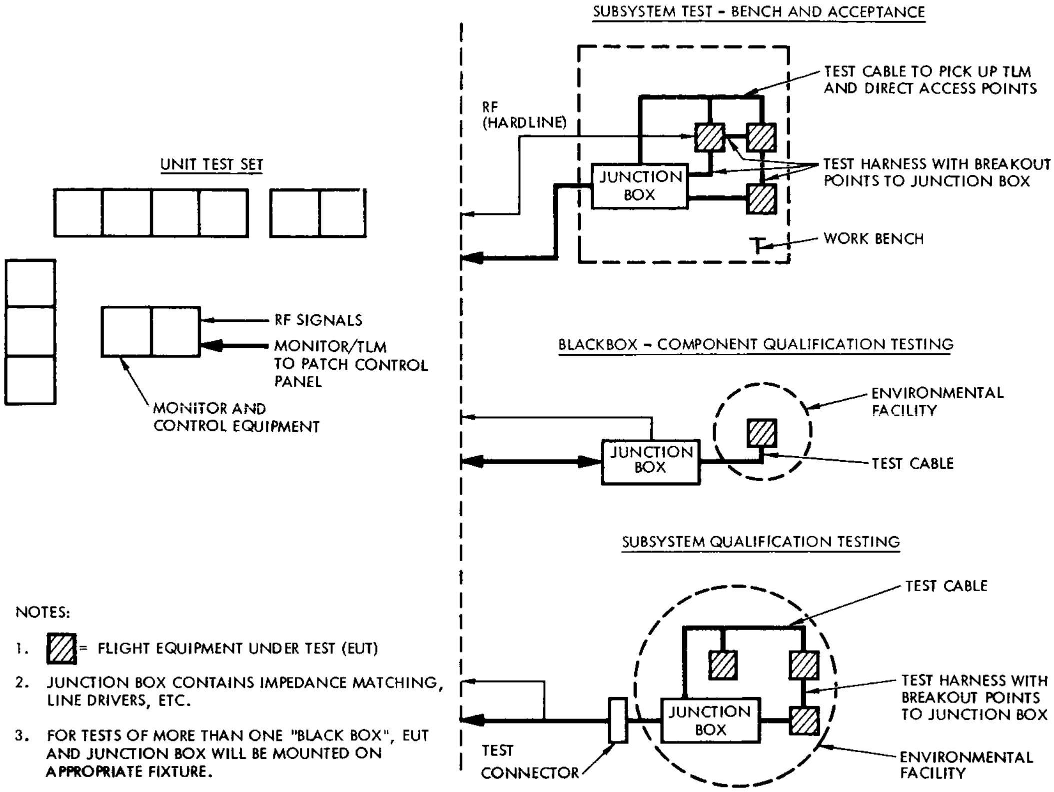


Figure 6. Typical Test Configuration

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The UTS operates from a power source as specified below:

Voltage	115 ± 10 vac
Frequency	60 ± 1 cps
Phase	Single

6.2 Service Environment

The UTS rack mounted equipment is designed to operate between 50 to 95°F. The equipment described in Section 5.1.1 operates satisfactorily over the range of environmental conditions specified for corresponding flight equipment.

6.3 Electrical Environment

The UTS configuration incorporates selected components and shielding techniques to minimization of RF radiation.

7. CONSTRAINTS

The UTS is designed to operate by manually introducing the input stimuli and transcribing the recorded output signals. Telemetry signals are automatically indexed by the scanning equipment and printed outputs available to the operator. Simultaneously, the telemetry signals are available in BCD format for transfer to remote processing.

Portions of the UTS which are in close proximity to flight equipment under test do not influence significantly the magnetic field measurements being made. All UTS equipment required to accompany flight equipment in environments other than ambients is designed and qualified accordingly.

8. INTERFACES

The following equipment is interfaced with the communications UTS:

a) Receivers

b) Receiver selector

c) Command detectors

d) Modulator exciters

e) Low power transmitter

f) Four-port circulator

g) Power amplifiers and supplies.

VHF COMMUNICATIONS UTS
OSE/VS-4-311-2

1. SCOPE

This document defines the functional requirements for the unit test set (UTS) used to test and evaluate the VHF communications equipment of the Voyager spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints
OSE/VS-4-310	Voyager Communications and Data Handling Subsystem
OSE/VS-3-120	OSE Automatic Data Handling System

3. FUNCTIONAL REQUIREMENTS

3.1 General

The UTS is designed to test and evaluate the Voyager VHF communication subsystem down to the "black box" level of a given subassembly. In the Voyager configuration, the VHF communications link consists of:

a) VHF receiver

b) FSK demodulator

c) Bandpass filter

d) VHF pre-amplifier.

3.2 Required Functions

3.2.1 Over-all

Over-all required functions of the VHF communications UTS are to provide:

a) Input power to equipment under test (EUT)

b) Input power measurement and recording

c) Stimulus, data paths, and analyzing equipment for all test modes

d) Monitoring and recording facilities for T/M data, direct access points and parameters being evaluated

e) Necessary interface equipment and test harnesses

f) Protection for EUT

g) OSE self-check and fault isolation capabilities

h) Adaptability with automatic data handling system (ADHS)

i) Additionally, the UHF communications UTS exercises system components over a test specified operating range.

3.2.2 VHF Communications Subsystem

The VHF communications subsystem provides the following:

a) Stable VHF transmitter for evaluation of flight receiver performance

b) VHF receiver for confirmation of UTS operational status (self-check)

c) Display and recording for waveform analysis.

It also generates a bit stream for end-to-end loop evaluation and supplies test equipment for measurement of subassembly characteristics.

4. DESIGN REQUIREMENTS

The UTS is designed to verify that the operation of the equipment under test is within design specified limits. Included in the UTS are special and commercial test equipments designed to give the degree of accuracy needed to establish confidence in EUT test results.

4.1 End-to-End Loop Analysis

A simulated capsule T/M (10 bit/sec) bit stream is generated in the UTS; verification of output is provided automatically utilizing the UTS test receiver and demodulator. The bit stream is analyzed at the output of the FSK demodulator flight equipment. A digital comparator and trace recorder comparison of output versus input signals determines error rate and permits waveform analysis to be performed.

4.2 Bandpass Filter

4.2.1 Bandwidth

The bandpass filter is required to have a 3db BW of 2MC (136-138MC nominal). The UTS performs this measurement with sweeping capabilities of 0.5 M to 10 MC or better. The accuracy of measurement is 1KC or better.

4.2.2 Insertion Loss

The insertion loss of the BPF is specified as 0.2db maximum. The UTS is capable of making this measurement over the range of 0.1 to 3db with an accuracy of 0.05db or better.

4.3 Pre-Amplifier

4.3.1 Bandwidth

The bandwidth of the pre-amplifier is specified as 1MC at the 3db points. The UTS measures to the accuracy given in 4.2.1.

4.3.2 Noise Figure

The noise figure of the pre-amplifier is specified as 3.5db maximum. The UTS performs this test over a range of 2 to 10db with an accuracy of 0.25db or better.

4.3.3 Gain

The gain of the pre-amp is specified as 10db minimum. The UTS performs this test over the range of 0 to 60db with an accuracy of 0.5db or better.

4.4 Receiver

4.4.1 IF Bandwidth

The bandwidth of the receiver is specified as 44KC. The UTS measures this parameter with sweeping capabilities of 1 to 100KC. The accuracy of measurement will be 1KC or better.

4.4.2 Local Oscillator Stability

The stability of the receiver LO is specified as 3×10^{-5}. The UTS performs this test with an accuracy of 1×10^{-7} or better.

4.5 FSK Demodulator

4.5.1 Channel Frequencies

The FSK data channel is centered at 10MC. The "mark" channel fc is specified as 10.011MC and the "space" channel fc is specified as 9.989MC. The UTS performs tests which resolve these channels with an accuracy of ±100 cps.

4.5.2 Error Rate

The error rate of the demodulator is specified as 10^{-3} at a C/N of -14 db in a 44KC BW. The UTS tests for error rates of 10^{-5} or less with accuracies of within 1 percent of the stated value.

5. FUNCTIONAL DESCRIPTION

The VHF communications UTS contains standard commercial and special test equipment specifically designed for testing the Voyager VHF communications package. The equipment is configured to provide hard-line access for all end-to-end loop measurements and a central patch/control panel for all interconnections at test levels lower than subsystem.

Physically, the integrated UTS includes:

a) Rack-mounted special and commercial test equipment

b) Test harnesses, both simulated flight and test point monitoring

c) Interface equipment both in the flight configuration and the EUT-OSE area.

A functional block diagram of the VHF communications UTS is shown as Figure 1. Only special test equipments and the prime input-output data paths are shown. The block representation of patch panel interconnections refer to all sublevels of test.

The physical configuration of the UTS test layout detailing the rack-mounted equipment locations, is shown in Figure 2. Consisting of five racks, three single and one double unit, the UTS is functionally arranged so that with a minimum of supplementary standard test equipment detached units may be operated to permit parallel testing of the subassemblies comprising the VHF communications link.

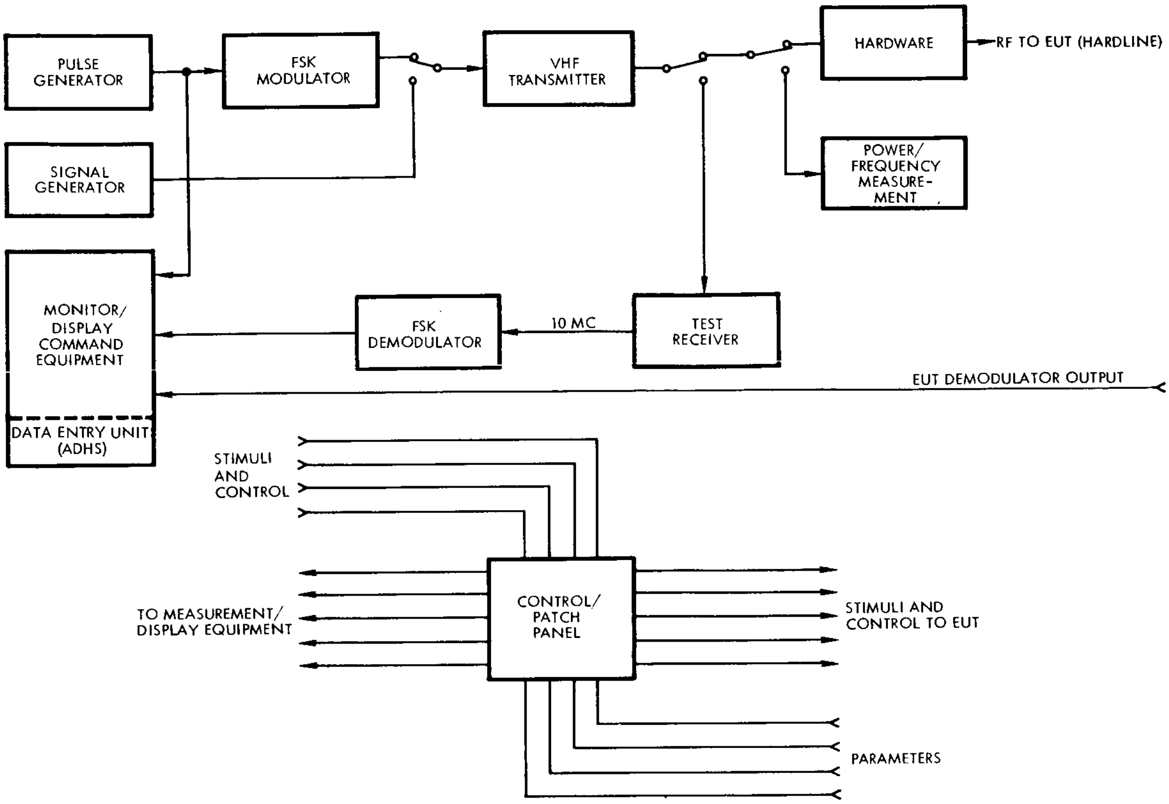


Figure 1. UHF Communications Unit Test Set, Block Diagram

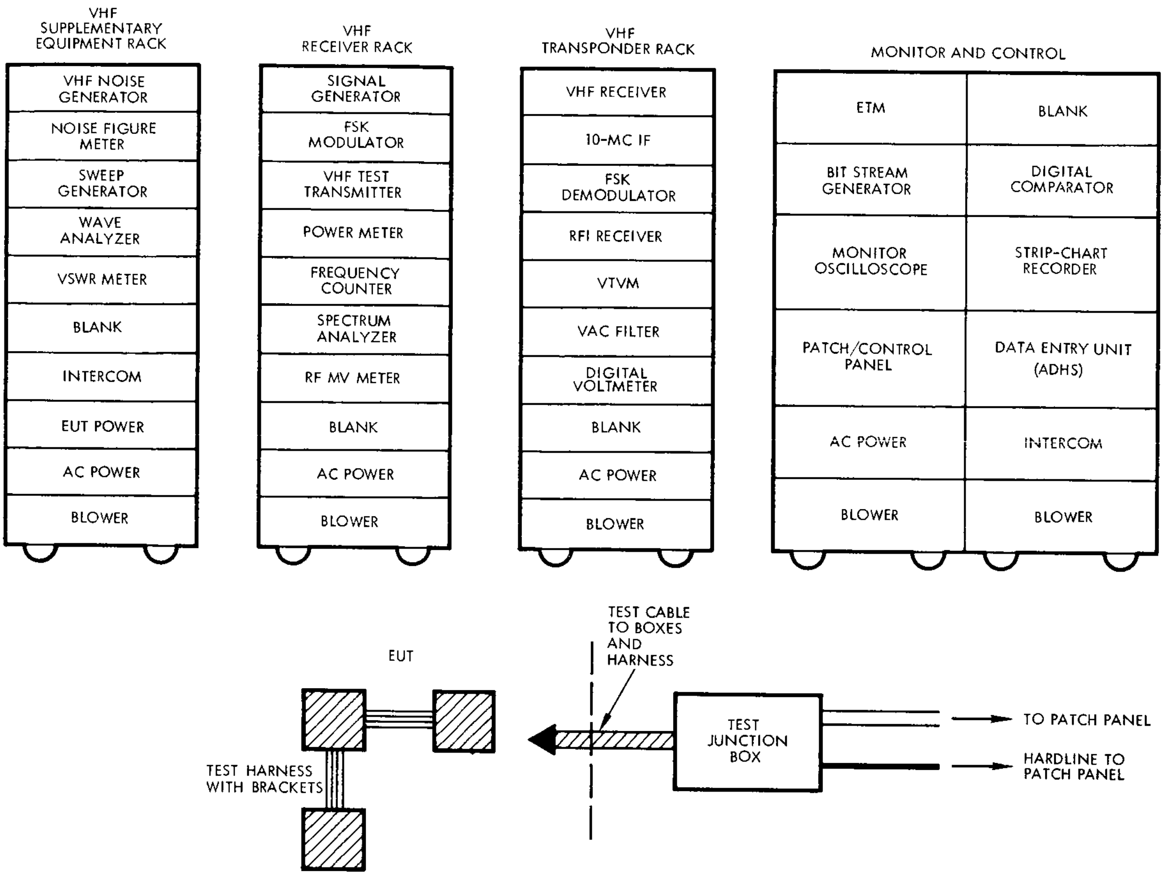


Figure 2. UHF Communications Unit Test Set, Rack Layout

5.1 Functional Breakdown

5.1.1 Interface Equipment

a. Junction Box

The junction box provides both active and passive elements to provide terminations, loading, impedance matching, line drivers, and signal paths. All signals to the equipment under test are routed through the junction box.

For proper performance, the junction box is located in close proximity to the EUT. As such, the unit is designed and qualified to flight standards for operation in environments other than standard.

b. Test Cables

To provide the evaluation data, T/M points, direct access points, and pertinent signals, both from inter-box harness breakouts and from the equipment directly, are routed by the test cable to the junction box.

The test cable is designed and qualified to adhere to flight specifications for operation at non-ambient environmental conditions.

c. Test Harness

The interwiring of the EUT should be physically and electrically identical to the flight harness within limits imposed by requiring access to signals within the harness configuration. The test harness is designed and qualified to flight environmental specifications and, as nearly as possible, to flight configuration. Sufficient isolation is provided at the breakout points to insure against disturbing the electrical characteristics at that point.

5.1.2 Rack-Mounted Equipment

a. VHF Receiver Rack

Test equipments in this portion of the UTS provide the RF input stimuli and measurement equipment to check the RF performance of the flight receiver. The main components are as follows.

FSK Modulator. Electrically identical with the capsule flight modulator, this unit is designed to provide the FSK modulated 10MC IF signal to the test transmitter.

VHF Test Transmitter. This transmitter is a piece of special test equipment designed to electrically simulate the capsule transmitter. The output of the test transmitter is nominally 136 to 138 MC.

RF Measurement Equipment. These items are designed to measure, with the required accuracy, the RF characteristics of the VHF receiver.

b. VHF Transmitter Rack

The VHF transmitter rack contains the equipment necessary to verify operation and accuracy of the UTS generated test signals used to evaluate the EUT. The main components are as follows.

VHF Receiver. Encompassing the design characteristics of the flight reciever, the UTS receiver accepts the transmitter output (136 to 138MC) and converts it to the 10MC IF signal.

FSK Demodulator. Electrically identical with the flight demodulator, the FSK demodulator provides the channel filtering (10.011 and 9.989 MC) and the envelope detectors required to produce the reconstructed 10 bit/sec capsule data.

RF Equipment. This equipment consists of the necessary hardware and RF measuring equipment of the required accuracy to make the specific measurements given in Section 4.

c. VHF Supplementary Equipment

This rack contains the remainder of the commercial test equipment required to test the VHF communications RF characteristics.

d. Monitor and Control

The functions of the various equipment in the monitor and control unit of the VHF communications UTS are to provide the following:

a) Simulated capsule signals and analysis and recording equipment for end-to-end loop analysis

b) Permanent records of parameters

c) Patching and control for all test modes

d) Provision for transfer of data to ADHS.

The major equipment included in the monitor and control cabinets is as follows.

Bit Stream Generator. This equipment produces the simulate capsule data bit stream, a 10 bit/sec NRZ code.

Digital Comparator. This item provides bit by bit comparison of input code versus OSE and EUT demodulated signal. An events count will be provided to determine loop error rates.

Strip Chart Recorder. A multi-channel hot stylus trace recorder is provided to provide a record of input and output digital signals for waveform analysis and permanent records. Continuous monitoring of selected parameters will be performed.

Monitor Oscilloscope. The oscilloscope is used for monitoring and fault analysis purposes. Inputs are selected on patch panel.

Patch/Control. All test signal paths are routed through the patch/control panel where the test configuration is determined. Monitoring points are provided at all connection points for removal of data. The patch panel provides the necessary isolation and terminations for signal paths; coupling devices mounted in the rear of the panel area provide interface with various RF equipments.

ADHS Buffering. Stimuli, selected monitor points, and acquired data are converted to a form compatible with the ADHS input requirements and stored in the ADHS buffering unit; transfer of data is by command.

5.2 Specific Equipment Description

5.2.1 Major Commercial Test Equipment

Type	Typical Model
Counter	HP 5245L
Counter plug-in	HP 5254A
Oscilloscope	HP 175A
Scope plug-ins	HP 1754A, HP 1752A
Spectrum analyzer	HP 851A with 8551A Head
RF millivoltmeter	HP 411A
Power meter	HP 431B

Type	Typical Model
VTVM	HP 400LR
Noise figure meter	HP 340B
VHF noise source	HP 343A
VSWR meter	Narda 441C
10MC IF	RIAG L1003
RFI receiver	Polaroid Model CFI
Signal generator	TF 144 H/4
Wave analyzer	HP 310A
Digital voltmeter	HP 3460A
Variable electronic filter	SKL 302
Sweep generator	Telonic SM-2000, L-6 Plug-in
VSWR detector	Telonic Series TRB
Trace recorder	Sanborn Multichannel

5.2.2 Special Test Equipment

VHF Test receiver
FSK demodulator
VHF test transmitter
Digital comparator
Data rate generator
Dummy loads
Calibrated attenuators

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The VHF communications UTS operates from a primary power source as specified below:

Voltage:	115 ±10V AC
Frequence:	60 ±1 cps
Phase:	Single

6.2 Service Environment

The rack mounted portions of the UTS operate satisfactorily at an ambient temperature of 50 to 95°F.

The junction box, test harnesses, and test cables as described in Section 5.1.1 operate satisfactorily over the range of environmental conditions as specified for the flight hardware.

7. CONSTRAINTS

There are no constraints on this UTS other than those already put forth in previous sections of this document.

8. INTERFACES

To be specified.

COMMAND DETECTOR UNIT TEST SET
OSE/VS-4-311-3

1. SCOPE

This document defines the functional requirements for the unit test set (UTS) required to evaluate the performance of the Voyager spacecraft command detector.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL DSIF PN System

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints
OSE/VS-4-310	Voyager Communications and Data Handling Subsystem
OSE/VS-3-130	Voyager Mission Dependent Equipment

3. FUNCTIONAL REQUIREMENTS

3.1 General

The command detector UTS provides the facility for testing and recording the performance of a complete command detector or any of its major submodules. The UTS also has a self-check capability for readiness and confidence checks and performs in conjunction with other UTS during integrated communications subsystems tests.

3.2 Required Functions

The command detector is required to provide the following:

a) Suitable input signals and output loads for both the command detector and its major submodules

b) Appropriate power for normal and marginal condition testing

c) Semi- or completely automatic verification of input-output operation, including a permanent record of both the input and output

d) Appropriate recording and monitoring equipment to perform normal testing as well as fault isolation down to the circuit board level

e) Means of checking out redundant subsystems individually as well as together, and any automatic switching associated with them

f) Adaptability with ADHS.

The UTS is suitably fused and isolated so that any failure of it will not cause any damage to a unit being tested.

4. DESIGN REQUIREMENTS

4.1 General

In order to perform the functions described in paragraphs 3.1 and 3.2, the UTS satisfies the design requirements outlined in the following paragraphs.

4.1.1 Input Signals

In the redundant configuration, the command detectors are selected by PN code format identification. The UTS provides multiple code formats, modulated by simulated command information, for automatic selection verification and input-output correlation checks.

4.1.2 Interfaces

The UTS provides simulated loads and source impedances for command detector checkout.

4.1.3 Input-Output Verification

The UTS provides automatic recording and indicating equipment for the purposes of verifying the output command bit stream against the input command bit stream (prior to PN encoding).

4.1.4 Junction Box

A junction box, containing both passive and active components, is provided to ensure proper impedance matching and to minimize signal degradation due to long cable lengths. The junction box is located as close to the command detector under test as possible. It is designed and qualified for all environmental conditions to which the command decoder will be exposed during tests (e. g., thermal/vacuum). In addition, the junction box does not interfere with EMI or magnetic tests.

The junction box also contains break-in monitoring points (properly isolated for protection) for use in sub-module testing and fault isolation.

4.2 Testing Requirements

4.2.1 Detector Selection

Proper command detector selection is verified by application of appropriate PN code format input signals.

4.2.2 VCO Null Frequency Measurement

The UTS measures the VCO null frequencies to an accuracy of ±0.1 cps. A record of the VCO versus time/temperature is utilized for calibration.

4.2.3 Lock-In

At a minimum SNR of 19.5 db/cps, the command detector is designed to have a 1/3 probability of lock-in within one PN cycle. Using the UTS test generator, this probability will be verified using a sufficient number of cycles.

4.2.4 Error Rate

With the command detector input SNR of 18.5 ±1 db/cps, a sufficient number of input-output samples are compared to establish compliance with the design specification of 10^{-5} Pe^{b}.

4.2.5 VCO Pull-In, Hold-In Ranges

At nominal SNR's, the operating ranges of both command detectors are measured within ±1 cps.

4.2.6 Lock at Varying SNR

The UTS records the lock-in profile at various SNR's at the nominal input frequencies.

4.2.7 Multi-Channel Recording

The UTS provides a recording device to facilitate signal comparison and waveform analysis to monitor the following functions:

a) Detector Internal Frequencies - Four internal signals to be monitored for frequency and phase relationship to each other and to UTS generated fs

b) PLL Error Voltage - Monitoring of VCO input with varying UTS generated subcarrier rates

c) PN Generator Output - Internally generated PN code monitored for waveform analysis and tested against input PN code using the input-output verification comparator

d) Internal Bit Sync - Measured to establish coherency with internal PN code

5. FUNCTIONAL DESCRIPTION

5.1 General

The UTS is comprised of standard and special test equipment housed in a standard rack. In addition, special test harnesses and a junction box are provided.

A functional block diagram of the UTS is shown in Figure 1.

The UTS rack layout is shown in Figure 2. The UTS consists of two racks, a junction box, and associated cables. A desk top surface is provided in front of the racks for mounting the command decoder and junction box during non-environmental testing. Each rack has a blower and AC power circuit breakers.

5.2 Chassis Description

5.2.1 PN Code Generator

The PN code generator provides the various PN code formats required to test the command decoders. It accepts a bi-phase one bit/ second command signal input, which bi-phase modulates the 511 bit/ second PN code output signal. It also supplies a synchronizing signal to the command signal generator.

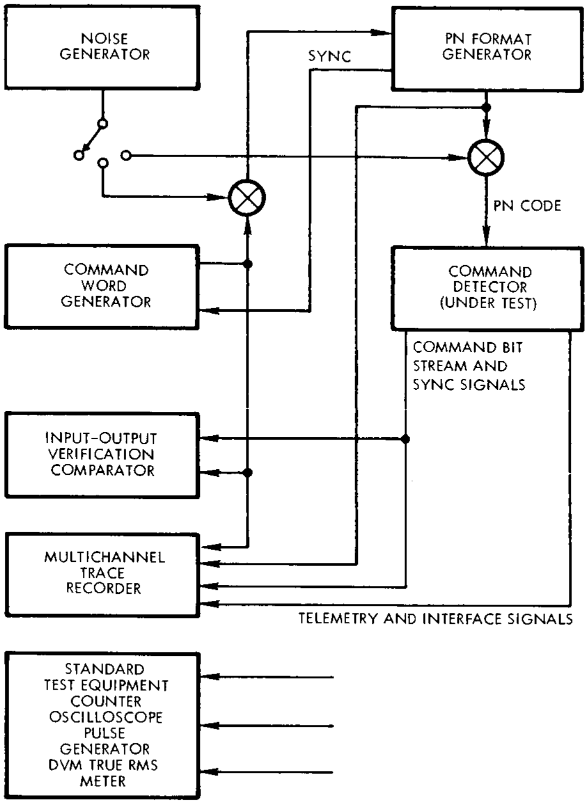


Figure 1. Command Detector Unit Test Set, Block Diagram

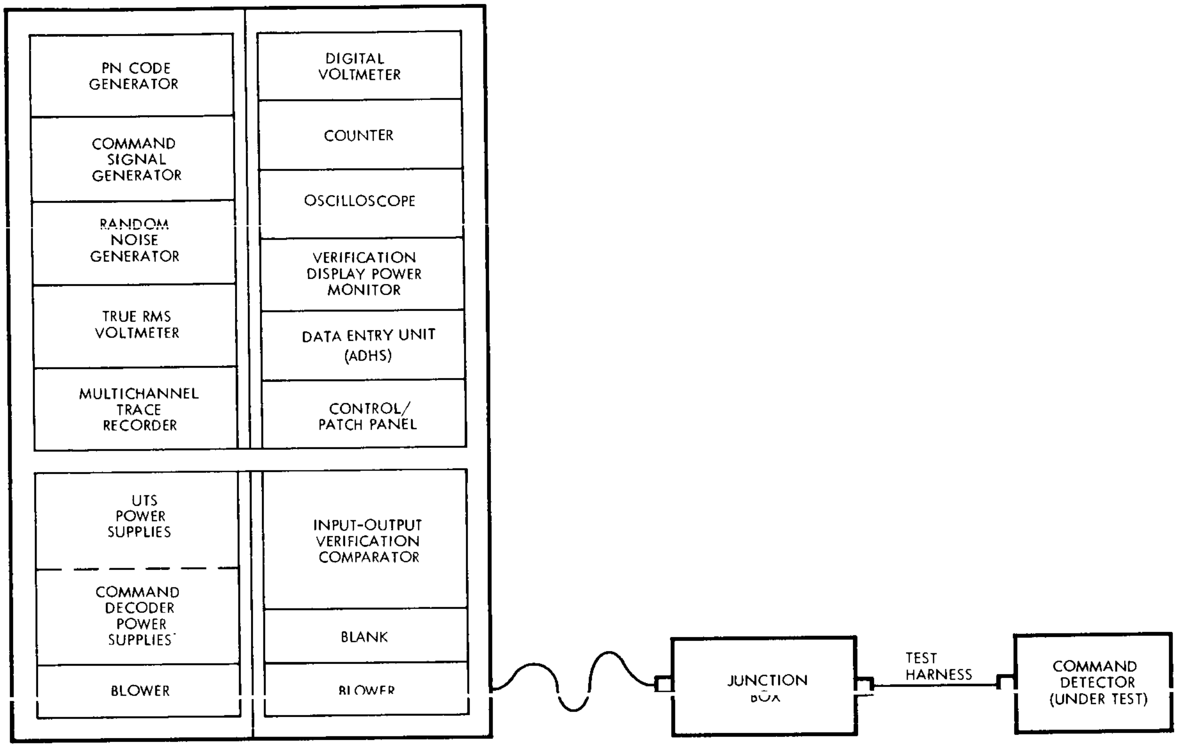


Figure 2. Command Detector Unit Test Set, Rack Layout

5.2.2 Command Signal Generator

The command signal generator generates a bi-phased one bit/second bit stream of at least 15 bits/cycle. The format of the bit stream is controlled by toggle switches, one for each bit in the cycle. The bit rate is synchronized by an external signal emanating from the PN code generator.

5.2.3 Random Noise Generator

A random noise generator is utilized during performance evaluation tests requiring specific S/N ratios for input signals. The noise introduction into the particular signal path is accomplished by components associated with a control panel located in the adjacent rack.

5.2.4 True RMS Voltmeter

This unit is required for making accurate SNR measurements when utilizing the random noise generator.

5.2.5 Multichannel Trace Recorder

This device simultaneously records significant input, output, test point, and telemetry waveforms associated with command detector testing. It contains at least eight channels and has a minimum bandwidth of DC to 5 KC (3 db) on at least two of the channels. The remaining channels need only have a DC to 150 cps frequency response.

5.2.6 Power Supplies

Power supplies (both DC and AC) are provided for operating the various chassis in the UTS as well as the command decoder. The command decoder power supply is a separate unit; however, it is safely interlocked with the UTS power supplies. In addition, it contains protective devices for over-voltage and current limiting.

5.2.7 Digital Voltmeter, Electronic Counter, Oscilloscope

These pieces of standard test equipment will be used for monitoring the various parameters described in previous sections, as well as for self-test of the UTS.

Access to the inputs of this gear are via a patch panel mounted in the same rack. A conventional oscilloscope probe is utilized when impedance or risetime measurements require it.

5.2.8 Verification Display and Power Mounter

This panel contains both the numerical display associated with the input-output verification comparator and the current and voltage monitoring meters associated with the power being supplied to the command detector. The power monitoring meters have an accuracy of ±2 percent of full scale reading. Test points are provided on the panel for obtaining readings of ±0.1 percent accuracy by using the digital voltmeter, when required.

5.2.9 ADHS Panel

Space is provided for an ADHS data input panel to be mounted in this rack.

5.2.10 Control/Patch Panel

This panel is comprised of those switches and controls necessary to program the UTS for the various test modes associated with testing the command detector. In addition, a patch panel is utilized for gaining access to the inputs and outputs of various test equipments in the UTS, and monitoring points of the unit under test (these include inputs, outputs, accessible monitoring prints, and telemetry points).

5.2.11 Input-Output Verification Comparator

The input-output verification comparator (IOVC) monitors the command bit stream emanating from the command signal generator and the command bit stream at the output of the command detector under test. It compares the two inputs and indicates the number of errors as well as the number of bits generated. It works in conjunction with a remote display mounted above in the same rack.

5.2.12 Junction Box

The junction box is designed to meet all of the requirements set forth in Section 4.1.4.

5.2.13 Cables and Test Harnesses

Cables and test harnesses are provided as required. The test harnesses coming into contact with the command detector will be fully flight qualified. The cables required to operate under extreme environmental conditions (other than ambient) will be qualification tested for these environments.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The command detector UTS operates from the following power source:

a) Voltage: 115 ±10 VAC

b) Frequency: 60 ±1 CPS

c) Phase: Single

d) Power: 5 KW

6.2 Operational Environment

The rack mounted portions of the UTS operate satisfactorily within the following ambient environmental conditions:

a) Temperature: 50 to 95°F

b) Relative Humidity: 0 to 60 percent

The junction box and test harnesses operate satisfactorily in the test environments imposed on the unit under test when they must accompany the unit in the environment.

7. PARAMETERS

The command detector measures and/or records the following parameters and operational characteristics of the command detector:

a) Detector selection

b) VCO frequencies

c) Lock-in time and range

d) Error rate, input versus output

e) Noise susceptibility

8. CONSTRAINTS

There are no constraints on the UTS other than those already put forth in previous sections of this document.

9. INTERFACES

To be specified.

DATA HANDLING UNIT TEST SET
OSE/VS-4-311-4

1. SCOPE

This document establishes the requirements for the data handling unit test set (UTS) used to evaluate the down-link data gathering and formatting system.

2. APPLICABLE DOCUMENTS

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110 OSE Mission Objectives and Criteria

OSE/VS-2-110 OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 General

The data handling unit test set is used to verify proper operation of the following:

a) Collection of data

b) Storage of data

c) Data formats

d) Internal power supply operation

e) Subunit operation.

3.2 Test Functions

The UTS individually tests the data storage unit and the bulk memory unit. The digital telemetry unit is tested when interconnected with the data storage unit and the bulk memory unit. The following portions of the data handling unit will be tested:

a) Data storage unit

b) Bulk memory unit

c) Digital telemetry unit (includes format control and subcarrier modulation operations)

d) Internal power supply operation.

4. DESIGN REQUIREMENTS

4.1 Data Storage Unit

To test the data storage unit, worst case patterns of data are stored and checked for fidelity. Manual entry and retrieval of data from any storage location will be possible.

4.2 Bulk Storage Unit

To test the bulk storage unit, a pseudo random code is recorded and played back for fidelity. Automatic error detection is included.

4.3 Digital Telemetry Unit

The digital telemetry unit is tested with a data storage unit and a bulk storage unit. Testing includes commanding format and data rates with the various data inputs programmed according to the desire of the experimenter. Worst case patterns are devised and used for testing.

4.4 Internal Power Supply Operation

Internal power supplies are monitored to assure proper operation.

4.5 Signal Levels

Signal levels, logic levels, etc., are monitored. Inputs are varied between specification limits for worst case testing.

5. FUNCTIONAL DESCRIPTION

5.1 General

The test set comprises standard commercial test equipment and specific equipment designed for testing functions peculiar to Voyager.

The test set is used for three specific tests. Figure 1 is a functional block diagram of a data storage unit test and Figure 2 is a functional block diagram of the bulk storage unit test. The largest test, or the one that uses the most equipment, is the digital telemetry unit test whose functional block diagram appears in Figure 3.

Figure 4 shows the three rack physical configuration of the data handling unit test set.

Figure 1. Memory Unit Test Set, Block Diagram

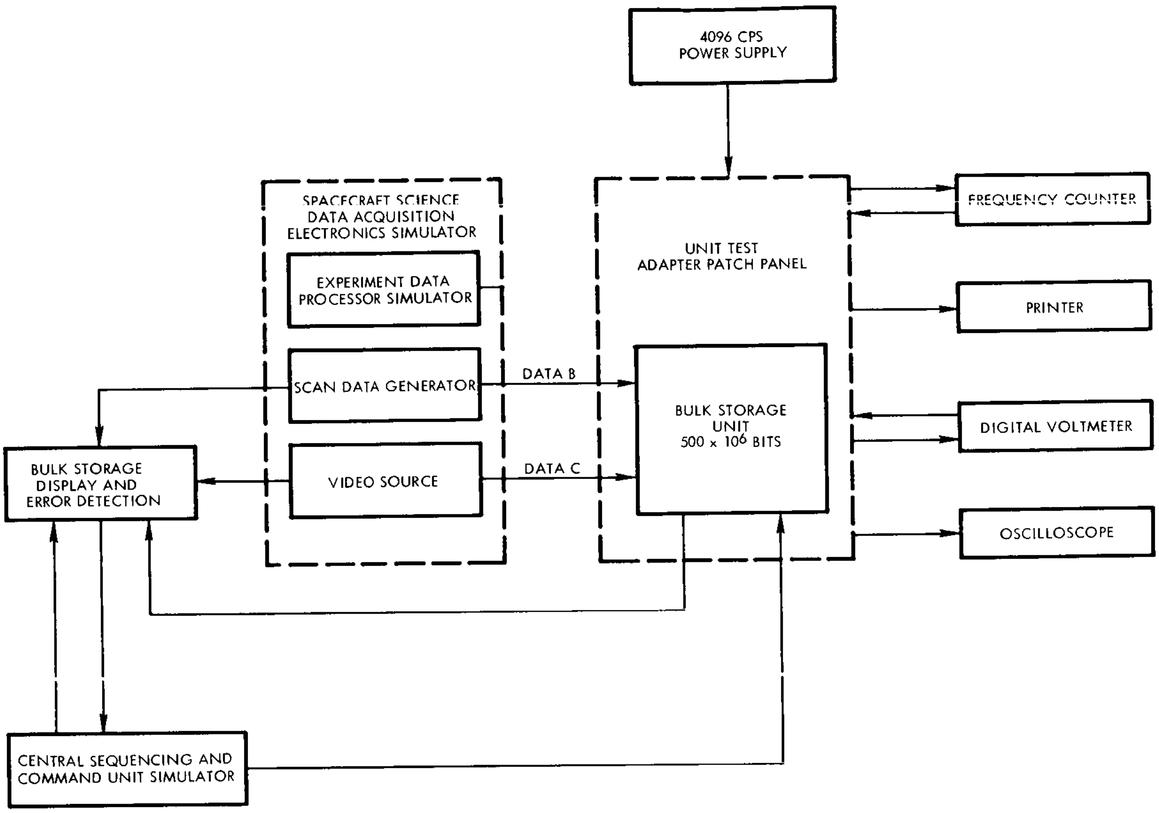


Figure 2. Bulk Memory Unit Test Set, Rack Layout

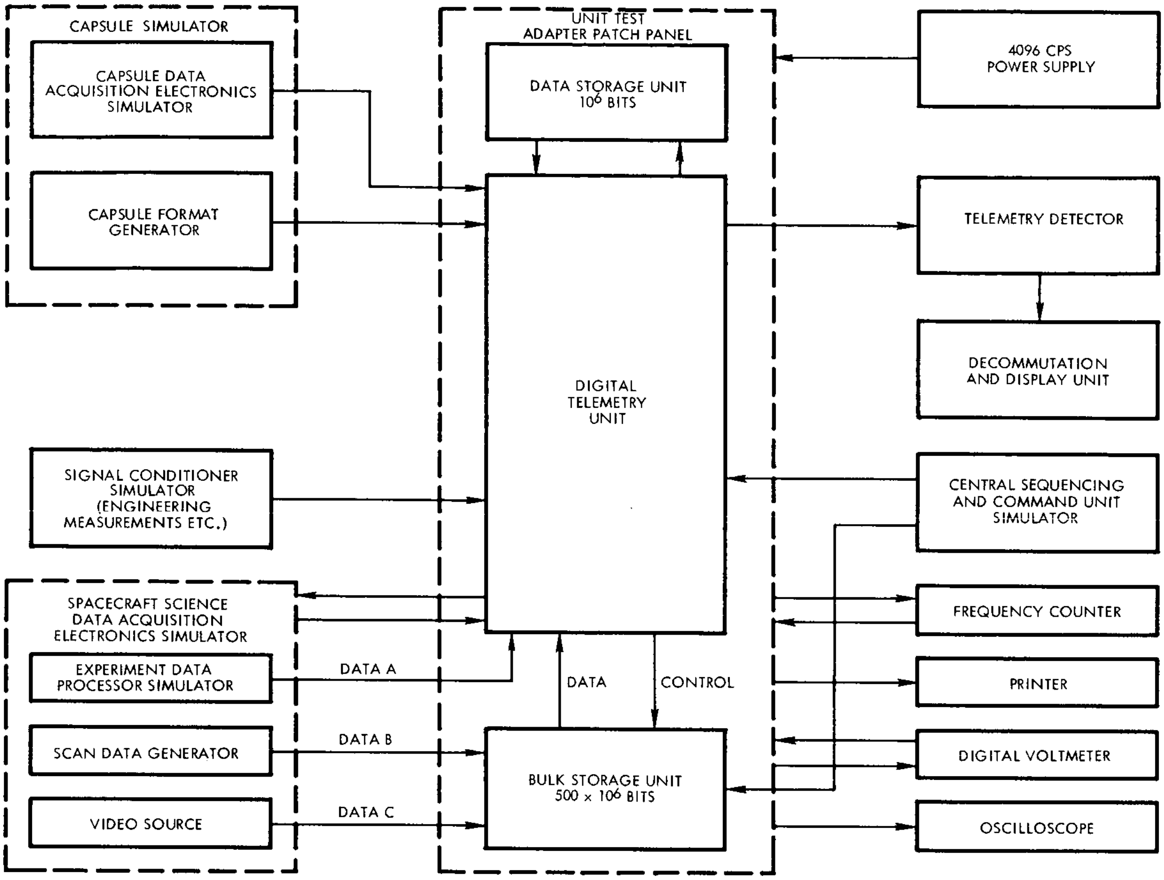


Figure 3. Digital Telemetry Unit Test Set, Block Diagram

5.2 Specific Equipment Description

5.2.1 Special Test Equipment

a. Memory Unit Tester

The digital storage unit mounts on the memory unit tester. The tester is capable of generating and entering various code patterns into the storage unit and checking circuitry is incorporated to detect errors in the bit pattern as the storage unit is read. Manual entry and retrieval of data will be possible.

b. Unit Test Adapter Patch Panel

The data storage unit, the bulk storage unit, and the digital telemetry unit are mounted to the patch panel for testing, except that the data storage unit is mounted elsewhere when tested alone. Power to the units is supplied through the panel. All unit test prints are available on the panel.

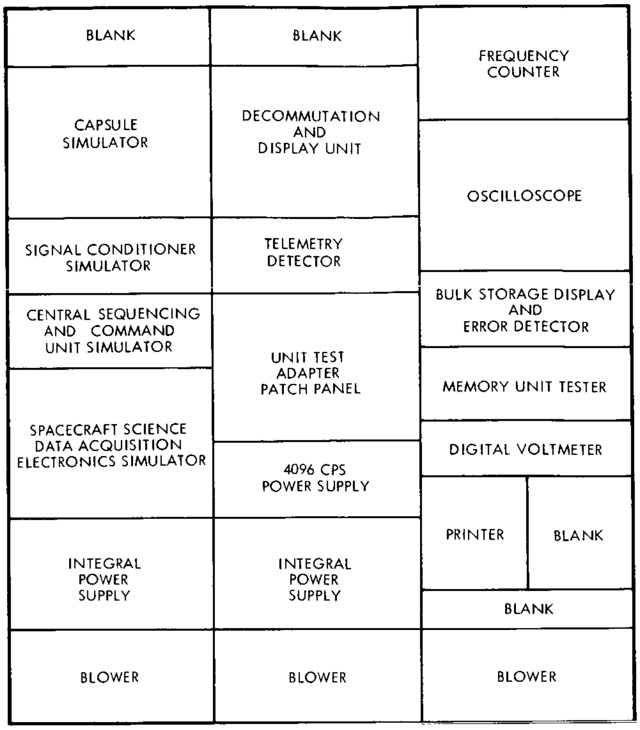


Figure 4. Data Handling Unit Test Set, Rack Layout

c. <u>Unit Power Supply</u>

All spacecraft units are powered from a 4096 cps square wave power supply. The supply is capable of being internally or externally synchronized. Low voltage level is 0V, high voltage level is +50V, and average power delivered is   watts. The unit power supply is capable of supplying all units connected to the patch panel (5.2.2b) as well as any connected to the memory tester (5.2.2a).

d. <u>Central Sequencer and Command Unit Simulator</u>

All signals normally supplied by the spacecraft central sequencer are generated by the simulator. These signals include discrete levels, power syncs, and elapsed time information. Also, any special control signals required by unit test gear will be provided.

e. Spacecraft Science Data Acquisition Electronics Simulator

Data generated and formatted from experiments aboard the spacecraft is generated by the science data acquisition simulator. Suitable information streams are generated to supply the bulk memory with test data and the digital telemetry unit with data it requires. The simulator is capable of regenerating data for comparison and error detecting purposes.

f. Bulk Storage Display and Error Detector

Information streams from the bulk storage unit and the science data acquisition simulator (5.2.2c) are compared by the bulk storage display and error detector and any mismatches indicated. Also, some provision for displaying selected contents of the bulk storage will be made.

g. Capsule Simulator

Characteristics of the capsule while connected to the spacecraft and after it has been launched are simulated by the capsule simulator.

h. Signal Conditioner Simulator

Data as normally presented from the signal conditioner is simulated by the signal conditioner simulator.

i. Demodulator/Synchronizer

The demodulator/synchronizer derives sync clocks and data streams from the down-link subcarrier. Bit rates from 4096 bits/second to 128 bits/second are accommodated. Bit and work sync is derived using a 63-count per work pseudo-noise code and loss of sync will be indicated.

j. Decommutation and Display Unit

The decommutation and display unit displays selected words from selected frames of data as indicated by manual controls on the front panel. Frame sync is supplied as an output to other devices, such as a computer complex. Loss of bit, work, or frame sync will be indicated.

5.2.2 Commerical Test Equipment

a. Frequency Counter

A frequency counter capable of measuring frequencies to 50.0 kc to an accuracy of parts in 10^{10} per hour is used. The counter is capable of driving a printer and is used in keeping track of bit rates, pseudo-noise rates, and in determining the accuracy and stability of any master clock sources.

b. Oscilloscope

A dual, or more, trace oscilloscope is used. Maximum vertical risetime is 20 μsec; minimum input resistance is 10 megohms; and maximum input capacitance to ground is 15 pf.

c. Digital Voltmeter

Resolution is digits. Value of the least significant digit on the lowest range is mv. Maximum voltage to be measured is 99 volts. Steady state accuracy is percent of full scale for that range. Automatic ranging is optional. Step function response time is no more than seconds. The digital voltmeter is able to follow a voltage ramp function of volts/second with no more than percent degradation in accuracy. The digital voltmeter is capable of driving a printer. Input resistance is greater than megohms and input capacitance is less than pf to ground.

d. Printer

A paper tape printer capable of printing data from the frequency counter and the digital voltmeter is used. Only one device need be connected to the printer at any one time.

5.4.3 Mechanical Design

Standard six-foot racks accepting 19-inch drawers are used. TRW standards are used where applicable.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The data handling unit test set operates from the following power source:

Voltage	115 ± 10 VAC
Frequency	60 ± 1 cps
Phase	Single
Maximum Average Power	watts

6.2 Service Condition

Ambient Temperature	0 to 40°C
Humidity	30 to 70 percent

7. PARAMETERS

(Details unknown)

8. CONSTRAINTS

Maximum use of developed hardware.

STABILIZATION AND CONTROL SUBSYSTEM
OSE/VS-4-410

1. SCOPE

1.1 Definition

This document defines the general requirements equipment list and applicable documents for stabilization and control subsystem MOSE required for the protection, transport, and storage of the stabilization and control subsystem equipment used in the Voyager program.

1.2 Identification

The models covered by this document conform to the requirements delineated herein and are identified as the VS-4-410 numbered series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

PPP-B-601A Amend. 2 16 August 1963	Boxes, Wood, Cleated - Plywood
MIL-D-3464B 31 October 1955	Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification
MIL-D-9959 Amend. 1 5 February 1963	Container, Flexible, Reusable, Water- Vaporproof

DAC/MSSD

Mechanical Support Equipment and Facilities Manual.

3. REQUIREMENTS

The stabilization and control subsystem MOSE items defined herein are designed to perform their functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

The end items defined within this documentation group are associated with the alignment and protection of the stabilization and control nozzles. The equipment enumerated in Table I accomplishes these major mechanical handling and support functions.

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The stabilization and control subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when used to support the stabilization and control subsystem components. All MOSE coupled to these components is operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is fabricated of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem components' physical envelope.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the stabilization and control subsystem components during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action are avoided.

Table I. Stabilization and Control Subsystem

Item No.	Nomenclature
4-410-1	Alignment Fixture, Stabilization and Control Nozzles
4-410-2	Protective Covers, Stabilization and Control Nozzles

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or land. This equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The equipment design requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use will there be degradation in the reliability of the stabilization and control subsystem equipment.

3.7 Maintainability

The MOSE is designed so that repairs, adjustments and overhaul can be readily accomplished by operating personnel using conventional general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification, with lift points, rated loads, hazard warnings, and special instructions noted.

ALIGNMENT FIXTURE, STABILIZATION AND CONTROL NOZZLES
OSE/VS-4-410-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the stabilization and control nozzles alignment fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-410	Voyager OSE, Stabilization and Control Subsystem

3. FUNCTIONAL REQUIREMENTS

The nozzle fixture supports level vials and alignment targets along an extension of the nozzle thrust axis.

4. DESIGN REQUIREMENTS

4.1 Accuracy

The fit of the fixture in the nozzles, the location of the targets, and the mounting of the level vials must be accomplished to an accuracy at least one order of magnitude greater accuracy than required by the alignment specifications of the nozzles.

4.2 Installation

The fixture is installed in the attitude control nozzles without deflecting them and without damage to the nozzle surface finish. The fixture clamps to the nozzles.

5. EQUIPMENT DESCRIPTION

5.1 General

The fixture consists of a conical plug tapered to match the inside of the nozzle. It has an extension from the large end which is on the geometric centerline of the nozzle plug. A ball level (or 2 vial levels) are mounted on this extension. These levels are accurate to the geometric centerline of the nozzle to ±0.25 degrees or less. The extension carries eight optical alignment targets which describe the longitudinal geometric axis of the nozzle. (See Figure 1.)

5.2 Interface Definition

The fixture mechanically interfaces with the nozzles and optically interfaces with the basic reference system.

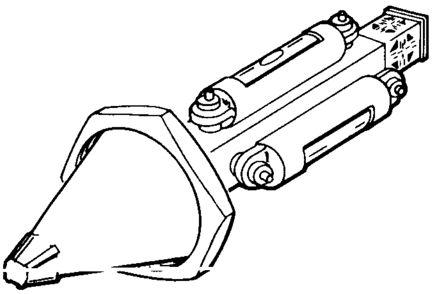


Figure 1. Alignment Fixture, Attitude Control Nozzles

PROTECTIVE COVERS, STABILIZATION AND CONTROL NOZZLES
OSE/VS-4-410-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the stabilization and control nozzles protective covers.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

<u>TRW</u> 1971 Voyager OSE Design Documents

OSE/VS-4-410	OSE Stabilization and Control Subsystem

3. FUNCTIONAL REQUIREMENTS

The stabilization and control nozzle protective covers provide physical protection against damage to the stabilization and control nozzles and retain a clean environment within the stabilization and control system.

4. DESIGN REQUIREMENTS

4.1 Mounting

The protective cover assembly completely covers the nozzle assembly and is attached to the nozzle block mounting boom by a friction grip method.

4.2 Load

The protective cover assembly withstands a 50 g impact load for a duration of 10 msec.

4.3 Deflection

The protective cover assembly will not deflect into the nozzles when subjected to an imposed load.

4.4 Nozzle Cover

Each nozzle is provided with a flexible foam cover capable of maintaining a clean environment within the nozzle assembly and the rest of the stabilization and control supply system.

4.5 Condensation

No moisture condensation is permitted within the protective covers.

4.6 Altitude

The protective covers are capable of functioning at all altitudes from sea level to 40,000 feet.

4.7 Reusability

The protective covers are reusable.

5. EQUIPMENT DESCRIPTION

5.1 General

The protective covers consist of an assembly cover enclosing the entire nozzle block assembly and covers for each nozzle.

The assembly cover is a rigid ribbed structure fabricated from thermoplastic acrylic butadiene styrene (ABS) material. It consists of two mating flanged halves bolted to one another for assembly. Clearance between the nozzle block assembly and the assembly cover is at least 2 inches. The cover extends beyond the nozzle block assembly and bears against the attach boom. Wing nuts allow the cover to be tightly snugged against the boom to provide a friction grip which prevents the assembly cover from moving. The contact points between the cover and boom are cushioned with felt.

Each nozzle cover is a tapered nylon reinforced, polyurethane, foam-lined, vinyl coated cover with an elasticized sleeve. The thickness of the cover is 1/2 inch. The elasticized cover fits snugly and the cover contains a relief valve. The cover is sealed by a flap containing a bonded ribbon of 2-inch wide nylon hook and pile tape (Velcro). This design concept is shown in Figure 1.

5.2 System Interface

This item has no physical or electrical interface with other MOSE.

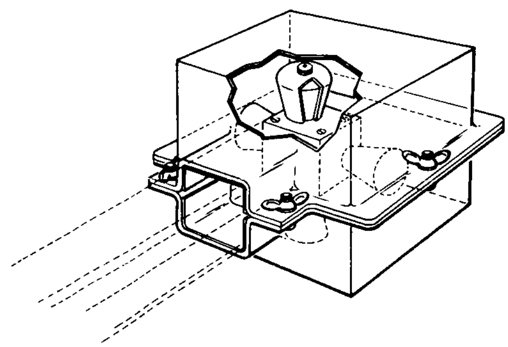


Figure 1. Protective Covers, Stabilization and Control Nozzles

RATE GYRO ASSEMBLY UNIT TEST SET
OSE/VS-4-411-1

1. SCOPE

This document covers the requirements for the rate gyro assembly unit test set used to evaluate the performance of the Voyager gyro assembly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The rate gyro assembly UTS is used to perform gyro unit tests consisting of:

a) Gyro drift

b) Mode control

c) Temperature control

d) Torquer calibration

e) Telemetry output signal output calibration.

3.2 Test Functions

The gyro assembly UTS provides:

a) Power to the gyro assembly unit

b) Analog torquer commands

c) Mode switching discretes

d) Accurate inertial angular rates

e) Accurate inertial angular positions

f) Loads for the analog telemetry output signals.

It also monitors:

a) Various DC voltages

b) Various AC voltages

c) Gyro torquer current

d) Gyro temperatures

e) Frequency of spin motor.

4. DESIGN REQUIREMENTS

To provide a complete test of the gyro assembly performance, the rate gyro assembly UTS performs the following specified tests:

4.1 Gyro Alignment

Gyro alignment is accomplished by sequentially applying inertial rates about the axis of each gyro in the gyro assembly and measuring the corresponding gimble position signal.

4.2 Gyro Drift

Gyro drift is dynamically measured by operating the gyros in a servo loop mode utilizing the servo table. Temperature control is monitored continuously and recorded on the strip chart recorder.

4.3 Torquer Calibration

Torquer calibration is measured by connecting the gyros in a closed servo table mode, applying an accurately calibrated current to the gyro torquers and timing servo table rotation.

5. FUNCTIONAL DESCRIPTION

5.1 Test Set Equipment

The test set comprises standard commercial test equipment and special equipment designed for testing functions peculiar to the Voyager rate gyro assembly.

Figure 1 is a functional block diagram of the test set.

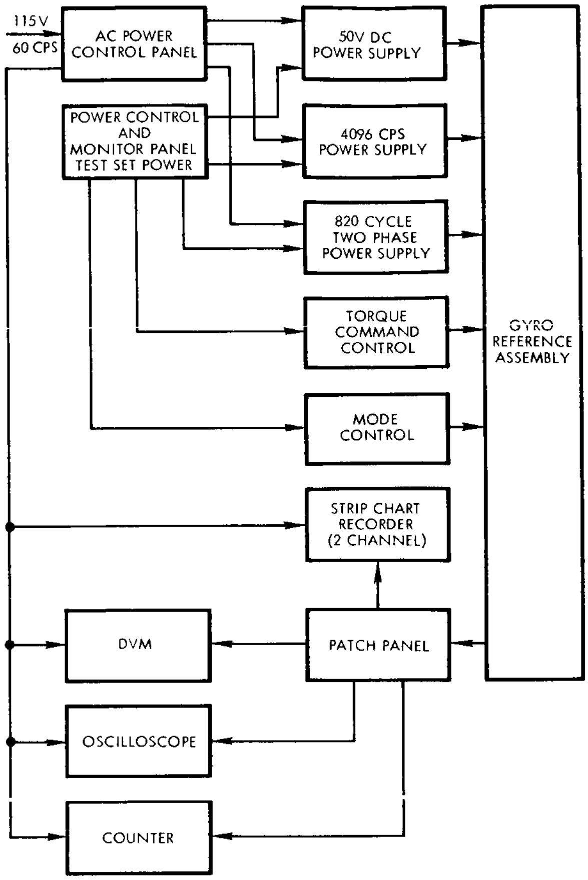


Figure 1. Rate Gyro Assembly Unit Test Set, Block Diagram

Figure 2 is a drawing of the configuration of the test set. The test set consists of one rack of equipment and a servo table with the gyro mount.

5.2 Test Equipment Description

5.2.1 Counter

5.2.2 Digital Voltmeter

5.2.3 Oscilloscope

5.2.4 Recorder

5.2.5 Test Control and Patch Panel

A test control and patch panel contains mode control switches, torquer command current control, and a suitable number of test points.

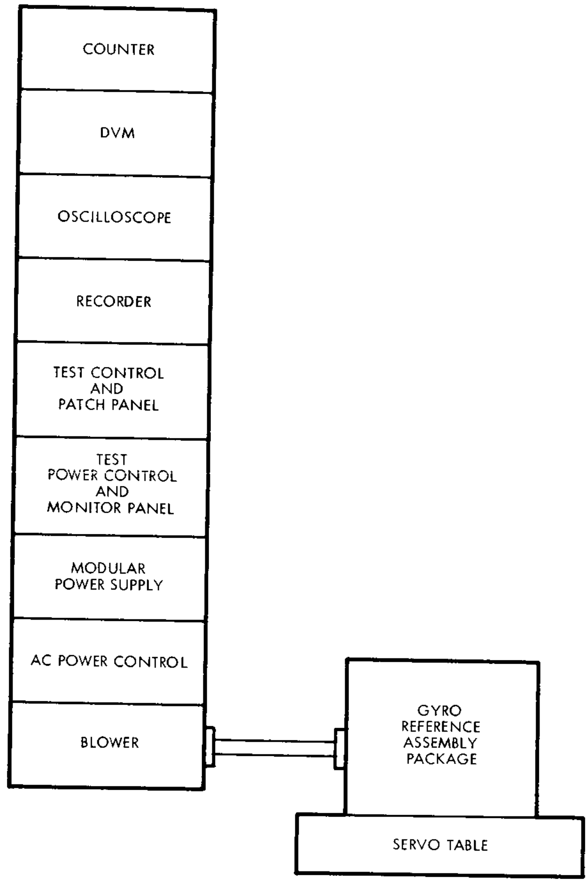


Figure 2. Rate Gyro Assembly Unit Test Set, Rack Layout

5.2.6 Power Control and Monitor Panel

A power control and monitor panel contains switches and meters to control and monitor the power applied to the gyro reference assembly.

5.2.7 Modular Power Panel

A modular power panel contains modular power supplies to duplicate the power necessary for the gyro assembly.

5.2.8 Cables and Mounting Fixture

Two test cables approximately 20 feet long are provided to connect the gyro assembly to the test set. A precision mounting fixture with three orthogonal sides is provided to mount the gyro assembly on a rate table, which is also provided.

5.3 Commercial Test Equipment

The following items are standard commercial equipment, the functions of which are discussed above.

a) Counter

b) Digital voltmeter

c) Oscilloscope

d) Recorder.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The test sets operate from a power source as specified below:

Voltage	115 ± 10 Volts AC
Frequency	60 ± 1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 - 90°F
Humidity	Less than 50 per cent

7. PARAMETERS

The following critical parameters of the spacecraft are tested by the rate gyro assembly UTS:

a) Gyro alignment

b) Gyro drift

c) Mode control

d) Temperature control

e) Torquer calibration

f) Telemetry signal calibration.

8. CONSTRAINTS

No constraints are imposed upon the test set.

9. INTERFACE

9.1 Inputs

The rate gyro assembly UTS interfaces with the following inputs:

a) Telemetry: To provide loads analog telemetry signal outputs from the spacecraft.

b) Gyro error signals: To provide a load and monitoring position.

c) Gyro torquer current: To provide measurement.

d) Gyro temperature signal: To monitor temperature.

e) Spin motion detectors signal: To monitor rotation.

9.2 Outputs

The test set provides the following outputs to the interface:

a) AC and DC power: To the gyro assembly.

b) Torque command: Command signals to the gyro assembly.

c) Mode command: Command discretes to the gyro assembly.

d) Servo-table feedback: To the gyro assembly.

SUN-SENSOR UNIT TEST SET
OSE/VS-4-411-2

1. SCOPE

This document covers the requirements for the sun sensor detector unit test set used to evaluate the performance of three of the Voyager control and stabilization subsystem components: coarse sun sensor, fine sun sensor, and near-earth detector.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The sun sensor unit test set is used to perform control and stabilization tests consisting of:

a) Null point (sun sensors)

b) Scale factor

c) Linearity.

3.2 Test Functions

The sun sensor unit test set provides the following functions:

a) AC power to the sun sensor assemblies and the near-earth detector

b) Collimated light to simulate sun or earth

c) Angular rotation of sensor assembly under test with respect to line of radiation from solar simulator

d) Monitor of sensor assembly for test error signals.

4. DESIGN REQUIREMENTS

To provide a complete test of the sun sensor and near-earth detector performance, the sun sensor unit test set performs the following specified tests:

4.1

4.2 TO BE SUPPLIED LATER

4.3

5. FUNCTIONAL DESCRIPTION

5.1 Test Set Equipment

The test set comprises standard commercial test equipment and special equipment designed for testing functions peculiar to the Voyager sun sensor and near earth detector.

Figure 1 is a functional block diagram of the test set.

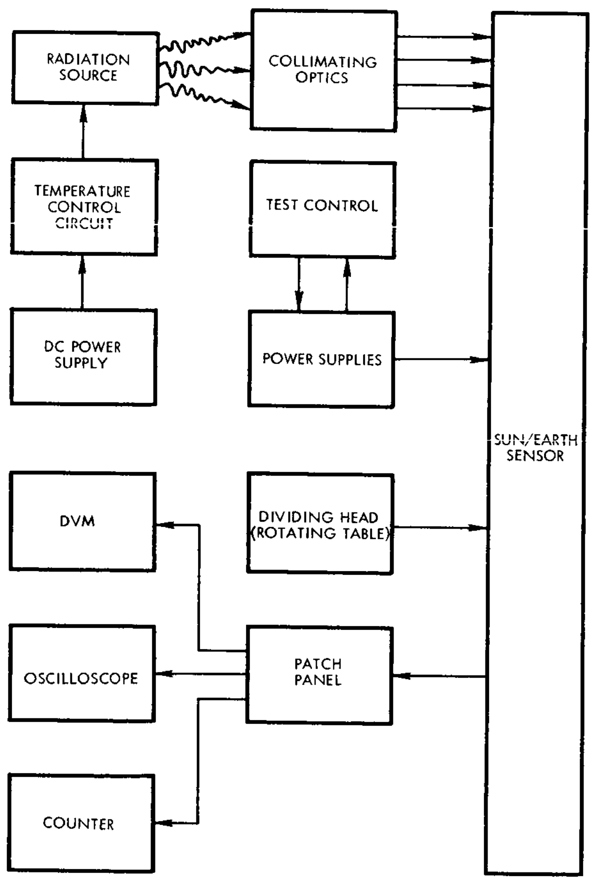


Figure 1. Sun Sensor Unit Test Set, Block Diagram

Figure 2 is a drawing of the configuration of the test set. The test set consists of one rack of equipment, a rotating table upon which to mount the sun sensor and near earth detector assembly, and the sun and earth simulator which contains the radiation sources assembly and collimating optics, and the base.

5.2 Test Equipment Description

5.2.1 Counter

5.2.2 Digital Voltmeter

5.2.3 Oscilloscope

5.2.4 Test Control and Monitor Panel

5.2.5 Power Control and Monitor Panel

The power control and monitor panel provides controls for the application of DC and AC power to the assembly under test and patch points to facilitate connection of commercial test equipment to the sensor assembly.

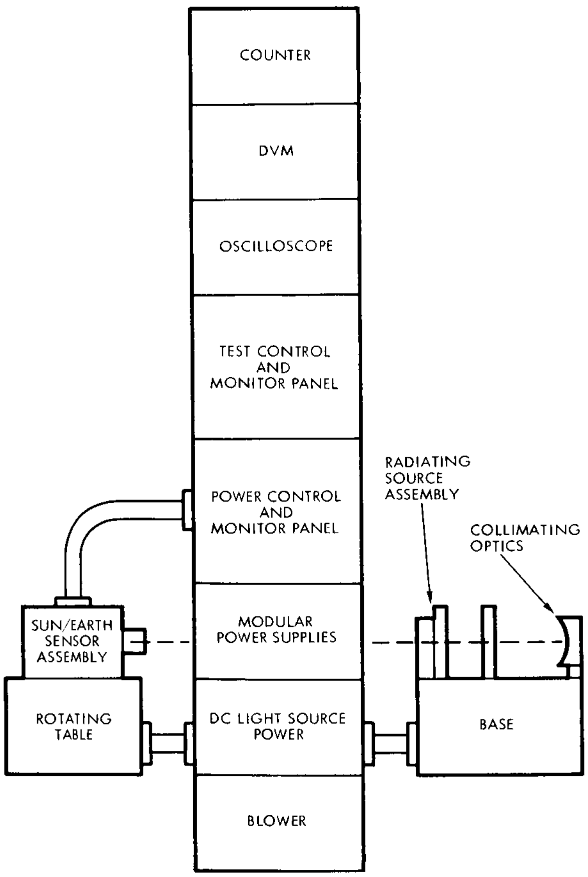


Figure 2. Sun Sensor Unit Test Set, Rack Layout

5.2.6 Modular Power Panel

The modular power panel provides the power required for operation of the assembly and the test set.

5.2.7 Sun and Earth Simulator

The sun and earth simulator consists of an arc light and collimating mirrors to provide a less intense light source to the earth detector assembly.

5.2.8 Rotating Table, Sensor Mount, and Cable

An accurately machined mounting bracket with an optical device for alignment purposes is provided to mount the assembly under test to a dividing head. The dividing head provides a calibrated angular rotation of the assembly under test with respect to the light source. A test cable approximately 20 feet long is provided to connect the sun sensor assembly to the test console.

5.3 Commercial Test Equipment

The following items are standard commercial equipment, the functions of which are discussed above:

a) Counter

b) Digital voltmeter

c) Oscilloscope.

6. BOUNDARY OF DEFINITIONS

6.1 Primary Power Source

The test set operates from a power source as specified below:

Voltage	115 Volts ± 10 Volts, AC
Frequency	60 cps ± 1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 to 90°F
Relative humidity	Less than 50 per cent

7. PARAMETERS

The following critical parameters of the spacecraft are tested by the sun sensor unit test set:

a) Null point determination

b) Scale factor

c) Linearity.

8. CONSTRAINTS

No constraints are imposed upon the test set.

9. INTERFACES

9.1 Inputs

The sun sensor unit test set provides interface with the sun sensor assembly unit error output signal.

9.2 Outputs

The test set provides DC and AC power as necessary to the interface for the sensor assembly.

STAR SENSOR UNIT TEST SET
OSE/VS-4-411-3

1. SCOPE

This document covers the requirements for the star sensor unit test set used to evaluate the performance of the Voyager star sensor assembly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The star sensor UTS is used to perform control and stabilization tests consisting of:

a) Null point

b) Scale factor

c) Linearity.

3.2 Test Functions

The star sensor UTS provides the following functions:

a) AC power to the star sensor assembly

b) Collimated light to simulate the star

c) Angular rotation of star sensor assembly under test (with respect to line of radiation from the star simulator)

d) Monitor of sensor assembly for test error signals.

4. DESIGN REQUIREMENTS

To provide a complete test of the star sensor assembly performance, the star sensor UTS performs the following specified tests:

4.1

4.2 TO BE SUPPLIED LATER

4.3

5. FUNCTIONAL DESCRIPTION

5.1 Test Set Equipment

The test set comprises standard commercial test equipment and special equipment designed for testing functions peculiar to the Voyager star sensor.

Figure 1 is a functional block diagram of the test set.

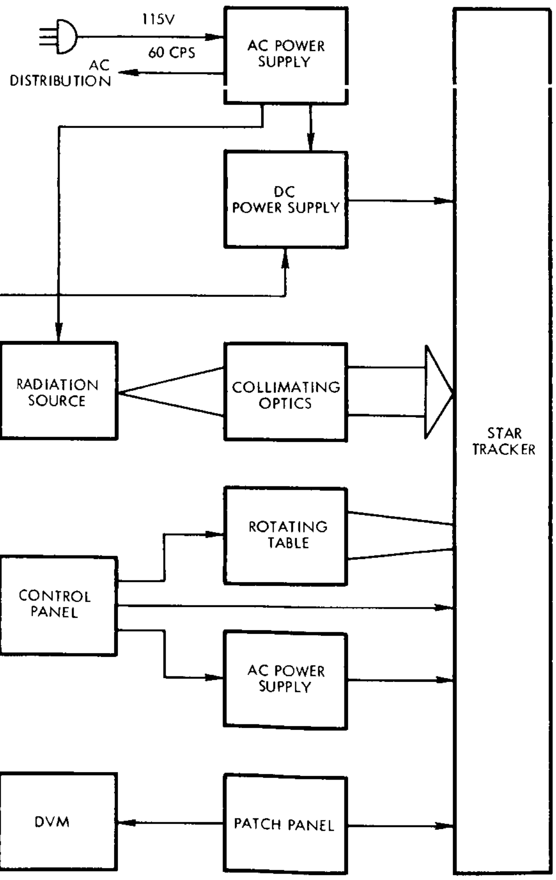


Figure 1. Star Sensor Unit Test Set, Block Diagram

Figure 2 is a drawing of the configuration of the test set. The test set consists of one rack of equipment, a rotating table upon which to mount the star sensor assembly, and the star simulator which contains the radiation source assembly and collimating optics and the base.

5.2 Test Equipment Description

5.2.1 Test Control and Monitor Panel

The test control and monitor panel provides controls for the application of DC and AC power to the assembly under test and patch points to facilitate connection of commercial test equipment to the sensor assembly.

5.2.3 Modular Power Panel

The modular power panel provides AC and DC power required for operation of the star sensor assembly and the test set.

5.2.4 Star Simulator

The star simulator consists of an incandescent light and collimating mirrors to provide a light of suitable intensity to the sensor assembly.

5.2.5 Rotating Table, Sensor Mount, and Cable

An accurately machined mounting bracket with an optical device for alignment purposes is provided to mount the star sensor assembly to a dividing head. The dividing head provides a calibrated angular rotation of the star sensor assembly with respect to the light source. A test cable approximately 20 feet long is provided to connect the star sensor assembly to the test set.

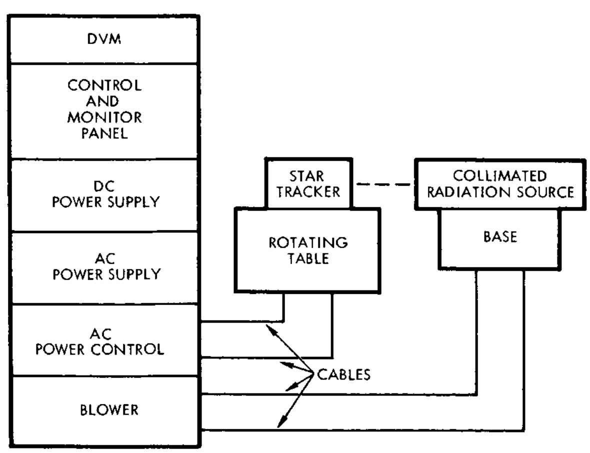


Figure 2. Star Sensor Unit Test Set, Rack Layout

5.3 Commercial Test Equipment

The digital voltmeter, the function of which is discussed above is standard commercial equipment.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Sources

The test set operates from power sources as specified below:

Voltage	115 volts ± 10 volts, ac.
Frequency	60 cps ± 1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 to 90°F
Relative Humidity	Less than 50 per cent

7. PARAMETERS

The following critical parameters of the spacecraft will be tested by the star sensor UTS:

a) Null point determination

b) Scale factor

c) Linearity.

8. CONSTRAINTS

No constraints are imposed upon the test set.

9. INTERFACES

9.1 Inputs

The star sensor UTS provides interface with the star sensor assembly to accept the tracking error signal.

9.2 Outputs

The test set interfaces with the sensor assembly to provide the following outputs:

a) AC and DC power as necessary

b) Angular motion to the assembly under test

c) Light radiation to the unit under test.

STABILIZATION CONTROL ELECTRONICS ASSEMBLY UNIT TEST SET
OSE/VS-4-411-4

1. SCOPE

This document covers the requirements for the control electronics assembly (CEA) unit test set used to evaluate the performance of the Voyager control electronics assembly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The CEA UTS is used to perform control electronics tests consisting of:

a) Power profile

b) Mode control

c) Command response

d) Telemetry signal conditioning

e) Phasing

f) Threshold levels

g) Actuator drive signals.

3.2 Test Function

The CEA UTS provides the following functions:

a) Power for the control electronics assembly

b) Control commands

c) Simulated sensor inputs to the control electronics

- Gyro signals
- Sun and earth sensor discretes
- Sun and earth error signals
- Actuator potentiometer signals
- Pneumatic pressure transducer signals,

d) Loads for all control electronic assembly outputs

- Gyro torques
- Gyro discretes
- Gyro power
- Sun, earth, and star sensor
- Actuator outputs
- Pneumatic valve
- Telemetry output signals.

e) Monitor the following control electronics assembly output:

- The discretes
- DC and AC voltages
- Frequency of input powers
- Wave shapes from the control electronics assembly
- Status of valve drive signals
- Status of control electronics output discreets.

4. DESIGN REQUIREMENTS

To provide a complete test of the control electronics assembly performance, the CEA UTS performs the tests specified in the following paragraphs.

4.1 <u>Power Profile</u>

4.2 <u>Mode Control</u>

4.3 Command Response

4.4 Telemetry Signal Response

4.5 Phasing

4.6 Threshold Levels

4.7 Actuator Drive Signals

5. FUNCTIONAL DESCRIPTION

5.1 Test Set Equipment

The test set comprises standard commercial test equipment and special equipment designed for testing functions peculiar to the Voyager control electronics assembly.

Figure 1 is a functional block diagram of the test set.

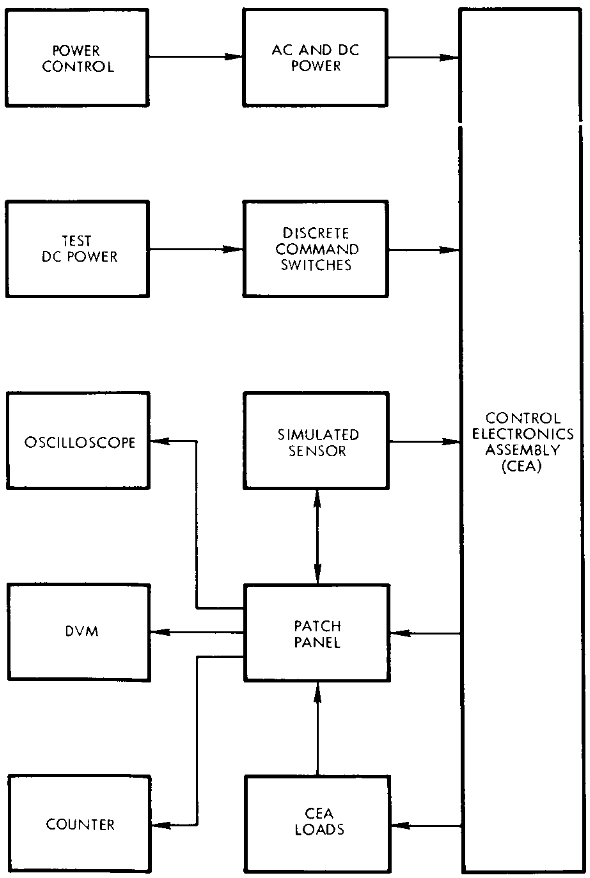


Figure 1. Control Electronics Assembly Unit Test Set, Block Diagram

Figure 2 is a drawing of the configuration of the test set. The test set consists of one console.

5.2 Test Equipment Description

5.2.1 Counter

5.2.2 Digital Voltmeter

5.2.3 Oscilloscope

5.2.4 Control and Monitor Panel

The control and display panel contains switches and potentiometers to generate discretes, quantized signals, simulated sensor AC and DC signals and simulated telemetry signals for application to the control electronics assembly.

Discrete commands are generated and applied to the control electronics assembly under test as required by the test procedure.

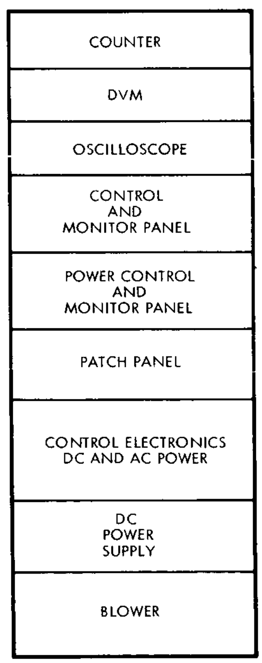


Figure 2. Control Electronics Assembly Unit Test Set, Rack Layout

5.2.5 Patch Panel

The patch panel contains test points to facilitate connecting of special commercial test equipment to appropriate control electronics assembly outputs for performance monitoring. This panel also contains the resistive and inductive loads for the outputs from the control electronics assembly.

5.2.6 Control Electronics Power Panel and Cables

The power panel contains modular power supplies to provide the control electronics assembly with correct input voltages.

Test cables approximately 20 feet long are provided to connect the test set to the control electronics assembly.

5.3 Commercial Test Equipment

The following items of equipment, the functions of which are discussed above, will be standard commercial equipment:

a) Counter

b) Digital voltmeter

c) Oscilloscope.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The CEA UTS operates from a power source as specified below:

Voltage	115 volts ±10 volts, ac
Frequency	60 cps ±1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 to 90°F
Relative Humidity	Less than 50 percent

7. PARAMETERS

The following critical parameters of the spacecraft are tested by the CEA UTS:

a) Power profile

b) Mode control

c) Command response.

8. CONSTRAINTS

No constraints are imposed upon the test set.

9. INTERFACES

9.1 Inputs

The CEA UTS interfaces with the control electronics assembly to accept the following inputs:

a) Gyro torquer loads

b) Gyro discrete loads

c) Sun sensor course and fine and near earth detector loads

d) Actuator loads

e) Telemetry output signal loads

f) Loads for all assembly output, discretes.

9.2 Outputs

The test set provides the following outputs at the interfaces with the control electronics assembly:

a) Gyro error signals

b) Sun sensor output signal

c) Fine sun sensor output signal

d) Near earth detector signal

e) Actuator signals

f) Antenna array direction signals

g) Simulated telemetry command signals.

ACTUATOR UNIT TEST SET
OSE/VS-4-411-5

1. SCOPE

This document covers the requirements for the actuator unit test set used to evaluate the performance of the Voyager de-boost engine thrust vector control actuator valve assemblies.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The actuator UTS is used to perform actuator tests as follows:

a) Drive capability

b) Feedback signal linearity

c) Injectant flow versus valve drive signal.

3.2 Test Functions

The actuator UTS provides the following functions:

a) Actuator drive signals

b) Excitation for actuator position potentiometers

c) Measurement of position potentiometer signals

d) Measurement of actuator mechanical motion.

4. DESIGN REQUIREMENTS

To provide a complete test of the actuator valve assembly performance, the actuator UTS performs the following specified tests:

4.1 Drive Capability

4.2 Feedback Signal Linearity

4.3 Injectant Flow Versus Valve Drive Signal

5. FUNCTIONAL DESCRIPTION

5.1 Test Set Equipment

The test set comprises standard commercial test equipment and special equipment designed for testing functions peculiar to the Voyager actuator valve assemblies.

Figure 1 is a functional block diagram of the test set.

Figure 2 is a drawing of the configuration of the test set. The test set consists of one console.

5.2 Test Equipment Description

5.2.1 Digital Voltmeter

5.2.2 Oscilloscope

5.2.3 Steering Signal Simulator

5.2.4 Actuator Assembly Control and Monitor Panel

The control and monitor panel contains switches to control application of power to the actuator and test points for monitoring.

5.2.5 Actuator Assembly Power Supply

The actuator assembly power supply contains modular power supplied to provide the actuator potentiometers with excitation and test points for monitoring.

5.3 Commercial Test Equipment

The following items of equipment, the functions of which are discussed above, are standard commercial equipment:

a) Digital voltmeter

b) Oscilloscope.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The actuator UTS operates from a power source as specified below:

Voltage	115 Volts ±10 Volts, AC
Frequency	60 cps ±1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 to 90°F
Relative Humidity	Less than 50 percent

6.3 Injectant Flow

The test set is used with an injectant flow facility which provides injectant flow calibration capability.

7. PARAMETERS

The following critical parameters of the spacecraft are tested by the actuator UTS:

a) Positioning accuracy

b) Feedback signal linearity

c) Injectant valve flow calibration.

8. CONSTRAINTS

No constraints are imposed upon the test set.

9. INTERFACES

9.1 Inputs

The actuator UTS accepts the actuator potentiometer position signals at an interface with the actuator.

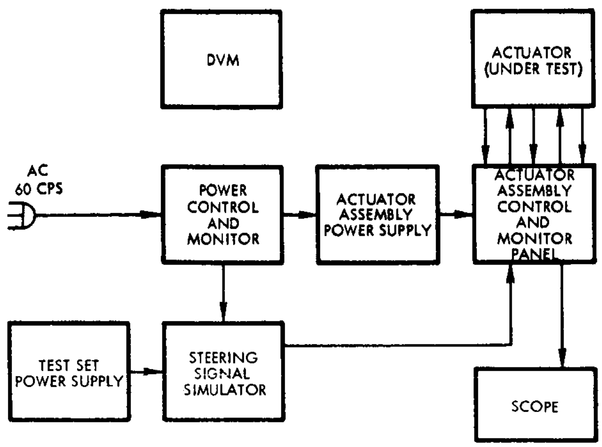


Figure 1. Actuator Unit Test Set, Block Diagram

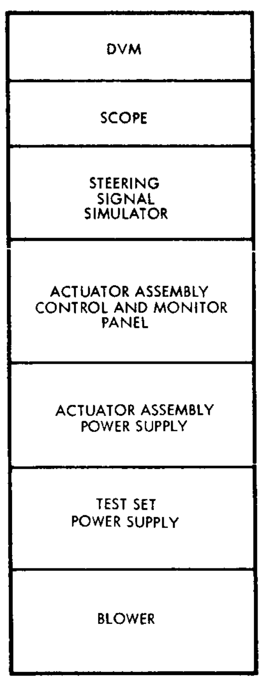


Figure 2. Actuator Unit Test Set, Rack Layout

9.2 <u>Outputs</u>

The test set provides the following outputs at an interface with the actuator:

a) Power as required

b) Actuator movement signals.

CENTRAL SEQUENCING AND COMMAND UNIT TEST SET
OSE/VS-4-451-1

1. SCOPE

This document establishes the requirements for the central sequencing and command unit test set used to evaluate the performance of the Voyager command decoding system.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 General

The central sequencing and command UTS is used to verify proper operation of the following:

a) Sync detection

b) Spacecraft address detection

c) Direct command interpretation

d) Quantitative command interpretation

e) Stored command sequences

f) Master clock stability and accuracy

g) Elapsed time clock

h) Timed discretes

i) Power sync

j) Telemetry bit rate

k) Telemetry data

l) Internal power supply operation

m) Subunit operation.

3.2 Test Functions

The UTS tests the following portions of the central sequencer while assembled as a total system:

a) Input decoders

b) Command decoders

c) Programmers (storage unit)

d) Master clock system

e) Internal power supplies.

4. DESIGN REQUIREMENTS

4.1 Command Encoder

The command encoder is designed to supply commands to control the spacecraft. Normal transmission of commands are under manual or computer control as selected by a front panel switch. The bit number and the entire command (spacecraft address, contents of quantitative data and any parity bits) are displayed on the front panel. An error detected in the command during transmission lights an error display.

4.1.1 Input

Inputs to the command encoder are manual, with front panel switches controlling the operation, or from a general purpose control computer. Mode of control is selected by a front panel switch.

4.1.2 Output Signals

a. Up-link Subcarrier

The encoder combines a 511 bit pseudo-noise (PN) code with each bit of command data to form a single line subcarrier. One of two PN codes, each unique to a spacecraft decoder, is selected by a front panel switch. The output signal of the encoder modulates the carrier signal of a transmitter. The subcarrier signal has the following characteristics:

Modulation	Phase shift keying (PSK)
PN code clock frequency	511 cps ± percent

PN code clock duty cycle	50% ± percent
PN code clock sine wave distortion	
PN code clock sine wave phase angle	
Switched at zero crossover ± usec	
Zero crossover and phase within ± usec of the symmetrical digital clock	
Bit rate	One bit per second ± percent
Value of command and sync attenuation constants	0 to 100 per cent
Amplitude	
Distortion	
Output impedance	
Bias	Bias adjustments independent of attenuation shall be provided for command and sync information streams

b. Central Sequencing and Command Test Signals

A command bit train and bit sync suitable for operating the command and sequencer unit is made available.

c. Computer Signals

A command bit train and bit sync is made available to the computer. Also, the encoder informs the computer when transmission of a command is complete.

4.1.3 General Modes of Operation

The command encoder operates in two general modes, normal transmit or test, regardless of manual or computer control.

a. Normal Transmit

For testing spacecraft reactions, correctly coded commands are issued, unless specifically overridden.

b. Self-test

For self-test and maintenance, additional controls for power and clock are provided. To prevent inadvertent unit shutdown, the power control is active only when a test-operate control is in the test position. Clock speed controls are provided to allow a standard one second bit rate, a fast clock for use with oscilloscope display, and a step control to allow manual step-through each of the sequences. The controls are active only when the operate-test control is in the test position. In the self-test mode, the PSK output is defeated so that commands will not be transmitted.

4.1.4 Error Detection

An error detection loop is an integral part of the encoder circuitry.

4.1.5 Error Generation

All possible codes are capable of being generated for testing purposes. No improper code is possible under normal transmit conditions unless specifically overridden.

4.1.6 Marginal Testing

a. Timing

Timing adjustments are provided on all outputs.

b. Signal Levels

Signal level adjustments are provided on all outputs.

4.1.7 Block Diagram

A functional block diagram appears in Figure 1.

4.2 Spacecraft Status Display

The status display monitors the level of each discrete output and gives a visual indication of those commands that are active. The status display also shows the value of elapsed time stored in the central sequencing and command unit.

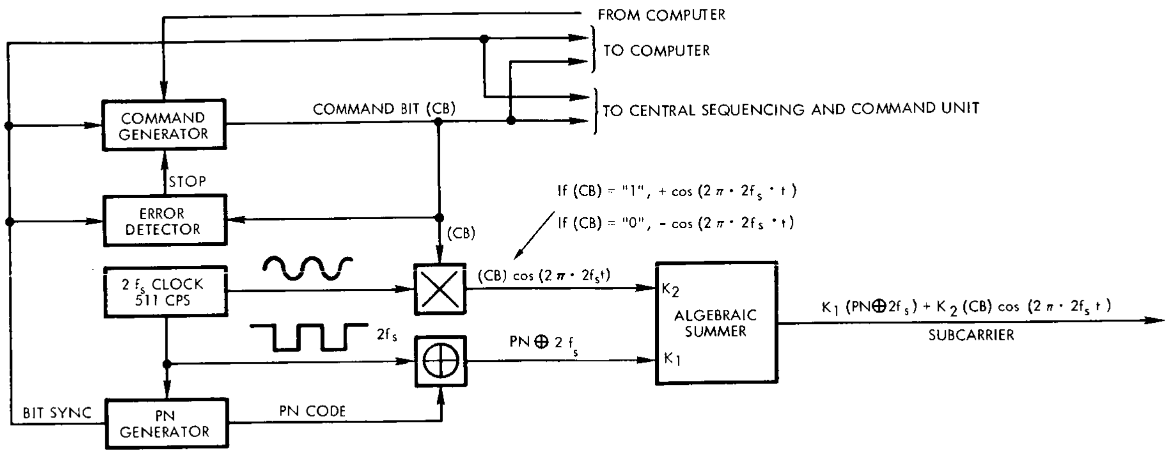


Figure 1. Command Encoder, Block Diagram

4.3 Event Recorder

An event recorder may be used for measuring relative timing of command discretes and/or quantitative data.

4.4 Strip Chart Recorder

A strip chart recorder is made available for measuring timing and signal levels of slow data outputs.

4.5 Oscilloscope

An oscilloscope is available for measuring rapidly changing signals such as master clocks, power syncs, etc.

4.6 Counter

A general purpose event-per-unit-time counter is available for determining master clock accuracy and stability.

4.7 Unit Test Adapter and Load Simulator Patch Panel

A panel to which the central sequencer and command unit can be mounted, is required. All input/output connects are brought out to a patch panel. Loads and minor interface circuits are available on the panel.

4.8 Digital Voltmeter

A digital voltmeter is required to accurately determine the state of the unit's internal power supply and any other levels.

4.9 Printer

A small line printer can be used to automatically record counter and/or voltmeter data.

4.10 Spacecraft Power Source

The central sequencer and command unit requires a 50V (0V to +50V) square wave power source at 4096 cps. Provision is made for the central sequencer and command unit to supply the proper sync frequency as required.

5. FUNCTIONAL DESCRIPTION

The test set comprises standard commercial test equipment and specific equipment designed for testing functions peculiar to Voyager.

A function block diagram appears in Figure 2.

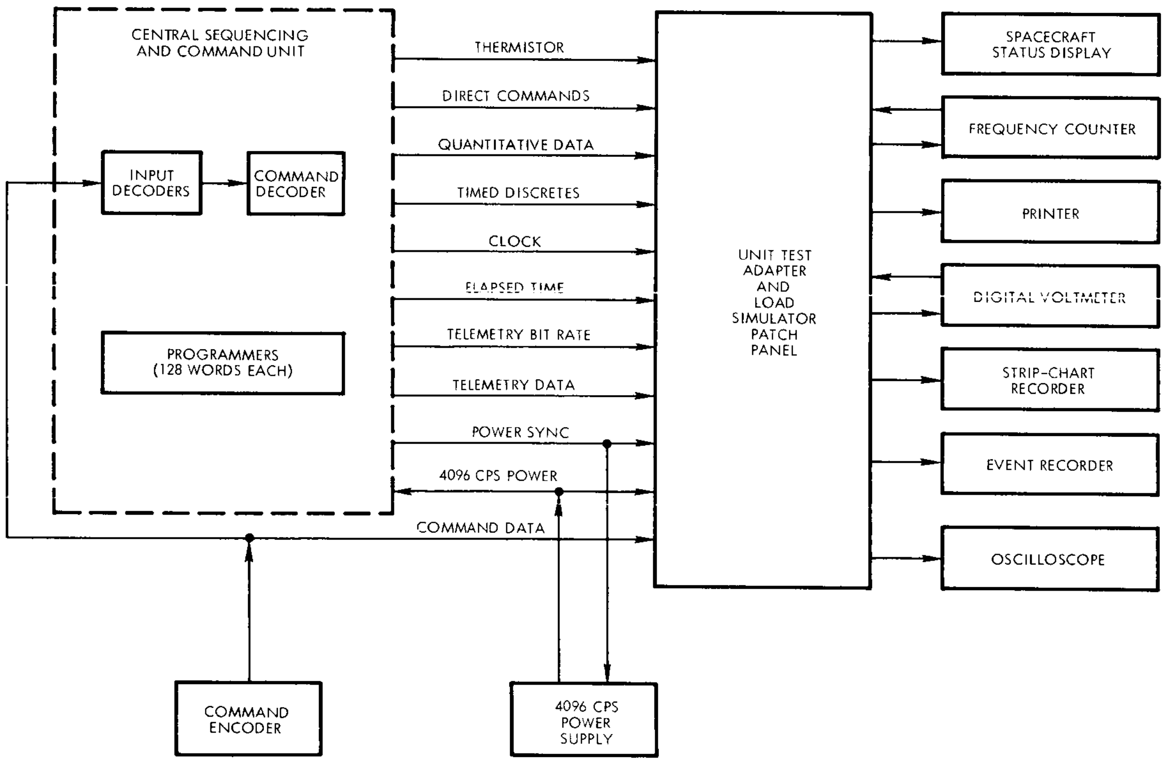


Figure 2. Central Sequencing and Command Unit Test Set, Block Diagram

Figure 3 shows the physical configuration of the central sequencing and command unit test set. It consists of two racks of equipment; the first rack contains special adapter and test equipment and the second contains general commercial gear.

5.2 Specific Equipment Description

5.2.1 Special Test Equipment

a. Command Encoder

All possible bit combinations of commands are possible for testing purposes. Signal levels as required by the central sequencing and command unit are available from the command encoder. For more detailed discussions see 4.1.

b. Spacecraft Status Display

The status display monitors the level of each discrete output and gives a visual indication of those commands that are active. Elapsed time is also displayed.

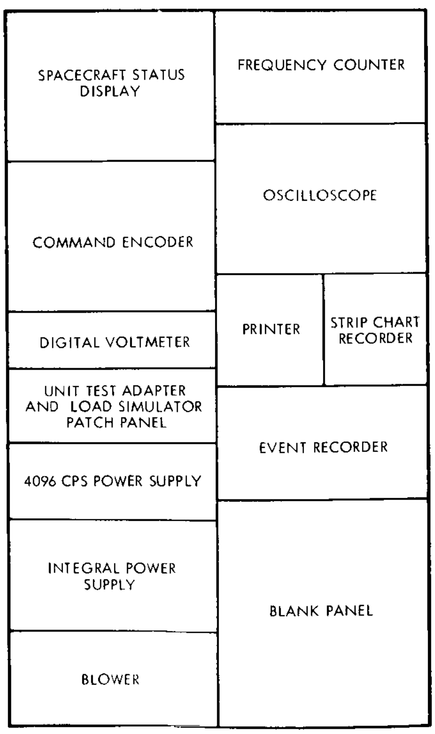


Figure 3. Central Sequencing and Command Unit Test Set, Rack Layout

c. Unit Test Adapter and Load Simulator Patch Panel

A mounting and electrical interface compatible with the central sequencer and command unit is provided on this panel. Innerconnections to the various pieces of test gear can be made from this panel. Any required loads or simple innerface networks are also provided.

5.2.2 Commercial Test Equipment

a. Frequency Counter

A frequency counter capable of measuring frequencies to 5.0 kc to an accuracy of parts in 10^{10} per hour will be used. The counter is capable of driving a printer.

b. Oscilloscope

A dual or more trace capability with a vertical risetime response of 20 usec maximum will be used.

c. Event Recorder

The event recorder follows number of events and is able to discern time differences of seconds, to an accuracy of seconds.

d. Strip Chart Recorder

The strip chart recorder is able to plot number of inputs against time. The time scale is variable between inches per second to inches per second. The deflection is calibrated within per cent of full scale. Response time is seconds (10 to 90 per cent full scale) with no overshoot.

e. Printer

A paper tape printer capable of printing data from the frequency counter and the digital voltmeter is used. Only one device need be connected to the printer at any one time, and this connection may be accomplished by manually changing connectors.

f. Digital Voltmeter

The digital voltmeter is able to follow a changing waveform of volts per second with no more than per cent degradation in accuracy. Steady state accuracy will be per cent full scale on any range.

Resolution will be digits. Value of the least significant digit on the lowest range will be mv. Maximum voltage to be measured is 99 volts. Automatic ranging optional. Response time seconds. Input impedance megohms shunted by pf.

5.4.3 Mechanical Design

Standard 6-foot racks accepting 19 inch drawers are used. TRW standards are used where applicable.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The central sequencing and command unit test set operates from a power source as specified below:

Voltage	115 ± 10 VAC
Frequency	60 ± 1 cps
Phase	Single
Maximum Average Power	watts

6.2 Unit Supply

The central sequencing and command unit is powered by a 4096 cps square wave. Low level: 0V; high level: +50V. Average power delivered: watts.

6.3 Service Condition

Temperature	0 degree C to 40° C
Humidity	30 to 70 per cent

7. PARAMETERS

7.1 Input

7.1.1 Timing and Synchronization

7.1.2 Signal Levels

7.2 Output

7.2.1 Timing and Synchronization

7.2.2 Signal Levels

8. CONSTRAINTS

Maximum use of any developed hardware.

9. INTERFACES

9.1 Command Encoder

The command encoder meets the input requirements of the central sequencing and command unit.

9.2 Outputs

9.2.1 Discrete Commands

9.2.2 Quantitative Commands

POWER SUBSYSTEM
OSE/VS-4-460

1. SCOPE

1.1 Content

This document defines the general requirements, equipment list and applicable documents for power subsystem MOSE required for the assembly, handling, protection, transport, shipment and storage of the power subsystem equipment used in the Voyager program.

1.2 Identification

The models covered by this document conform to the requirements delineated herein and are identified as the VS-4-460 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

JPL

JPL Specification 20064A -	General Specification for Packing Flight Equipment for Shipment within CONUS.

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

PPP-B-601A Amend. 2 16 August 1963	Boxes, Wood, Cleated - Plywood
MIL-P-116D	Preservation, Methods of
MIL-M-008090D	
MIL-D-3464B 31 October 1955	Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification
MIL-D-3716A Amend. 2 14 May 1962	Desiccants, Activated for Dynamic Dehumidification

MIL-E-5556B Amend. 1 15 March 1963	Enamel, Camouflage, Quick Dry
MIL-C-9959 Amend. 1 5 February 1963	Container, Flexible, Reusable, Water-Vaporproof

DAC/MSSD

Mechanical Support Equipment and Facilities Manual

3. REQUIREMENTS

The power subsystem MOSE items defined in the following paragraphs are designed to perform their specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

The end items defined within this documentation group are associated with the assembly, handling, positioning, testing, hoisting, shipping, protection, and storage of the power subsystem equipment. The equipment defined below accomplishes these major mechanical handling and support functions.

Table I presents the mechanical OSE for the power subsystem.

Table I. Mechanical OSE for the Power Subsystem

Item No.	Nomenclature
4-460-1	Assembly and handling frame, solar panel segment
4-460-2	Protective cover, solar panel segments
4-460-3	Shipping container, solar panel segments
4-460-4	Handling dolly, solar panel segment
4-460-5	Sling assembly, solar panel segment
4-460-6	Shipping container, battery
4-460-7	Shipping container, power amplifier

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The power subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when the equipment is used to support the power subsystem components. All MOSE coupled to these components are operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is fabricated of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the physical envelopes of the subsystem components.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the power subsystem components and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or over land. This equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured, and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use will there be a degradation in the reliability of the power subsystem equipment.

3.7 Maintainability

The MOSE is constructed so that repairs, adjustments, and overhaul can be readily accomplished by the operating personnel using conventional, general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification, with points, rated loads, hazard warnings, and special instructions be noted.

ASSEMBLY AND HANDLING FRAME, SOLAR PANEL SEGMENT OSE/VS-4-460-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the solar panel segment assembly and handling frame.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE Power Subsystem
OSE/VS-4-460-2	Protective Cover, Solar Panel Segment
OSE/VS-4-460-3	Shipping Container, Solar Panel Segment
OSE/VS-4-460-4	Handling Dolly, Solar Panel Segment
OSE/VS-4-460-5	Sling Assembly, Solar Panel Segment

3. FUNCTIONAL REQUIREMENTS

The solar panel segments require support during fabrication, assembly, testing, shipping and storage until installation on the spacecraft. The panels are protected from handling and transport racking loads. Panel cleanliness is maintained and interfacing machined surfaces are protected. The solar panel segments require orientation through 360 degrees around one axis for assembly and test functions.

4. DESIGN REQUIREMENTS

4.1 Minimum Dimensions

A wedge shaped panel segment size of 116 x 64 inches overall dimensions is accommodated by the frame.

4.2 Locking

When the frame and solar panel segment structure is mounted in the trunnions of the handling dolly (VS-4-460-4) it is capable of positive locking to a desired orientation. The locks are adjustable to permit panel orientation from 0 to 360 degrees in 10-degree increments.

4.3 Load

The frame is capable of carrying the weight of a fully assembled solar panel segment and provides sufficient stiffness to preclude any racking or bending loads being imposed on the solar panel segment assembly. The design meets load factors in accordance with OSE/VS-2-110.

4.4 Shackles

A minimum of four MS standard hoisting shackles are attached for handling the frame.

5. EQUIPMENT DESCRIPTION

5.1 General

The handling frame consists of a welded aluminum tube frame with mounting pads for tooling clamps to clamp the solar panel segment structure to the frame. The clamps are padded to prevent abrasion or scratching of the solar panel structure at its hard points. The tooling clamps are either vacuum or quick release mechanical type. Two trunnion shafts provide means for mounting the frame in the handling dolly. A gear tooth profile is attached to one trunnion shaft for indexing the panels in 10-degree increments when mounted in the handling dolly trunnions. Diagonal sway braces are provided to reduce racking distortion in the panel plane. This design concept is shown in Figure 1.

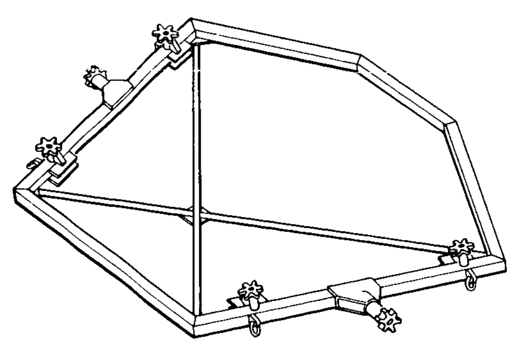


Figure 1. Assembly and Handling Frame, Solar Panel Segment

5.2 Interface Definition

The assembly and handling frame interfaces with the solar panel segment structure and mechanically function in conjunction with the solar panel sling assembly (VS-4-460-5), the solar panel segment shipping container (VS-4-460-3), and the solar panel segment handling dolly (VS-4-460-4). It is also used in conjunction with the solar panel segment protective covers (VS-4-460-2) but is not physically attached to them.

PROTECTIVE COVER, SOLAR PANEL SEGMENT
OSE/VS-4-460-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the solar panel segment protective cover.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE, Power Subsystem
OSE/VS-4-460-1	Assembly and Handling Frame, Solar Panel Segment
OSE/VS-4-460-3	Shipping Container, Solar Panel Segment
OSE/VS-4-460-4	Handling Dolly, Solar Panel Segment

3. FUNCTIONAL REQUIREMENTS

The solar panel segment protective cover provides physical protection to the solar cells on the solar panel structural segment assemblies during handling, storage, shipment, and spacecraft integration procedures.

4. DESIGN REQUIREMENTS

4.1 Mounting

The protective cover is mounted to the solar panel segment structural frame.

4.2 Load

The protective cover withstands a 50 g impact load for a duration of 10 msec, and meets the design load factors contained in OSE/VS-2-110.

4.3 Mechanical

The cover is compatible with, but not mated to the solar panel segments assembly and handling frame (VS-4-460-1). The protective cover does not interfere with the functions of the solar panel segment handling dolly (VS-4-460-4).

4.4 Deflection

The protective cover does not deflect into the solar panel cells when subjected to an imposed load.

4.5 Transparency

The material employed is transparent enough to permit limited inspection of the solar cell assemblies.

4.6 Condensation

No moisture condensation is permitted within the protective cover.

4.7 Altitude

The protective covers are capable of performing their functions at all altitudes from sea level to 20,000 feet.

4.8 Reusability

The protective cover is reusable.

5. EQUIPMENT DESCRIPTION

5.1 General

The protective cover is a ribbed structure fabricated from clear thermoplastic polyvinyl-butyrate material. It has a neoprene weather strip seal. Clearance between the solar cells and the protective cover will be a minimum of 2 inches. The protective cover is bolted to the solar panel on the solar cell side and remains in place during shipping, storage, installation, testing, and spacecraft integration. This design concept is shown in Figure 1.

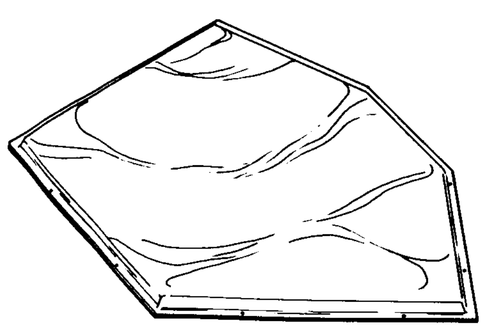


Figure 1. Protective Cover, Solar Panel Segment

5.2 System Interface

The protective cover is used in conjunction with the following: assembly and handling frame, solar panel segment (VS-4-460-1); shipping container, solar panel segment (VS-4-460-3); and handling dolly, solar panel segment (VS-4-460-4).

SHIPPING CONTAINER, SOLAR PANEL SEGMENT
OSE/VS-4-460-3

1. SCOPE

This document defines the functional and design requirements and equipment description for the solar panel segment shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE Power Subsystem
OSE/VS-4-460-1	Assembly and Handling Frame, Solar Panel Segment
OSE/VS-4-460-2	Protective Covers Solar-Panel Segment
OSE/VS-4-460-5	Sling Assembly, Solar Panel Segment

3. FUNCTIONAL REQUIREMENTS

The solar panel segment shipping container provides environmental protection for the solar panel segments during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the solar panel segments from physical damage during surface and air transportation and during periods of storage.

4.2 Weight and Size

The weight and size are minimum within the constraints of providing the desired protection. Each container accommodates two solar panels with their respective solar panel segment assembly and handling frames (VS-4-460-1) and protective covers (VS-4-460-2) installed.

4.3 Environment

4.3.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the solar panels to less than that occurring during flight environments.

4.3.2 Temperature

The temperature within the container is maintained within a range of 0 to 130°F.

4.3.3 Humidity

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.4 Altitude

The shipping container is capable of operating at altitudes consistent with commercial and military air transportation.

4.5 Venting

When required, venting provisions are incorporated for air-transport operating modes for altitudes from sea level to 20,000 ft. Venting occurs through desiccants.

4.6 Transportability

The container is capable of being transported by rail, truck, or air.

4.7 Reusability

The container is reusable.

4.8 Hoisting

Hoist points are appropriately located.

4.9 Attachments

The device used to attach the container cover to the skid is capable of withstanding normal road hazards and environments without degradation.

4.10 Fork Lift and Tiedowns

The container includes tiedown rings for truck and air transportation and accommodates a standard fork lift.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping equipment consists of a shock mitigating system, frame and an environmentally controlled shipping container.

The shipping container is made of modular aluminum-faced honeycomb panels capable of withstanding altitudes from sea level to 20,000 feet. This container is painted white and the paint conforms to MIL-E-5556 with an emissivity of 0.8.

The shipping container consists of a hood and a skid with forklift capabilities. The hood is secured to the skid by latching devices and has a perimeter of weather stripping to insure an environmental seal. The hood has a relief valve to allow for pressure venting and dry nitrogen or dry air purging. A static desiccant system containing desiccant conforming to MIL-D-3716 type IV is mounted on the hood to ensure a relative humidity of less than 20 percent.

The skid consists of two aluminum honeycomb panels separated by a shock-mitigating system. The shock system is either polyurethane foam or rubber shear mounts bonded or mounted to the honeycomb skid panels. The skid contains aluminum channels for fork-lift capability. Aluminum brackets capable of holding two solar-panel segments with their attached solar panel segment assembly and handling frames (VS-4-460-1) and protective covers are mounted to the skid. The solar-panel segments are held in the aluminum brackets by the assembly and handling frames which are bolted to the aluminum brackets. Tiedown fittings are provided on the skid.

Prior to shipment, the container is purged to a dew point of $0^{\circ}F$. If an unpressurized airplane is used for transportation, the container is purged to a dew point of $-32^{\circ}F$.

The modular honeycomb panels including the specified mountings (desiccant canister and relief valve) are hermetically sealed. The equipment concept is shown in Figure 1.

5.2 Equipment Interface

The shipping container accommodates the solar panel segment assembly and handling frame (VS-4-460-1), the solar panel segment protective covers (VS-4-460-3) and the solar panel sling assembly (VS-4-460-5).

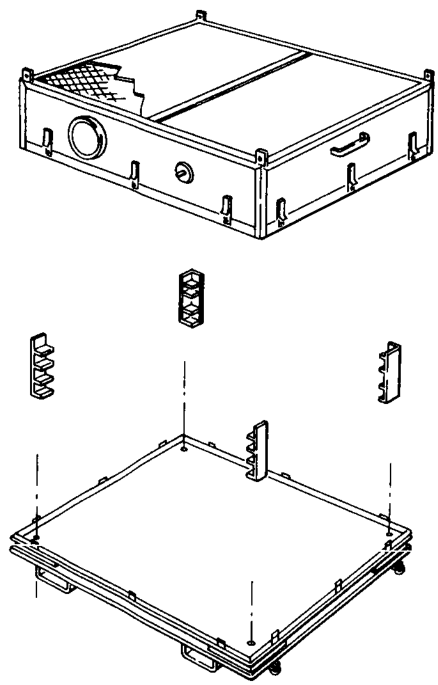


Figure 1. Shipping Container, Solar Panel Segment

HANDLING DOLLY, SOLAR PANEL SEGMENT
OSE/VS-4-460-4

1. SCOPE

This document defines the functional and design requirements and equipment description for the solar panel segment handling dolly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE Power Subsystem
OSE/VS-4-460-1	Assembly and Handling Frame, Solar Panel Segment
OSE/VS-4-460-3	Protective Cover, Solar Panel Segment
OSE/VS-4-460-5	Sling Assembly, Solar Panel Segment

3. FUNCTIONAL REQUIREMENTS

The solar panel segment handling dolly provides support, access, and mobility for the solar panels when installed in their assembly and handling frames (VS-4-460-1) during assembly, power tests, inspection, maintenance, and intrashop transportation.

4. DESIGN REQUIREMENTS

4.1 Mobility

The dolly is designed to conform to Type I, Class I, mobility requirements of MIL-M-008090D. The dolly running gear consists of four swivel casters with swivel locks and step on parking brakes. Shock absorbers and a spring-suspension system are not required. The caster wheels are made of rubber or equivalent tread material for shock-absorbing purposes. Towbars are not provided.

4.2 Loads

The dolly is capable of supporting a solar panel when mounted in its assembly and handling frame (VS-4-460-1) and covered with the solar panel protective covers (VS-4-460-3). All other load factors are in accordance with OSE/VS-2-110.

4.3 Rotational Capabilities

The dolly provides hand powered roll capabilities with limits of ±180 degrees. Positive locking provisions are provided to position the solar panel in an orientation of 0 to 180 degrees in 10-degree increments.

4.4 Dimensional Characteristics

The dolly is capable of accommodating the length, width, and height of the solar panel assembly in the handling frame. The loaded dolly is capable of passing through a 7-foot door.

4.5 Stability

The center of gravity of the loaded dolly is low enough and the under carriage frame wide enough, to create a stable condition during solar panel rotation and during normal factory movement.

5. EQUIPMENT DESCRIPTION

5.1 General

The handling dolly consists of a rectangular aluminum under-carriage frame which supports an A-frame at each end. The base frame has four swivel casters mounted on it. Each A-frame contains at its apex a split pillow block assembly. A pawl, or spring-loaded pin, is mounted within the pillow block and interface with the trunnion shaft gear of the solar panel assembly and handling frame (VS-4-460-1). A handwheel is attached to the trunnion shaft for manual rotation of the solar panel to any desired orientation, with positive lock positions every 10 degrees. The spring loaded pin, working against the trunnion shaft gear, provides the means for position locking. The handwheel is attached to the dolly frame by means of a retaining chain. The upper half of the trunnion pillow blocks includes vertical strut members which tie to a crossbar to provide additional rigidity to the assembled dolly and afford protection to the solar panel segment. The handling dolly concept is shown in Figure 1.

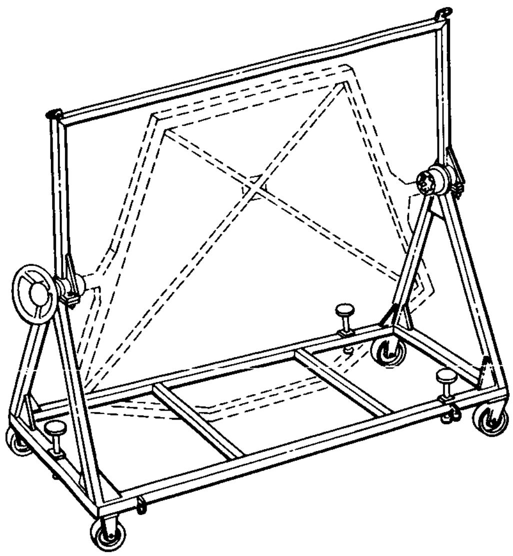


Figure 1. Handling Dolly, Solar Panel Segment

5.2 Interface Definition

The dolly pillow blocks interface with the trunnion shafts of the solar panel assembly and handling frame (VS-4-460-1). The dolly functions in conjunction with the solar panel sling assembly (VS-460-5).

SLING ASSEMBLY, SOLAR PANEL SEGMENT
OSE/VS-4-460-5

1. SCOPE

This document defines the functional and design requirements and equipment description for the solar panel segment sling and assembly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE, Power Subsystem
OSE/VS-4-460-1	Assembly and Handling Frame, Solar Panel Segment
OSE/VS-4-460-2	Protective Cover, Solar Panel Segment
OSE/VS-4-460-3	Shipping Container, Solar Panel Segment
OSE/VS-4-460-4	Handling Dolly, Solar Panel Segment

3. FUNCTIONAL REQUIREMENTS

The solar panel segment, in its assembly and handling frame (VS-4-460-1), requires hoisting and handling during various fabrication, testing, and installation operations. The solar panel segment and its handling frame are also hoisted and positioned in the handling dolly (VS-4-460-4) for assembly and checkout. Following fabrication and test, the combined assembly of the solar panel segment, its handling frame, and its protective covers (VS-4-460-2) is installed in the shipping container (VS-4-460-3). The solar panel sling assembly are used to hoist the shipping container hood and place it on the skid section, prior to shipment.

4. DESIGN REQUIREMENTS

A general-purpose sling and spreader bar is designed to comply with the functional requirements specified in Paragraph 3. above. The following design requirements apply.

4.1 Load

The sling lifts the combined maximum load of the solar panel segment installed in its handling frame with protective covers attached, the shipping container hood, or the solar panel segment handling dolly, whichever is heaviest. The anticipated maximum load is estimated at 300 pounds.

4.2 Load Factors

The load and handling factors are in accordance with OSE/VS-2-110.

4.3 Fasteners

Fasteners are of the quick release type and are attached to the cable-sling ends. All end fittings are vinyl-coated to prevent abrasion to the solar panels.

4.4 D-Ring

An MS standard D-ring is used in the sling assembly.

5. EQUIPMENT DESCRIPTION

5.1 General

The solar panel sling consists of an MS standard D-ring supporting four stainless steel wire ropes. Each cable has standard quick release end fittings to be attached to pickup points on the various items to be hoisted. They are vinyl-coated for protection against abrasion. This sling assembly design concept is shown in Figure 1.

5.2 Equipment Interface

The sling mechanically interfaces with the solar panel assembly and handling frame (VS-4-460-1), the solar panel segment protective cover (VS-4-460-2), the solar panel shipping container hood (VS-4-460-3), and the solar panel segment handling dolly (VS-4-460-4).

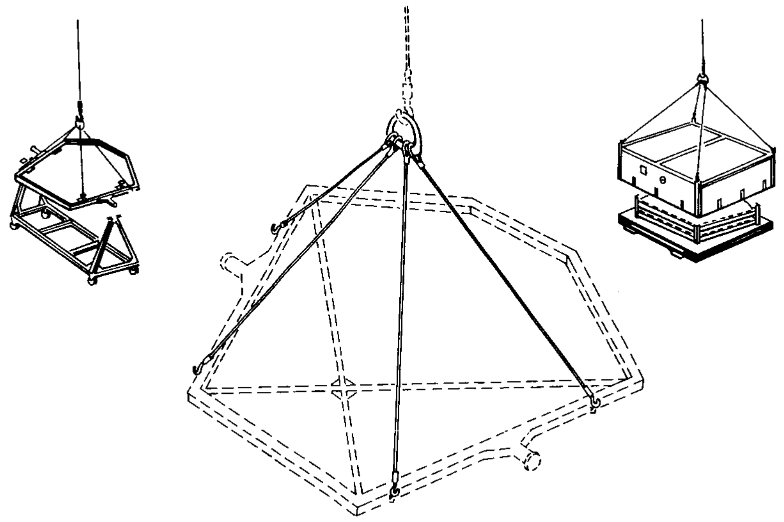


Figure 1. Sling Assembly, Solar Panel Segment

SHIPPING CONTAINER, BATTERY
OSE/VS-4-460-6

1. SCOPE

This document defines the functional and design requirements and equipment description for battery shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE, Power Subsystem
OSE/VS-3-140-5	Shipping Container Group, Standard Modules

3. FUNCTIONAL REQUIREMENTS

The battery shipping container provides environmental protection for battery units during transportation and storage.

4. DESIGN REQUIREMENTS

The design requirements of this container are equivalent to MOSE item OSE/VS-3-140-5. The batteries are shipped in a dry condition.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping container consists of a shock-mitigating system, an environmental cover (barrier material), an intermediate container, and an exterior shipping container. The battery is completely encapsulated in 2 pound-density polyethylene foam, which nests the battery in such a manner that the load is distributed equally.

The encapsulated battery is placed in a reusable thermoplastic acrylic butadiene styrene container. The container has hinges made from polypropylene material with nickel-plated hasps.

The battery, prior to encapsulation, is placed in a barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963, with a water-vapor transmission value of 0.05 to 0.085 $g/100\ in^2/24\ hr$. The barrier material is made of scrim foil, nylon-reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. The barrier material contains desiccant bags conforming to MIL-D-3464B with a humidity indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 30 percent. Prior to shipment, the barrier material is purged with dry nitrogen or dry air to a $0^{\circ}F$ dew point, desiccated, and evacuated.

The battery, encapsulated in foam and enclosed in its barrier material and intermediate ABS container, is placed in a reusable cleated-plywood exterior container conforming to PPP-B-601. The 2-inch void separating the interior container from the exterior container is filled with 2 pound-density polyethylene foam. This shipping container design concept is shown in Figure 1.

5.2 Interface Definition

The container interfaces with the Voyager spacecraft battery. No other interface with MOSE exists.

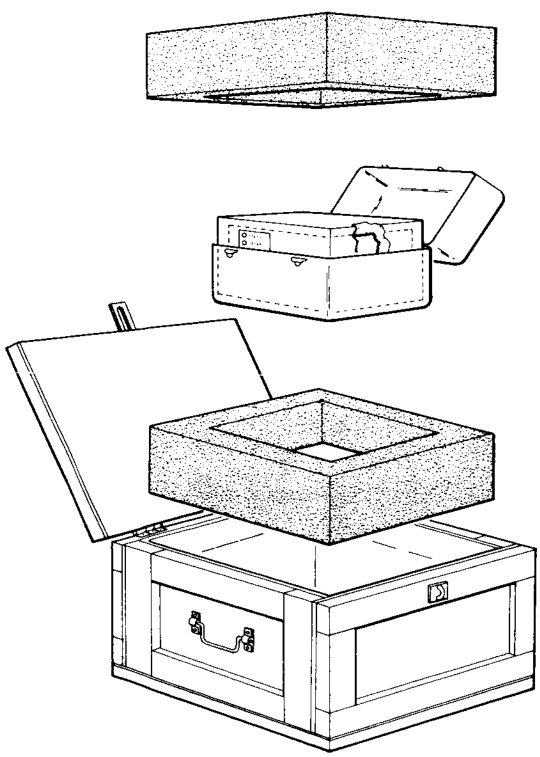


Figure 1. Shipping Container, Battery

SHIPPING CONTAINER, POWER AMPLIFIER
OSE/VS-4-460-7

1. SCOPE

This document defines the functional requirements and equipment description for the power amplifier shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-460	Voyager OSE Power Subsystem
OSE/VS-3-140-5	Shipping Container Group, Standard Modules

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the power amplifier during transportation and storage.

4. DESIGN REQUIREMENTS

The design requirements of this container are equivalent to MOSE item OSE/VS-3-140-5.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping container consists of a shock mitigating system, an environmental cover (barrier material), an intermediate container and an exterior shipping container. The power amplifier is completely encapsulated in polyethylene or polyurethane foam, which nests the power amplifier in such a manner that the load is distributed equally.

The encapsulated power amplifier is placed in a reusable, thermoplastic, acrylic butadiene styrene container. The container contains hinges made from polypropylene material with nickel-plated hasps.

The power amplifier, prior to encapsulation, is placed in a barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963 with a water vapor transmission value of 0.05 gm to 0.085 gm/100 in^2/24 hours. The barrier material is made of one of the following materials: scrim foil, nylon-reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. The barrier material contains desiccant bags conforming to MIL-D-3464B, with a humidity indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 20 percent. The desiccant quantity required is calculated in accordance with MIL-P-116D, paragraph 3.5.6. Prior to shipment, the barrier material is purged with dry nitrogen to a $0^{\circ}F$ dew point, desiccated, and evacuated.

The power amplifier, encapsulated in foam and enclosed in its barrier material and intermediate ABS container, is placed in a reusable cleated plywood exterior container conforming to PPP-B-601. The 2-inch void separating the interior container from the exterior container is filled with foam (polyethylene or polyurethane).

5.1 System Interface

The container has no physical or electrical interface with other operational support equipment.

SOLAR PANEL UNIT TEST SET
OSE/VS-4-461-1

1. SCOPE

This document covers the requirements for the solar panel unit test set used to test and monitor the performance of the solar panels for the Voyager spacecraft program.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 General

The solar panel UTS provides the facilities for checking and recording the performance of complete solar panels, and functional portions of each panel, down to the series string level. The UTS provides all necessary stimuli, loads, and instrumentation to perform these functions during the various phases of testing imposed on the solar panels.

The solar panel UTS in no way influences the parameters being measured on the solar panel under test (such as magnetic fields). In addition, the UTS provides fail-safe protection against solar panel damage due to any UTS malfunction.

Human engineering principles are utilized in order to achieve simple and logical operational modes. In addition, chassis layout and modular construction facilitate ease at maintenance.

3.2 Testing Capabilities

The solar panel UTS has the capabilities of measuring the following:

a) I-V characteristics under artificially illuminated conditions, at various temperatures

b) I-V characteristics under "dark current" conditions

c) Insulation resistance between solar cells and substrate

d) Forward and reverse characteristics of blocking diodes

e) Magnetic field of solar panel while illuminated. (Measurements made with separate magnetometer, UTS only provides illumination and loads.)

Additionally, the solar panel UTS provides a calibrated illumination source and positioning device for the solar panel.

4. DESIGN REQUIREMENTS

4.1 Instrumentation

4.1.1 Readout Devices

a. Voltage and Current

Voltage and current measurements are made using a digital voltmeter and appropriate shunts. These readings are accurate to within ±0.5 percent.

b. I-V Plotting

An x-y plotter is utilized for plotting the I-V characteristics of the unit under test. These plots are accurate to within ±1.0 percent.

c. Temperature Indicator

The solar panel under test and the calibration standard are temperature monitored. The readings of temperature are accurate to within ±0.5°C.

d. Color Temperature

The color temperature monitoring equipment has a reading accuracy to within ±50°K, and a repeatability of ±5°K.

e. Resistance Meter

Insulation resistance measurements are made to an accuracy of ±5 percent.

f. Multichannel Trace Recorder

A multichannel trace recorder is provided to permit simultaneous recording of significant parameters (temperature, voltage, current, etc.) during an entire test cycle. The calibration accuracy of this device is ±5 percent.

4.1.2 Variable Load

a. Power Rating

The variable load has a power dissipative capacity of 200 watts minimum.

b. Resistance

The variable load has a resistance range from 500 ohms at the high end, to less than 0.1 ohm at the low end (approaching zero ohms).

c. Power Supplies

Power supplies are provided with appropriate voltage and current range to facilitate dark current solar panel and diode characteristics tests.

4.2 Illuminator and Test Stand

4.2.1 Intensity

The intensity of illumination at the panel surface is an equivalent air mass zero intensity of 140 mw/cm^2.

4.2.2 Uniformity of Intensity

The illumination intensity is uniform over the test area within ±5 percent when measured with the calibration cell in the plane of the solar panel to be tested.

4.2.3 Test Area

The illuminator is capable of illuminating an area equal to at least one half of a solar panel, consisting of three series string circuits.

4.2.4 Stability of Intensity

Voltage regulators are provided to control total variations of intensity due to line transients and fluctuations within ±2 percent for a one hour period.

4.2.5 Test Stand

The test stand is the main mechanical structure for supporting the panel mount and light source. It is of rigid construction to ensure accurate and repeatable positioning of the solar panel under test and of the calibration cell.

4.2.6 Panel Mount

The panel mount consists of an accurate indexed horizontal x-y positioning carriage upon which the solar panel to be tested is mounted. The mounting interface of the carriage is appropriate for the type of handling jig to be used with the solar panel. The panel mount also includes an indexed vertical positioning mechanism for adjusting the intensity of illumination. The vertical adjustment is smoothly controllable over a distance of at least 12 inches under the light source.

4.2.7 Calibration Cell Positioner

Provision is made for indexing the water cooled calibration cell anywhere within the illumination area by means of an indexed x-y pantograph. The calibration cell is used in the horizontal plane at which the solar panel is illuminated, and removed from the illuminated area during actual testing.

4.3 Temperature Control Equipment and Protective Interlocks

4.3.1 Air Conditioner

The air conditioner has sufficient cooling capacity to maintain the solar panel under test at a minimum temperature of $30^\circ C \pm 5^\circ C$, under a maximum ambient temperature of $95^\circ F$ with a 60 percent maximum relative humidity, and under the highest intensity of irradiation used for solar panel testing. The solar panel temperature is manually controllable for higher temperatures required for testing.

4.3.2 Chiller Unit

The chiller unit provides cold water circulation for maintaining the illuminator chamber at a nominal temperature of 35°C maximum, and the calibration solar cell at the same temperature as the solar panel under test. Control for the latter is automatic.

4.3.3 Protective Interlock

Protective interlocking circuits are employed for automatic turn off (or prevent the turn on) of the illuminator light source if either the air conditioner or chiller unit fails to maintain specified temperature control.

5. FUNCTIONAL DESCRIPTION

The solar panel UTS is comprised of standard commercial and specially designed equipment for testing and monitoring the performance of the Voyager solar panels.

A functional block diagram appears in Figure 1.

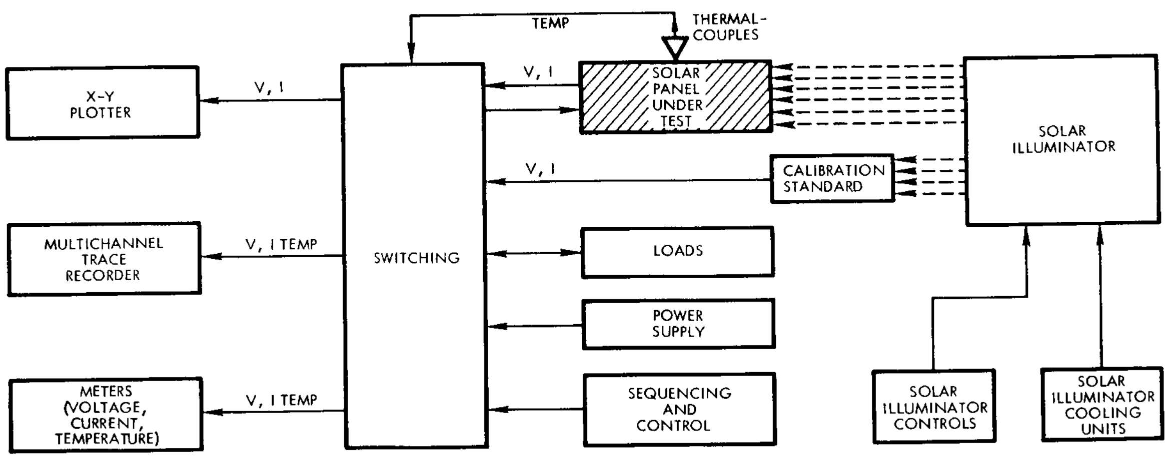


Figure 1. Solar Panel Unit Test Set, Block Diagram

Figure 2 shows the physical configuration of the test set. It consists of three racks of equipment, an illuminator, consisting of a light source and test stand, and cooling equipment. The racks are functionally grouped as follows:

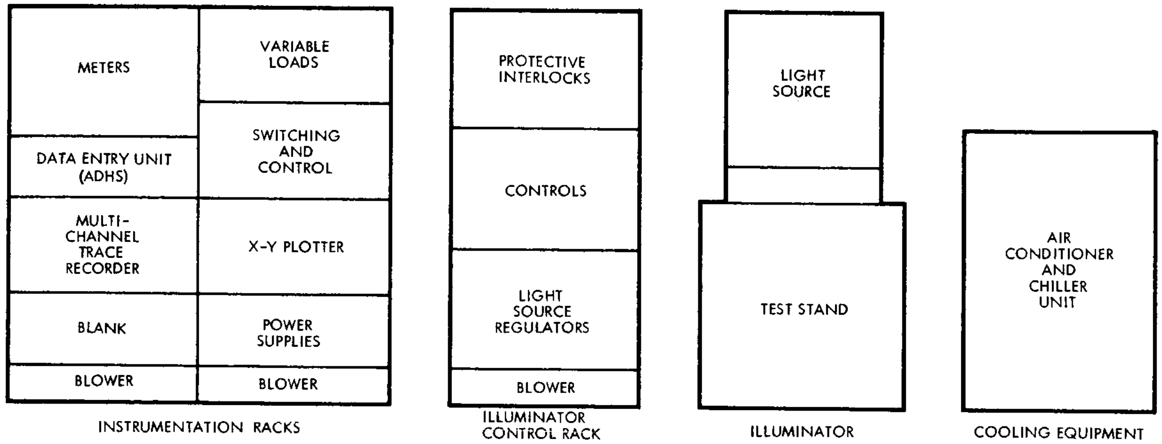


Figure 2. Solar Panel Unit Test Set, Rack Layout

a) Two instrumentation racks, containing voltage, current, temperature, color temperature and resistance meters; a multichannel trace recorder; an x-y plotter; power supplies; variable loads; and appropriate switching and controls to perform the specified tests.

b) One illuminator control rack, containing light source voltage regulators, protective interlock circuitry for light source, air conditioner, and chiller unit; and air conditioner and chiller temperature controls (both manual and automatic).

c) The illuminator is made up at two basic parts: the test stand, and the light source. The test stand supports the light source, the panel mount, and the calibration cell positioning mechanism. The light source consists of an array of tungsten iodine quartz lamps mounted above a high efficiency light diffusion plenum chamber. Blowers and circulating chilled water are employed to cool the lamps, sockets, plenum and calibration cell. Air conditioning is utilized for maintaining the solar panel at the proper temperature.

d) The cooling equipment consists of two separate units: the air conditioning unit, and the chiller unit.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The solar panel UTS operates from a power source as specified in the following paragraphs.

6.1.1 Instrumentation Racks

Voltage	115 ±10 vac
Frequency	60 ±1 cps
Phase	Single
Power	5 kw

6.1.2 Illuminator and Illuminator Control Rack

Voltage	230 ±15 vac
Frequency	60 ±1 cps
Phase	Single
Power	-

6.1.3 Air Conditioner and Chiller Unit

Voltage	230 ±15 vac
Frequency	60 ±1 cps
Phase	Single
Power	-

6.2 Operational Environment

The solar panel UTS operates satisfactorily within the following ambient environmental conditions:

Temperature	60 to 90°F
Relative Humidity	0 to 50 per cent

7. PARAMETERS

The following is a tabulation of the critical parameters of the solar panel measured by the UTS:

a) I-V characteristics - illuminated

b) I-V characteristics - "dark current" conditions

c) Insulation resistance between solar cells and substrate

d) Forward and reverse characteristics of blocking diodes

e) Magnetic field produced by solar panel under illuminated conditions.

8. CONSTRAINTS

For practical purposes, the compressor portions of the cooling equipment associated with the solar panel UTS are located outside of the test area (preferably in the out-of-doors). This requires routing of special refrigerant carrying lines from the compressors to the cooling units. It may also be desirable to draw in and exhaust to the outside the large volume of air required for the air conditioning, thus requiring large diameter ducts or flexible hoses to be installed.

9. INTERFACES

The main interface existing between the solar panel UTS and the solar panel under test is the output terminals of each series string. In addition to these, access to the blocking diode terminals and to the substrate surface is required for complete testing.

POWER INVERTER UNIT TEST SET
OSE/VS-4-461-2

1. SCOPE

This document covers the requirements for the power inverter unit test set used to evaluate the performance of the following types of Voyager power inverters:

a) Main AC power inverter

b) 410 cycle single phase power inverter

c) 820 cycle two phase power inverter.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

<u>TRW</u> 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The power inverter unit test set is used to perform tests on the main AC power inverter unit, the 410 cycle single phase inverter unit, and the 820 cycle two phase inverter unit.

3.2 Test Functions

The power inverter unit test set provides the following functions to either the main AC power inverter, the 410 cycle single phase inverter or the 820 cycle two phase inverter:

a) Variable DC input power

b) Variable output loads.

4. DESIGN REQUIREMENTS

To provide a complete test of the main AC power inverter, the 410 cycle single phase inverter, and the 820 cycle two phase inverter, the power inverter unit test set performs the following specified tests:

a) Output voltage variation as function of input DC variations, load constant

b) Output voltage variation as function of load variation

c) Examination of output wave shape as function of load.

5. FUNCTIONAL DESCRIPTION

The test set is comprised of standard commercial test equipment and special equipment designed for testing functions of the three types of Voyager power inverters.

Figure 1 is a functional block diagram of the test set.

Figure 2 is a drawing of the configuration of the test set.

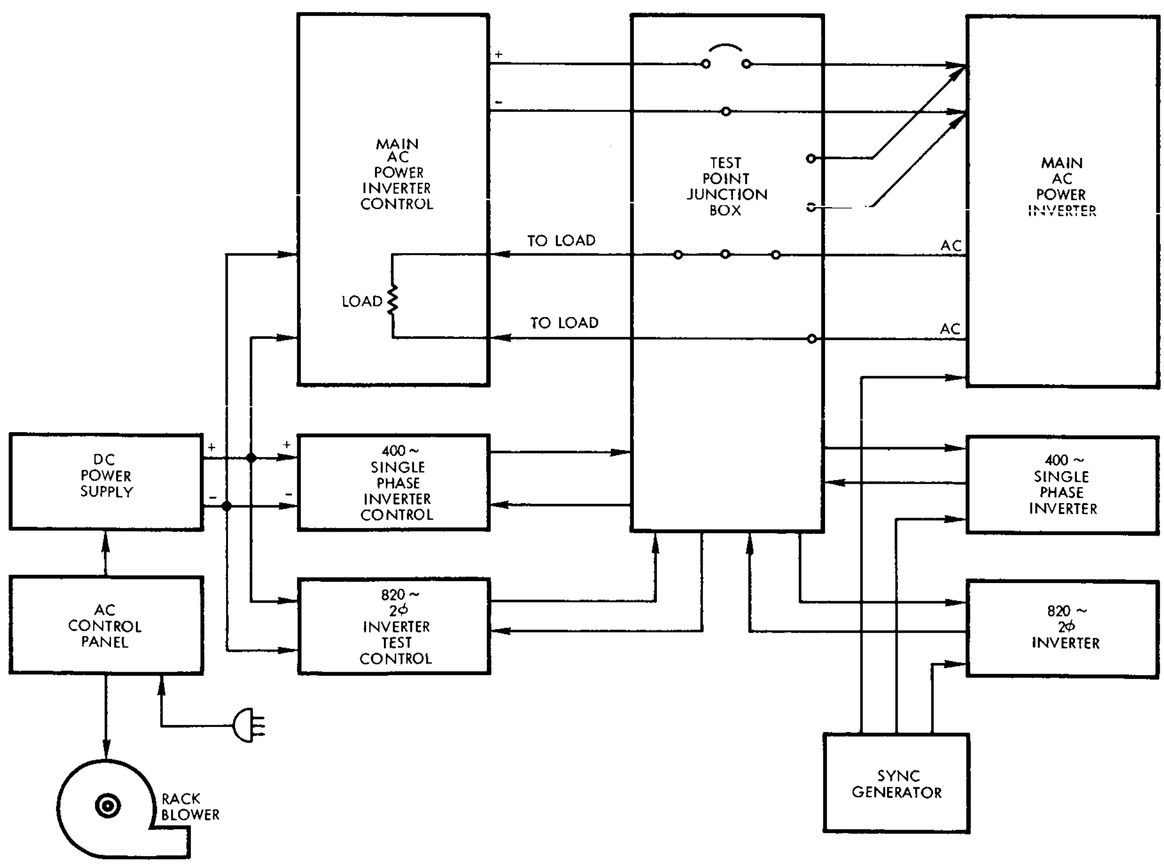


Figure 1. Power Inverter Unit Test Set, Block Diagram

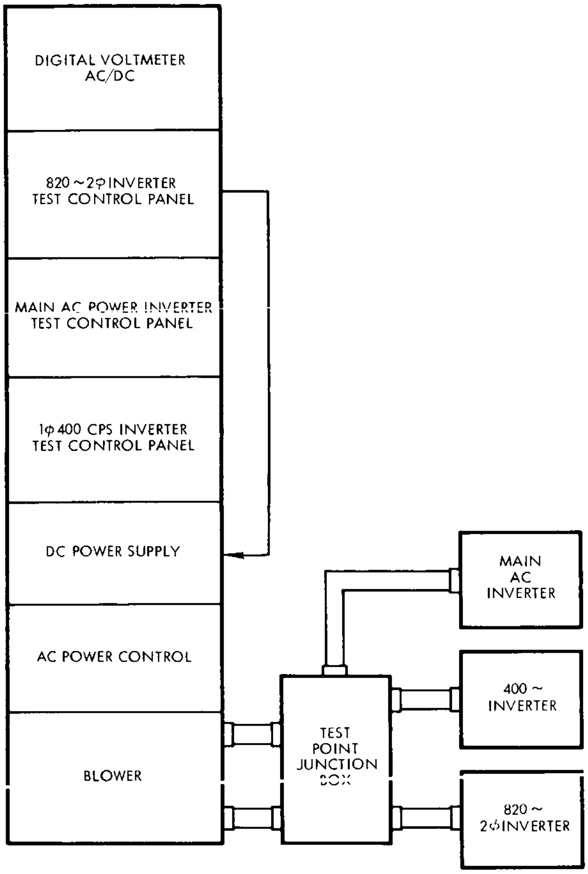


Figure 2. Power Inverter Unit Test Set, Rack Layout

5.1 Test Equipment Description

5.1.1 Panel #1, Rack Blower

This drawer provides cooling air for the inverter loads and the DC power supply.

5.1.2 Panel #2, AC Power Control

This is a panel with a master AC power control switch.

5.1.3 Panel #3, DC Power Supply

Commercial rack mounted DC power supply capable of supplying the inputs to the inverters either singly or simultaneously.

5.1.4 Drawer #4

This drawer provides the 410 cycle single phase inverter input and output control switches and load switches. Varying loads of zero, 50, 75, and 100 percent are provided by switch selection, although only a single load resistor is depicted in Figure 2.

5.1.5 Drawer #5

This drawer provides the main AC inverter input and output control switches and load switches. Varying loads of zero, 50, 75, and 100 percent are provided by switch selection.

5.1.6 Test Point Junction Box #6

This junction box provides test points for accurate voltage and current capital equipment meters. This is a junction box rather than a rack panel for convenience. The table or desk which support the unit in test supports the capital equipment test meters and the test box, which avoids long leads from the test points to the test meters.

5.1.7 Digital Voltmeter

5.2 Commercial Test Equipment

The following items of equipment, the functions of which are discussed above, are standard commercial equipment:

a) DC power supply:

b) Digital voltmeter:

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The test set operates from a power source as specified below:

Voltage	115 ±10 volts AC rms
Frequency	60 ±1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 - 90°F
Humidity	Less than 50 percent

7. PARAMETERS

Critical inverter(s) parameters defined by subsystem specifications will be listed in this section.

8. INTERFACES

8.1 Inputs

a) The test set provides a load for the output of the main AC inverter

b) The test set provides a load for the output of the 410 cycle single phase inverter.

8.2 Output Requirements

a) The test set provides to the main AC power inverter and the 410 cycle single phase inverter DC power

b) The test set provides a conversion synchronization signal output to the main AC power inverter

c) The test set provides a conversion synchronization signal to the 410 cycle single phase inverter.

BATTERY CONTROL UNIT TEST SET
OSE/VS-4-461-3

1. SCOPE

This document covers the requirements for the battery control unit test set used to evaluate the performance of the Voyager battery control unit.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

<u>TRW</u> 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 <u>Description</u>

The battery control unit test set is used to perform tests on the battery control unit consisting of:

3.2 <u>Test Functions</u>

This test set provides a DC input and a resistive load which simulates the spacecraft 50 volt ±1 percent power bus to test the capability of the the solar array booster to raise power at its input voltage range to 50 volts. A battery is provided to test the battery charge control circuitry's capability to regulate a test battery similar to the spacecraft battery. The battery supplied is capable of the same charge and discharge rates as the spacecraft battery but is smaller in ampere-hour capacity.

4. DESIGN REQUIREMENTS

To provide a complete test of the battery control unit, the battery control unit test set tests the following:

a) The ability of the battery booster converters to maintain constant output (50 V DC ± 1 percent) as a function of input variation, constant load; input variation, varying load; and input constant, varying load.

b) The ability of the battery charge control converter to charge a simulated spacecraft battery at either of two rates, 2 amps and 4 amps, the latter on command, under varying input/output parameters.

c) The ability of the shunt regulator to regulate the output of the solar array to 50 V DC ±1 percent, under varying input/output parameters.

5. FUNCTIONAL DESCRIPTION

The test set is comprised of standard commercial test equipment and special equipment designed for testing functions of the battery control unit.

Figure 1 is a functional block diagram of the test set.

Figure 2 is a drawing of the configuration of the test set.

5.1 Test Equipment Description

5.1.1 Blower Panel

The blower cools the power supplies, 50 volt bus simulation resistors, and the test battery discharge load bank.

5.1.2 AC Power Control

This panel holds the AC master control switch.

5.1.3 DC Power Supply

The rack mounted commercial power supply provides DC output power to simulate the output of the solar array.

5.1.4 Test Battery Discharge Load

This is a load bank of selectable resistors for discharging the test battery at the initiation of a test.

5.1.5 Control Panel

This provides switches for control of the battery control unit and the test battery.

5.1.6 Test Battery

A battery used to simulate the spacecraft battery to the battery charge control circuits of the Battery Control Unit.

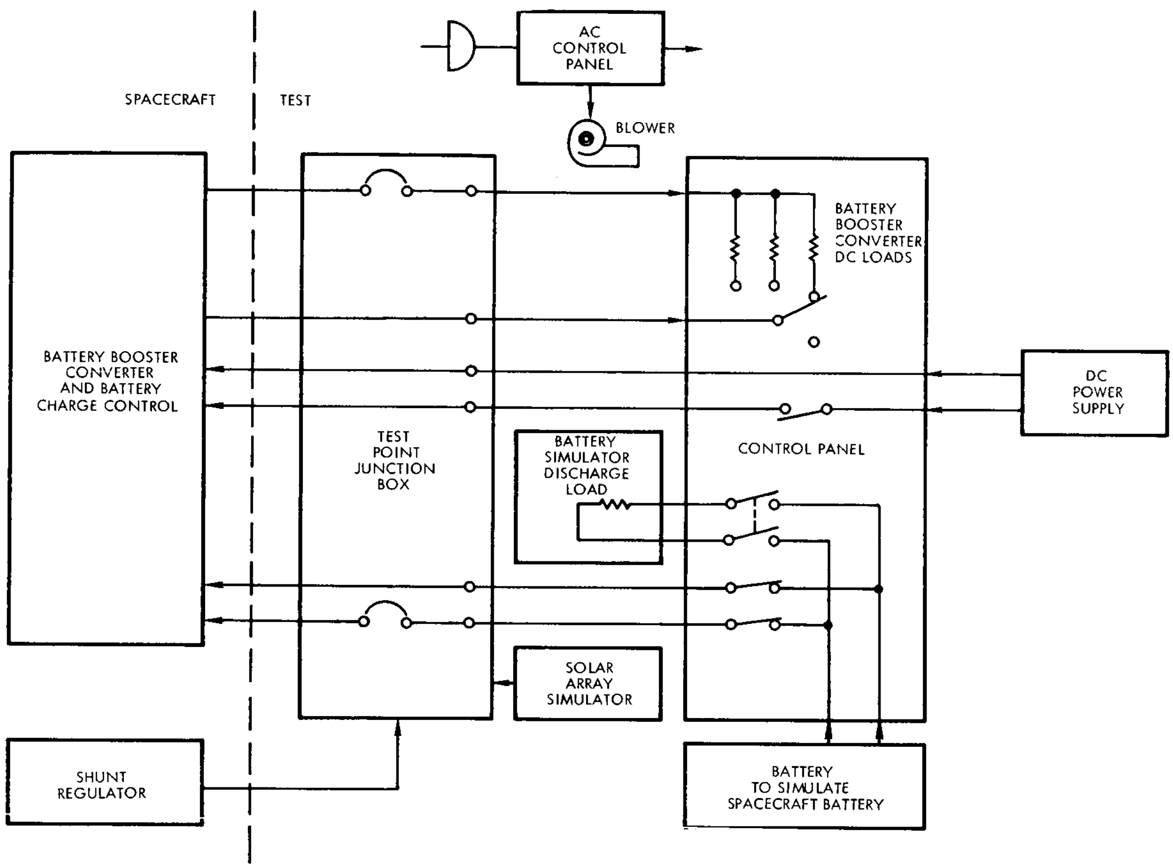


Figure 1. Battery Control Unit Test Set, Block Diagram

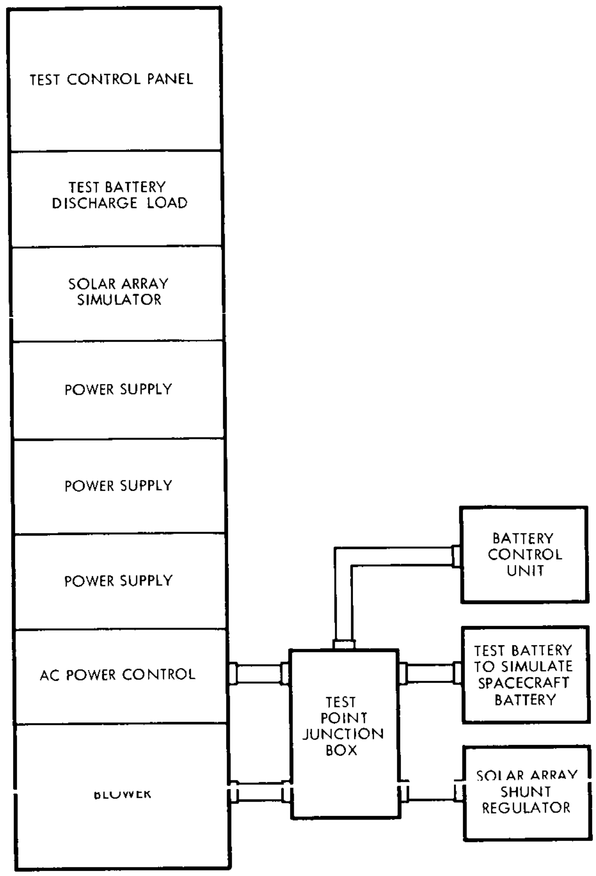


Figure 2. Battery Control Unit Test Set, Rack Layout

5.1.7 Test Point Junction Box

This box provides voltage and current monitoring points by external, accurate meters. Monitoring of every battery control unit connector pin is available.

5.1.8 Digital Voltmeter

5.2 Commercial Test Equipment

The following items of equipment, the functions of which are discussed above, are standard commercial equipment:

a) DC power supplies:

b) Digital voltmeter:

POWER CONTROL ELECTRONIC ASSEMBLY (PCEA) UNIT TEST SET
OSE/VS-4-461-4

1. SCOPE

This document covers the requirements for the control electronic assembly (CEA) unit test set used to evaluate the performance of the Voyager power CEA.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objective and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

3.2 Test Functions

This test set tests the power control unit electronic assembly. It tests the following:

a) The ability to count the external sync input signal down to those frequencies necessary to be distributed to the inverters and battery control for their internal sync

b) The ability to sense loss of external sync. input and to switch to an internal oscillator to provide sync. distribution as in (a) above

c) The ability to encode and output the power subsystem telemetry data

d) The internal oscillator frequency.

4. DESIGN REQUIREMENTS

To provide a complete test of the CEA, the CEA unit test set will perform the following specified tests:

5. FUNCTIONAL DESCRIPTION

The test set is comprised of standard commercial test equipment and special equipment designed for testing functions of the CEA.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The test set operates from a power source as specified below:

Voltage	115 ±10 volts AC rms
Frequency	60 ±1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 - 90°F
Humidity	Less than 50 percent

7. PARAMETERS

Critical inverter(s) parameters defined by subsystem specifications will be listed in this section.

8. INTERFACES

8.1 Inputs

a) The test set provides a load for the output of the battery booster converter circuits

b) The test set provides a battery to simulate the spacecraft battery. This battery receives charge from the battery charge control circuit.

8.2 Outputs

a) The test set provides a charge rate command

b) The test set provides an adjustable DC voltage to simulate the output of the solar array to the battery control unit with characteristics as follows:

Voltage	0 - 60 DC
Watts (max.)	

Figure 1 is a functional block diagram of the test set.

Figure 2 is a drawing of the configuration of the test set.

5.1 Test Equipment Description

5.1.1 Panel #1 AC Power Panel

This panel provides a master AC power switch.

5.1.2 Panel #2 DC Power Supply

This is a modular multiple voltage power supply used to simulate the presence of the following listed spacecraft voltages to the power control unit:

a) 50 volt spacecraft bus

b) Solar array output

e) Battery.

5.1.3 Panel #3, Synchronization Generator

This is a signal generator used to simulate the spacecraft synchronization. Signal input to the power control unit.

5.1.4 Panel #4, Control Panel

A panel with control switches for selecting the inputs which the power control unit under test receives.

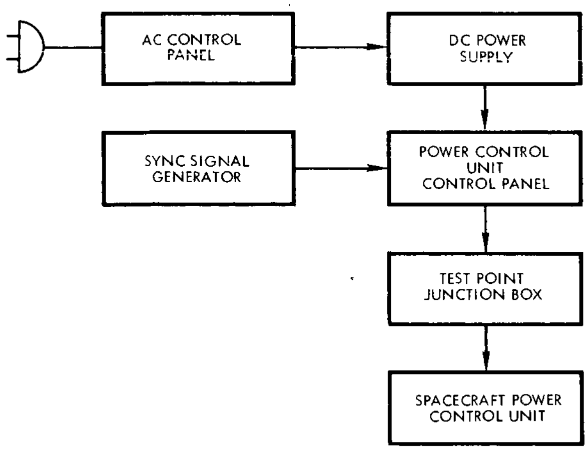


Figure 1. Power Control Electronics Assembly Unit Test Set, Block Diagram

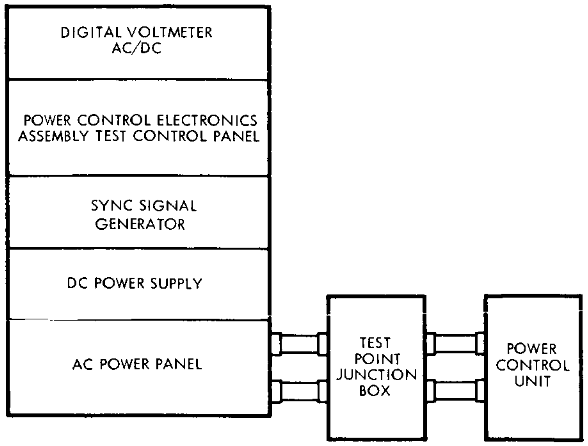


Figure 2. Power Control Electronics Assembly Unit Test Set, Rack Layout

5.1.5 Box #5 Test Point Junction Box

This box provides voltage and current monitoring points for accurate external meters. Monitoring of each power control unit connector pin is available.

5.1.6 Digital Voltmeter

5.2 Commercial Test Equipment

The following items of equipment, the functions of which are discussed above, are standard commercial equipment:

a) Power supplies:

b) Digital voltmeter:

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The test set operates from a power source as specified below:

Voltage	115 ± 10 volts AC rms
Frequency	60 ± 1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 - 90°F
Humidity	Less than 50 percent

7. PARAMETERS

Critical inverter(s) parameters defined by subsystem specifications will be listed in this section.

8. INTERFACES

8.1 Inputs

The test set provides the following:

a) Test points and indicator monitors for the spacecraft power control unit telemetry, status, and command output signals

b) Three monitoring test points for power control unit converter and inverter synchronization signals.

8.2 Outputs

The test set provides the following:

a) Simulated spacecraft DC power

b) Simulated spacecraft solar array output voltage

c) Simulated spacecraft battery voltage output

d) A simulated spacecraft synchronization signal.

BATTERY UNIT TEST SET
OSE/VS-4-461-5

1. SCOPE

This document covers the requirements for battery unit test set used to evaluate the performance of the Voyager battery.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

3.1 Description

The test set includes the following capabilities:

a) Battery under and overvoltage protection, display, and alarm

b) Continuous battery and/or all voltage printout

c) Time of day to one minute accuracy

d) Battery charge and discharge cycle programming, and battery operating capability

e) Individual cell sensing and control to ±1 mv accuracy from 60 to 100°F.

f) Battery sensor functional checkout and display.

3.2 Test Functions

The test set includes as a minimum the following functions:

a) Battery under and over voltage protection, display and alarm

b) Battery charge and discharge cycle programming, and battery operating capability

c) Individual battery cell sensing and control to ±1 mv accuracy from 60 to 100°F

d) Battery sensor functional checkout and display

e) Automatic tester protection and isolation in case of power, component, or battery failures.

4. DESIGN REQUIREMENTS

5. FUNCTIONAL DESCRIPTION

This test set consists of those functional and instrumentation capabilities required to fully test a nickel-cadmium battery if used by a trained battery technician, but is not designed to be "fully safe" since the performance of batteries is not predictable and acceptance testing includes necessary trend surveillance based on experience.

The test set includes as a minimum the following instrumentation and data displays:

a) Continuous battery and/or all voltage printout

b) Precision digital readout with printer output of all battery voltages, currents, and sensor inputs and outputs

c) Visual indication of charge/discharge current and power supply voltage

d) Tester operating time cumulative

e) Time of day to one minute accuracy minimum.

Figure 1 is a functional block diagram of the test set.

Figure 2 is a drawing of the configuration of the test set.

5.1 Test Equipment Description

The consoles contain five special panels, two commercial power supplies, and a modified digital instrumentation and printout system and print chart recording system.

5.1.1 Battery Control Chassis

The battery monitoring panel contains data selection switches and battery voltages and current controls. It also provides monitoring of charge and discharge state, cell mode of operation or battery mode of operation, and contains the operate and standby control for the battery.

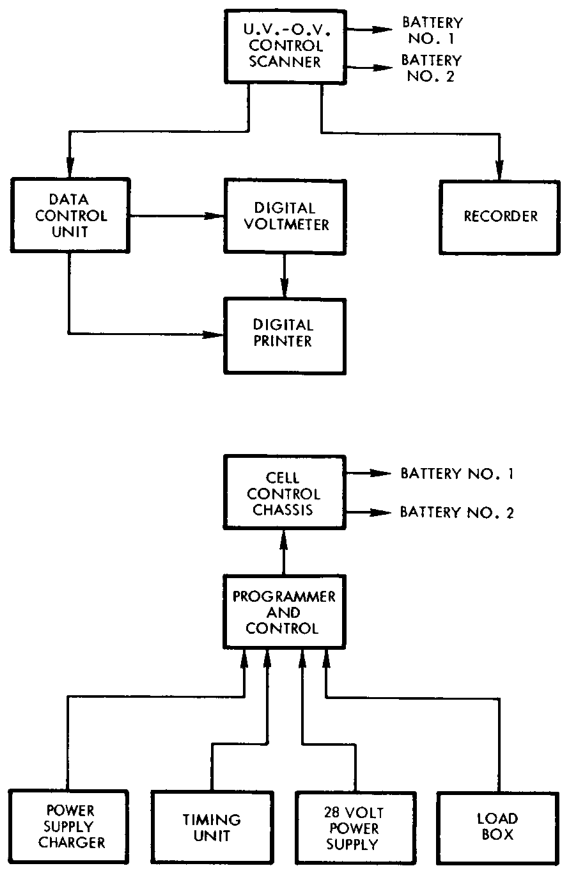


Figure 1. Battery Unit Test Set, Block Diagram

5.1.2 Cell Override Chassis

The cell override chassis and panel contains visual display of scanner/cell position and a manual takeout capability for removing individual cells from the test circuit.

5.1.3 Automatic Manual Control

The automatic manual control chassis provides a capability for going to manual control or automatic timing of the cell over-ride scanner. It also includes a scanning reset capability and monitors for the battery thermal sensors.

5.1.4 Digital Control Unit

The digital control unit provides the necessary logic to sequence the digital monitoring and printout system through the data requirements for

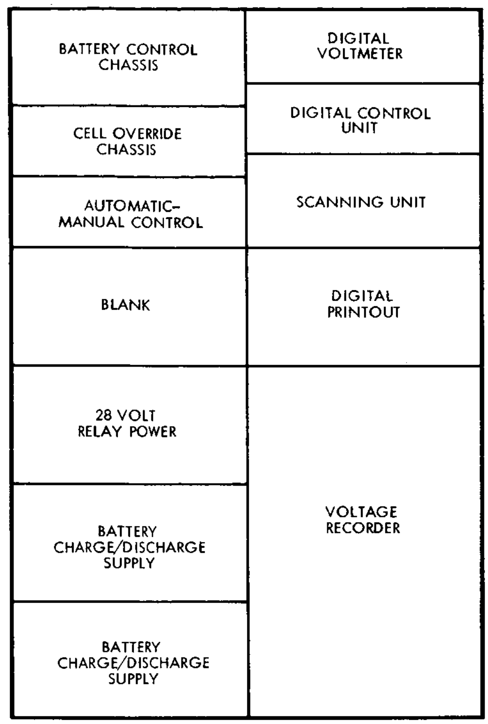


Figure 2. Battery Unit Test Set, Rack Layout

two batteries operated simultaneously. This unit scans individual cells and display cell position and provides the capabilities for monitoring one battery only or two batteries with either automatic scanning or manual data selection. This panel also includes the battery undervoltage and overvoltage alarm and interruption switches.

5.1.5 Scanning Unit

The scanning unit provides necessary scanner and controls to sequentially scan all the cells and battery data points required to test two batteries simultaneously. This panel provides power on and off capabilities, scan control capability, scan over-ride capability, and the capability for calibrating and setting the under-voltage, overvoltage controls.

5.1.6 Power Supply Units

The power supply units in this tester include a 28-volt relay power supply and two battery charge/discharge power supplies modified to react as battery chargers. These units remotely programmable for both charge and discharge current control.

5.1.7 Digital Voltmeter and Recorder

The digital voltmeter and digital printout recorder are provided as standard purchased items modified to be programmed by the test console sensing and control equipment and the digital control and scanning units. The voltage print chart recorder is a standard 24-point selectomatic recorder with a modified reference and high-speed printout capability compatible with impedance and response-time characteristics of a battery and with the interface requirements of the conditioner test set.

5.1.8 Cables

Battery charging and discharging are accomplished by connecting test cables, provided with the units, to the battery connector. The conditioner is then programmed to completely charge and discharge the batteries and provide full-time data monitoring of battery performed and battery sensor outputs.

5.2 Commercial Test Equipment

a) Digital voltmeter:

b) Digital printout:

6. BOUNDARY DEFINITIONS

6.1 Primary Power Sources

The test set operates from a power source as specified below:

Voltage	115 ±10 volts AC rms
Frequency	60 ±1 cps
Phase	Single

6.2 Service Environment

The service environment is a laboratory type having the following characteristics:

Temperature	60 - 90°F
Humidity	Less than 50 percent

7. PARAMETERS

Critical inverter(s) parameters defined by subsystem specifications will be listed in this section.

8. INTERFACES

8.1 Inputs

a) Battery voltage is monitored by means of a digital voltmeter and/or a strip chart recorder

b) Battery individual cell sensing will be monitored to ±1 mv by the digital voltmeter and/or a strip chart recorder

c) Battery sensor signal.

8.2 Outputs

The test set will provide DC charging current to the spacecraft battery.

ELECTRICAL DISTRIBUTION UNIT TEST SET
OSE/VS-4-471-1

1. SCOPE

This document establishes the requirements for the electrical distribution unit test set used to evaluate performance of the Voyager electrical distribution subsystem.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Mission Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

The electrical distribution unit test set is used to test the Voyager electrical distribution subsystem by supplying command discretes, simulated loads, and square wave audio power compatible with those experienced by the electrical distribution subsystem in operational use. It provides the following:

a) Discrete commands on an individual basis to the ordnance initiate circuits

b) Passive networks to simulate ordnance loads

c) Monitor lights to indicate ordnance circuit activation

d) Discrete commands on an individual basis to the power command relays

e) Passive networks to simulate power command relay loads

f) Monitor lights to indicate power command relay activation

g) Means of measuring circuit continuity

h) Square wave audio power to the electrical distribution power supply distribution unit

i) Passive networks to simulate power supply distribution unit loads.

4. DESIGN REQUIREMENTS

To provide a complete test of the electrical distribution subsystem performance, the electrical distribution unit test set performs the tests specified in the following paragraphs.

4.1 Command Discretes

4.1.1 Ordnance Initiate Commands

The test set supplies a maximum of 25 manual initiated command discretes that can be individually selected. These discretes are identical and have the following characteristics:

Voltage	25 to 32 volts DC
Duration	100 ±10 milliseconds
Maximum loading	100 milliamperes per circuit

4.1.2 Power Relay Commands

The test set supplies a maximum of 50 manually initiated command discretes that can be individually selected. These discretes are identical and have the following characteristics:

Voltage	25 to 32 volts DC
Duration	100 + 10 milliseconds
Maximum loading	100 milliamperes per circuit

4.1.3 Maximum Simultaneous Switching

The test set is capable of simultaneously commanding a minimum of 5 discretes.

4.2 Dummy Loads

The test set provides passive dummy loads to simulate those external loads encountered by the EDU in operational use. Load simulation will be of sufficient accuracy to insure that the test results will not be degraded.

4.3 Monitor Lights

The test set has monitor lights to indicate test status. 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 and/or test configuration

4.4 Electrical Distribution Subsystem Input Power

The test set supplies audio square wave power to the electrical distribution subsystem. This input power has the following characteristics:

Voltage	Fixed between 28 and 50 volts peak-to-peak
Waveform	Square wave
Distortion	Rise and fall time not to exceed 10 percent of pulse duration
Frequency	Adjustable -2000 to 5000 cps
Frequency stability	±1 percent
Power	100 watts maximum

4.5 Circuit Continuity

Provisions are made in the test set for measuring switching element resistances to an accuracy of 10 percent.

4.6 Voltage Monitoring

Provisions are made in the test set for monitoring voltage levels and wave shapes.

4.7 Protective Devices

Protective features are incorporated in the test set to protect both the test set and the unit under test against catastrophic damage due to a malfunction of either device.

5. FUNCTIONAL DESCRIPTION

5.1 General

The test set comprises standard commercial test equipment and specific equipment designed for testing functions peculiar to the Voyager electrical distribution subsystem.

A functional block diagram appears in Figure 1.

Figure 2 shows the physical configuration of the electrical distribution unit test set. It consists of one rack of commercial and special purpose equipment.

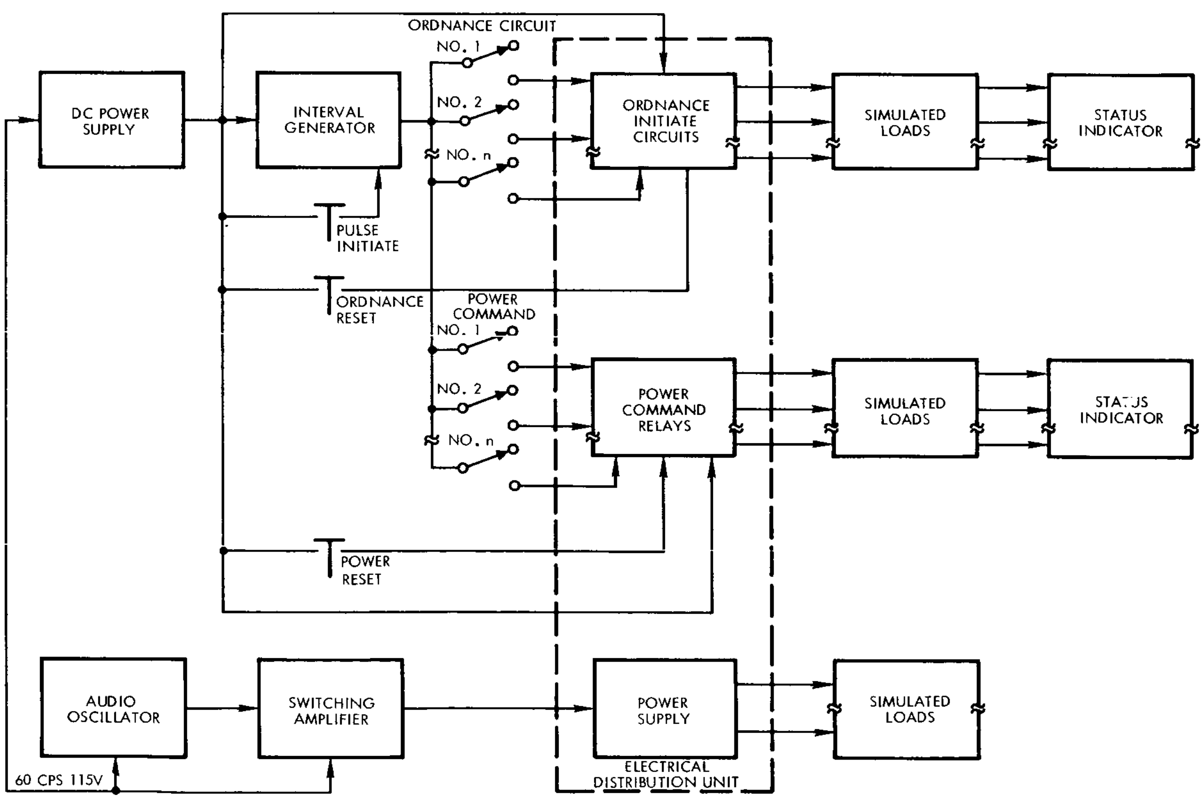


Figure 1. Electrical Distribution Unit Test Set, Block Diagram

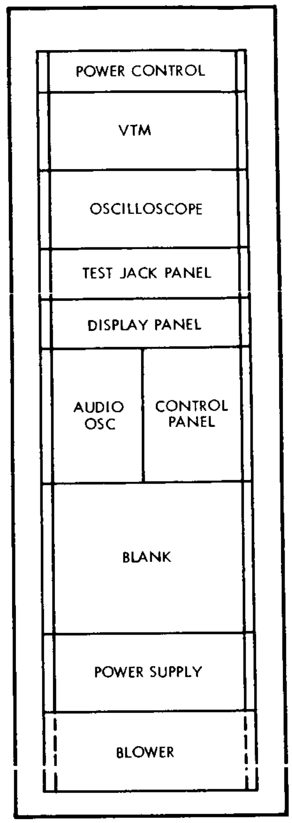


Figure 2. Electrical Distribution Unit Test Set, Rack Layout

5.2 Specific Equipment Description

5.2.1 Commercial Test

a) Oscilloscope - Hewlett Packard, Model 120 BR, or equivalent

b) Audio Oscillator - Hewlett Packard, Model 201 CR, or equivalent

c) Vacuum Tube Voltmeter - Hewlett Packard, Model 410 BR, or equivalent

d) Power Supply - Harrison Laboratories, Model 809 A, or equivalent.

5.3.2 Mechanical Design

a) Racks are in accordance with OSE/VS-2-110.

b) Jacketed and Molded Cables are in accordance with OSE/VS-2-110.

c) Soldering is in accordance with OSE/VS-2-110.

d) Colors conform to OSE/VS-2-110.

e) Abbreviations are in accordance with MIL-STD-12.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The electrical distribution unit test set operates from a power source as specified below:

Voltage	115 ±10 vac
Frequency	60 ±1 cps
Phase	Single

6.2 Service Conditions

The electrical distribution unit test set operates in the environment specified below:

Temperature	60-90°F
Humidity	Less than 50 per cent
Altitude	Less than 5000 feet

THERMAL CONTROL SUBSYSTEM
OSE/VS-4-510

1. SCOPE

This document defines the general requirements, equipment list, and applicable documents for thermal control subsystem MOSE required for the assembly, handling, protection, transport, shipment, and storage of the thermal control subsystem equipment used in the Voyager program.

The models covered by this document conform to the requirements delineated herein and are identified as the VS-4-510-numbered series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

MIL-C-9959 Amend. 1 5 February 1963	Container Flexible, Reusable, Water-Vaporproof
MIL-D-3464B 31 October 1955	Desiccant (Activated) in Bags; For Static Dehumidification and Packaging
MIL-P-1160 MIL-P-9024C	Preservation, Methods of
PPP-B-601A 16 August 1963	Box, Wood, Cleated-Plywood
PPP-B-636C 12 June 1965	Box, Fiberboard
MIL-P-9024C 30 April 1965	Packaging, Air Weapons Systems, Specifications and General Design Requirements for

DAC/MSSD

Mechanical Support Equipment and Facilities Manual

3. REQUIREMENTS

The thermal control subsystem MOSE items defined in the following paragraphs are designed to perform their specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

The end items defined within this documentation group are associated with the assembly, handling, shipping, protection, and storage of the thermal control subsystem equipment. The equipment enumerated below accomplishes these support functions.

Thermal Control Subsystem - OSE/VS-4-510

Item No.	Nomenclature
4-510-1	Assembly and Handling Fixture, Spacecraft Louvers
4-510-2	Shipping Container, Spacecraft Louvers
4-510-3	Handling and Shipping Container, Insulation

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The thermal control subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when used to support the thermal control subsystem components. All MOSE coupled to these components are operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is made of non-magnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem components' physical envelope.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the thermal control subsystem components and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used, unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or over land. The equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured, and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use will there be degradation in the reliability of the thermal control subsystem equipment.

3.7 Maintainability

The MOSE is designed so that repairs, adjustments, and overhaul can be readily accomplished by operating personnel using conventional general-purpose tools and equipment.

3.8 Identification and Marking

All MOSE carry adequate marking for identification with lift points, rated loads, hazard warnings, and special instructions noted.

ASSEMBLY AND HANDLING FIXTURE; SPACECRAFT LOUVERS
OSE/VS-4-510-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the spacecraft louver assembly and handling fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-510	Voyager Operational Support Equipment, Thermal Control Subsystem
OSE/VS-4-510-2	Shipping Container, Spacecraft Louvers

3. FUNCTIONAL REQUIREMENTS

The spacecraft louvers require physical protection and support during louver assembly, test, shipment, and storage, and protection during and after spacecraft integration.

4. DESIGN REQUIREMENTS

4.1 Minimum Dimensions

The dimensions of the assembly and handling fixtures vary for each louver frame assembly required. The louver assemblies are approximately 6, 5, 3 and 2 square feet in area. Three assemblies are approximately 34 inches wide, and one approximately 19 inches wide. Each panel assembly is approximately 3 inches deep and the mounting bolt hole line is 3/4 inch inside the overall dimensions. The assembly and handling fixtures, therefore, are not interchangeable for identical assemblies for the various spacecraft test models.

4.2 Handles

Handles or handholds are provided for fixture handling.

4.3 Load Factors

The design load and handling factors are in accordance with OSE/VS-2-110. Strength and rigidity are provided to withstand the design conditions during usage. The fixture supports the load of the louver assemblies during all handling and shipping operations.

5. EQUIPMENT DESIGN

5.1 General

The spacecraft louver assembly and handling fixture consists of a rectangular frame large enough to fit around the 3 inch-deep section of the louver panel assembly. Each corner of the frame mounts a 90-degree corner type clamp which attaches the assembly and handling fixture to the louver assembly chassis. Each corner clamp is padded and has its adjustment limited by mechanical stops to limit bearing on the louver assembly chassis. The mounting flange hole pattern on the chassis is thus exposed for convenience during assembly to the spacecraft. For protection during shipment, interchangeable fiberglass covers are attached to the frame on both front and rear sides of the louver assemblies. The cover which protects the rear side is removed prior to louver mounting to the spacecraft cold plate. Carrying handles are fastened to the fiberglass covers for lifting and handling the entire assembled package. This design concept is shown in Figure 1.

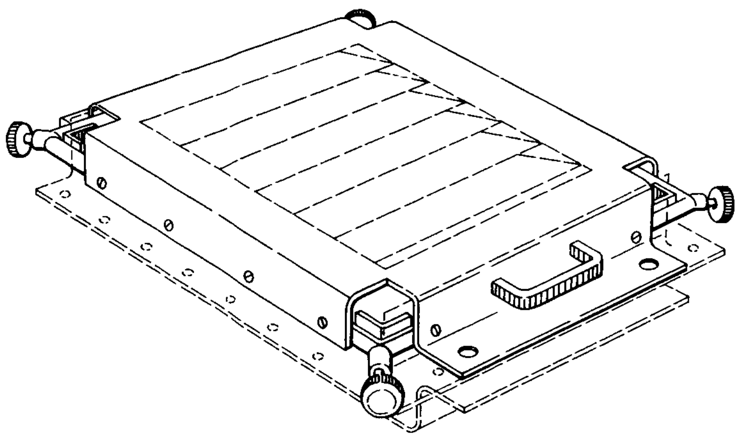


Figure 1. Assembly and Handling Fixture, Spacecraft Louvers

5.2 Interface Definition

The spacecraft assembly and handling fixture interface with the spacecraft louver assembly. During shipment, the assembly and handling fixture are compatible with the spacecraft louver shipping container.

SHIPPING CONTAINER, SPACECRAFT LOUVERS
OSE/VS-4-510-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the spacecraft louver shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-510	Voyager OSE Thermal Control Subsystem
OSE/VS-4-510-1	Assembly and Handling Fixture, Spacecraft Louvers

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the spacecraft louvers during transportation, and storage, and shipment.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the spacecraft louvers from physical damage during surface and air transportation, and during periods of storage.

4.2 Weight and Size

The weight and size are minimum within the constraints of providing the desired protection. At least four separate and distinct louver assembly sizes are accommodated.

4.3 Environment

4.3.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the louver assemblies to less than that occurring during flight environments.

4.3.2 Humidity and Temperature

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.3.3 Condensation and Corrosion Control

No moisture condensation nor corrosive atmosphere are permitted within the container.

4.4 Altitude

The shipping container is capable of operating at altitudes consistent with commercial and military air transportation.

4.5 Transportability

The container is capable of being transported by rail, truck or air.

4.6 Reusability

The shipping container is reusable.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping equipment consists of an interior container, a shock mitigating system, an environmental cover (barrier material), and an exterior shipping container.

The interior container is fabricated from corrugated fiberboard conforming to PP-B-636.

The shock mitigating insert to the interior container is 2.0 to 3.8 pounds density polyurethane foam ester base or rubberized hair of suitable thickness. If rubberized hair is used, a dust shroud of polyethylene is placed around the spacecraft louvers.

The spacecraft louver assemblies, with their respective assembly and handling fixtures, shock mitigation system and interior container are enclosed in a reusable barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment I, 5 February 1963, with a water-vapor transmission value of 0.05 to 0.085 g/100 sq. in./25 hrs. The barrier material is made of one of the following materials: scrim foil, nylon reinforced polyvinylchloride, fluorohalocarbon, or combinations thereof. The barrier material contains desiccant bags conforming to MIL-D-3464B, with a humidity-indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 20 percent. The required desiccant quantity is calculated in accordance with MIL-P-116D, paragraph 3.5.6.

Prior to shipment, the barrier material is purged with dry nitrogen to a 0-degree dew point, desiccated, and evacuated.

The interior container and barrier material are placed in a reusable cleated plywood container conforming to PPP-B-601.

The void separating the interior container (PPP-B-636) from the exterior container (cleated plywood) is filled with 2.4 to 3.8 density polyurethane foam or rubberized hair. This design concept is shown in Figure 1.

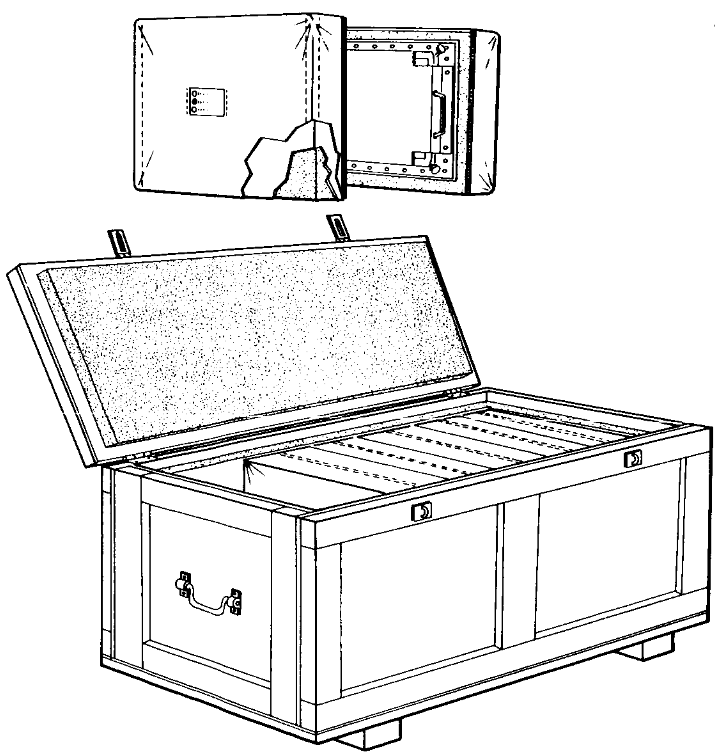


Figure 1. Shipping Container, Spacecraft Louvers

5.2 Equipment Interface

The shipping container interfaces with the spacecraft louver assembly and handling fixture.

HANDLING AND SHIPPING CONTAINER, INSULATION
OSE/VS-4-510-3

1. SCOPE

This document defines the functional and design requirements and equipment description for the insulation handling and shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-510	Voyager OSE, Thermal Control Subsystem

3. FUNCTIONAL REQUIREMENT

The handling container supports and protects the spacecraft insulation during in-plant handling and storage. The handling container is overpacked for protection during interfacility shipment.

4. DESIGN REQUIREMENTS

4.1 Cleanliness

No foreign matter (fingerprints, dust, grease, etc.) is permitted within the container.

4.2 Configuration

The insulation is folded (aluminized mylar) or coiled (refrasil). The container is large enough to accept the insulation so prepared.

4.3 Physical Protection

The handling container protects the insulation from physical damage during in-plant handling and storage. The shipping container protects the insulation from physical damage during interfacility transportation.

4.4 Shock and Vibration

The insulation is protected from normal shock and vibration environments encountered during handling and transportation. The environments are identified in MIL-P-9024 C.

5. EQUIPMENT DESCRIPTION

5.1 General

The folded (aluminized mylar) or coiled (refrasil) insulation is placed in a clean polyethylene or nylon bag, purged with clean dry nitrogen or air, and heat sealed. The enclosed insulation is placed in a hinged and latched plastic (polyethylene, acrylic butadiene styrene) container which has a smooth interior surface to prevent punctures of the encapsulating bag or the insulation sheets.

For shipment, the plastic container, or several containers, is overpacked in an exterior cleated plywood container conforming to PPP-B-601. The container is closed by means other than nailing. Adequate foam dunnage (polycerethane) is used within the shipping container to protect the insulation from handling and shipping damage. Fork lift capability is provided with the container if dimensionally required. This design concept is shown in Figure 1.

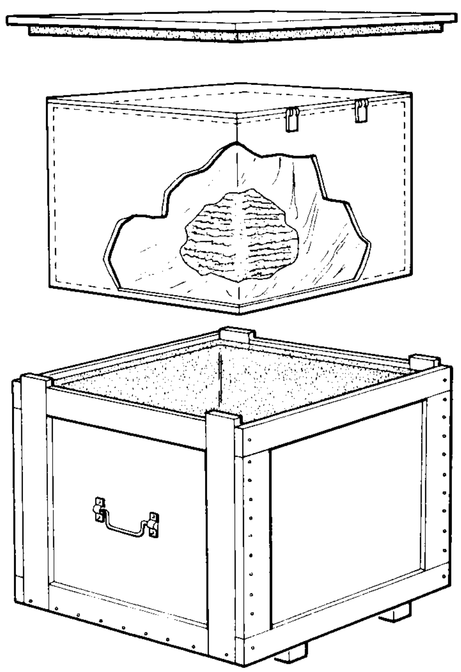


Figure 1. Handling and Shipping Container, Insulation

5.2 Equipment Interface

The handling and shipping containers are used to store and ship the required spacecraft insulation. The handling and shipping containers have no physical or electrical interface with other operating support equipment.

STRUCTURAL SUBSYSTEM
OSE/VS-4-520

1. SCOPE

This document defines the general requirements equipment list and applicable documents for structural subsystem mechanical operating support equipment required for fabrication, protection, handling, transport, shipment and storage of the structural subsystem equipment used in the Voyager program.

The models covered by this document conform to the requirements delineated herein and are identified as the VS-4-520 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

PPP-B-601A Amend. 2 16 August 1963	Boxes, Wood, Cleated - Plywood
PPP-B-621A Amend. 2 12 April 1963	Boxes, Wood, Nailed and Lock-Corner
MIL-M-008090D 21 February 1961	Mobility Requirements, Ground Support Equipment, General Specifications For
MIL-P-9024B 2 June 1959	Packaging, Air Weapons Systems, Specifications and General Design Requirements For
MIL-STD-1186 28 October 1963	Cushioning, Anchoring, Bracing, Blocking, and Waterproofing, With Appropriate Test Methods

DAC/MSSD

Mechanical Support Equipment and Facilities Manual.

3. REQUIREMENTS

The structural subsystem mechanical operating support equipment items defined in the following paragraphs are designed to perform specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs, as prime considerations.

The end items defined within this documentation group are associated with the fabrication, shipping, protection, handling, and storage of the structural subsystem equipment. The equipment enumerated below accomplishes these major mechanical handling and support functions.

Item No.	Nomenclature
4-520-1	Dolly, Structural Sections
4-520-2	Shipping Container, Miscellaneous Spacecraft Structure
4-520-3	Sling, Propulsion/Pneumatic Structural Section
4-520-4	Interface Match Tool, Spacecraft Flight Capsule
4-520-5	Interface Match Tool, Spacecraft Centaur Adapter

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The structural subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when used to support the structural subsystem components. All MOSE coupled to these components is operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is made of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem components' physical envelope.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to structural subsystem components and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or land. The equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured, and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use will there be degradation in the reliability of the structural subsystem equipment.

3.7 Maintainability

The MOSE is designed so that repairs, adjustments, and overhaul can be readily accomplished by operating personnel using conventional, general-purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification with lift points, rated loads, hazard warnings, and special instructions noted.

DOLLY, STRUCTURAL SECTIONS
OSE/VS-4-520-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the dolly structural sections.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-520	Voyager OSE, Structural Subsystem
OSE/VS-3-140-15	Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle

3. FUNCTIONAL REQUIREMENT

The structural sections dolly provides support for the hexagonal section of the spacecraft structure and any installed subsystem equipment during intraplant handling prior to final spacecraft structural assembly. It also provides intraplant support for the propulsion/pneumatic section of the spacecraft structural subsystem during assembly, checkout, and local movement at the static test firing facility.

4. DESIGN REQUIREMENTS

4.1 Mobility

The structural sections dolly is designed in accordance with the requirements for type I, class II mobility, per MIL-M-008090. The running gear consists of four swivel casters with swivel locks and step-on parking brakes. No spring suspension systems or shock absorbers are required. The dolly is equipped with a towbar or heavy-duty shackle capable of attaching to a shop mule for intrafacility movement.

4.2 Loads

The dolly is capable of supporting the weight of the hexagonal section of the spacecraft structure, or the propulsion/pneumatic section mounted on the universal mounting ring, OSE/VS-3-140-15, whichever is heavier. No deflection at limit or applied loads detrimental to the performance characteristics of the equipment are permitted. The dolly frame is sufficiently rigid and the supporting base provides adequate bearing surface to prevent deformation of the spacecraft structural sections. All other load factors are in accordance with OSE/VS-2-110.

4.3 Physical Characteristics

The structural sections dolly provides a base sufficiently large to accommodate a structure 120 inches across the hex points, 105 inches across flats, and 59 inches high maximum.

4.4 Access

Assembly and checkout functions require complete access around the periphery of the dolly. The dolly is designed so that its height provides access for operating personnel to various elements of the structural subsystem.

4.5 Support and Attitude

The dolly supports the structural sections in their normal launch orientation and attaches to the universal mounting ring, OSE/VS-3-140-15 which in turn mounts and provides protection and rigidity to the structural sections.

4.6 Hoisting Provisions

Hoisting provisions are provided for lifting the dolly using ordinary floor or overhead hoists.

5. EQUIPMENT DESCRIPTION

5.1 General

The structural sections dolly consists of a hexagonal aluminum frame fabricated of standard structural shapes. The upper surface mounts six cantilevered pads for supporting the universal mounting ring. Six quick-disconnect attachments are mounted in the area of each support pad for fastening the universal mounting ring and the structural section being handled to the dolly.

The bottom of the frame mounts four swivel casters. The frame is equipped with a standard towbar for attachment to a prime mover for towing, and is provided with 4 rings for hoisting purposes. This design concept is shown in Figure 1.

5.2 Interface Definition

The structural sections dolly interfaces with the structural hard-points of the universal mounting ring, OSE/VS-3-140-15. The dolly has no other interface with MOSE.

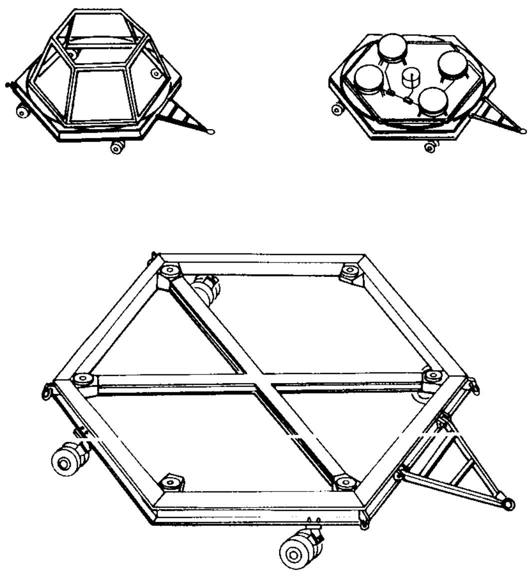


Figure 1. Dolly, Structural Sections

SHIPPING CONTAINER, MISCELLANEOUS SPACECRAFT STRUCTURE
OSE/VS-4-520-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the shipping container and miscellaneous structure elements.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-520	Voyager OSE, Structural Subsystem

3. FUNCTIONAL REQUIREMENTS

The shipping container provides physical protection for the spacecraft miscellaneous support structure which is shipped separate from the spacecraft.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the spacecraft miscellaneous support structure from physical damage during surface and air transportation.

4.2 Weight and Size

The weight and size are as small as possible to provide the desired protection.

4.3 Kit

The container accommodates the structural supports and miscellaneous bracketry for the solar array panels, antenna and POP systems.

4.4 Shock and Vibration

The support structure is protected from normal shock and vibration environment encountered during handling and transportation. The environments are identified in MIL-P-9024.

5. EQUIPMENT DESCRIPTION

5.1 General

Each structural part or assembly is separated from other parts by foam or individual inner containers. All parts or assemblies are placed in one outer wooden container, conforming to PPP-B-621 or PPP-B-601 as applicable. The parts or assemblies are secured within the outer container to prevent movement. Blocking and bracing within the outer container are in accordance with MIL-STD-1186.

5.2 Interface Definition

The shipping container is used to transport the miscellaneous spacecraft structural parts or assemblies but has no physical or electrical interface with other operating support equipment.

SLING, PROPULSION/PNEUMATIC STRUCTURAL SECTION OSE/VS-4-520-3

1. SCOPE

This document defines the functional and design requirements and equipment description for the propulsion/ pneumatic structural section hoist sling.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-520	Voyager OSE, Structural Subsystem
OSE/BS-4-520-1	Dolly, Structural Sections
OSE/VS-3-140-1	Transporter, Flight Spacecraft
OSE/VS-3-140-15	Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle

3. FUNCTIONAL REQUIREMENTS

The propulsion/pneumatic structural section sling supports the propulsion/pneumatic structural section when mounted on the universal mounting ring during installation or removal from the flight spacecraft transporter, the structural section dolly, various test fixtures, and the static firing stand.

4. DESIGN REQUIREMENTS

4.1 Loads

The sling supports the propulsion/pneumatic structural section with propellants and pressurized gas fully loaded when mounted on the universal mounting ring. Design load factors are in accordance with OSE/ VS-2-110.

4.2 Stability

The sling provides stability to the propulsion/pneumatic structural section during all hoisting and translational movement requirements.

4.3 Fasteners

All fasteners are the quick-release type and are attached to the sling cables.

5. EQUIPMENT DESCRIPTION

5.1 General

The sling consists of three individual stainless steel cables and a lifting ring. One end of each cable is connected to the lifting ring; the other end contains a clevis which attaches to the hoist points of the universal mounting ring. The sling design is illustrated in Figure 1.

5.2 Interface Definition

The sling connects to the universal mounting ring, OSE/VS-3-140-15, which is adapted to support the propulsion/pneumatic structural section. The sling operates compatibly with overhead traveling cranes or mobile hoists.

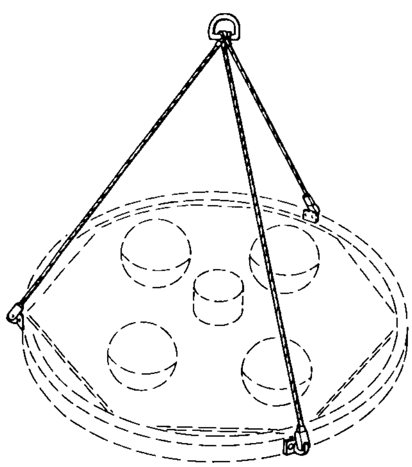


Figure 1. Sling, Propulsion/Pneumatic Structural Section

INTERFACE MATCH TOOL, SPACECRAFT/FLIGHT CAPSULE
OSE/VS-4-520-4

1. SCOPE

This document defines the functional and design requirements and equipment description for the interface match tool, spacecraft/flight capsule.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-520	Voyager OSE, Structural Subsystem

3. FUNCTIONAL REQUIREMENTS

The interface and hole pattern between the Voyager flight spacecraft and the flight capsule requires control since the two program elements are separate systems provided by separate contractors. To assure the compatibility of this interface, a set of matched gages or tools is required at each contractor's manufacturing and assembly facility. These tools augment the normal interface drawing control system, and provide means for physically assuring matching compatibility, although the two systems will not be physically mated until a later time.

4. DESIGN REQUIREMENTS

4.1 Minimum Dimensions

The tool dimensions are compatible with the nominal 80 inch diameter flight spacecraft/flight capsule interface plane.

4.2 Loads

The tool accepts induced hoisting loads and nominal loads imposed during normal tool usage.

4.3 Hoisting

Provisions for hoisting the tool are provided on the tool ring periphery.

4.4 Load Factors

All load and handling factors are in accordance with OSE/VS-2-110.

4.5 Fasteners

Fasteners are of the quick-release type and are attached to the tool structure by a vinyl-coated chain or cable.

4.6 Shackles

MS standard hoisting shackles (three minimum) are attached for tool handling.

4.7 Precision Hole Sleeves

Precision ground (OD and ID) stainless steel sleeve bushings are pressed into the tool aluminum frame to preserve hole dimensions.

4.8 Electrical

The gage provides the matching electrical connector location size and pin information duplicating the capsule umbilical.

5. EQUIPMENT DESCRIPTION

5.1 General

The spacecraft/flight capsule interface match tool consists of an aluminum structure stressed for minimum deflections. Precision-ground stainless steel insert bushings are pressed into special cast blocks on the tube inner periphery. The flight capsule interface hole pattern is determined and each bushing reamed to the final fit. The interfacing flange surface is ground undersize and a controlled thickness teflon shim bonded to it for abrasion protection. MS standard shackles are attached to welded lugs for hoisting points. Welded mounting pads are provided for mounting the tool to assembly jigs, and pip pins provide temporary coupling for stability. Precision reference surfaces are ground at four 90-degree positions to provide height gage blocks or optical tooling calibration scribe marks. The tool provides appropriate bolt and shear pin match holes, as well as a matching hole pattern for all electrical, and any required pneumatic connections between the flight spacecraft and the flight capsule. This design concept is shown in Figure 1.

5.2 Interface Definitions

The spacecraft/flight capsule interface match tool interfaces with the 80-inch nominal diameter interface plane, the 120-degree bolt-hole pattern, and the shear pin hole pattern which is offset 60 degrees from the bolt holes and are also 120 degrees apart. The tool attaches to assembly fixtures at both DAC/MSSD facilities and the flight capsule contractor assembly facilities, as well as the TRW assembly, inspection and verification test fixture.

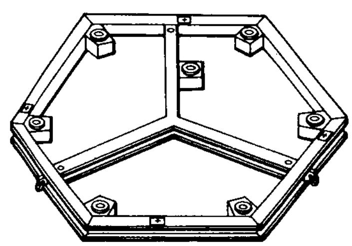


Figure 1. Interface Match Tool, Spacecraft/Flight Capsule

INTERFACE MATCH TOOL, SPACECRAFT/CENTAUR ADAPTER
OSE/VS-4-520-5

1. SCOPE

This document defines the functional and design requirements and equipment description for the interface match tool, for the spacecraft/Centaur adapter.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-520	Voyager, OSE, Structural Subsystem

3. FUNCTIONAL REQUIREMENTS

The interface and hole pattern between the Voyager flight spacecraft and the Centaur adapter requires control since the two program elements are separate systems provided by separate contractors. To assure the compatibility of this interface, a set of matched gages or tools is required at each contractor's manufacturing and assembly facility. These tools augment the normal interface drawing control system, and provide the means for physically assuring matching compatibility, although the two systems are not physically mated prior to their final assembly at the launch complex.

4. DESIGN REQUIREMENTS

4.1 Minimum Dimensions

The tool dimensions are compatible with the 120-inch diameter flight spacecraft/Centaur adapter interface plane.

4.2 Loads

The tool accepts induced hoisting loads and nominal loads imposed during normal tool usage.

4.3 Hoisting

Provisions for hoisting the tool are provided on the tool ring periphery.

4.4 Load Factors

All load and handling factors are in accordance with OSE/VS-2-110.

4.5 Fasteners

Fasteners are the quick release type and attached to the tool structure by a vinyl-coated chain or cable.

4.6 Shackle

MS standard hoisting shackles (4 minimum) are attached for tool handling.

4.7 Precision Hole Sleeves

Precision-ground (OD and ID) stainless steel sleeve bushings are pressed into the tool aluminum frame to preserve hole dimensions.

5. EQUIPMENT DESCRIPTION

5.1 General

The flight spacecraft/Centaur adapter interface match tool consists of an aluminum structure stressed for minimum deflections. Precision-ground stainless steel insert bushings are pressed into specially cast blocks on the tool frame periphery. The Centaur adapter hole pattern is determined and each bushing reamed to the final fit. The interfacing flange surface is ground undersize and a controlled thickness teflon shim bonded to it for abrasion protection. MS standard shackles are attached to welded lugs for hoisting points. Welded mounting pads are provided for mounting the tool to assembly jigs, and pip pins provide temporary coupling for stability. Precision reference surfaces are ground at four 90 degree positions to provide appropriate bolt and shear-pin match holes, as well as provide a matching interface pattern for all electrical, pneumatic and air conditioning interstage connections between the launch vehicle system and the planetary vehicle. This design concept is shown in Figure 1.

5.2 Interface Definition

The spacecraft/Centaur adapter interface match tool interfaces with the 120-inch diameter interface plane, the 120-degree bolt hole pattern, and the 3 shear-pin hole pattern, which is offset 60 degrees from the bolt holes and are also 120 degrees apart. The tool attaches to assembly fixtures at both DAC/MSSD facilities and the launch vehicle contractor assembly facilities, as well as the TRW assembly, inspection and verification test fixture.

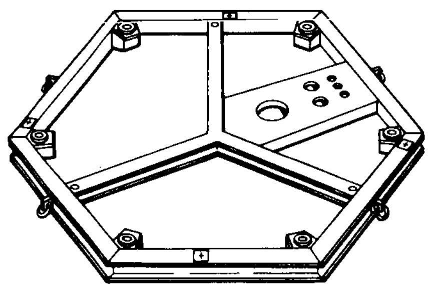


Figure 1. Interface Match Tool, Spacecraft/Centaur Adaptor

PYROTECHNIC SUBSYSTEM
OSE/VS-4-530

1. SCOPE

This document defines the general requirements, equipment list and applicable documents for pyrotechnic subsystem mechanical operating support equipment required for the assembly, handling, transport, and storage of the pyrotechnic subsystem equipment used in the Voyager program.

The models covered by this document conform to the requirements delineated herein and are identified as the VS-4-530 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

MIL-P-116D	Preservation, Methods of
ICC Tariff No. 15	Regulation for Transportation of Explosives and other Dangerous Articles
Air Force Manual 71-4	Packaging and Handling of Dangerous Material for Military Aircraft

DAC/MSSD

Mechanical Support Equipment and Facilities Manual

3. REQUIREMENTS

The pyrotechnic subsystem mechanical operating support equipment items defined in the following paragraphs are designed to perform their specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

The end items defined within this documentation group are associated with the handling, shipping, protection, and storage of the pyrotechnic subsystem equipment. The equipment enumerated in Table I accomplishes these major mechanical handling and support functions.

Table I. Pyrotechnic Subsystem

Item No.	Nomenclature
4-530-1	Shipping Container - Explosive Train
4-530-2	Handling Case, Arming Kit

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The pyrotechnic subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when used to support the pyrotechnic subsystem components. All MOSE coupled to these components is operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is fabricated of non-magnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem physical envelopes of the subsystem components.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the pyrotechnic subsystem components and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosion action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or land. The equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use shall there be degradation in the reliability of the pyrotechnic subsystem equipment.

3.7 Maintainability

The MOSE is designed so that repairs, adjustments, and overhaul can be readily accomplished by operating personnel using conventional general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification with lift points, rated loads, hazard warnings, and special instructions noted.

SHIPPING CONTAINER, EXPLOSIVE TRAIN
OSE/VS-4-530-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the shipping container, explosive train.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-530 Voyager OSE Pyrotechnic Subsystem

3. FUNCTIONAL REQUIREMENTS

The explosive train shipping container, provides environmental protection for the explosive train during handling, transportation and storage.

4. DESIGN REQUIREMENTS

The explosive train is packaged and packed and the container marked in accordance with Interstate Commerce Commission (ICC) requirements, using standard nomenclature and classification for shipment of explosive material for all modes of transportation.

4.1 Container Fabrication and Explosive Weight Limitations

Explosive weight limitations and container design and fabrication for explosive items are in accordance with the applicable ICC documents governing specific transportation modes.

4.2 Transportation

All explosive items are transported in accordance with the following transportation regulations governing specific transportation modes:

a) Interstate Commerce Commission Tariff No. 15, for land and water transportation.

b) Packaging and Handling of Dangerous Material Air Force Manual (AF 111) 71-4 for military aircraft.

4.3 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the explosive train to less than that occurring during flight environment.

4.4 Humidity

The relative humidity is less than 20 percent within a temperature range of 0 to 130°F.

4.5 Altitude

The shipping container and its barrier are capable of operating at altitudes consistent with military air transportation.

4.6 Transportability

The container is capable of being transported by rail, truck, or air.

5. EQUIPMENT DESCRIPTION

5.1 General

The explosive train is classified as an explosive mine, class A explosive, in accordance with ICC regulations. The explosive train is preserved in accordance with Method II of MIL-P-116D, using a conductive film as a moisture barrier. The conductive film consists of 3 layers (a conductive polyethylene film, aluminum foil, and a conductive polyethylene film) bonded to one another. The barrier contains a humidity indicator window capable of being easily inspected. The enclosed explosive train is encapsulated in 2.4 to 3.8-lb-density polyurethane foam of suitable thickness for shock isolation. The shipping container is fabricated to comply with paragraph 73.56 of ICC Tariff No. 15 for transportation by land and water. This paragraph requires the use of a strong wooden container conforming to ICC Specification 15A, 15B, or 15C for shipment and transportation. All container marking is in accordance with paragraph 73.56 of ICC Tariff No. 15. Shipment of the explosive train by Military Air Transport is in accordance with paragraph 5.29 of AFM 71-4. These requirements are equivalent to ICC Tariff No. 15.

Shipment of the explosive train by commercial air freight and passenger cargo transportation is not allowed. A pictorial representation of this shipping container is shown in Figure 1.

5.2 Equipment Interface

The shipping container is used to store and ship the explosive train. The container has no physical or electrical interface with other operating support equipment.

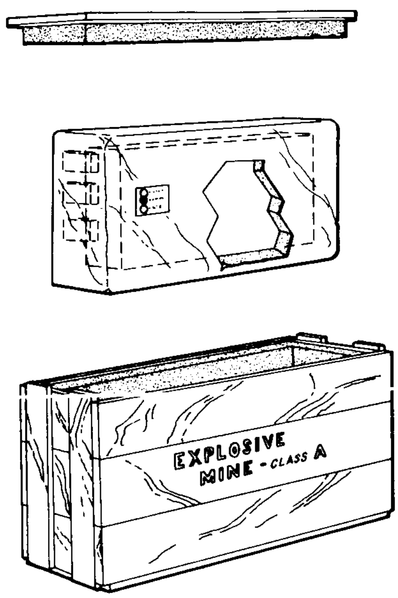


Figure 1. Shipping Container, Explosive Train

HANDLING CASE, ARMING KIT
OSE/VS-4-530-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the handling case, arming kit.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-530	Voyager OSE, Pyrotechnic Subsystem

3. FUNCTIONAL REQUIREMENTS

An arming kit handling case is provided for final arming and installation of category A squibs and detonators.

4. DESIGN REQUIREMENTS

4.1 Accountability

The arming kit handling case allows for positive accountability of all items in the arming kit.

The handling case contains separate positions for the following accountable items:

a) Category A squibs

b) Detonators

c) Hand tools

d) Continuity electrical meter

e) Voltage meter.

4.2 Safety Plugs

Safety plugs replace the squibs and detonators in the handling case after installation of the pyrotechnics.

4.3 Conductivity

Conductive material is used.

4.4 Grounding

The handling case is capable of being grounded.

4.5 Portability

The handling case is portable and is equipped with carrying handles.

5. EQUIPMENT DESCRIPTION

5.1 General

The handling case equipment consists of a wiring shelf, tooling tray, die-cut foam pad, and an aluminum handling case. The wiring shelf is fabricated from an aluminum sheet and is hinged and latched to the top of the handling case. The depth of the shelf is a minimum of 4 inches. The tooling tray is fabricated from aluminum and contains an aluminum handle across the entire length of the tray for ease in removal and a 2.4-pound-density polyurethane foam pad is bonded to the bottom of the tray.

The die-cut foam pad is fabricated from 2.4-pound-density polyurethane foam. Each squib, meter, and detonator is placed in an individual cavity and each cavity is separated from adjacent cavities by a minimum of 1/2 inch. The safety plugs are placed in the applicable cavity after installation of the squibs and detonators. A carbonized conductive polyethylene film (Velstat) is placed around the entire die-cut foam pad to prevent a static-charge buildup.

The tooling tray and the die-cut polyurethane foam pads are placed in the bottom half of the aluminum handling case. The aluminum handling case contains latches and handles. The lid of the aluminum handling case is hinged. The handling case is grounded in accordance with safety regulations prior to use. A pictorial representation of the arming kit handling case is shown in Figure 1.

5.2 Equipment Interface

The handling case equipment is used to install and maintain accountability for ordnance devices. The handling case has no physical or electrical interface with other operating support equipment.

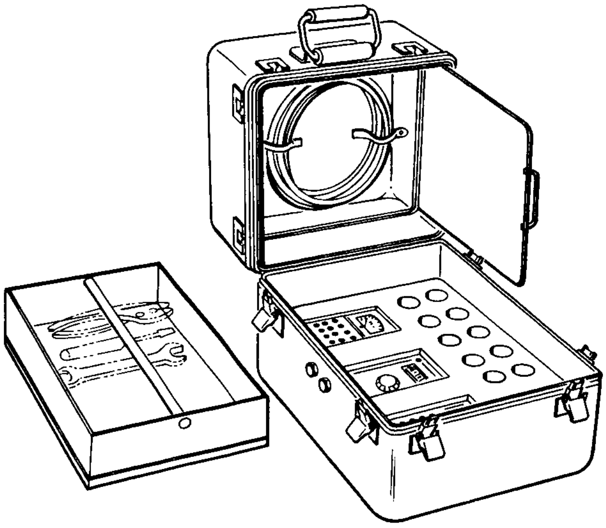


Figure 1. Handling Case, Arming Kit

PLANET ORIENTED PACKAGE SUBSYSTEM
OSE/VS-4-580

1. SCOPE

This document defines the general requirements, equipment list, and applicable documents for planet oriented package (POP) subsystem MOSE support equipment required for the protection, transport, and storage of the POP subsystem equipment used in the Voyager program.

The models covered by this specification conform to the requirements delienated herein and are identified as the OSE/VS-4-580 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

MIL-P-116D	Preservation, Methods of
MIL-B-131 15 November 1963	Barrier Material, Water Vaporproof, Flexible
MIL-B-26195A Amended 25 May 1962	Boxes, Wood-Cleated, Skidded, Load Bearing Base
MIL-D-3464B Amended 31 October 1955	Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification
MIL-M- 008090D Amended 21 February 1961	Mobility Requirements, Ground Support Equipment, General Specification for
PPP-B-601A Amendment 2 16 August 1963	Box, Wood, Cleated - Plywood

DAD/MSSD Documents

Mechanical Support Equipment and Facilities Manual

3. REQUIREMENTS

The POP subsystem MOSE items defined in the following paragraphs are designed to perform their specified functions, with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

Planet Oriented Package Subsystem, OSE/VS-4-580

Item No.	Nomenclature
4-580-1	Assembly Fixture and Dolly, POP
4-580-2	Shipping Container, POP
4-580-3	Hoise Beam, POP

The items defined within this specification group are associated with the handling, shipping, protection, and storage of POP subsystem equipment. The equipment enumerated above accomplishes these support functions.

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The POP subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when used to support the POP subsystem components. All MOSE coupled to these components is operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is fabricated of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem components' physical envelope.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the POP subsystem components and injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosive action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or over land. The equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured, and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology, and at no time during use shall there be degradation in the reliability of the POP subsystem equipment.

3.7 Maintainability

The MOSE is designed so that repairs, adjustments, and overhaul can be readily accomplished by operating personnel using conventional, general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification with lift points, rated loads, hazard warnings, and special instructions noted.

ASSEMBLY FIXTURE AND DOLLY, POP
OSE/VS-4-580-1

1. SCOPE

This functional description presents the requirements and preliminary design of the assembly fixture and dolly for the POP.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-4-580	1971 Voyager OSE, Planet-Oriented Package

3. FUNCTIONAL REQUIREMENTS

The POP assembly fixture and dolly provides intrafacility transportation for the planet-oriented package during assembly, precision alignment, and functional testing prior to spacecraft installation.

4. DESIGN REQUIREMENTS

4.1 Mobility

The POP assembly fixture and dolly is designed in accordance with Type I, Class 2, mobility as defined in MIL-M-008090D. The running gear consists of four rubber tread casters, two of which are capable of full swivel. All casters are equipped with step-lock parking brakes.

No spring suspension or shock absorbers are required. The dolly is equipped with a towbar or heavy duty shackle for attaching to a shop mule for in plant moving.

4.2 Loads and Load Factors

The dolly is capable of supporting the weight of the planet-oriented package which is approximately 125 pounds. The dolly frame is rigid and two supporting cradles provide adequate bearing surface to prevent deformation of the planet-oriented package structure. Design load factors are in accordance with OSE/VS-2-110.

4.3 Access

Access is provided to the sides of the POP structure and the dolly design allows the POP to be inverted on the support cradles.

4.4 Support and Attitude

The POP assembly fixture and dolly supports the POP and its gimbal arm in a horizontal attitude.

5. EQUIPMENT DESCRIPTION

5.1 General

The dolly consists of a rectangular base frame mounted on four casters. The POP support structure consists of two cushioned cradles and a support post which are welded directly to the dolly base frame. The two cradles support the main POP structure and the support post supports the gimbal arm structure. A clamp is mounted on the post for securing the boom structure.

Floor jack pads are located near each caster to provide stability during POP installation. The frame also contains a standard tow-bar for attachment to a prime mover. The dolly design concept is shown in Figure 1.

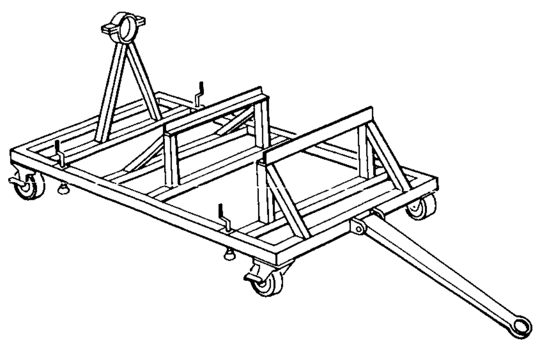


Figure 1. Assembly Fixture and Dolly, POP

5.2 Interface Definition

The POP assembly fixture and dolly interfaces with the planet-oriented package and operates compatibly with the shop mule for in-plant moving. It is also designed to function compatibly with the POP hoist beam.

SHIPPING CONTAINER, POP
OSE/VS-4-580-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the planet oriented package shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-580	1971 Voyager OSE, POP Subsystem
OSE/VS-4-580-3	Hoist Beam, POP

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection against the elements for the POP during transportation and storage of spare or replacement POP units.

4. DESIGN REQUIREMENTS

4.1 Physical Constraints

The shipping container protects the POP from physical damage during transportation and shipment. It is commensurate with the over-all POP dimensions of approximately 26 x 26 x 25 inches (main structure), and the 48 inch gimbal arm extension. The POP unit weight is approximately 125 pounds.

4.2 Environment

4.2.1 Shock and Vibration

Shock and vibration isolation is provided to reduce imposed loads on the POP to less than that occurring during flight environments.

4.2.2 Humidity

The maximum relative humidity within the shipping container is less than 20 per cent at 20 to 110°F.

4.2.3 Condensation

No moisture condensation is permitted within the shipping container during transportation or storage.

4.2.4 Corrosion

No corrosive atmosphere is permitted within the shipping container during transportation and storage.

4.2.5 Dust

The shipping container protects the POP from dust particle contamination. Particle sizes in excess of 100 μ are excluded by positive filtering or other protective means.

4.2.6 Altitude

The shipping container equipment suffers no functional deterioration when subjected to altitudes experienced during air shipments.

4.3 Venting

Venting provisions are incorporated for air transport to withstand altitude of 20,000 feet. Venting occurs through desiccants to prevent moisture influx.

4.4 Hoisting

Hoist or lift points are appropriately located for use in handling the container.

4.5 Load Factors

The shipping container is designed to load and handling factors specified in OSE/VS-2-110.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping equipment consists of a dust cover, a shock mitigating system, an environmental cover, and an exterior shipping container.

The POP main structure and the gimbal arm extension are placed in a clean plastic bag (polyamide, fluorohalocarbon or polyethylene) purged with clean, dry air or nitrogen, and evacuated. The POP and gimbal arm extension are encapsulated in foam (polyurethane or polyethylene). The foam nests the POP and gimbal arm extension in such manner as to distribute the load equally. The encapsulated POP is enclosed in a barrier material conforming to MIL-B-131, and heat sealed. The barrier material contains desiccant bags conforming to MIL-D-3464B with a humidity-indicator window capable of being easily inspected. The required desiccant quantity is calculated in accordance with MIL-P-116D, paragraph 3.5.6. The desiccant is changed when the indicator shows a relative humidity of more than 20 per cent. Prior to shipment the barrier material is purged with dry air or nitrogen to a $+20^{\circ}F$ dew point. The POP main structure and the gimbal arm extension are placed in a reusable wooden shipping container conforming to MIL-B-26195 or PPP-B-601, depending on dimension and weight limitations. The wooden container provides dunnage to prevent damage to the environmental cover. Hoist points and forklift capabilities as required are appropriately located. This design concept is shown in Figure 1.

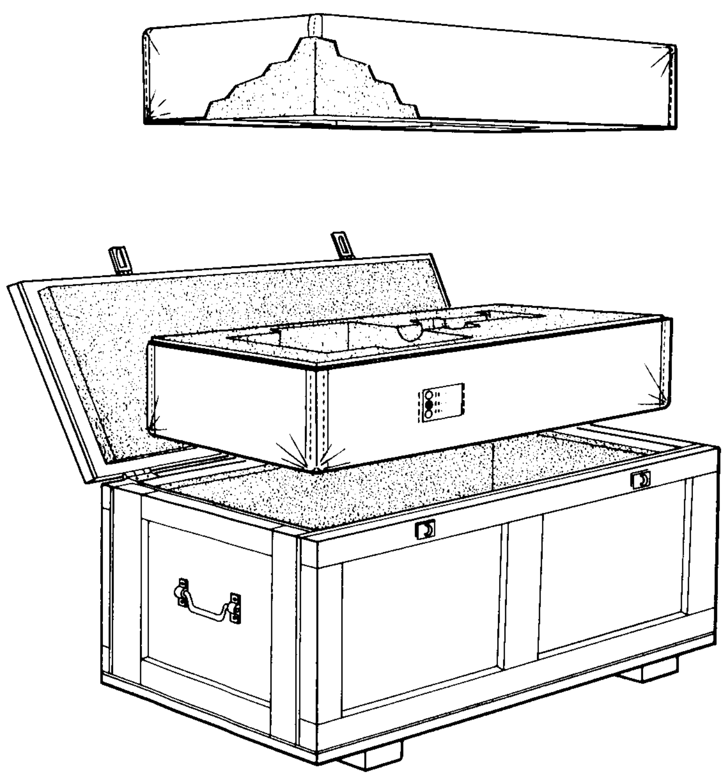


Figure 1. Shipping Container, POP

5.2 Interface Definition

The shipping container is compatible with the use of the POP hoist beam (VS-4-580-3) and the weight and dimensions of the POP unit. No other interface with MOSE is required.

HOIST BEAM, POP
OSE/VS-4-580-3

1. SCOPE

This document defines the functional and design requirements and equipment description for the planet oriented package hoist beam.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents.

OSE/VS-4-580	Voyager OSE, POP Subsystem
OSE/VS-4-580-1	Assembly Fixture and Dolly, POP
OSE/VS-4-580-2	Shipping Container, POP

3. FUNCTIONAL REQUIREMENTS

The planet oriented package is hoisted and handled during various operations. The handling includes lifting the POP from alignment benches and its assembly fixture and dolly, maintaining a specific orientation. The gimbal arm mounting plate is exposed for access during installation to the spacecraft.

4. DESIGN REQUIREMENTS

4.1 Minimum Dimensions

The hoist beam dimensions are compatible with the POP dimensions of 26 x 26 x 25 inches (POP structure), and the 48 inch gimbal arm.

4.2 Loads

The sling is designed for an applied load of approximately 125 pounds.

4.3 Load Factors

The load and handling factors are in accordance with OSE/VS-2-110.

4.4 Fasteners

Fasteners are the quick-release type.

5. EQUIPMENT DESCRIPTION

5.1 General

The POP hoist beam consists of a 2 piece aluminum-channel frame yoke, welded to a plate spreader bar. The yoke straddles the POP gimbal frame; the bottom portion mounts a fixed, soft cushioning pad which assumes the weight of the POP structure. Two adjustable clamps with cushioning pads (1 on each side) are attached to the upper half of the yoke at the upper portion of the POP structure to stabilize the POP within the yoke. The spreader bar extends to provide an additional support point for the gimbal frame arm. A single-point shackle centered over the assembly c.g. is attached to the spreader bar and serves as the lift point. The hoist beam is attached to the POP assembly by installing the bottom portion of the yoke under the POP, and bolting the top half to it. The adjusting clamps are then tightened, and a support bracket is secured to the gimbal-frame arm and spreader bar extension, thus securing the hoist beam to the POP assembly. The POP hoist beam design concept is shown in Figure 1.

5.2 Interface Definition

The hoist sling interfaces with the POP assembly, and is used in conjunction with the POP assembly fixture and dolly and the POP shipping containers. No other interface with MOSE is indicated.

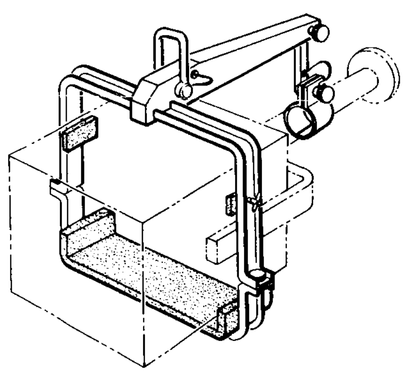


Figure 1. Hoist Beam, POP

PLANET ORIENTED PACKAGE UNIT TEST SET
OSE/VS-4-581-1

1. SCOPE

This document establishes the requirements for the planet oriented package (POP) unit test set used to evaluate performance of the POP prior to the installation of the experiments.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

OSE/VS-1-110	OSE Objectives and Criteria
OSE/VS-2-110	OSE Design Characteristics and Restraints

3. FUNCTIONAL REQUIREMENTS

The POP unit test set is used to test the POP in the alignment and servo modes.

The unit test set tests these portions of the POP:

a) Mars sensor

b) Gimbal drive and pickoff

c) Gimbal electronics.

4. DESIGN REQUIREMENTS

4.1 Mode Control

The POP unit test set provides mode control discretes to the POP and tests its response to verify the ability to operate in the alignment mode and the servo mode.

4.2 Mars Sensor Alignment

The POP unit test set provides a simulated Mars horizon for testing the Mars sensor alignment. The simulated horizon is accurately aligned with respect to the Mars sensor at points through the range of ±180 degrees to an accuracy of ± 20 arc sec. and the Mars sensor error signal is measured through the range of ± volts to an accuracy of ± .03 per cent with a voltmeter having an input impedance of 100 k ohms minimum. For this test, the POP is in the alignment mode to prevent the gimbal electronics from nulling the Mars sensor error signal and the POP is tested 1 axis at a time.

4.3 Gimbal Positioning Accuracy

The POP unit test set tests the gimbal positioning accuracy in both the alignment mode and the servo mode.

4.3.1 Alignment Mode Positioning Accuracy

With the POP in the alignment mode, the POP test set provides a serial pulse train and an associated clock signal to load the POP alignment register with test angles. The position pickoff transducers on the POP gimbals are monitored to verify the positioning accuracy of the POP. The serial pulse train consists of a binary word of 12 bits maximum in conventional NRZ format. The pulse train has the following characteristics:

Amplitude	
True Level	+6.0 volts to +7.7 volts
False Level	Zero volts to +0.5 volts
Rise Time (False to True)	100 ns max., 10 to 90 per cent points
Fall Time (True to False)	150 ns max., 90 to 10 per cent points
Bit Rate	

The clock signal will have the following characteristics:

Amplitude	
True Level	+6.0 volts to +7.7 volts
False Level	Zero volts to +0.5 volts
Rise Time	30 ns max.
Fall Time	30 ns max.
Width	500 ± 10 ns

The position pickoff on POP gimbals is monitored by means of a reversible counter.

4. 3. 2 Servo Mode Positioning Accuracy

With the POP in the servo mode, the POP unit test set provides a simulated Mars horizon at various angles and the POP positioning accuracy is checked by monitoring the gimbal pickoffs and the Mars sensor error signals. This test is performed 1 axis at a time. The simulated Mars horizon is manually varied through the range of ± 180 degrees to an accuracy of 20 arc sec. The pickoff on the POP gimbals is monitored with a reversible counter and the Mars sensor error signals are monitored on a voltmeter.

4. 4 Telemetry Output Data

The POP unit test set checks the accuracy of the POP telemetry output data at appropriate times in the tests described above.

The contents of the two POP gimbal angle 12 bit registers will be displayed on status lights in binary form. In addition, the POP test set will provide a simulated telemetry readout clock to check the "readability" of the POP gimbal angle registers.

The POP analog signals to telemetry (voltages, temperatures, etc.) are checked with a voltmeter or ohmmeter.

5. FUNCTIONAL DESCRIPTION

The test set comprises standard commercial test equipment and specific equipment designed for testing functions peculiar to the Voyager POP.

A functional block diagram appears in Figure 1.

Figures 2 and 3 show the physical configuration of the POP unit test set. It consists of one rack of equipment and a bench mounted Mars horizon simulator.

5. 2 Specific Equipment Description

5. 2. 1 Special Test Equipment

a. Test Control and Monitor Panel

The test control and monitor panel contains switches and lights to control the application of power and provide mode control discretes to

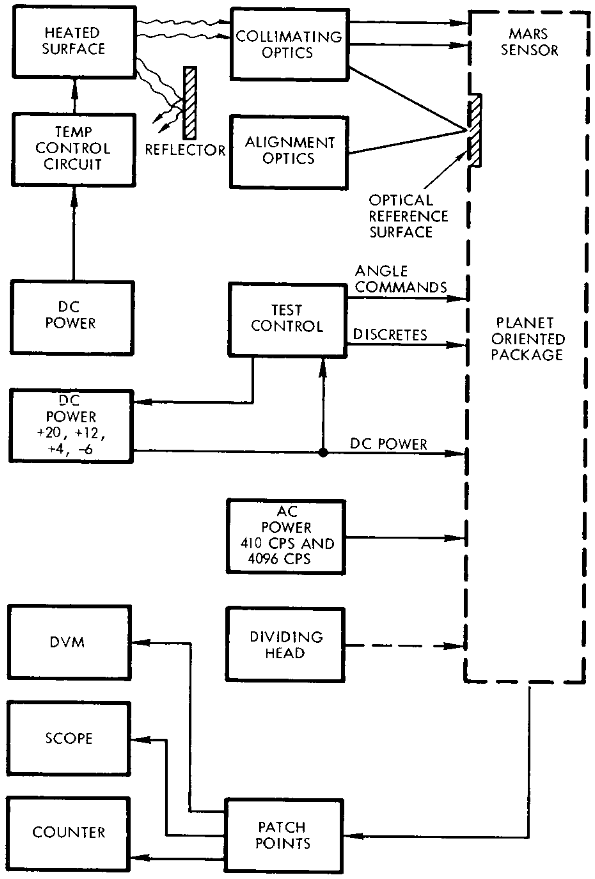


Figure 1. Planet-Oriented Package Unit Test Set, Block Diagram

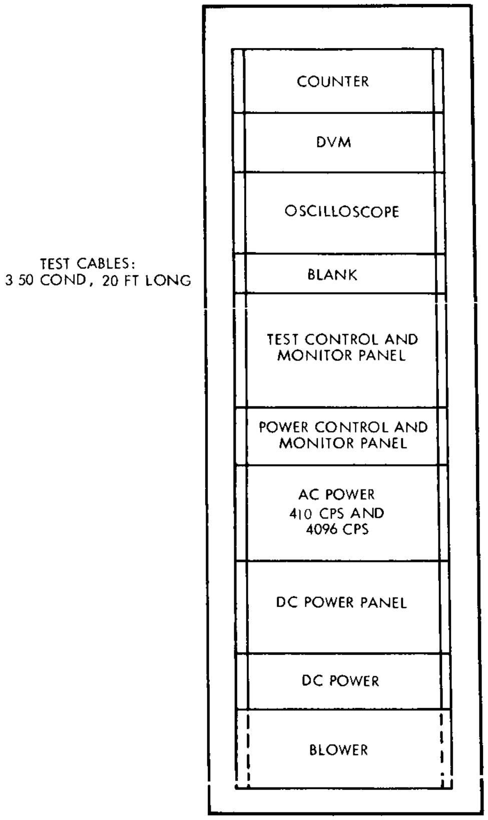


Figure 2. Planet Oriented Package Test Console

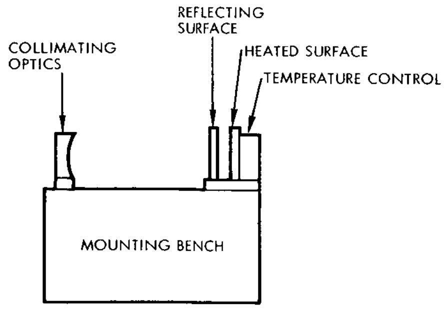


Figure 3. Artificial Horizon

the POP. Test patch points are provided on this panel to facilitate the application of the commercial test equipment to the POP. This panel includes circuits to generate a serial pulse train for commanding various POP gimbal angles. Front panel switches are provided to select the desired value of the angle and status lights are provided to display the angle stored in the POP telemetry register.

b. Power Control and Monitor Panel

The power control and monitor panel contain switches to control the application of power to the test set and the test item. Meters are included for coarse monitoring of the test item voltage and current.

c. AC Power Panel

The AC power panel generates 410 cps power for the gimbal drive motors and nominal 4096 cps power for the POP input power. The 410 cps power supply is as follows:

Voltage	±
Frequency	410 cps ±
Phases	Two at 90° ±
Power	—
Wave shape	Square wave

d. DC Power Panel

The DC power panel provides DC power for use in the test set as well as DC power required by the POP. The POP DC power is as follows:

Voltage	±
Ripple	—
Amps	—

e. Artificial Horizon

The artificial horizon provides a collimated source of heat with a step function temperature gradient of approximately 0 to 200 ± $^{\circ}F$. The higher temperature is radiated from an area (after collimation) approximately 2.5 x 5 degrees. Angular displacement of the simulated Mars horizon is provided by rotating the POP. A conventional rotary table with an angular reading accuracy of 20 arc sec is provided for this purpose. A mounting fixture having two orthogonal sides is provided to mount the POP to the rotary table.

5.2.2 Commercial Test Equipment

a. Frequency Counter

The frequency counter is capable of measuring frequencies between 10 cps and 2 mc to an accuracy of 3 parts in 10^6.

b. Digital Voltmeter/Ohmmeter

The digital voltmeter is capable of measuring voltages in the range of .001 volts to 999.99 volts DC to an accuracy of ± .03 per cent on the low ranges and ± .01 per cent on the high ranges. The input impedance is 100 k ohms minimum. An ohms converter is included for measuring temperature transducer resistance.

c. Oscilloscope

The scope is capable of measuring pulses as specified in 4.3 and 4.4 as well as various square wave audio signals.

6. BOUNDARY DEFINITIONS

6.1 Primary Power Source

The POP test set operates from a power source as specified below:

Voltage	115 ± 10 vac
Frequency	60 ± 1 cps
Phase	Single

6.2 Service Condition

Environment - Standard laboratory conditions.

7. PARAMETERS

a) Mode Control

b) Mars Sensor Alignment

c) Gimbal Positioning Accuracy

d) Telemetry Data

8. CONSTRAINTS

The test set is designed for use by trained electronic technicians.

PROPULSION SUBSYSTEM
OSE/VS-4-610

1. SCOPE

This document defines the general requirements, equipment list, and applicable documents for the propulsion subsystem mechanical operating support equipment required for the handling, testing, protection, transport, shipping, and storage of the propulsion subsystem equipment used in the Voyager program.

The models covered by this document conform to the requirements delineated herein and are identified as the VS-4-610 series.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-2-110	OSE Design Characteristics and Restraints

Government

Federal

PPP-B-621A 12 April 1963	Box, Wood, Nailed and Lock-Corner

Military

MIL-P-116D	Preservation, Methods of
MIL-D-3464B 31 October 1955	Desiccant (Activated) in Bags; for Static Dehumidification and Packaging
MIL-D-3716A Amendment 2 14 May 1962	Desiccants, Activated, for Dynamic Dehumidification
MIL-E-5556B Amendment 1 15 March 1963	Enamel, Camouflage, Quick Dry
MIL-M-008090D 21 February 1961	Mobility Requirements, Ground Support Equipment, General Specification for

MIL-C-9959 Amendment 1 5 February 1963	Container, Flexible, Reusable, Water-Vaporproof
MIL-P-27401B 19 September 1962	Propellant Pressurizing Agent, Nitrogen

DAC/MSSD

Mechanical Support Equipment and Facilities Manual

Other

ASME Pressure Vessel Code

3. REQUIREMENTS

The propulsion subsystem mechanical operating support equipment items defined in the following paragraphs are designed to perform their specified functions with simplicity of design and operation, adequate service life, and low manufacturing costs as prime considerations.

The items defined within this documentation group are associated with the handling, shipping, protection, testing, and storage of the propulsion subsystem equipment. The equipment enumerated below accomplishes these support functions.

Propulsion Subsystem OSE/VS-4-610

Item No.	Nomenclature
4-610-1	Sling, Retropropulsion Motor
4-610-2	Dolly, Retropropulsion Motor
4-610-3	Alignment Fixture, Retropropulsion Motor
4-610-4	Alignment Fixture, Midcourse Engine
4-610-5	Shipping Container, Retropropulsion Motor
4-610-6	Shipping Container, Midcourse Engine
4-610-7	Pneumatic Test Set
4-610-8	Pneumatic Fill Cart
4-610-9	Propellant Transfer and Handling Cart
4-610-10	Alignment Fixture, Midcourse Engine/Steering Vanes
4-610-11	Universal Handling Fixture, Hydrazine/Helium Tank
4-610-12	Sling, Hydrazine/Helium Tanks

3.1 Safety Requirements

3.1.1 Electrostatic Protection

The propulsion subsystem MOSE incorporates safety features to eliminate the hazards of static electricity when used to support the propulsion subsystem components. All MOSE coupled to these components is operated at the same ground potential.

3.1.2 Magnetic Fields

The equipment is fabricated of nonmagnetic materials or magnetic material which constrains the maximum magnetic environment to less than 80 oersteds at or around the subsystem components' physical envelope.

3.1.3 Personnel and Equipment Safety

All equipment includes safety features to preclude damage to the propulsion subsystem components or injury to operating personnel during functional performance of the equipment.

3.2 Material and Processes

3.2.1 Electrolytic Corrosion

The use of dissimilar metals in immediate contact which may result in corrosion by electrolytic action is avoided.

3.2.2 Fungi and Moisture Resistance

Those materials which resist the corrosive action of a moisture, saline, or fungi entrained environment are used unless otherwise required by design considerations.

3.3 Transportability and Storage

The equipment is designed for transportability by air or overland. The equipment is designed to perform after limited periods of storage in the natural environment of CONUS without rehabilitation.

3.4 Interchangeability

The design of the equipment requires tolerances no more stringent than are necessary to achieve interchangeability without departure from specified performance. All replaceable mechanical components of like part numbers are dimensionally and functionally interchangeable.

3.5 Workmanship

All MOSE is designed, manufactured, and assembled using workmanship consistent with the interests of economy and quality production methods.

3.6 Reliability

The MOSE is designed to provide the maximum degree of reliability consistent with program cost, schedule, and intended use of equipment. Designs are based upon proven methods and technology and at no time during use will there be degradation in the reliability of the propulsion subsystem equipment.

3.7 Maintainability

The MOSE is resigned so that repairs, adjustments, and overhaul can be readily accomplished by operating personnel using conventional, general purpose tools and equipment.

3.8 Identification and Marking

All MOSE carries adequate marking for identification with lift points, rated loads, hazard warnings, and special instructions noted.

SLING, RETROPROPULSION MOTOR
OSE/VS-4-610-1

1. SCOPE

This document defines the functional and design requirements and equipment description for the retropropulsion motor sling.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem
OSE/VS-4-610-5	Shipping Container, Retro-propulsion Motor

3. FUNCTIONAL REQUIREMENTS

The retropropulsion motor is hoisted and handled while being installed in, or removed from, its shipping container VS-4-610-5 and handling dolly, VS-4-610-2 and during mating to the flight spacecraft. Complete freedom of the motor case mounting flange and the spacecraft mounting structure interface is maintained during the mating operations.

4. DESIGN REQUIREMENTS

4.1 Loads

The sling is designed to handle the fully loaded motor weight of 3150 pounds.

4.2 Load Factors

The design load and handling factors are in accordance with OSE/VS-2-110.

4.3 Fasteners

Fasteners are compatible with the threads and hold pattern provided in the retropropulsion motor mounting flange.

4.4 Shackles

Standard shackles (three minimum) are used for cable and fittings.

4.5 Wire Rope

Wire rope is corrosion resistant and protected by vinyl coating.

5. EQUIPMENT DESCRIPTION

5.1 General

The retropropulsion motor sling consists of a channel ring bolted to the retropropulsion motor mounting flange. The ring prevents the mounting flange from deflecting locally at the hoist points. Three shackle attach brackets are welded to the channel ring at 120-degree intervals. A 3-point suspension cable sling shall attach to the shackles. The cables, D-ring, end fittings, and the channel ring interface surfaces are provided with vinyl coating to prevent abrasion to the motor case and its flange. This design concept is shown in Figure 1.

5.2 Interface Definition

The sling interfaces with the hooks of overhead or floor hoists and the retropropulsion motor mounting flange.

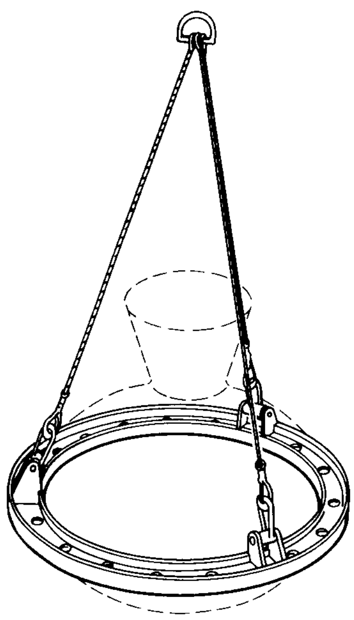


Figure 1. Sling, Retropropulsion Motor

DOLLY, RETROPROPULSION MOTOR
OSE/VS-4-610-2

1. SCOPE

This document defines the functional and design requirements and equipment description for the retropropulsion motor dolly.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents

TRW/ 1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem
OSE/VS-4-610-1	Sling, Retropropulsion Motor

3. FUNCTIONAL REQUIREMENTS

The retropropulsion motor dolly provides support for the retropropulsion motor in an inverted (thrust chamber up) attitude during non-firing tests, handling, storage, and intraplant transportation.

4. DESIGN REQUIREMENTS

4.1 Mobility

The retropropulsion motor dolly is designed for Type I, Class 2 mobility specified in MIL-M-008090D. The running gear consists of 4 rubber treaded casters with step-lock parking brakes. At least 2 adjacent casters provide full swivel. Shock absorbers or a suspension system is required to mitigate shock loads imposed on the motor during transportation. A tow bar is provided to accommodate manual and/or shop mule towing.

4.2 Loads and Load Factors

The dolly is capable of supporting the weight of the retropropulsion motor, approximately 3150 pounds. The dolly frame is rigid and the motor support structure provides adequate bearing surface to prevent deformation of the retropropulsion motor mounting flange.

The dolly conforms to the design loads and handling factors of OSE/VS-2-110.

4.3 Support and Attitude

The dolly supports the retropropulsion motor at its circular mounting flange with the thrust chamber pointing up. Adequate clearance is provided between the bottom of the motor case and the dolly frame structure.

5. EQUIPMENT DESCRIPTION

5.1 General

The retropropulsion motor dolly consists of a rectangular aluminum frame, which mounts four shock-mounted casters. Four floor pads will be provided for stabilizing the dolly during installation of the motor. The base frame also contains a military standard towbar assembly for attachment to a prime mover. The motor support structure consists of a circular aluminum ring supported by a truss structure welded to the dolly frame. The ring contains a teflon, vinyl, or equivalent bonded surface to prevent abrasion to the motor mounting flange, with suitable holes for securing the motor. The motor mounting flange rests on the ring, and quick disconnect fasteners are used to secure the motor. An aluminum tubular framework is provided for protection of the thrust chamber. This framework is installed after the solid motor has been secured to the mounting ring. This design concept is shown in Figure 1.

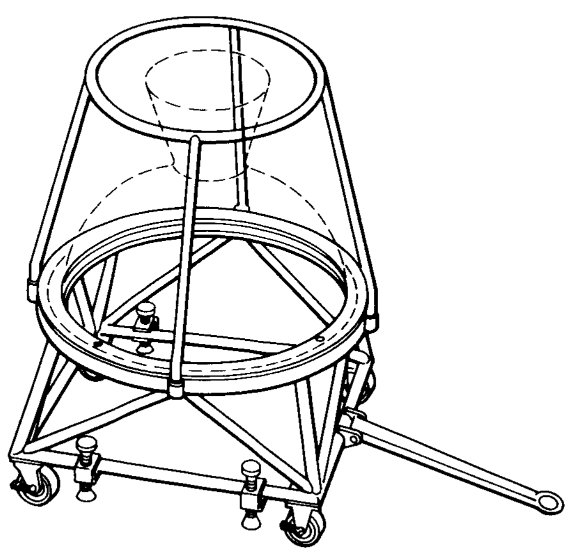


Figure 1. Dolly, Retropropulsion Motor

5.2 Interface Definition

The dolly interfaces with the mounting flange of the retropropulsion motor. The dolly is used in conjunction with the retropropulsion motor sling during installation.

ALIGNMENT FIXTURE, RETROPROPULSION MOTOR
OSE/VS-4-610-3

1. SCOPE

This document defines the functional and design requirements and equipment description for the retropropulsion motor alignment fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-4-410	Voyager OSE Support Equipment, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

The alignment fixture supports level vials and alignment targets along an extension of the retromotor thrust vector.

4. DESIGN REQUIREMENTS

4.1 Accuracy

The fit of the fixture in the nozzles, the location of the targets, and the mounting of the level vials are accomplished to an accuracy at least 1 order of magnitude greater than required by the alignment specifications of the retropropulsion motor.

4.2 Installation

The fixture is installed in the nozzle without deflecting the nozzle and without damage to the nozzle surface finish. The fixture clamps to the nozzle.

5. EQUIPMENT DESCRIPTION

5.1 General

The fixture consists of a conical plug whose taper matches the inside of the nozzle. It has an extension from the large end which is on the geometric centerline of the nozzle. A ball level (or 2 level vials) is mounted on this extension. The levels are accurate to the geometric centerline of the nozzle to $\pm$ 0.02 degrees or less. The extension carries 8 optical alignment targets which describe the longitudinal geometric axis of the nozzle. This design concept is shown in Figure 1.

5.2 Interface Definition

The fixture interfaces mechanically with the nozzle and optically with the basic reference system.

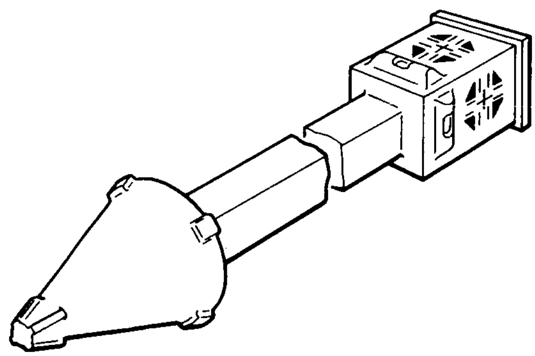


Figure 1. Alignment Fixture, Retropropulsion Motor

ALIGNMENT FIXTURE, MIDCOURSE ENGINE
OSE/VS-4-610-4

1. SCOPE

This document defines the functional and design requirements and equipment description for the midcourse engine alignment fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-4-410	Voyager Operational Support Equipment, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

The alignment fixture supports level vials and alignment targets along an extension of the midcourse engine thrust vector.

4. DESIGN REQUIREMENTS

4.1 Accuracy

The fit of the fixture in the nozzles, the location of the targets, and the mounting of the level vials are accomplished to an accuracy at least one order of magnitude greater than required by the alignment specifications of the midcourse engine.

4.2 Installation

The fixture is installed in the nozzle without deflecting the nozzle and without damage to the nozzle surface finish. The fixture clamps to the nozzle.

5. EQUIPMENT DESCRIPTION

5.1 General

The fixture consists of a conical plug whose taper matches the inside of the nozzle. It has an extension from the large end which is on the geometric centerline of the nozzle. A ball level (or 2 level vials) are mounted on this extension. The levels are accurate to the geometric centerline of the nozzle to $\pm$ 0.02 degrees or less. The extension carries

8 optical alignment targets which describe the longitudinal geometric axis of the nozzle. This design concept is shown in Figure 1.

5.2 Interface Definition

The fixture interfaces mechanically with the nozzle and optically with the basic reference system.

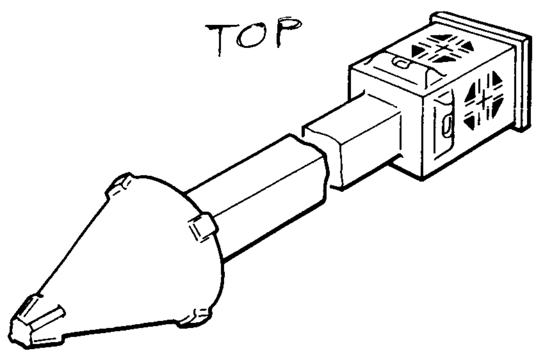


Figure 1. Alignment Fixture, Midcourse Engine

SHIPPING CONTAINER, RETROPROPULSION MOTOR
OSE/VS-4-610-5

1. SCOPE

This document defines the functional and design requirements and equipment description for the retropropulsion motor shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/ 1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem
OSE/VS-4-610-1	Sling, Retropropulsion Motor

3. FUNCTIONAL REQUIREMENTS

The retropropulsion motor shipping container provides environmental protection for the retropropulsion motor during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the retropropulsion motor from physical damage during truck and air transportation and during periods of storage.

4.2 Weight and Size

The weight and size are minimal within the constraints of providing the desired protection. The retropropulsion motor weight is approximately 3150 pounds and measures about 53 inches in length and 48 inches in diameter.

4.3 Orientation

The thrust chamber is oriented upward in the container.

4.4 Storage

The retropropulsion motor is protected within the container for a minimum of 2 years.

4.5 Altitude

The shipping container is capable of operating at altitudes consistent with air transportation.

4.6 Venting

When required, venting provisions are incorporated for air transport at altitudes from sea level to 20,000 feet. Venting occurs through desiccants.

4.7 Transportability

The container is capable of being transported by truck or air.

4.8 Fork Lift and Tiedowns

The container contains tiedown rings for truck and air transportation and accommodates a standard forklift.

4.9 Environment

4.9.1 Shock and Vibration

Shock and vibration isolation is provided to reduce the imposed loads on the motor to less than that occurring during flight environments.

4.9.2 Humidity

The relative humidity is less than 50 percent within a temperature range of 30 to 100°F.

4.9.3 Condensation and Corrosion

No condensation or corrosive atmosphere is permitted within the container.

4.9.4 Temperature

The temperature within the container is maintained within a range of 30 to 100°F.

4.10 Load Factors

All load and handling factors are in accordance with OSE/VS-2-110.

4.11 Reusability

The shipping container is reusable.

4.12 Hoist Points

The shipping container is capable of being hoisted by standard cables and handling slings.

4.13 Conductivity

Conductive material is used.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping container is fabricated from aluminum honeycomb panels. The container exterior is painted white. The paint conforms to MIL-E-5556 and shall have an emissivity of 0.8.

The container has a rectangular configuration. The top half of the container is latched securely to the bottom half. A static, free-breathing desiccant canister using desiccant conforming to MIL-D-3716, Type IV, is mounted to the end of the container. The breather has a wire mesh screen to prevent any desiccant influx to the container. In addition, a relief valve for nitrogen purging and pressure relieving, a connection to accommodate a heater when required, and a box for inspection records are mounted to the ends of the container. The container lid opens so the retropropulsion motor mounting flange is accessible to lift the motor from the container. Four metal lift and tiedown rings, and aluminum reinforced channels for forklift capabilities are mounted to the container base structure.

The motor is placed in the container with the thrust chamber oriented upward. The motor case is encapsulated with 1 of the following foam materials: polyurethane, polyethylene, or rubberized hair. The cushions (2 halves) are bonded to the walls of the shipping container. The cushion thickness is determined by the shock and vibration dampening requirements. The cushioning material is covered with a carbonized conductive polyethylene (velostat) material to prevent static charge buildup. The foam nests the motor to distribute the load equally.

The container halves, including its attachments (desiccant breather, relief valve and heater connection), are hermetically sealed; breathing is accomplished through the desiccant canister.

Prior to shipment, the container is purged to a dew point of $-32^{\circ}F$. A heater (government furnished equipment) accompanies the shipment of each motor case when the transportation or storage environment exceeds the temperature requirements of this specification. All closure fastenings on the container are such that only common hand tools are required to open and close the container. The container is secured to the transporting vehicle through its tiedown rings. In addition, the motor is grounded to the shipping container at all times. This design concept is shown in Figure 1.

5.2 Interface Definition

The shipping container, depending on transportation and storage conditions, is interface with a Government furnished heater. No physical or electrical interface with other support equipment is required. The container design allows compatible use of the retropropulsion motor sling.

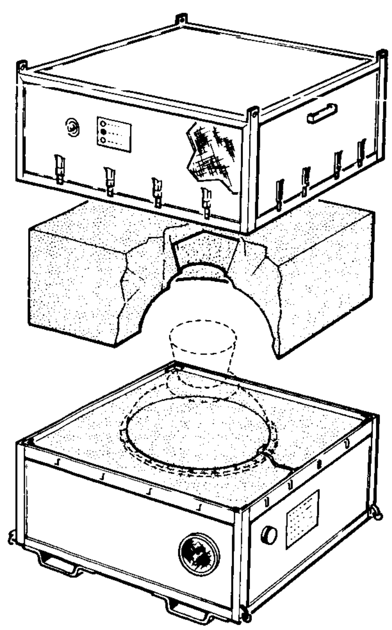


Figure 1. Shipping Container, Retropropulsion Motor

SHIPPING CONTAINER MIDCOURSE ENGINE
OSE/VS-4-610-6

1. SCOPE

This document defines the functional and design requirements and equipment description for the midcourse engine shipping container.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

The shipping container provides environmental protection for the midcourse engine during transportation and storage.

4. DESIGN REQUIREMENTS

4.1 Physical Protection

The shipping container protects the midcourse engine from physical damage during truck and air transportation and during periods of storage.

4.2 Weight and Size

The weight and size are minimal within the constraints of providing the desired protection. The midcourse engine weight is less than 10 pounds and presents an envelope size of approximately 6 x 4 x 4 inches.

4.3 Altitude

The shipping container is capable of operating at altitudes consistent with air transportation.

4.4 Transportability

The container is capable of being transported by truck or air.

4.5 Reusability

The shipping container is reusable.

4.6 Environment

The container protects the engine against corrosion and condensation.

4.7 Load Factors

All load and handling factors are in accordance with OSE/VS-2-110.

5. EQUIPMENT DESCRIPTION

5.1 General

The shipping equipment consists of foam chocks, an environmental cover (barrier material), and an exterior shipping container.

The midcourse engine is supported by foam chocks fabricated from 2.4 pounds density polyurethane foam.

The midcourse engine is enclosed in a barrier material conforming to MIL-C-9959, Class II, Grade B, Amendment 1, 5 February 1963. The barrier material contains desiccant bags conforming to MIL-D-3464, with a humidity indicator window capable of being easily inspected. The desiccant is changed when the indicator shows a relative humidity of more than 20 percent. The required desiccant quantity is calculated in accordance with MIL-P-116D, paragraph 3.5.6. Prior to shipment, the barrier material is purged with dry nitrogen to a 0° dew point, desiccated, and evacuated. The midcourse engine in its foam chocks, and enclosed in its barrier, is placed in a reusable wooden container conforming to PPP-B-621. The foam chocks are bonded to the wooden container to allow ease of midcourse engine removal. This design concept is shown in Figure 1.

5.2 Interface Definition

The shipping container is used to store and transport the midcourse engine, but has no physical or electrical interface with other operating support equipment.

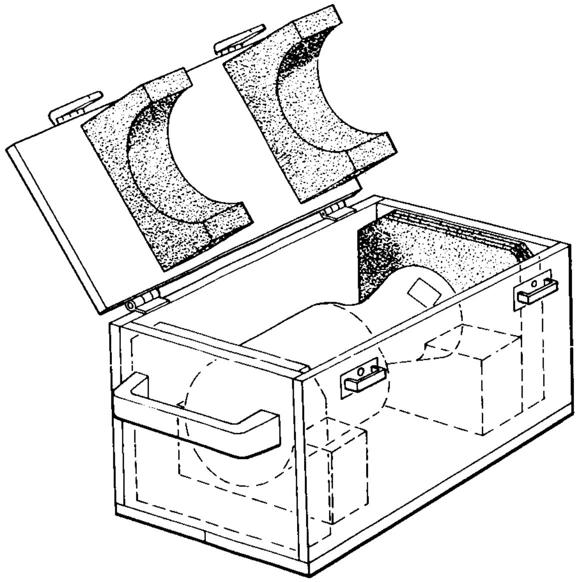


Figure 1. Shipping Container, Mid-Course Engine

PNEUMATIC TEST SET
OSE/VS-4-610-7

1. SCOPE

This document defines the functional and design requirements and equipment description for the Pneumatic Test Set.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW/1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

Internal and external leakage tests are performed on the Voyager mid-course propulsion subsystem and stabilization and control subsystem, and the engine solenoid valves are operated to verify function and timing characteristics. Provision is required for applying, directing, and containing internal pneumatic pressure in the range of 50 to 400 psig in the propellant passages of the midcourse propulsion system to perform the leakage tests and to verify operation of the pressure transducer and plenum chamber pressure switches. A nozzle throat plug is provided to contain this pressure in the Midcourse Engine during engine leak check. A requirement also exists to provide Nitrogen at a maximum pressure of 3000 psig for leak tests of the SCS.

The pneumatic test set detects and measures the leakage rate (based on a maximum allowable rate). Adapters, hoses, and electrical cables are supplied to attach to the engine and facility interfaces.

4. DESIGN REQUIREMENTS

The pneumatic test set is grouped by subfunction, including pneumatic console, nitrogen supply, helium supply, Freon-12 supply, throat plug, and interconnecting plumbing and cabling.

4.1 Pneumatic Console

The console contains the instruments, control valves, and pneumatic and electrical connectors required to pressurize the propulsion system and operate the engine control valves. Readout and indication of engine pressure transducer operation and plenum chamber pressure switch actuation are also provided.

4.1.1 Instrumentation

Instruments are contained in the console to measure:

Pressure:	0-100 psig gas pressure (1 percent accuracy)
	0-500 psig gas pressure (1 percent accuracy)
	0-5,000 psig gas pressure (1 percent accuracy)

4.1.2 Valve Actuation

Controls and indicators are required to actuate the airborne valves and verify their operation.

4.1.3 Valve Actuation Rate

The test set incorporates provisions for measuring propellant valve actuation rate for both closed-to-open and open-to-closed cycles.

4.1.4 Engine Pressure Transducer Operation

The test set incorporates provisions for verifying engine pressure transducer operational accuracy.

4.1.5 Panel Layout

The panel layout indicates interconnections between panel-located items.

4.2 Internal and External Leakage

Provisions for pressurizing the spacecraft propulsion system and detecting internal and external leakage are incorporated in the test set. Detection is based on a specified maximum allowable leakage rate.

4.2.1 Gas Supply

Nitrogen and/or halogen gas, as well as helium and/or halogen gas, are supplied from commercial bottles. Provisions are made to regulate, filter, and measure gas pressure.

4.2.2 Nozzle Plug

A nozzle throat plug (capable of withstanding 250 psig) is required to seal the thrust chamber.

4.2.3 Leakage Detector

A commercial halogen detector with a sensitivity of 10^{-4} standard cm^3 of Freon 12 per second is required.

4.3 Mobility

The pneumatic test set is designed in accordance with the requirements stipulated for Type I, Class I mobility per MIL-M-008090D. The running gear consists of four swivel casters with swivel locks and parking brakes. No spring suspension systems or shock absorbers are required.

4.4 Load Factors

The equipment is designed in accordance with the OSE/VS-2-110, where applicable.

5. EQUIPMENT DESCRIPTION

5.1 General

The pneumatic test set contains all components required to leak-test the spacecraft midcourse propulsion and stabilization and control subsystems, and to control and indicate engine valve actuation.

Visual readout of engine pressure transducer operation is provided. The test set consists of a wheel-mounted pneumatic console, an electrically operated portable halogen leak detector, a nozzle throat plug, Freon-12 and dry-nitrogen and helium gas supplies, and the required flexible pneumatic hoses and electrical cabling.

The pneumatic console is designed for manual movement. The console contains the required regulators, valves, and gages to control and monitor pneumatic pressurization of the spacecraft propulsion system during leak-check operations.

Pneumatic test line connections are located on the top of the cabinet structure for ease of connection and disconnection. An electrical control panel provides control and readout of spacecraft engine valve actuation, main power status (current and voltage), engine pressure transducer operation, and main power shutoff. Electrical connectors are located on the rear of the cabinet for electrical interconnection with the spacecraft, power supply, and the facility visirecorders. These recorders are used to determine the timing characteristics of spacecraft valve operation. The console cabinet has two front mounted doors for access to the console pneumatic and electrical systems. The console provides storage space for the following items when not in use:

a) The halogen leak detector

b) Required pneumatic flex hoses and electrical cabling

c) Nozzle throat plug.

The halogen leak detector is a standard commercial unit (hand-carried "gun" type) capable of leak detection sensitivity of 10^{-4} standard cm^3 of Freon 12 per second.

The nozzle throat plug allows thrust chamber pressurization through the plug body.

Freon-12 and helium gas supplies are from commercial K-bottles which shall be emplaced in an area compatible with local safety regulations. Nitrogen is from the facility high-pressure nitrogen supply.

The pneumatic test set concept is shown in Figure 1. A schematic of the system is shown in Figure 2.

5.2 Interface Definition

The Pneumatic Test Set interfaces with the spacecraft electrical and instrumentation systems, facility electrical power, facility visirecorders, spacecraft pneumatic and propellant fittings, and the nitrogen and Freon 12 gas supplies.

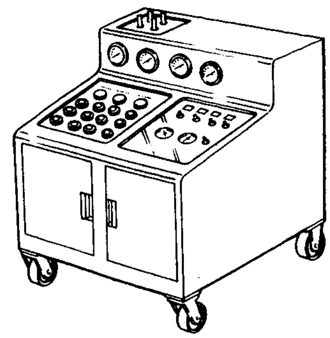


Figure 1. Pneumatic Test Set

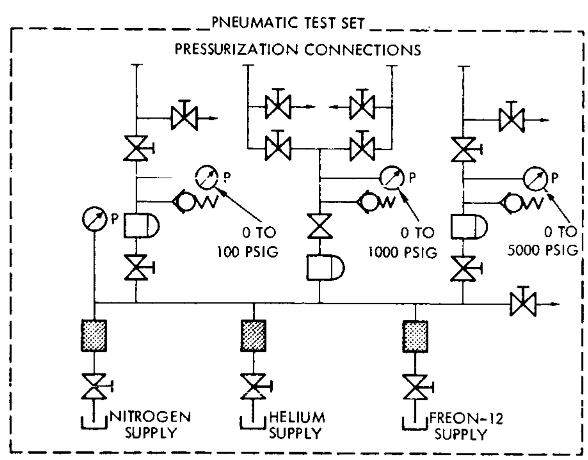


Figure 2. Pneumatic Test Set, Schematic

PNEUMATIC FILL CART
OSE/VS-4-610-8

1. SCOPE

This document defines the functional and design requirements and the equipment description for the pneumatic fill cart.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-610 Voyager OSE, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

The stabilization and control gas system (SCS) and propellant pressurization system require loading equipment. The equipment ensures that an adequate supply of nitrogen and helium gas has been safely loaded on the spacecraft. Loading operations are conducted at the safe assembly facility at AFETR and at contractor facilities.

4. DESIGN REQUIREMENTS

4.1 Components

The pneumatic fill cart provides:

4.1.1 Pressure

The pressurization system pressurizes the spacecraft stabilization and control gas system to 3000 psig and the propellant pressurization system to 350 psig.

4.1.2 Regulation

The fill system regulates proper loading pressure.

4.1.3 Interface

The fill system connects to the SCS and propulsion pressurant fill valve connection and the spacecraft instrumentation system.

4.1.4 Fluids

Dry nitrogen per MIL-P-27401A and Bureau of Mines and Grade A helium are the pressurants.

4.1.5 Nitrogen Supply Tank

An ASME-certified nitrogen tank is used.

4.1.6 Helium Supply Tank

Approved commercial helium K-bottles are used.

4.2 Safety

The high-pressure helium and nitrogen tanks are mounted in accordance with AFETR safety regulations, and relief valves are provided in the fill cart and at the spacecraft fill fitting.

4.3 Mobility

The pneumatic fill cart is designed for Type I, Class I mobility per MIL-M-008090D. The running gear consists of four swivel casters with swivel locks and parking brakes. No spring suspension systems or shock absorbers are required.

5. EQUIPMENT DESCRIPTION

5.1 General

The pneumatic fill cart consists of 4000-psig nitrogen supply tank, a 2200-psig helium supply (K-bottles), appropriate safety devices, tank and supply line pressure gages, spacecraft bottle temperature and pressure monitor gages, and a portable cabinet assembly.

The nitrogen supply tank is fabricated and certified per the ASME pressure vessel code and shall be mounted per AFETR safety regulations. The helium supply is from commercially available K-bottles mounted per AFETR regulations.

Safety devices minimize safety hazards due to supply tank, line, and Spacecraft overpressurization. They include regulator and supply line and supply tank relief valves.

Filtration of nitrogen and helium gas is accomplished by in-line 10-micron filters.

The cart cabinet assembly includes a control panel which shall govern pressure regulation and valve operations. The panel also mounts gages for readout of the nitrogen supply tank pressure, helium supply pressure, line pressure, and spacecraft nitrogen and helium bottle temperatures and pressures. The latter two readouts are required during loading operations to ensure correct gas quantity loading, and preclude too-rapid gas loading which could result in spacecraft bottle overheating and possible bottle failure. Calibration curves are required to relate various bottle temperatures and pressures to mass of gas loaded.

Miscellaneous equipment items not an integral part of the pneumatic fill cart include a flexible supply hose, a line-mounted relief valve, suitable spacecraft and fill cart connector couplings, temperature and pressure instrumentation cabling, and electrical power cabling. A schematic of the pneumatic fill cart is shown in Figure 1, and the cart design concept is shown in Figure 2.

5.2 Interface Definition

The flexible nitrogen supply hose mates compatibly with the SCS fill valve. The flexible helium supply hose mates compatibly with the propulsion pressurant fill valve. The instrumentation cabling is suitable for connection to the spacecraft instrumentation system. The electrical power cabling is suitable for connection to facility power supplies.

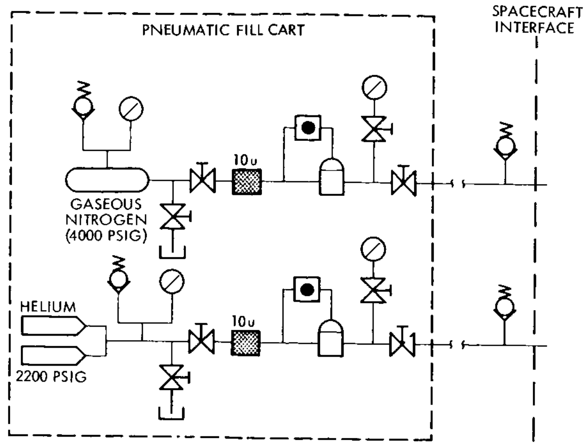


Figure 1. Pneumatic Fill Cart, Schematic

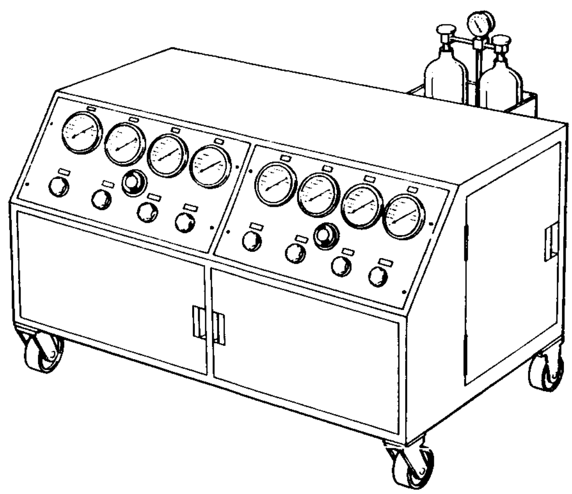


Figure 2. Pneumatic Fill Cart

PROPELLANT TRANSFER AND HANDLING CART
OSE/VS-4-610-9

1. SCOPE

This document defines the functional and design requirements and equipment description for the propellant transfer and handling cart for the midcourse propulsion system.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-610 Voyager OSE, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

The propellant transfer and handling cart loads propellant at the AFETR safe assembly facility before the spacecraft is transported to the launch site and at the static firing facility. This unit performs the following functions:

a) Store an adequate amount of hydrazine (N_2H_4) to fill the spacecraft propellant tanks

b) Transfer propellant from the cart to the spacecraft and load the spacecraft with approximately 250 lbs of hydrazine to an accuracy of 1 pound

c) Evacuate the fill line and hydrazine tank bladders to ensure that no trapped gas is present during filling operations

d) Provide a means to purge the transfer lines of propellant after loading operations

e) Provide propellant transfer recirculation to ensure filling of the transfer line prior to spacecraft tank filling.

f) Provide propellant filtration prior to filling spacecraft tankage

g) Measure propellant bulk temperature to ±2°F accuracy

h) Provide safe venting of hydrazine vapors from the unit during postloading and emergency operations.

4. DESIGN REQUIREMENTS

4.1 Propellant Loading

The propellant loading design provides for the following.

4.1.1 Evacuation

A vacuum system is required to evacuate the propellant side of the propellant tank bladders, propulsion system piping, and ground transfer lines to ensure that no gas is entrapped during the loading operation.

4.1.2 Recirculation

A propellant recirculation system is required to ensure that the transfer line is completely filled with propellant up to the spacecraft propellant-fill valve. This is required to guarantee loading accuracy.

4.1.3 Purge

A purge system is required to empty the propellant lines prior to disconnection after loading.

4.2 Cleanliness

All pneumatic and propellant components and subassemblies used in fabrication of the propellant transfer and handling cart are processed prior to assembly per applicable cleanliness requirements necessary for the safe and reliable operation of the spacecraft system. Before and after use, all flex hoses, disconnect fittings, etc., require recleaning to preclude contamination of the spacecraft pneumatic and propellant system.

4.3 Defueling

The cart defuels the fully loaded spacecraft. This is done by applying helium purge pressure to the helium side of the hydrazine tank bladders and forcing the propellant back into the transfer and handling cart. Evacuation and purge of these lines and the propellant tanks are required after off-loading.

4.4 Filtering

Propellant supply lines are equipped with filters at the transfer and handling cart, which will effect 10-micron filtration.

4.5 Venting

Means for venting the propellant lines and tankage to a safely remote area are provided. The vacuum pump exhaust is also directed to this remote area.

4.6 Fittings

All handling cart lines with external fittings are provided with caps to be installed when the unit is not in use.

4.7 Material Compatibility

The materials selected for use in all components in contact with hydrazine are of acceptable compatibility.

4.8 Traps

Plumbing design minimizes the possibility of trapping air or contaminants in the plumbing and allows venting or circulation to eliminate trapped air.

4.9 Safety

All motors, electrical starters, controls, and instruments are explosion-proof and approved for use in the safe assembly facility at AFETR.

4.10 Mobility

The propellant transfer and handling cart is designed for Type I, Class 1 mobility per MIL-M-008090D. The running gear consists of 4 swivel casters and swivel locks and parking brakes. No spring suspension system or shock absorbers are required.

4.11 Load Factors

The equipment is designed to load factors in accordance with OSE/VS-2-110.

5. EQUIPMENT DESCRIPTION

5.1 General

The propellant transfer and handling cart is a wheel mounted unit capable of loading approximately 250 pounds of hydrazine on board the spacecraft propulsion system. The unit consists of a cabinet containing a 50-gallon stainless steel propellant supply tank, a 20-gallon stainless steel waste tank, a vacuum pump, and propellant, vacuum, and pneumatic lines and controls.

The stainless steel propellant supply tank includes pressure and temperature gages for parameter readout. The tank is approximately 1.25 feet diameter and 5.4 feet long. The tank is mounted vertically (on end) and fitted with a top-mounted vent and pressurization fitting and a bottom-mounted propellant feed and drain fitting. Fabrication is per the ASME pressure vessel code; tank design working pressure is approximately 100 psig. A vertical sight glass is installed for the full length of the tank with a calibrated scale for determining the amount of propellant in the tank. The tank geometry is such that each 1/8 inch increment of liquid seen in the sight glass will represent a propellant volume of less than 1 pound in weight, thereby providing the means to gage propellant loading within ±1 pound accuracy. Sightglass tank pickup points are at the tank top and bottom and lie on the tank vertical centerline (Z axis). Sight-glass sensing at these points minimizes the loading error if the tank level is disturbed slightly during loading operations. A liquid level sensor provides an additional means of determining the spacecraft propellant tank load. Readout is on the cart control panel. The waste tank is a receiver for propellant during recirculation. This tank is fabricated of stainless steel and has a capacity of 20 gallons. Integral tank ports are provided for recirculation, venting, and drainage. Both tanks include vent systems for venting gas outside of the safe assembly facility to a remote area.

The transfer cart contains an evacuation system consisting of a vacuum pump and controls for drawing the transfer lines and spacecraft midcourse engine propellant system down to a vacuum of 1 mm of Hg in a time compatible with the loading schedule. The vacuum pump exhaust is directed outside of the safe assembly facility.

A pneumatic control system regulates and controls the helium gas required for propellant supply tank pressurization and purge operations.

Miscellaneous items required for use with the propellant transfer and handling cart include hydrazine-compatible flexible hoses, an adapter fitting for use at the spacecraft fill fitting, a relief valve, and a sight glass.

The propellant transfer and handling cart is fueled with hydrazine from existing support facilities; emplaced near the spacecraft, leveled, and the required hookups made. The spacecraft fuel tank bladders and propellant piping are evacuated with the cart purge-and-evacuation system. The purge gas in the bladder is vented and the vacuum pump shut off when a vacuum of approximately 1 mm of Hg has been established in the system.

The bulk temperature of the hydrazine in the propellant supply tank is measured, and the volume of hydrazine to be loaded is determined.

Controls are activated to pressurize the propellant supply tank. With the spacecraft fill valve closed, the propellant circulates from the propellant supply tank to the waste tank until bubble-free hydrazine is flowing through the sight glass at the vehicle and the pre-established propellant tank sight glass level has been attained.

The spacecraft fill valve is slowly opened and the filling operation initiated. When the desired level is indicated on the propellant supply tank sight glass, the spacecraft valve is closed and the propellant supply tank vented. A helium purge forces residual propellant in the transfer lines back into the propellant supply and waste tanks.

If the spacecraft is to be defueled, purge gas is introduced on the helium side of the propellant tank bladders, forcing the propellant back into the cart tankage. The propellant cavity and tankage are evacuated to allow vaporization of residual hydrazine. Dry helium purge gas is introduced to the propellant cavity for another evacuation cycle. The propellant tanks are then pressurized with a blanket helium pressure to prevent entrance of moisture into the spacecraft system. A schematic of the propellant handling and transfer cart is shown in Figure 1 and a pictorial representation of the cart design concept is shown in Figure 2.

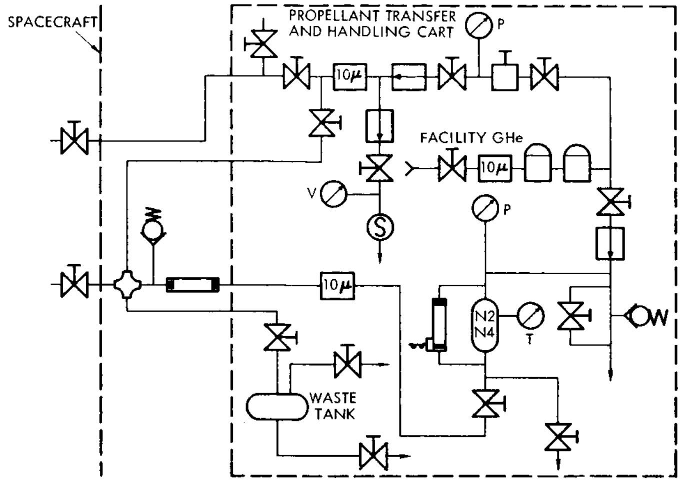


Figure 1. Propellant Transfer and Handling Cart, Schematic

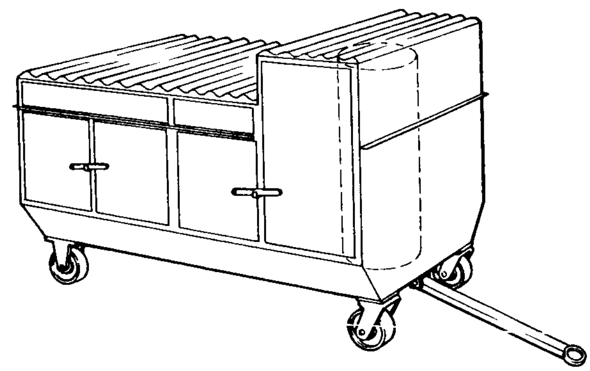


Figure 2. Propellant Transfer and Handling Cart

5.2 Interface Definition

The propellant transfer and handling cart interfaces with the airborne fill and drain valve and gaseous helium pressurant vent valve; the facility gaseous vent system; facility electrical power; and the facility propellant servicer.

ALIGNMENT FIXTURE, MIDCOURSE ENGINE/STEERING VANES
OSE/VS-4-610-10

1. SCOPE

This document defines the functional and design requirements and equipment description for the midcourse engine/steering vanes alignment fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem

3. FUNCTIONAL REQUIREMENTS

The fixture supports a ball level (or 2 level vials) attached to the steering vane subassembly.

4. DESIGN REQUIREMENTS

4.1 Accuracy

The fixture is fabricated to an accuracy level at least 1 order of magnitude greater than required by the specifications for the alignment of the steering vane subassembly.

4.2 Installation

The fixture is installed without interference with the steering vanes and without damage to the vanes surface finish. The fixture is clamped or bolted to the steering vane subassembly.

5. EQUIPMENT DESCRIPTION

5.1 General

The fixture consists of a spider assembly which bolts to the steering vane structure and which supports a ball level or orthogonally oriented level vials to indicate zero deflection of the vane subassembly. The alignment fixture design concept is shown in Figure 1.

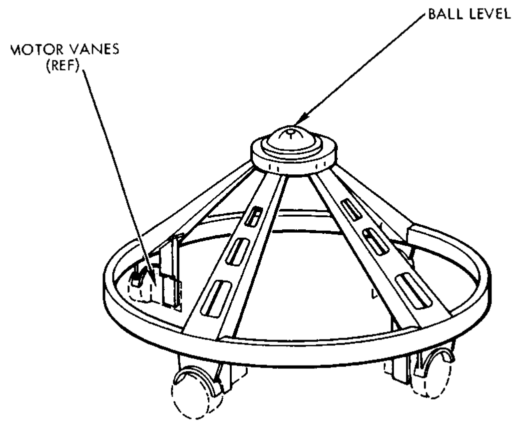


Figure 1. Alignment Fixture, Midcourse Engine/Steering Vanes

5.2 Interfaces

The fixture interfaces mechanically with the steering vane subassembly.

UNIVERSAL HANDLING FIXTURE, HYDRAZINE/HELIUM TANK
OSE/VS-4-610-11

1. SCOPE

This document defines the functional and design requirements and equipment for the hydrazine/helium tank universal handling fixture.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem
OSE/VS-4-610-10	Sling, Hydrazine/Helium Tank

3. FUNCTIONAL REQUIREMENTS

The hydrazine/helium tank universal handling fixture provides support and protection to the tanks during testing, storage, and intraplant transportation. It is adaptable to handling the attitude control gas tanks.

4. DESIGN REQUIREMENTS

4.1 Physical Characteristics

The fixture provides continuous support for the spherical tank 45 degrees below the tank equator. The tank diameter is 24 inches.

4.2 Load Factors

The design load and handling factors are in accordance with OSE/VS-2-110. The supporting cradle and straps provide adequate bearing surface to prevent deformation of the tank.

5. EQUIPMENT DESCRIPTION

5.1 General

The hydrazine/helium tank universal handling fixture consists of an aluminum sheet metal cradle, lined with a soft, cushioning material. The cradle is supported by stiffeners welded to 4 tubular legs. Two fabric strap assemblies are attached to the upper surface of the cradle. These straps contain a buckle type clinching mechanism for retention of the tank sphere in the cushioned cradle. A tubular cage structure is attached to the 4 supporting legs with quick disconnect fittings. An insert adapter allows handling of the smaller (15 in. diam.) attitude control high-pressure gas tanks. This fixture design concept is shown in Figure 1.

5.2 Interface Definition

The hydrazine/helium tank universal handling fixture interfaces with the tanks and is used compatibly with the hydrazine/helium tank sling, OSE/VS-4-610-12.

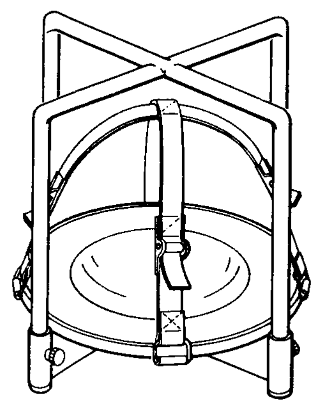


Figure 1. Universal Handling Fixture, Hydrazine/Helium Tank

SLING, HYDRAZINE/HELIUM TANK
OSE/VS-4-610-12

1. SCOPE

This document defines the functional and design requirements and equipment description for the hydrazine/helium tank sling.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are:

TRW 1971 Voyager OSE Design Documents

OSE/VS-4-610	Voyager OSE, Propulsion Subsystem
OSE/VS-4-610-10	Universal Handling Fixture, Hydrazine/Helium Tank

3. FUNCTIONAL REQUIREMENTS

The 24 inch diameter hydrazine/helium tank shall be lifted and supported during assembly, testing, and mating operations to the propulsion/pneumatic structural section.

4. DESIGN REQUIREMENTS

4.1 Loads

The sling is designed to carry 50 pounds.

4.2 Load Factors

The design load and handling factors are in accordance with OSE/VS-2-110.

4.3 Shackles

Standard shackles are used for cable end fittings.

4.4 Material

Wire rope is corrosion resistant and protected by vinyl coating.

5. EQUIPMENT DESCRIPTION

5.1 General

The hydrazine/helium tank sling consists of a marmon clamp handling ring that is padded to protect the tank surface and mechanically limited to a minimum bearing pressure. Three shackle attach brackets are welded to the clamp body at 120 degree intervals. A 3 leg cable sling attaches to the shackles. The cables are attached to a D-ring for hoist hook attachment. The cables, D-ring, end fittings, and handling ring surfaces are coated with vinyl to prevent scratching of the hydrazine/ helium tank. The sling design concept is shown in Figure 1.

5.2 Interface Definition

The sling is compatible with hooks of overhead hoists or portable floor hoists. It is designed to function compatibly with the hydrazine/ helium tank universal handling fixture.

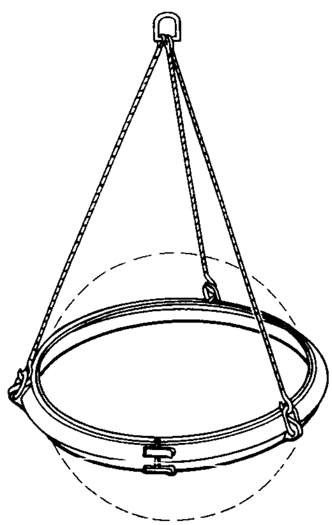


Figure 1. Sling-Hydrazine/Helium Tank

AUG 12 19[illegible]

SIGNIFICANT ERRATA. TRW Systems, Phase 1A Study Report, Voyager Spacecraft August 11, 1965

Volume 1. Summary

Substitute new p. 79 attached.

Volume 2. 1971 Voyager Spacecraft

p. 18. Item h) "necessary landed operations" should read "necessary lander operations."

p. 143. Section 3.4.1.a. second line should read "threshold of 0.25 gamma"

p. 282. Lines 3 and 4. Delete "or incorrect spacecraft address"

p. 284. Figure 5. Change "128 Word DRO Core Memory" to "256 Word DRO Core Memory"

p. 327. Denominator of second term on right hand side of equation should read

$$\left(\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1\right)\left(N - 1\right)$$

p. 351. Figure 1, Section F-F. "separation nut" should read "bolt catcher"

Volume 3. Voyager Program Plan

Substitute new p. 12 attached.

p. 13. Figure 2-3. PTM Assemblies in item 7 move 1.5 months to right

p. 16. Figure 2-6. First milestone date should be September 1, 1969, instead of mid-January 1970, and all subsequent dates should be correspondingly adjusted 4.5 months earlier.

p. 20. Table 2-2. Third item in 1969 column should read "coincident with completion of proof test model assemblies. Fifth item in this column change "2 weeks" to "3.5 months." Fourth item in 1971 column, change "4 months" to "5 months."

p. 67. Figure 5-2. Under Intersystem Interface Specification add a block entitled "Spacecraft to OSE Interface Specification"

p. 120. Last line of paragraph c should read "shown in Table 5-2."

p. 126. Figure 5-13. Year should be 1966 instead of 1965.

p. 153. Figure 5-18. Ignore all numbers associated with lines in figure.

p. 167. Figure 5-21. In line 20 change "design revisions" to "design reviews"

p. 254. Second paragraph, third line, "The capability of the transmitter to select" should read "The capability of the transmitter selector to select."

p. 258. Section heading n should read Experiment Data Handling

p. 604. Section 3.2.1 beginning of second paragraph should read "The hydrazine fuel . . . "

Volume 4. Alternate Designs: Systems Considerations

p. 103. Figure 3-19. Caption should read "Radial Center of Mass. . . "

p. 151. Last paragraph, second line, "For the baseline, the reliability. . ." should read "The reliability . . . "

p. 158. 8th line, replace "0.06 pound/watt" by "0.6 pound/watt"

p. 215. Figure 3-50. Dot in ellipse at right should be 0.

p. 230. Section 5.3.2, second paragraph, 7th line, should read "Figure 3-52."

p. 239. Second line, "with a variable V" should read "with a variable ΔV"

p. 247. First line, "3250 km/sec" should read "3.250 km/sec"

p. 261. Figure 3-64. Interchange coordinates, clock angle and cone angle

p. 293. Figure 3-81. An arrow should connect "Low-gain spacecraft antenna" and the dashed line at 73×10^6 km

Volume 4. Alternate Designs: Systems Considerations Appendix

p. 16. Figure A-2. The shaded portion under the lower curve should extend to the right only as far as 325 lb.

p. 9. Table A-1, part (1). In last column heading change "W_3" to "W_1". In part (4) last column heading change "W_3" to "W_4"

p. 22. Second line below tabulation, replace "575 × 35" by "570 × 35"

p. 29. Tabulation at bottom of page, change "18" to "30" and "400" to "240"

p. 207. Numerator of equation for λ best at bottom of page should read "0.0201," and numerator of equation for λ worst should read "9.21"

p. 209. Table 5B, fifth line. Delete " × 10¯." Also p. 213, Table 7A, seventh line, and p. 232, Table 3B, fifth line.

p. 217. Top portion of Table 9B should be labeled "primary mode" instead of "other modes"

p. 326. In equation following words "clearly" and "thus" insert " > " before second summation.

Volume 5. Alternate Designs: Subsystem Considerations

p. 3-15 Fifth line, "... is extended, spacecraft" should read "... is extended, two spacecraft"

p. 3-38 Last line, change " $= \frac{32}{4500} = M$" to "$\left(\frac{32}{4500}\right)\left(M\right)$"

p. 3-51 Two equations at bottom of page should read

$$D = 4\pi A/\lambda^2$$

$$A = \frac{D\lambda^2}{4\pi} = \frac{1000\,\lambda^2}{4\pi}$$

p. 3-67 Third line, last parenthesis "$\left(\frac{\pi}{2} + \phi\right) -$"

p. 3-82 6th line should read "50 degrees" instead of "50-140 degrees," and seventh line should read "140 degrees" instead of "50-140 degrees"

p. 3-111 Last line, change "50 Mc" to "1 Mc"

p. 3-137 Item g) for "... followed by 5 frames of real time" substitute "... followed by 11 frames of low rate science data and 5 frames of real time" pp. 3-150 and 3-151 are interchanged.

p. 3-156 Last line, should read "gates, a 7 bit"

p. 5-21 Second paragraph, third line, for "others since they are" substitute "others which are"

p. 5-33 Bjork equations should identify 0.18 as an exponent, and the exponent for (ρ_p/ρ_t) in the Hermann and Jones equation should be 2/3 in both cases.

p. 5-33 Figure 5-12 should be replaced with Figure C-7 of Appendix C.

p. 5-40 Three lines above Table 5-10 substitute "permanent set" for "experiment"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix I

p. B-11 Bottom of page, for "$r^{2/3}$" substitute "$(V/C)^{2/3}\,r$"

p. C-4 The title of Figure C-2 should read "Figure C-2. Meteoroid Influx Rate Circular Orbit Mars", and the title of Figure C-3 should read "Figure C-3. Meteoroid Influx Rate Cruise"

p. C-5 At bottom of page, add the following: "*Within 50,000 km of Mars"

p. C-6 Line 13 should read: "... of low density ($\rho_p < 2.4\ gm/cm^3$..."

p. C-6 Figure C-4. The ordinate "2" should read "100"

pp. C-17, C-21 The figures C-6 and C-7 on pages C-17 and C-21 should be reversed.

p. C-28 The title of Figure C-8 should read "Meteoroid Shield Test Specimen"

p. C-29 The title of Figure C-9 should read "Cutaway of Meteoroid Shield Test Specimen

p. C-34 In Section 1.8 the first sentence should be replaced by the following two sentences: "Preceding sections of this appendix contain derivations of the probability of penetrations of the spacecraft outer skin by meteoroids. It is clear that to design an outer skin of sufficient thickness to reduce the probability of no penetrations to a low level, such as 0.05 to 0.01, would be prohibitive in terms of the weight required."

p. C-35 In the first equation, the expression "(t in m^2)" in two places should read "(t in cm)" and "A" in two places should read "(A in m^2)"

p. C-38 In Table C-2, all values in inches should be in centimeters. A zero should be inserted immediately following the decimal point, for example: (0.020-inch) = 0.05080, (0.020-inch) = 0.06096, (0.020-inch) = 0.04064, etc.

p. C-40 In Section 1.8.7 Computation of R's, the sixth line should read "... than 10^6 are neglected"

p. C-45 In listing under "Values of t Used for Extreme Environment Analysis," under Inch, the first number should read 0.020 instead of 0.202

p. C-52 In 1.10 NOMENCLATURE, "K_2" should be defined as "$K^{-2/3}(4 \pm 2)$" and "B" should be

$$\frac{1000\, \rho_t V^2}{9.806\, H_t}$$

pp. C-150 and C-151 should be reversed.

p. C-208 Along the ordinate in the graph, "Stress $\times 10^{-3}$" should read "Stress $\times 10^{-2}$"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix II

p. F-23 Lines 7 and 10 change all subscript τ to T

p. F-24 Line 14, change "ME_i" to "mE_i"

p. F-29 Figure F-9 title should be "Reflection Phase Angle ϕ (deg)" and Figure F-10 title should be "Reflection Magnitude R"

p. F-30 Last line, change "0.27" to "0.175"

p. F-31 Lines 14 and 15, change "14,700 ft/sec to 460 ft/sec" to 14,700 ft/sec minus 460 ft/sec" and "14,700 ft/sec to 10,000 ft/sec" to "14,700 ft/sec minus 10,000 ft/sec"

p. F-32 Last line in item 4), change "27 per cent" to "17.5 per cent"

p. F-35 Table F-4, under Assumed Parameter for item 2 insert "$\pm 2 \times 10^{-5}$", for item 3 insert "$\pm 3 \times 10^{-5}$", and for item 4 insert "$\pm 2 \times 10^{-5}$"

p. F-53	Item d. Noise Figure, change "4 db" to "3.5 db"; Gain, change "20 db" to "10 db", last line change "10 db" to "4 db"
p. F-58	Figure F-21. Change 102 kc to 112 kc.
p. F-59	Line 22, change to "M_1 = 21.5 deg or 0.375 radians (rms, peak)"
p. F-60	Line 2, change to

$$"M_2 = \sqrt{(1.1)^2 - (0.375)^2}"$$

p. F-60	Line 3, change to "M_2 = 1.03 radians (rms) or 1.46 radians (peak)"
p. G-6	Paragraph 1.4, second line, change "from $E_M = 10^1 E_o$ to $10^4 E_o$ . . ." to read "from $E_M = 10^{-1} E_o$ to $10^4 E_o$ . . ."

Volume 6. Operational Support Equipment

p. 25	Figure 6. Caption should be "Typical Grounding Scheme"
p. 39	Section 1.3.3, change opening of first sentence to read "Launch pad equipment consists of the ground power and RF consoles and the test flight program power and control equipment . . ."
p. G-31	Figure 1. Lines enclosing Data Format Generator should be solid.
p. G-102	Last line substitute "4500" for "45"
p. G-113	In Section 4.4.2, change "25 per cent" to "250 per cent"
p. G-184	Section 4.5, substitute "6.5 feet" for "six feet"
p. G-311	Fifth line, change "30 per cent" to "20 per cent"
p. G-398	Section 4.2 should begin with "The hoist beam is . . ."
p. G-419	Second line "4 optical alignment targets" instead of 8. Same correction top of p. G-421.
p. G-423	Section 4.9.2, substitute "20 per cent" for "50 per cent"

Volume 7. <u>1969 Flight Test Spacecraft and OSE</u>

p. 90 First line should read "Launch pad equipment consists of the ground power and RF consoles and ..."

p. 107 Last line, change Volume 5 to Volume 6.