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VOYAGER SPACECRAFT SYSTEM PRELIMINARY DESIGN

1971 OPERATIONAL SUPPORT EQUIPMENT DESIGN

VOLUME C

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CALIFORNIA INSTITUTE OF TECHNOLOGY
JET PROPULSION LABORATORY
4800 OAK GROVE DRIVE
PASADENA, CALIFORNIA

GENERAL ELECTRIC
SPACECRAFT DEPARTMENT
A Department of the Missile and Space Division
Valley Forge Space Technology Center
P. O. Box 8555 • Philadelphia, Penna. 19101

TABLE OF CONTENTS

VOLUME C

1971 VOYAGER SPACECRAFT OPERATIONAL SUPPORT EQUIPMENT

TABLE OF CONTENTS (Continued)

CII - VB260SR101

SYSTEM TEST COMPLEX

AND

SUBSYSTEM TEST EQUIPMENT

TEST OBJECTIVES AND DESIGN CRITERIA

Index

1.0 SCOPE

It is the intent of this document to define those test objectives and design criteria which should be applied to the Operational Support Equipment (OSE) which forms the System Test Complex and the Subsystem Test Equipment necessary to support the Voyager 1971 test and evaluation program. The System Test Complex will be the basic test equipment used in the tests and operations at the Spacecraft Checkout Facility, Eastern Test Range, and at the Spacecraft Checkout Facility, Jet Propulsion Laboratories, for verification of the adequacy of the spacecraft's design, fabrication, assembly and flight readiness. The System Test Complex (STC) will also be used, prior to its delivery, at the Spacecraft contractor's plant, for spacecraft performance testing. Subsystem Test Equipment will also be employed at each of these locations.

2.0 APPLICABLE DOCUMENTS

Jet Propulsion Laboratory

V-MA-004-002-14-03	Voyager 1971 Mission Guideline
V-MA-004-001-14-03	Voyager 1971 Mission Specification (Preliminary)

3.0 OBJECTIVES

3.1 SPACECRAFT SYSTEM TESTS

The principal objective of the STC is to allow overall tests of the spacecraft operating as an entity, to be conducted. This requires the STC to have the capability of controlling and conducting simulated mission testing of the spacecraft. The simulated mission testing is the control and evaluation of the spacecraft, during the accelerated sequence of the events planned for the mission. During system tests, with the spacecraft being operated in a simulated flight mode, the System Test Complex must permit the performance of each flight subsystem to be evaluated. This evaluation is for the dual purpose of spacecraft performance evaluation, subsystem-by-subsystem as well as overall, and the isolation of faults to a subsystem.

The support of Spacecraft system tests is an objective applicable to the Spacecraft Checkout Facility at ETR, the Spacecraft Checkout Facility at JPL and the manufacturer's plant.

3.2 SPACECRAFT FLIGHT SUBSYSTEM TESTS

The Subsystem Test Equipment (SSTE) will have the capability of supporting tests of the flight spacecraft's on-board subsystems. These subsystem tests are intended to verify the performance capability of each flight subsystem, considered separate from the other flight subsystems on the spacecraft. The objectives of the tests are subsystem performance capability verification, and fault isolation to replaceable assemblies within the flight subsystem.

The SSTE will provide the required power, loading and stimuli to the flight subsystem, while the subsystem is functionally isolated from the rest of the spacecraft's subsystem. Additionally, the SSTE will provide the required measurement, control and recording/displaying capability required by the subsystem's test sequence. The SSTE consists of that portion of the STC equipment which tests a particular flight subsystem supplemented, as required, by ancillary equipment to allow operation independent of the STC.

3.3 LAUNCH CONTROL SUPPORT

A secondary objective of the STC is to provide back-up measurement and evaluation capability and transmit S/C commands to support and augment the Launch Complex Equipment (LCE), which is the Operational Support Equipment to be utilized at the Explosive Safe Facility and at the Launch Complex. The STC objective is the evaluation of performance of the spacecraft subsystems by monitoring and analizing the data obtained via the RF TLM signal and to transmit RF commands to the S/C, following the fueling/pyrotechnic loading and during the launch operation.

3.4 PROOF TEST MODEL SPACECRAFT SUPPORT

An objective of the STC and SSTE is the support of the JPL operations on the Proof Test Model spacecraft. This support is anticipated to be required for the detailed evaluation of design, construction and long term operating performance of the spacecraft Proof Test Model, at the SCF, JPL. This support would encompass the system test and subsystem test objectives, as well as constituting the additional objectives of supporting the inter-subsystem tests and environmental tests which may be conducted by JPL.

3.5 FACTORY AND VENDOR SUPPORT

SSTE must be able to support subsystem tests at the manufacturer's or vendor's plant. In addition, the STC and SSTE will support system, subsystem, inter-subsystem and environmental tests of the Spacecraft, at the Spacecraft manufacturer's facility.

3.6 COMPATIBILITY

An objective of the STC is to achieve compatibility between procedures and equipment used by the DSN for operations, and the procedures and equipment used for test.

4.0 DESIGN CRITERIA

4.1 STC DESIGN CRITERIA FOR SUBSYSTEM TESTING

Each flight subsystem will have its STC equipment allocated to consoles and relay racks intended for the testing of that particular subsystem. The intent of this criterion is to enable the cognizant subsystem engineer to conduct tests of his spacecraft subsystem, or of its assemblies, independent of tests being conducted on other subsystems.

Each subsystem test position in the System Test Complex will have test point access to the spacecraft, for purposes of response measurement. These test points shall be brought out of the spacecraft by a ring harness. Umbilical connections and the normal interface connections to other flight subsystems are to be used for control, stimulation and loading. Special access connectors are to be kept to a minimum.

Each subsystem test position in the STC should ensure the capability of providing electrical isolation between the subsystem under test and other subsystems. Isolation should be capable of being provided between the STC and the subsystem to be tested, during STC self test, to preclude jeopardizing spacecraft subsystem functional capability. Electrical isolation, between the spacecraft and the STC shall be sufficient to preclude spacecraft damage in the event of STC short circuits or overvoltages.

Each set of OSE in the STC which tests a particular flight subsystem, should be capable of removal from the STC and independent operation. The use of ancillary equipment, not normally required when in the STC configuration, is permissible to meet this criterion. The OSE, when removed from the STC and augmented, as required, by ancillary equipment, becomes SSTE.

4.2 STC DESIGN CRITERIA FOR SYSTEM TESTING

System tests of the spacecraft are tests which are conducted upon spacecraft subsystems, operating, in successively more extensive combinations, leading to interdependent operation of all spacecraft subsystems in a simulated mission.

Recording of test data, acquired from the subsystem test positions via hardline, and required for post-event analyses of systems test, is required. This recording capability is described in Functional Description VB260FD107.

Test measurements, made at the subsystem STC stations (the SSTE), are to be processed by the Computer Data System described by Functional Description VB260FD103. This test data is to be displayed at a master printer, and appropriate parts of the data printed for subsystem test stations displays. This Computer Data System constitutes the principle mode of integrating the subsystem positions of the STC to permit systems tests and simulated mission tests to be conducted. It is also to be used to implement such computer control as is required to conduct a system test involving a simulated flight mission.

Power for all parts of the STC, is to be provided on an integrated basis, as described in Functional Description VB260FD102.

Central Timing and synchronization, for all of the STC, including subsystem stations is required, as described in Functional Description VB260FD106.

A position for the Test Director, described in Functional Description VB260FD104, should provide the capability of centralized control of spacecraft system testing and monitoring the conductance of subsystem tests, where appropriate, and monitoring the status of subsystems.

Spacecraft simulation, to permit the validation of the STC to be accomplished without jeopardizing the spacecraft, is required. The simulation is described in Functional Description VB260FD108.

To achieve complete compatibility between operational control and test control, during system test the command word stream which stimulates the spacecraft should be generated by a Command Verification Equipment analagous to the DSIF CVE and the identical software. This is described in Functional Description VB263FD102.

4.3 GENERAL DESIGN CRITERIA

The STC and its component SSTE shall use the design approach implemented in the Mariner "C" program, where practicable. Where test requirements of the Voyager substantially differ from Mariner "C", such as number of commands and responses in a simulated mission, the Mariner "C" design approach should be extended to serve the increased requirements. An example of such extension may be the utilization of the Computer Data System in controlling sequences for the Command portion of the Telecommunication Subsystem STC position.

The primary STC equipment design criterion, to the extent determined above, is compatibility of design of Mariner "C" STC components with the design of the Voyager STC. This compatibility should exist in order to utilize Mariner "C" STC components in the Voyager STC, should they become available as GFE.

The design and layout of the STC should accomodate the utilization of Voyager Assembly Handling and Shipping Equipment (AHSE) at the Spacecraft Checkout Facility, at ETR, and at the Spaceflight Checkout Facility at JPL. This compatibility criterion is of primary importance in the case of such AHSE as the spacecraft handling fixture, and the alignment equipment.

The STC should be capable of incorporating a subsystem test position for the support of the Science Payload of the Bus, a GFE item for the STC. The STC design should accomodate this equipment with regard to its interfaces with the parts of the STC for which criteria are listed below, and its interface with the umbilical and spacecraft direct access test harness.

The system testing capability of the STC should also be capable of accomodating a set of subsystem test equipment for the capsule.

The SSTE and STC shall be capable of conducting subsystem tests and system tests involving the following methods of spacecraft test point access: umbilical "J" box, direct access harness, reduced power radiation to the on-board radio portion of the telecommunications subsystem, telemetry data reduction, and flight harness connectors. The following criteria indicate the order of preference:

System Test Access Criteria

Control and Stimulation	RF, umbilical, special access
Monitoring	Direct access, Telemetry, Umbilical

Subsystem Test Access Criteria

Control and Stimulation	Umbilical, Flight Access, Special Access
Monitoring	Direct Access, Umbilical, Flight Access, Special Access

SYSTEM TEST COMPLEX

DESIGN CHARACTERISTICS AND RESTRAINTS

Index

This document presents a general description of the System Test Complex for the Voyager Spacecraft. The STC is to be used at the Eastern Test Range to conduct system tests on the spacecraft and subsystem tests on the spacecraft's subsystem. It will also be utilized to support testing at the contractor's plant. Since the scope of utilization of the STC at the contractor's plant includes, but exceeds, its ETR use, the utilization of the STC at the contractor's plant is used in this document as the context to describe the general design characteristics of the STC.

The method by which the STC test objectives will be attained is described. The components of the STC, and their general functional requirements are also described. This description of the design characteristics of the STC, and the included description of design restraints on the STC, constitute controlling documents for the functional descriptions of STC equipment.

1.0 GENERAL DESCRIPTION

1.1 SYSTEM TEST COMPLEX

The Voyager System Test Complex will be based upon an extension of the design of the Mariner C System Test Complex, to the additional measurement and control requirements of Voyager Spacecraft tests. The configuration of operational support equipment constituting the STC shall be an extension of the Mariner C STC configuration insofar as STC functional capability and equipment design characteristics are concerned. The Voyager STC should be capable of incorporating those components of the Mariner C STC which may be available for use on Voyager and which are capable of satisfactory achievement of the test requirements of analogous Voyager spacecraft subsystems and components.

The Voyager STC should have the capability of testing the Science Payload, as well as the Spacecraft bus. The STC is required to assimilate OSE furnished for the Science Package on the same basis as the OSE to support the other spacecraft flight subsystems. The STC may also be required to assimilate OSE furnished for the Capsule.

The requirements of the following documents are applicable to this document:

a.	V-MA-004-002-14-03	Voyager 1971 Mission Guideliner
b.	V-MA-004-001-14-03	Voyager 1971 Mission Specification (Preliminary)
c.	VB260SR101	1971 Voyager STC Test Objective and Design Characteristics

1.2 PRE-LAUNCH FACTORY TEST PROFILE

1.2.1 STC VALIDATION TESTS

The STC, prior to its use in spacecraft acceptance tests, is required to perform a series of validation tests in order to establish confidence that the STC will inflict neither catastrophic nor subtle deteriorating influences upon the flight spacecraft. These tests include STC visual inspection to verify the status of the STC for test, closed loop tests of STC circuits working into dummy loads to establish correct electrical voltage application, and ground loop tests to establish the degree of suppression of spurious measurements.

1.2.2 STC-SC COMPATIBILITY TESTS

The STC, once validated for use in testing a spacecraft, is used to establish STC-SC compatibility. This is accomplished by a series of tests involving both the STC and SC. This series includes combined ground loop measurement and elimination, experimental verification of the capabilities of the STC to deliver the proper power to the spacecraft subsystems and of the spacecraft subsystems to employ the power. It also includes live power tests of the spacecraft electronics.

1.2.3 SUBSYSTEMS TESTS

The System Test Complex is required to be able to test each of the flight subsystems. This testing involves performance measurement of the flight subsystem utilizing the Subsystem Test Equipment (SSTE) which is that part of the OSE peculiar to that subsystem, for the control, stimulation, measurement and display required by the subsystem test. The subsystem test requires that subsystem-peculiar parts of the STC be augmented by the system test portions of the STC. The subsystems are required to be tested by the STC, with fault isolation capability to the level of a functional group of replaceable modules within the flight subsystem or bay. During these subsystem tests, the subsystem is functionally isolated from the rest of the spacecraft and tied to the STC by direct access test points, umbilical J-boxes and by connection to flight harness connectors. The subsystem tests are required to be conducted in series, in order to maximize test integrity.

1.2.4 INTER-SUBSYSTEM TESTS

The STC is required to have the capability of testing spacecraft subsystems operating in an integrated fashion, to determine whether the subsystems are performing properly, across the spacecraft's subsystem-to-subsystem interfaces. These tests are required to be conducted using the additional subsystem test peculiar STC equipment and also utilizing appropriate general purpose STC equipment for synchronization, control and test data processing. The inter-subsystem tests are required to be performed on a cumulative basis, starting with the on-board power system and building up to the test of all subsystems and all interfaces.

1.2.5 SYSTEM TESTS

The STC is required to have the capability of conducting system tests upon the spacecraft. These system tests involve all of the spacecraft flight subsystems and consists of simulating, in accelerated time, all the actions and commands involved in a complete mission, with measurement of the degree to which commanded or programmed actions have been implemented. To perform a simulated mission system test, the STC is required to be able to provide certain external stimuli, transmit simulated commands to the spacecraft, and measure and evaluate the spacecraft responses.

1.2.6 CERTIFICATION TESTS

The STC is required to have the capability of conducting special tests intended to certify the spacecraft. These tests duplicate the inter-subsystem and system tests described above, conducted in a stressed environment. The telemetry calibration is a prerequisite to these certification tests, as is a dummy run of prelaunch and countdown run in conjunction with the Launch Complex Equipment (LCE). The certification tests will include inter-subsystem and system tests under conditions of power and frequency variation established by the STC, system tests using telemetry and external stimulators only, with simulated solar irradiation, and systems tests under thermal, and vibration extremes.

2.0 BASIC DESIGN CHARACTERISTICS

The basic design characteristic of the System Test Complex is its division into two types of equipment; one type peculiar to a spacecraft flight subsystem on a one-for-one basis, and the other type being general purpose in that it is designed to be used in the system testing of all spacccraft subsystems.

a. Subsystem Test Equipment - The STC equipment peculiar to the test of a flight subsystem is designated "Subsystem Test Equipment - (SSTE)". It shall be designed so that the cognizant subsystem engineer can determine the adequacy of design, performance and construction of the subsystem, by using their equipment. The equipment shall be designed to power, control, stimulate, and load and evaluate the subsystem through direct test point and umbilical access (and RF in the case of radio flight equipment). The subsystem test equipment design characteristics shall be such as to enable these functions to be accomplished by the cognizant engineer, using these test consoles and appropriate capabilities of the STC general purpose equipment. The subsystem test equipment design should permit these functions to be implemented in a console or rack group allocated specifically to that flight subsystem. The subsystem test equipment configuration includes that equipment normally used during system tests to monitor and evaluate subsystem performance, plus the additional equipment needed to stimulate and load the subsystem when it is electrically isolated from the rest of the spacecraft.

b. General Purpose System Test Equipment - The STC equipment utilized for inter-subsystem tests and simulated mission system tests will include general purpose, system test equipment. This equipment is required to be designed so that analog data from the spacecraft subsystems can be recorded, digital and analog test data can be processed and printed, time and synchronizing information can be distributed to the STC, and test control and commands for system tests can be generated. The general purpose system test equipment should have the design characteristics required to enable support to be given to the subsystem test equipment which may require it.

Both the Subsystem Test Equipment and the System Test Equipment parts of the System Test Complex are required to have the following design characteristics:

a. Interface with Test Conductor's Console - The Test Conductor's console will serve as the control area for the STC communications network, which will consist of 3 1/2-inch communications panels located in the OSE S/S consoles. There will be a minimum of one communication panel for each test set or one for each three racks of a console.

   The status of many subsystem functions, both hardwire and TM will be available for display to the Test Conductor. This will be accomplished on alphanumeric displays and lamp indicators. A page printer which will be the primary computer output will be located sufficiently close to the Test Conductor so that a computer analysis of the hardwire and TLM data will be available for his use.

   The TC will have limited control over the testing. This will include emergency control over the STC power and a CDS test stop. For obtaining detailed test data or making modifications to the test sequence, a teletype interface will be provided to the CDS.

b. Depth or Fault Isolation and Parts Replacement - Each subsystem OSE console must be capable of isolating a fault to a subsystem level when the STC is in a system test mode. The suspected subsystem will then be electrically isolated by disconnecting it from the system harness, and use of flight connector cables for application of stimuli and loads. The fault will then be traced to the replaceable group of functional modules which can be replaced from spares.

c. Power Control - Each vehicle subsystem will receive power through the S/C ring harness from the vehicle power subsystem or from the OSE subsystem console. The OSE must be capable of supplying the power so that subsystem testing can be accomplished without dependence upon other vehicle subsystems. It is very important that both power sources are not connected to the subsystem at the same time. To accomplish this, there will only be one connector provided for each subsystem which will be exclusively for power. An OSE power cable will be provided with each test group; so the power source can be applied only by the appropriate cabling.

d. Direct Access Connector - Direct Access Connectors will be located on the electronic bays for use during system and subsystem testing. They will be connected into the subsystem via a removable test harness. All signals should be buffered within the S/C so that the vehicle will not be damaged because of an OSE equipment or personnel failure.

e. OSE Interchangeability - All OSE consoles, or test sets will be directly interchangeable with like consoles in other STC's. This will apply to all proof-test models and prime units.

f. Installation Compatibility - The STC equipment will require the following estimated facilities:

1. An air conditioned room 30' x 40'

2. Minimum ceiling height 11'

3. An 80 KVA 208/120, 3Ø, 4 W 60 ~ power and a 15 KVA, 208/120, 3Ø 4 W, 400 ~ power source.

4. A false floor for routing interconnecting cables.

5. Center of the room should be within 100 feet of the vehicle during testing for operating without line drivers for the cabling. Use of the STC or facilities with greater cable runs may require line drivers.

6. Building ground rods within 10 feet of the power and distribution consoles.

7. EMR will require provisions for roof-mounted antenna for LC support.

8. The consoles will be mounted on casters, and will disassemble into units which will not exceed two bays in width ($\approx$ 54") for shipping and installation purposes.

g. DSIF Interface and Compatibility - The Voyager program will require mission dependent software at the DSIF stations. To validate the interface compatibility the DSIF equipment must be used in conjunction with the MD software in the STC. The DSIF Station Telemetry Command and Data Processors (TCD) which use the MD program has many features that make its presense in the STC and its use desirable during system testing. For example, it can decommutate the TM data for computer processing and the CVE portion of the TCD can be slaved to the computer for use as a command generator. Therefore, the hardware and software used in the STC shall be functionally identical to the MD hardware and software, and shall be physically identical where possible.

h. LC Support - While the spacecraft is at the pad and the Explosive Safe Area, the STC will have RF capability to command and reduce TLM data. Also, a hardwire command verification link will exist for comparing the commands which the spacecraft receives with the transmitter commands. With this capability, the STC with its automatic data processing becomes a functional part of the LC.

3.0 GENERAL PURPOSE STC EQUIPMENT DESIGN CHARACTERISTICS

The following paragraphs describe the general design characteristics of the general purpose equipment in the STC. Figure 3-1, an overall functional block diagram of the STC, illustrates the functional relationship of the general purpose type equipment in the STC to the subsystem test equipment in the STC.

3.1 RECORDING AND DISPLAY

3.1.1 CENTRAL RECORDER

The central recorder will provide a continuous monitor of the vehicle status during system and subsystem testing. It will consist of a direct-write 50-channel oscillograph, with variable gain input amplifiers and a patch panel for changing its operating mode from system test to any one of several S/S tests. The frequency response of the galvanometers will be 0 to 150 cps; however, additional galvanometers will be provided which will have an undamped natural frequency of 1,500 cps.

3.1.2 TELETYPE WRITERS

Remote computer peripheral equipment will be located throughout the STC; this will provide the operating S/S personnel with the capability of control of automatic computer sequence (not in the system test mode) and obtain computer processed data for their subsystem. To accomplish this, teletype writers will be placed as defined in Figure 3-2.

3.1.3 LINE PRINTER

The computer will interface with a remote line printer which will be located near the Test Conductor's console. This will be the primary computer output for visual display, and will provide the test conductor with detailed information regarding the vehicle status.

3.2 CENTRAL TIMING GENERATOR

A central timing generator will be provided for each test facility. It has the capability to supply six system test complexes with timing signals for processing by a Timing Isolation Unit (TIU). The TIU will buffer the Timing Generator from the individual recorders that use the signals. A large central display of days, hours, minutes and seconds will be located in each STC where easily visible to all personnel. Also, a similar panel display will be located on the Test Conductor's console.

3.3 UMBILICAL JUNCTION BOX

An umbilical distribution box will be located in each STC. It will interface with the OSE subsystem consoles and the vehicle as defined in Figure 3-2.

3.4 COMPUTER DATA SYSTEM

The Computer Data System will have the capability of automatically sequencing the vehicle through executive control of command generation while monitoring and analyzing many of the output hardwires and TM data points. Additional control will parallel a few of the OSE console panel switches in order to permit a time coordination of certain external stimuli. While performing a test, the subsystem engineers will communicate with the computer via teletype writers and the vehicle status will be displayed on the five remotely located units in the STC. Also, the complete vehicle status will be periodically printed out by a line printer that will be located on the Test Conductor's console, and a magnetic tape of the test results will be produced by the computer.

3.5 POWER DISTRIBUTION

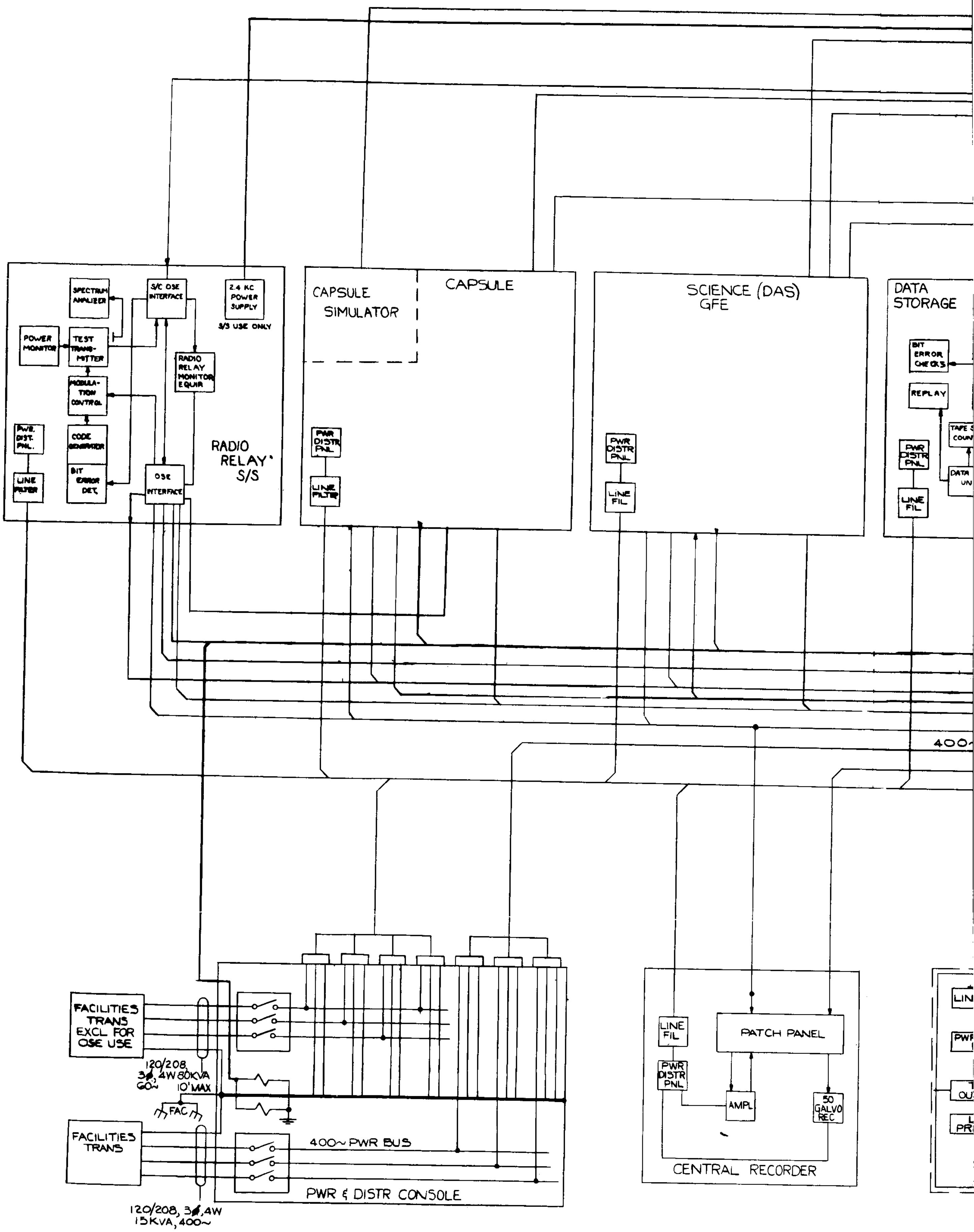
The power distribution console will interface between a facilities transformer which will be exclusively for OSE, and all the OSE consoles. It will distribute 120 VAC, 1Ø, 60-cps power to the test sets through 30-20A and 6-30A circuit breakers. The 400-cps power will be distributed through circuit breakers as required. The Test Conductor will have the capability to turn off all A-C power should an emergency arise.

3.6 SPACECRAFT SIMULATOR

The portable S/C simulator will be used to simulate all the umbilical functions, both in the LC and the STC. Its primary purpose will be to validate the cabling between the umbilical connector and the S/S OSE consoles. To do this, there will be an exact simulator of the vehicle sources, or loads, in the simulator.

3.7 TEST CONDUCTOR'S CONSOLE

The Test Conductor's console will serve as the central control and data area. It will have a limited amount of control over the computer data system and a few emergency controls. A complete computer analysis of TLM and hardwire data will be available to the Test Conductor on an alphanumeric display and the computer line printer. Other subsystem status indicators and communication controls will also be located on the console.



SPECTRUM ANALIZER
S/C OSE INTERFACE
2.4 KC POWER SUPPLY
S/S USE ONLY
POWER MONITOR
TEST TRANS-MITTER
RADIO RELAY MONITOR EQUIP
MODULA-TION CONTROL
PWR. DIST. PNL.
CODE GENERATOR
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OSE INTERFACE
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CENTRAL RECORDER

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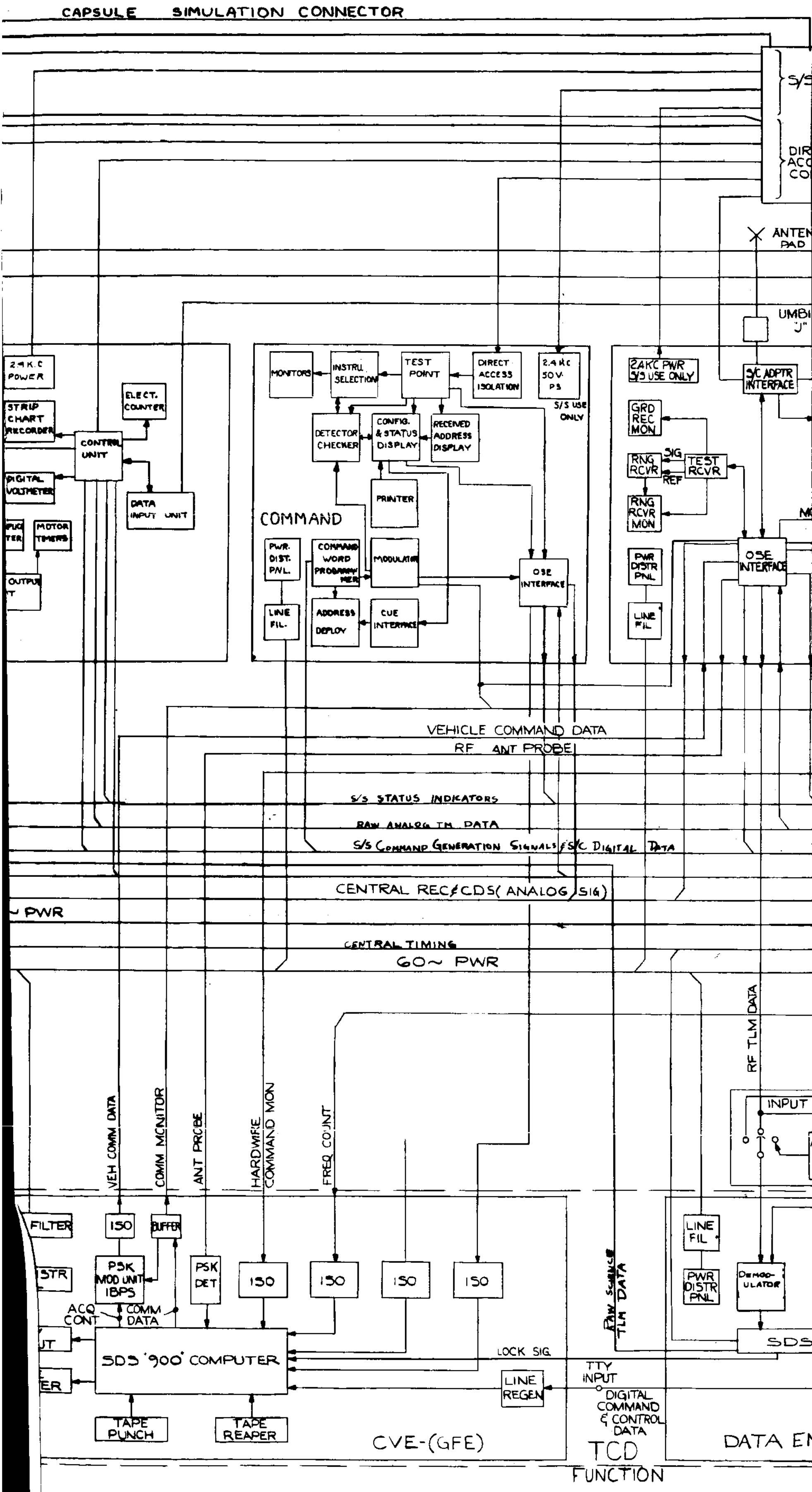
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MONITORS
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TEST POINT
DIRECT ACCESS ISOLATION
2.4 KC 50 V. PS
S/S USE ONLY
DETECTOR CHECKER
CONFIG. & STATUS DISPLAY
RECEIVED ADDRESS DISPLAY
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PWR. DIST. PNL.
COMMAND WORD PROGRAMMER
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ADDRESS DEPLOY
CUE INTERFACE
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S/S STATUS INDICATORS
RAW ANALOG TM DATA
S/S COMMAND GENERATION SIGNALS/S/C DIGITAL DATA
CENTRAL REC/CDS(ANALOG SIG)
~ PWR
CENTRAL TIMING
60~ PWR
RF TLM DATA
INPUT
VEH COMM DATA
COMM MONITOR
ANT PROBE
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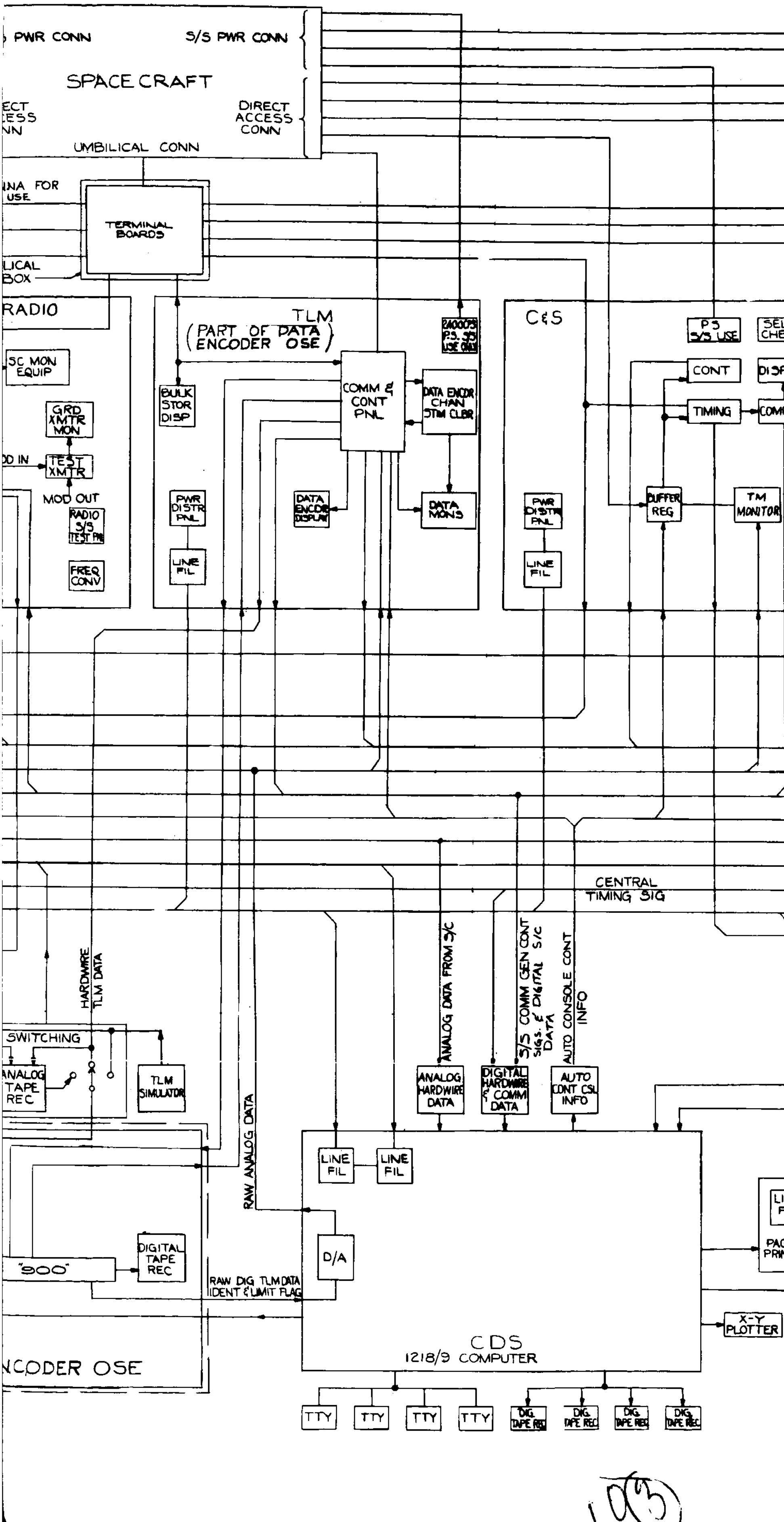

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ANALOG DATA FROM S/C
S/S COMM GEN CONT SIGS. & DIGITAL S/C DATA
AUTO CONSOLE CONT INFO
SWITCHING
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TLM SIMULATOR
ANALOG HARDWIRE DATA
DIGITAL HARDWIRE & COMM DATA
AUTO CONT CSL INFO
RAW ANALOG DATA
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"900"
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RAW DIG TLM DATA IDENT & LIMIT FLAG
CDS
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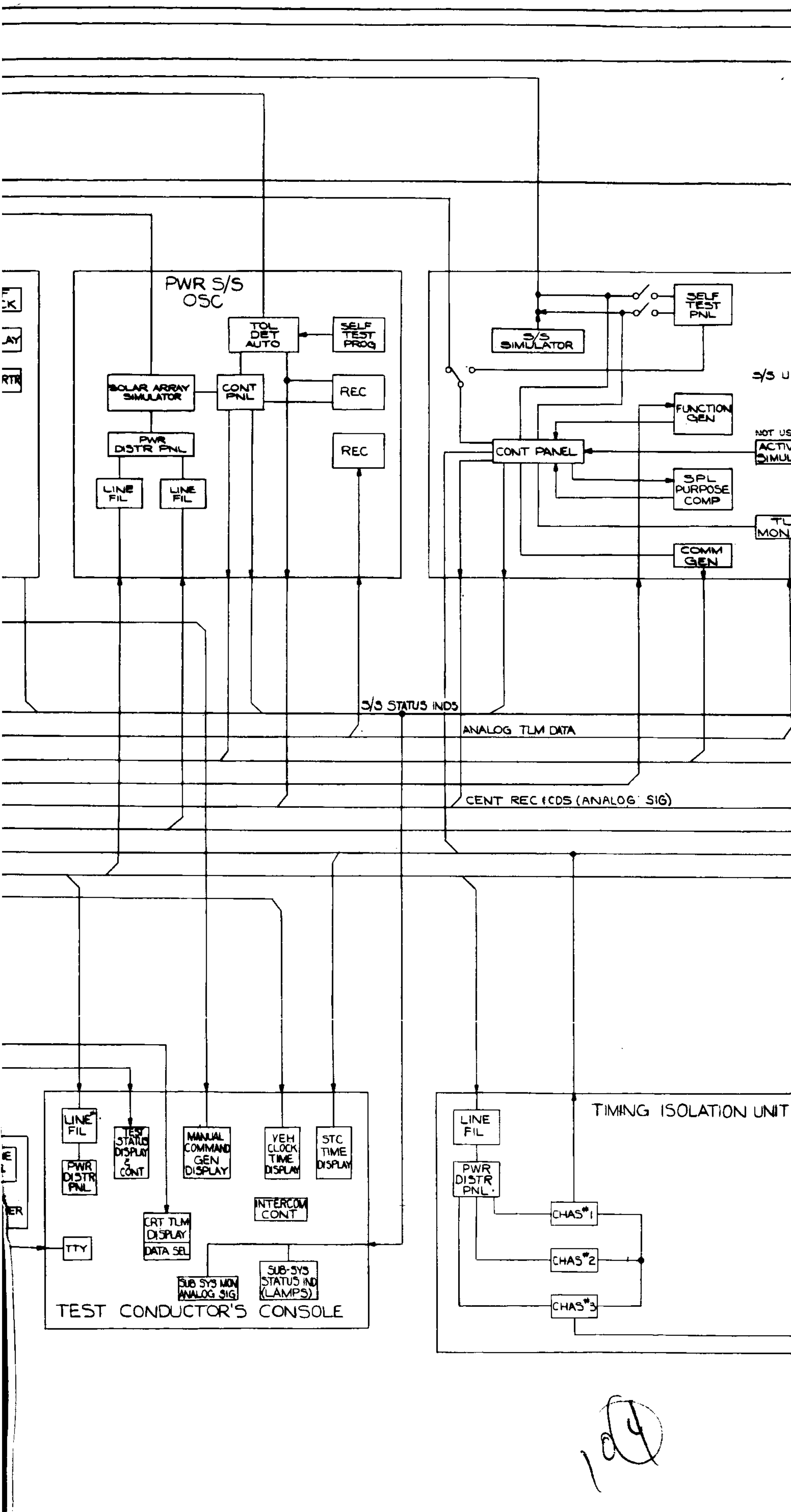
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S/S SIMULATOR
SELF TEST PNL
FUNCTION GEN
CONT PANEL
SPL PURPOSE COMP
COMM GEN
S/S STATUS INDS
ANALOG TLM DATA
CENT REC & CDS (ANALOG SIG)
LINE FIL
PWR DISTR PNL
TEST STATUS DISPLAY & CONT
MANUAL COMMAND GEN DISPLAY
VEH CLOCK TIME DISPLAY
STC TIME DISPLAY
INTERCOM CONT
CRT TLM DISPLAY
DATA SEL
TTY
SUB SYS MON ANALOG SIG
SUB-SYS STATUS IND (LAMPS)
TEST CONDUCTOR'S CONSOLE
TIMING ISOLATION UNIT
LINE FIL
PWR DISTR PNL
CHAS #1
CHAS #2
CHAS #3
104

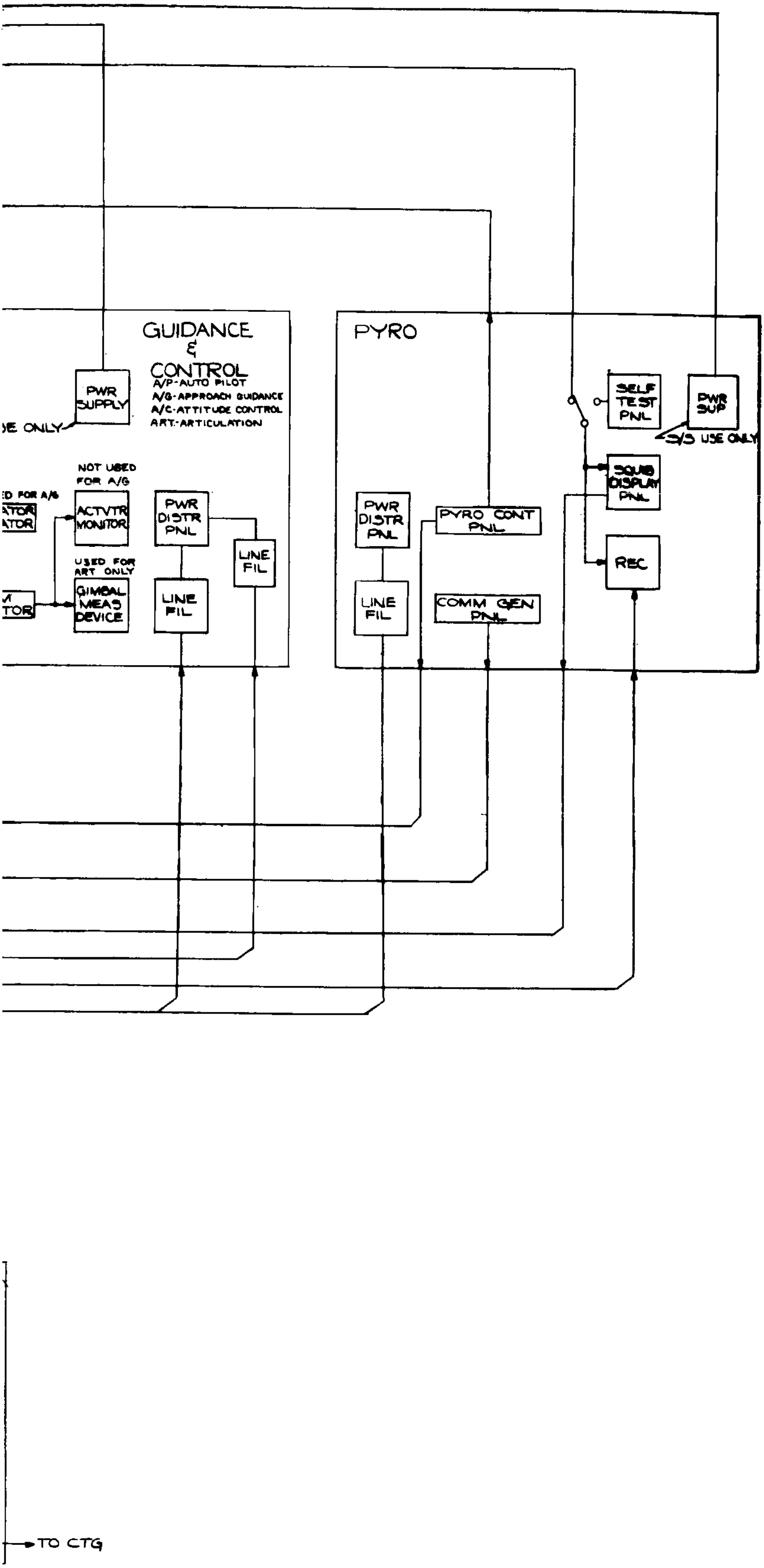


Figure 3-1. System Test Complex, Block Diagram

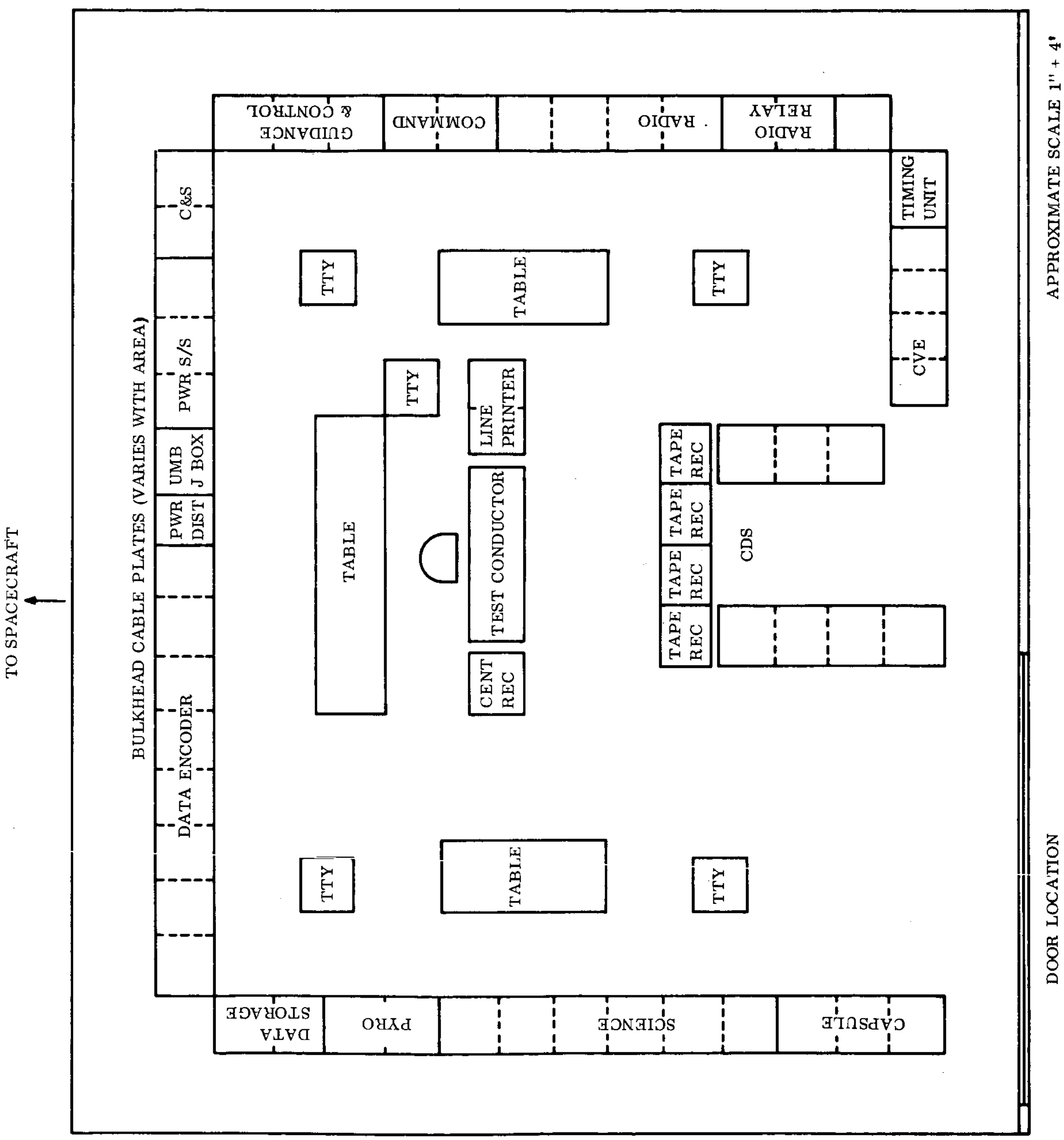


Figure 3-2. System Test Complex, Floor Plan

4.0 SPACECRAFT SUBSYSTEM-PECULIAR STC DESIGN CHARACTERISTICS

All subsystem-peculiar OSE consoles should conform to the design constraints outlined in this document. These consoles, such as the Science, Data Encoder, Data Storage Capsule, Attitude Control, Pyrotechnic, Controller and Sequencer, Radio, Radio Relay, Power and Command OSE must also conform to the following design characteristics:

a. Monitor all hardline S/C outputs.

b. Stimulate all S/S sensors (if possible), when conducting subsystem tests.

c. Stimulate all hardline input to the S/C, when conducting subsystem tests.

d. Provide for S/S failure detection.

e. Provide a record of S/CS/S operating time.

f. Contain an internal self-test mode which uses all interfaces cables.

g. Be capable of troubleshooting the isolated S/CS/S to the replaceable group of functional modules level.

5.0 DESIGN CONSTRAINTS

5.1 ELECTRICAL

5.1.1 OSE

All OSE circuits shall have the following characteristics:

a. Digital logic shall be standardized, and be compatible with the spacecraft wherever practicable.

b. Circuit standardization shall be optimized, thus making use of a maximum number of common circuits.

c. Transient and worst-case analysis will be required on all circuits.

d. Proven high-reliability circuits will be used if possible.

5.1.2 ISOLATION

The vehicle must be protected at all OSE interfaces against possible damage by equipment or personnel failures. All such functions shall withstand the following tests without detrimental effects.

a. Shorting of any interface functions to the common spacecraft electrical ground.

b. All OSE monitoring circuits (and cables) shall not affect the vehicle performance.

The following constraints apply only to umbilical functions.

c. They shall be capable of being shorted together in any possible combination without detrimental affects on the S/C operation.

d. All functions shall be capable of withstanding a 75V capacitively coupled pulse, five microseconds in duration, between each function and ground.

5.1.3 INSTRUMENT CALIBRATION

Instruments such as meters, scopes, counters, etc. should hold calibration for six-month intervals. The test consoles will contain a self-calibrating mode for special purpose circuits which require frequent calibration.

5.1.4 RFI

RFI effects should be minimized in all OSE test sets. This can be accomplished by using diodes across relay coils, line filters on all A-C power lines and the use of other standard RFI suppression methods.

5.1.5 SELF TEST

All OSE consoles will contain a means of self-checking the operating circuits. This may be either manual or automatic, depending primarily on the complexity of the test set; however, it should not require special laboratory test equipment. Scopes, multimeters, etc. are considered normal test equipment. The test should verify vehicle interface cables such as direct access, umbilical and S/S power.

5.1.6 CONVENIENCE OUTLETS

There will be a minimum of one 120V, 60-cps duplex convenience outlets located in each two racks of equipment.

5.1.7 SHIELDING

a. Shielded leads (in cables) between the spacecraft and the OSE consoles will have their shields common at the S/C end only, and they will be grounded only to the S/C structure.

b. Shielded leads will have insulation covering the shield.

c. Isolate all coaxial connectors from the chassis and use the appropriate ground for the shield.

5.1.8 STC POWER SOURCES

a. 120VAC, 1∅, 60 cps

b. 208V, 3∅, 60 cps

c. 120 VAC, 1∅, 400 cps

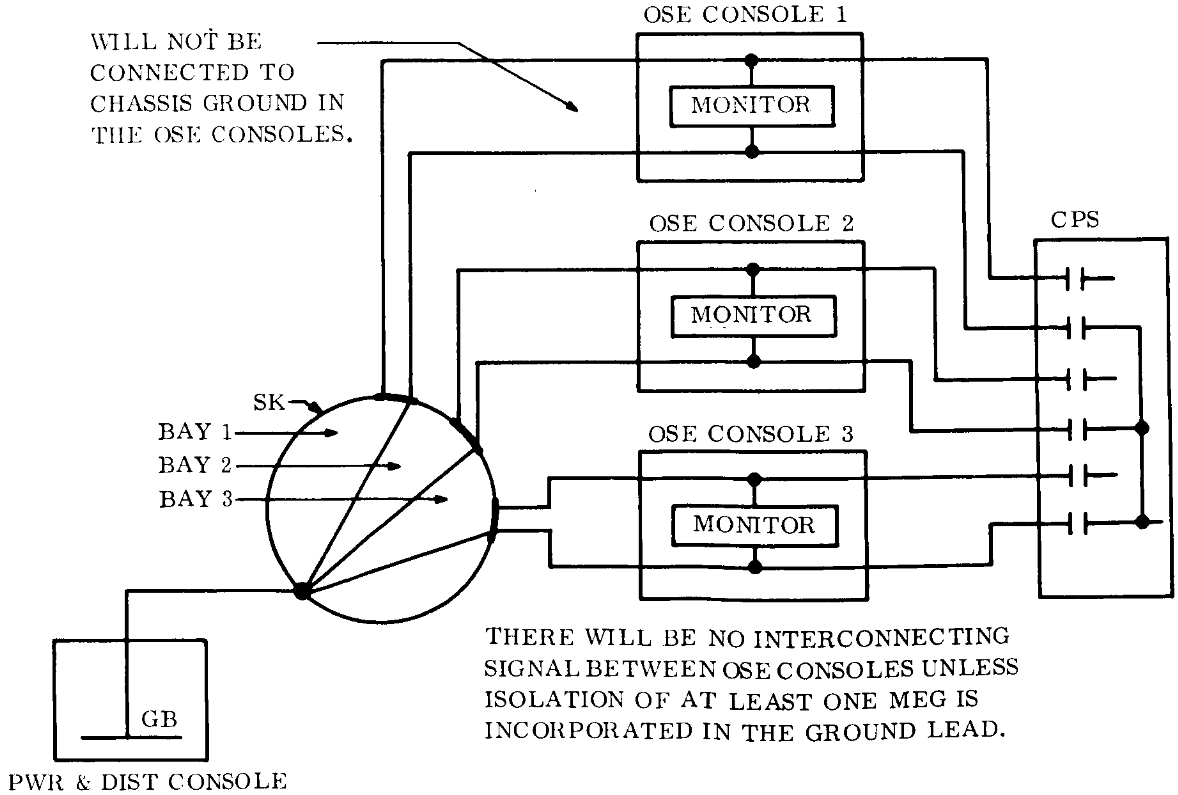
d. 208V, 3∅, 400 cps

5.1.9 GROUNDS

A copper bus bar will be contained in the power and distribution console which will have a cross sectional area equivalent to or greater than a 2/0 cable. It will serve as a common ground point for the STC. The power and distribution unit will be located as closely as possible to the S/C, and the ground bus will be connected to grounding rods by cables no greater than 10 feet in length.

5.1.9.1 SPACECRAFT GROUNDING

During system testing, the S/C frame will be considered as the zero voltage reference for all measurements. The OSE will be designed so that grounding integrity of the S/C and the STC will be maintained.



5.2 ENVIRONMENTAL

5.2.1 VIBRATION AND SHOCK

All OSE consoles will be operating in an area which will be free from shock and vibration; therefore, no requirements will be placed upon the equipment while operating. However, the consoles must be built to withstand the shock and vibration which is encountered during shipment in padded vans.

5.2.2 OPERATING TEMPERATURE AND HUMIDITY

The STC equipment will be in a controlled environment. The design ambient temperature range should be 65 to 80°F with a 50% relative humidity. Equipment which operates in a non-airconditioned building will be designed for an operating ambient temperature range of 25 to 120°F at a 95% humidity.

5.2.3 STORAGE TEMPERATURE

All equipment should be capable of withstanding a storage environment of 32 to 120°F at a 10 to 95% relative humidity.

5.2.4 RF ENVIRONMENT

The RF environment will vary at the different test facilities with the worst area being Cape Kennedy. Therefore, the equipment should be designed to operate in an ambient environment of high-power radars and transmitters.

5.3 RACK AND ACCESSORIES

5.3.1 STANDARD RACKS

The standard OSE equipment rack will be as specified in JPL Specification 30609. For shipping purposes, large consoles will separate into groups consisting of a maximum of two racks which will be mounted on casters.

5.3.2 PAINT

Paint type and color are specified in JPL Specification 30600. All OSE equipment requiring painting shall be in conformance with this specification.

5.3.3 CONSOLE AND PANEL ACCESSORIES

The accessories shall conform to the following specification.

a. Standard Control Knob - JPL Specification 30603

b. Meter - JPL Specification 30604

c. Writing Desk - Test console may contain a writing surface; however work tables will be provided in the STC.

d. Engraving and Identification - Engraving will be as per JPL Specification 30610. Dials, lights, switches, knobs, meters and other similar indicating devices shall be identified so that their function is clearly indicated to the operator.

5.3.4 COOLING

Sufficient cooling shall be provided in the racks to assure longevity of the equipment. Normally, this will be accomplished by fans located in the bottom of the racks which expel air through the top of the cabinet. The intake louvers, or grills will be located in the front or back of the rack near the bottom. A filter will be provided over the exhaust port to keep dirt from settling in the cabinet and maintaining a positive internal pressure.

5.3.5 IDENTIFICATION

a. Element Identification shall be as per JPL Specification MC-4-130 or equal.

b. Assembly Identification shall be as per JPL Specification 30506.

5.4 WIRING AND CABLES

All wiring shall conform to the following constraints:

a. Voltage drops shall be held to a minimum.

b. Current carrying capacity of wire and connectors shall be selected to satisfy the anticipated thermal operating conditions.

c. Voltage rating of the insulation shall be at least twice the anticipated peak voltage on the conductor, but not less than 600V.

d. The insulation resistance of assembled cables shall be a minimum of 100 megohms between each unshielded conductor and all other conductors and shields connected in parallel; a minimum of 20 megohms between shielded conductors and their associated shields; and a minimum of 100 megohms between each conductor and the connector shells; or any other point where continuity should not exist.

e. All wire bundles which are bent or twisted in normal use shall have the required flexibility.

f. Wires and cables shall be routed to avoid contact with rough or irregular edges. Where wires run through holes in metal, they shall be protected by suitable grommets or bushings.

g. Strain relief shall be provided at each solder point on electrical connector contact.

h. All signals should be useful after being transmitted through 150 feet of twisted pair shielded cable into a 1,000-ohm load.

5.4.1 CLASSES OF CABLES

There shall be two classes of cables, power and switching transients, and low-level signals. These functions will be separated from each other or appropriate buffering of the low-level signal will be provided.

5.4.2 DIRECT ACCESS CABLES

Direct access cables will be provided for connecting the hardwire functions to the OSE subsystem console. (See Table 5-1 for complete listing of functions.)

5.4.3 INFLIGHT DISCONNECT CABLES

Inflight disconnect cables are the cables which run between the vehicle's inflight disconnect and the STC umbilical "J" box. The "J" box then acts as a distribtuion center.

5.4.4 SPARE LEADS

A minimum of 15% spare leads will be incorporated into all OSE cables.

5.4.5 COAXIAL CABLE

Miniature coaxial cables such as Microbot coax shall be used in STC cables.

5.4.6 INTERCONNECTING STC CABLES

All interconnecting STC cables will be designed so that no two cables on the same OSE console will contain the same type connectors. This will prevent the accidental crossing of cables when connecting to a console.

5.5 CONNECTORS

5.5.1 CONNECTOR POLARITY CONVENTION

All connector junctions between the spacecraft and the OSE shall have female type connectors on the vehicle. Each cable will have a male and female connector with the male mating to the S/C and the female to the OSE.

5.5.2 POWER CONNECTOR

The A-C power cables will use a Hubbell type 3334-3W (or equivalent).

5.5.3 INTERCONNECTION STC CABLE CONNECTORS

Each cable will contain one male and one female connector and the corresponding type will be used on all cables.

5.6 SPECIAL EQUIPMENT

Special equipment shall be defined as anything which is necessary for testing of the S/C or the STC, that is not supplied as a part of an OSE console or facility test equipment.

a. "T" Boxes will be supplied for inserting in all S/C connector interfaces (excess umbilical). They will allow the vehicle to operate while monitoring the signals.

b. A special raised flooring must be provided in the STC. There should be clearance between the structural floor and the false floor in order that the interconnecting cables may be randomly routed between the consoles.

c. Dummy loads will be provided with, or in, each OSE console so that during the first power turn-on testing, the vehicle subsystems can be isolated while the power subsystem is being verified. This will prevent damage to both the power S/S and the other S/C subsystem during the initial turn-on period.

5.7 DOCUMENTATION

5.7.1 DRAWINGS

All drawings shall be of sufficient detail so that production of hardware can be easily accomplished.

5.7.2 STC DOCUMENTATION FILE

Each STC shall contain the following minimum documentation for each OSE console.

a. Schematic

b. Assembly Drawing

c. Wire List

d. Test Procedure

e. Operating Instruction

Other required documents will be STC interconnecting cable drawings, STC interface drawings and an STC cable layout drawing.

Table 5-1. Direct Access Functions

COMMAND S/S DIRECT ACCESS FUNCTIONS		
1.	DET A	Composite Command Signal
2.		Detector 4 f_s
3.		Sub-Bit Sync.
4.		Sub-Bit Sync Delayed
5.		Command sub bits
6.		Sync channel limiter emitter follower
7.		Loop phase detector
8.		Command Phase detector
9.		Detector lock signal phase detector
10.		Detector lock signal
11.	PROG CONT. A	Program Counter set pulses
12.	DECODER A TEST	Command
13.		Event Pulse to Data Encoder
14.	TR A	+28 VDC
15.		+V
16.		-V
17.		Return
18.	DET B	Composite command signal
19.		Detector 4 f_s
20.		Sub-Bit Sync.
21.		Sub-Bit Sync Delayed
22.		Command sub bits
23.	DET B	Sync channel limiter emitter follower
24.		Loop phase detector
25.		Command Phase detector
26.		Detector lock signal phase detector
27.		Detector lock signal
28.	PROG CONT. B	Program Counter set pulses
29.	DECODE B	Test Command

Table 5-1. Direct Access Functions (Continued)

30.		Event Pulse to Data Encoder
31.	TR B	+28 VDC
32.		+V
33.		-V
34.	DET C	Composite command signal
35.		Detector 4 f_S
36.		Sub-Bit Sync
37.		Sub-Bit sync delayed
38.		Command sub bits
39.		Sync channel limiter emitter follower
40.		Loop phase detector
41.		Command phase detector
42.		Detector lock signal phase detector
43.		Detector lock signal
44.	PROG CONT.	Program Counter set pulses
45.	Access Unit	Decoder Bit Sync
46.		Decoder QC bit sync
47.		Decoder Command bits
48.		Decoder Matrix Interrogate
49.		Decoder inhibit
50.		End of C&S Word
51.		Detector select A
52.		Detector select B
53.		Detector select C

TOTAL 53 WIRES

C&S

NONE

Table 5-1. Direct Access Functions (Continued)

GUIDANCE & CONTROL		WIRES
1.	Acquisition Sun Sensor simulation inputs 3 per 2 axes	6
2.	Cruise Sun Sensor simulation inputs 3 per 2 axes	6
3.	Canopus simulation input 3 per 1 axis	3
4.	Threshold Detector Outputs 6 per 3 axes	18
5.	Majority gates output 4 per 3 axes	12
6.	Solenoid Drive Current 4 per 3 axes	12
7.	Roll Integrator Output	1
8.	Sun gate Amplifier Output	2
9.	Gyro Output Amplifier 1 per 3 axes	3
10.	Accelerometer Output Amplifier	1
11.	Accelerometer Torque Input	1
12.	Gyro Logic Circuits	7
13.	Accelerometer Logic Circuits	2
14.	MCS Engine Control Error signals	8
15.	MCS Engine Control Feedback signals	8
16.	Velocimeter Engine Cutoff	1
17.	Gimbals Motor Temperature (1-5)	5
18.	Gimbals Motor Pressure (1-5)	5
19.	Gimbals Motor Voltage +(1-5)	5
20.	Gimbals Motor Voltage -(1-5)	5
21.	Gearbox Pressure (1-5)	5
22.	Gearbox Temperature (1-5)	5
23.	Stepping flip flops	10
24.	Horizon Sensor Video (Coax)	1
25.	Approach guidance amplifier	1

Table 5-1. Direct Access Functions (Continued)

GUIDANCE & CONTROL		WIRES
26.	Sweep circuit monitor (coax)	1
27.	High voltage	1
28.	Approach Guidance temperature	3
	TOTAL	140

POWER S/S		WIRES
1.	Main Regulator Voltage	2
2.	Main Regulator Current	1
3.	2.4 KC Buss Current	1
4.	Capsule current	1
5.	Xmtr current	2
6.	Battery No. 1 current	1
7.	Battery No. 2 current	1
8.	Battery No. 2 current	2
9.	Array current	1
10.	Charge Reg No. 1 - off A, B or C setting	5
11.	Charge Reg No. 2 - off A, B or C setting	5
12.	Charge Reg No. 3 - off A, B or C setting	5
13.	Main Regulator No. 1 on/off	1
14.	Main Regulator No. 2 on/off	1
15.	2.4 KC inverter No. 1 on/off	1
16.	2.4 KC inverter No. 2 on/off	1
17.	400 CPS inverter No. 1 on/off	2
18.	400 CPS inverter No. 2 on/off	1
19.	Back-up oscillator input voltage	2
20.	Stimulate fault sensor - Main Regulator	1
21.	Stimulate fault sensor - 2.4 KC inverter	1
22.	Stimulate fault sensor - 400 CPS inverter	2
23.	K_1 switch monitor	1

Table 5-1. Direct Access Functions (Continued)

POWER S/S		WIRES
24.	K_2 switch monitor	1
25.	K_3 switch monitor	1
26.	K_4 switch monitor	1
27.	K_5 switch monitor	1
28.	K_6 switch monitor	1
29.	K_7 switch monitor	1
30.	K_8 switch monitor	1
31.	Battery cell voltage monitors (every third cell)	30
TOTAL		78

PYRO S/S	
1 - 100	Bridge Wire Monitors
101	Capacitor Bank 1 Voltage
102	Capacitor Bank 2 Voltage
103	Capacitor Bank 3 Voltage
104	Capacitor Bank 4 Voltage
105	Command Power Voltage Level
106	DC Return
107	AC input voltage
108	AC Return

Total 108 Wires

RADIO SUBSYSTEM DIRECT ACCESS FUNCTION LIST	
FUNCTION	
1.	AGC - Receiver No. 1
2.	AGC - Receiver No. 2
3.	AGC - Receiver No. 3
4.	SPE - Receiver No. 1
5.	SPE - Receiver No. 2

Table 5-1. Direct Access Functions (Continued)

6.	SPE - Receiver No. 3	
7.	DPE - Receiver No. 1	
8.	DPE - Receiver No. 2	
9.	DPE - Receiver No. 3	
10.	+15V - Receiver No. 1	
11.	+15V - Receiver No. 2	
12.	+15V - Receiver No. 3	
13.	-15V - Receiver No. 1	
14.	-15V - Receiver No. 2	
15.	-15V - Receiver No. 3	
16.	-15V - Exciter No. 1	
17.	-15V - Exciter No. 2	
18.	-15V - Exciter No. 3	
19.	-25V - Exciter No. 1	
20.	-25V - Exciter No. 2	
21.	-25V - Exciter No. 3	
22.	Helix Current	P.A. No. 1
23.	Anode + 100V	P.A. No. 1
24.	Collector - 520	P.A. No. 1
25.	Cathode - 1610	P.A. No. 1
26.	Helix Current	P.A. No. 2
27.	Anode + 100V	P.A. No. 2
28.	Collector - 520	P.A. No. 2
29.	Cathode - 1620	P.A. No. 2
30.	Heater - 3V	P.A. No. 3
31.	Anode + 100V	P.A. No. 3
32.	Collector - 520	P.A. No. 3
33.	Cathode - 1610	P.A. No. 3
34.	Ground	
35.	Exciter 1 Power Output Monitor	

Table 5-1. Direct Access Functions (Continued)

36.	Exciter 2 Power Output Monitor
37.	Exciter 3 Power Output Monitor
38.	Power Amplifier 1 Power Output Monitor
39.	Power Amplifier 2 Power Output Monitor
40.	Power Amplifier 3 Power Output Monitor
41.	Ranging Monitor

A total of 41 wires are required for the S/S.

RADIO RELAY S/S DIRECT ACCESS FUNCTIONS		
FUNCTION		NO. OF WIRES
1.	Receiver 1 AGC	1
2.	Receiver 1 SPE	1
3.	Receiver 1 SDE	1
4.	Receiver 1 TR Voltage +15	1
5.	Receiver 1 TR Voltage -15	1
6.	Receiver 1 Return	1
7.	Receiver 2 AGC	1
8.	Receiver 2 SPE	1
9.	Receiver 2 DPE	1
10.	Receiver 2 TR Voltage +15	1
11.	Receiver 2 TR Voltage -15	1
12.	Receiver 2 Return	1
13.	TLM Mod No. 1 TR Voltage +6	
14.	TLM Mod No. 1 TR Voltage -6	
15.	TLM Mod No. 2 TR Voltage +6	
16.	TLM Mod No. 2 TR Voltage -6	
17.	Output Selector +6	
18.	Output Selector -6	

Table 5-1. Direct Access Functions (Continued)

		NO. OF WIRES
19.	Lock Indicator	
20.	Data	
21.	Sync	
TOTAL		21

SCIENCE

No data is presently available; 170 wires will be assumed.

TELEMETRY SUBSYSTEM		
DATA STORAGE		WIRES
1.	+3.5 Volts	1
2.	+28 V	1
3.	-3 V	1
4.	Motor Temperature	1
5.	Tape Motion indicator	1
6.	Power Amplifier output	1
7.	Loop error signal	1
8.	Scan Data Output	1
9.	Scan Data Gate	1
10.	Dump Data Output	1
11.	Driver Current Waveform	1
12.	CRB Data Output	1
13.	CRB Filled Signal	1
14.	Monitor A/D converter No. 1	1
15.	Monitor A/D converter No. 2	1
16.	Monitor A/D converter No. 3	1
17.	Monitor P/N Generator No. 1	1
18.	Monitor P/N Generator No. 2	1

Table 5-1. Direct Access Functions (Continued)

		WIRES
19.	Monitor P/N Generator No. 3	1
20.	Monitor Programmer No. 1	1
21.	Monitor Programmer No. 2	1
22.	Monitor Programmer No. 3	1
23.	-20V TR monitor	1
24.	+3.5V TR monitor	1
25.	Data Encoder return	1
TOTAL		25

Table 5-2. In-Flight Disconnect Functions

RADIO SUBSYSTEM	
1.	(Coax) High Gain Antenna Probe
2.	(Coax) Medium Gain Antenna Probe
3.	(Coax) Primary Low-Gain Antenna Probe
4.	(Coax) Secondary Low-Gain Antenna Probe

RELAY RADIO SUBSYSTEM

1. (Coax) Antenna Probe

COMMAND SUBSYSTEM

None

POWER SUBSYSTEM	
1.	Array/Battery Bus Voltage
2.	Array Enable Switch SW-1 Monitor
3.	Turn-on Enable Switch, SW-1
4.	Turn-off Enable Switch, SW-1
5.	Enable Switch, SW-1, Return
6.	External Power, 44-55 VDC, 15 Amps
7.	External Power Return
8.	Battery No. 1 Voltage
9.	Battery No. 2 Voltage
10.	Battery No. 3 Voltage
11.	Battery No. 1 Temperature
12.	Battery No. 2 Temperature
13.	Battery No. 3 Temperature
14.	Battery Sensor Temperature Return
15.	400 cps, Phase 1 Voltage
16.	400 cps, Phase 2 Voltage
17.	400 cps, Phase 3 Voltage

Table 5-2. In-Flight Disconnect Functions (Continued)

POWER SUBSYSTEM (Continued)	
18.	2.4 KC Voltage
19.	2.4 KC Voltage Return
20.	Raw Battery Bus Voltage
CONTROLLER AND SEQUENCER	
1.	Alert Signal
2.	Sync Signal
3.	Command Information
4.	Engine Burn Enable State
5.	Speed-Up Timing (138.1)
6.	Inhibit Master Timer
7.	Update Master Timer
8.	Clean Timer
9.	Inhibit Power Supply No. 1
10.	Inhibit Power Supply No. 2
DATA ENCODER	
1.	Modulated Subcarrier A
2.	Modulated Subcarrier B
3.	Modulated Subcarrier C
4.	Work Sync
5.	Bit Sync
6.	Frame Sync
7.	Mode II Command
8.	Data Encoder Return
9.	Recorders to Launch Mode
10.	2 f_s

Table 5-2. In-Flight Disconnect Functions (Continued)

GUIDANCE AND CONTROL	
1.	Gyro Heater Power
2.	Gyro Rebalance Amplifier 1
3.	Gyro Rebalance Amplifier 2
4.	Gyro Rebalance Amplifier 3
5.	Common Return
6.	Shield Return
7.	Approach Guidance Update 1
8.	Approach Guidance Update 2
PYROTECHNIC SUBSYSTEM	
1.	Continuity
2.	Continuity Return
3.	Arm Signal 1
4.	Arm Signal 2
DATA STORAGE SUBSYSTEM	
None	
SCIENCE	
Undetermined. Ten umbilical lines are assumed for this subsystem.	
CAPSULE	
1.	Ground Power Main Bus
2.	Ground Power Main Bus Monitor
3.	Internal/External Power Transfer
4.	Internal/External Power Monitor
5.	Barrier Pressure Monitor
6.	Checkout Sequence Signal
7.	Spacecraft/Capsule External Clock Select
8.	Telemetry Mode 1 Select

Table 5-2. In-Flight Disconnect Functions (Continued)

CAPSULE (Continued)	
9.	Telemetry Mode 2 Select
10.	Telemetry Mode 3 Select
11.	Telemetry Mode 4 Select
12.	Telemetry Mode 5 Select
13.	UHF Transmitter On/Off
14.	UHF Power Amplifier On/Off
15.	VHF Transmitter On/Off
16.	Sequencer Enable
17.	Sequencer Rate Select
18.	Ground Telemetry Clocks
19.	External/Internal Telemetry
20.	Telemetry On/Off
21.	Tape Recorder On/Off
22.	Tape Recorder Record/Playback Select
23.	Telemetry Bit Rate Output
24.	Telemetry Frame Rate Output
25.	Telemetry Encoder Output
26.	Sensor Power Supply Monitor
27.	Pyrotechnic Continuity
28.	Telemetry Mode Monitor
29.	Tape Recorder Reset

CII - VB260FD102

SYSTEM TEST COMPLEX

GROUND POWER DISTRIBUTION

Index

1.0 SCOPE

This document is a functional description of the Ground Power Distribution equipment required to activate a System Test Complex for the 1971 Voyager Spacecraft.

2.0 APPLICABLE DOCUMENTS

GE VB260SR101	1971 Voyager Spacecraft System Test Complex Test Objectives and Design Criteria
GE VB260SR102	1971 Voyager Spacecraft System Test Complex Design Characteristics and Constraints

3.0 DESCRIPTION

Each System Test Complex is required to have an A-C power distribution console incorporated within itself. Installation of multiple System Test Complexes may have their respective power distribution consoles arranged in physical proximity to each other, or have their power distribution components integrated physically into a master power distribution network. Regardless of the physical configuration used, each power distribution network should provide the functions described herein, for each System Test Complex.

The A-C power distribution and power grounding are shown in Figure 3-1. Incoming power, obtained from range facilities in the case of the STC in the Spacecraft Check-out Facility at the Eastern Test Range, will be regulated at the source, to the degree negotiated with the power supplier. Should their degree of regulation prove to be insufficient, any further regulation will be implemented by incorporating suitable regulation transformers in the subsystem consoles of the STC. The power distribution consoles will however, incorporate sufficient meters to indicate voltage, current, and power levels.

A copper ground bus, which will be equivalent to or greater than 2/0 cable, will be contained within the console, and will be used as a common tie point for all the OSE. The power distribution console will be located as closely as possible to the S/C, and the ground bus will be connected to grounding rods, so that the maximum impedance to earth is less than three ohms. Also, the building structure will be connected to suitable earth ground rods. Finally, to ensure RF grounding, the ground bus will be connected to a water main or other structure which has a large capacitive coupling to earth.

As indicated in Figure 3-1, a remotely activated switch capability is required, so that both 60-cycle and 400-cycle power can be shut down by the Test Conductor in an emergency.

A seperate circuit breaker will be associated with each output power line shown in Figure 3-1. Frequency, voltage, current and power meters are required only at the input bus.

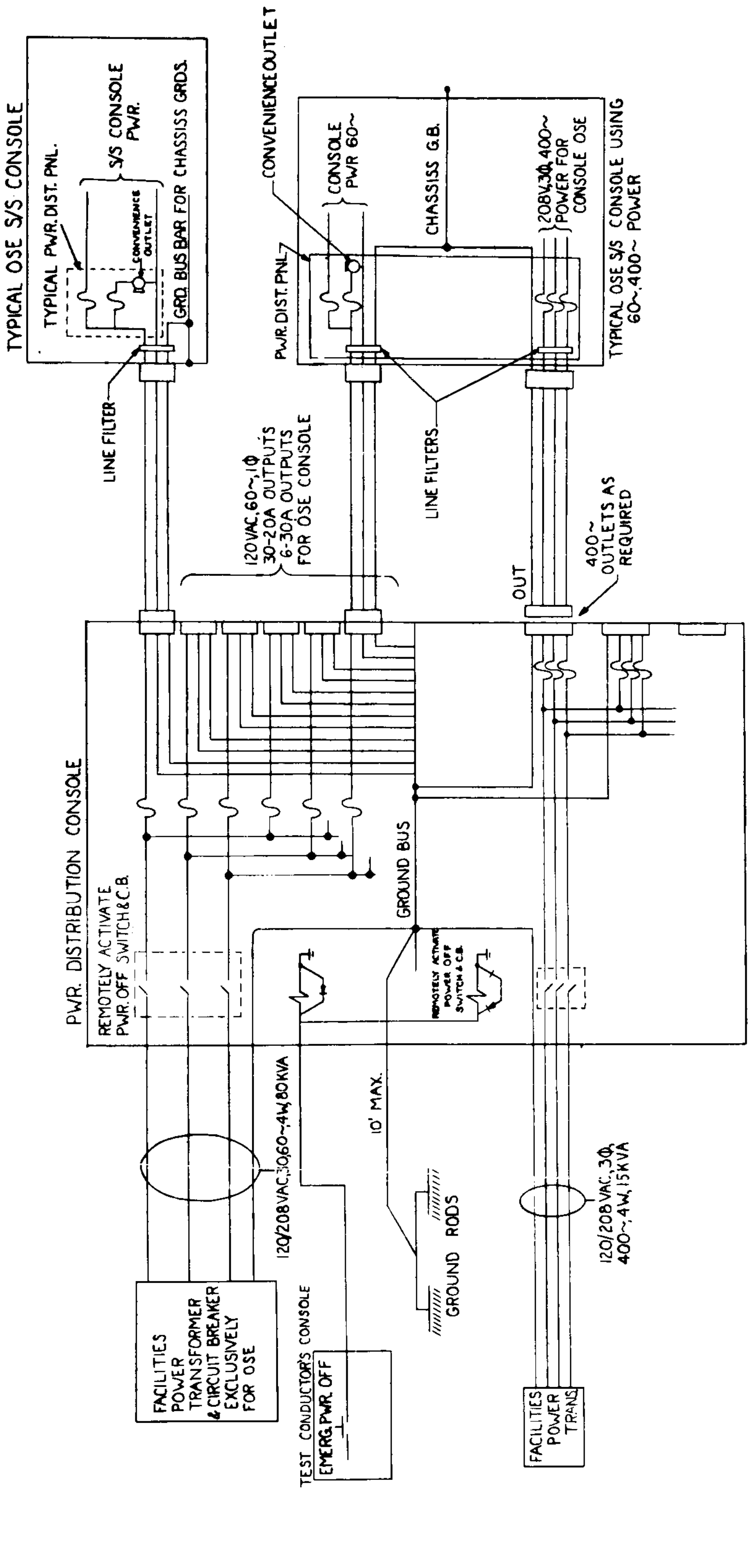


Figure 3-1. Power Distribution and Grounding System for the STC

The power distribution equipment will include master on-off switches at the input which can be overriden by the remote emergency power shut down switch.

4.0 INTERFACE DEFINITION

4.1 ELECTRICAL INTERFACES

a. Input from Facilities

1. 120/208 VAC, 60 cycle, 3 ohms, 4W, 80 KVA

2. 120/208 VAC, 400 cycle, 3 ohms, 4 W, 15 KVA

b. Output to STC

1. 30 output lines, 60 cycles, 20 A
   6 output lines, 60 cycles, 30 A

2. Output lines, number undetermined, 400 cycles.

c. Remote power switch control from Test Conductor's Console.

5.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Ground Power Distribution for a System Test Complex can be packaged into one standard console, which would conform to STC design constraints. The design should be for a controlled, interior environment.

6.0 PERFORMANCE CHARACTERISTICS

a. Meter accuracy - $\pm$ 1%

b. RFI - In accordance with STC Design Constraints

7.0 SAFETY PROVISIONS

It is considered that the breakers and grounding provisions will be adequate protection against surges, overvoltages and lightening strokes. Interlocks, and heavy screening and precaution warning plates, common in the design of power handling equipment, will constitute adequate provision for personnel safety.

The safety factor of most concern in the control of electromagnetic radiation, which tends to affect the performance of associated equipment. EMI suppression techniques should be emphasized in the console design.

CII - VB260FD103

SYSTEM TEST COMPLEX

COMPUTER DATA SYSTEM

Index

1.0 SCOPE

This document describes the functions of an automatic on-line data acquisition, processing and control system designed for supporting spacecraft system and subsystem tests. It is a part of the System Text Complex (STC) where it will be used to acquire, monitor, record and display selected spacecraft and OSE functions; and to control portions of the test.

2.0 APPLICABLE DOCUMENTS

The following documents pertain to this specification:

General Electric Publications

VB260SR101	STC Test Objectives and Design Criteria
VB260SR102	STC Design Characteristics and Restraints
VB260FD105	Functional Description, Voyager System Test Complex Printers and Displays.

Other Documents

Univac MO5262A	General Description — Univac 1218
Univac MO7564B	General Description — Univac 1219
Univac MO6763	Technical Description — Univac 1218
Univac MO8864	Technical Description — Univac 1219
Univac MA3545	Description of Input/Output Operation
Univac PX3450	Univac 1232 Input/Output Console
Univac PX3221	Univac 1240 Magnetic Tape Unit
Univac MO8664	Univac 1469 High Speed Printer

JPL Spec GMG-50109-DSN-A — Design Spec — CVE

3.0 DESCRIPTION

3.1 GENERAL

The Computer Data System (CDS) is designed to acquire, process and printout data from spacecraft system and subsystem tests and to assist in the control of these tests. During these tests, data is acquired from the Ground Telemetry System and/or from the Flight Telemetry Subsystem OSE and from spacecraft direct-access and umbilical circuits routed through junction boxes and the subsystem OSE.

3.2 BLOCK DIAGRAM

Figure 3-1 is a block diagram of the CDS showing the physical layout of the computer facility and the System Test Complex acquisition and printing equipment which is required to support system test.

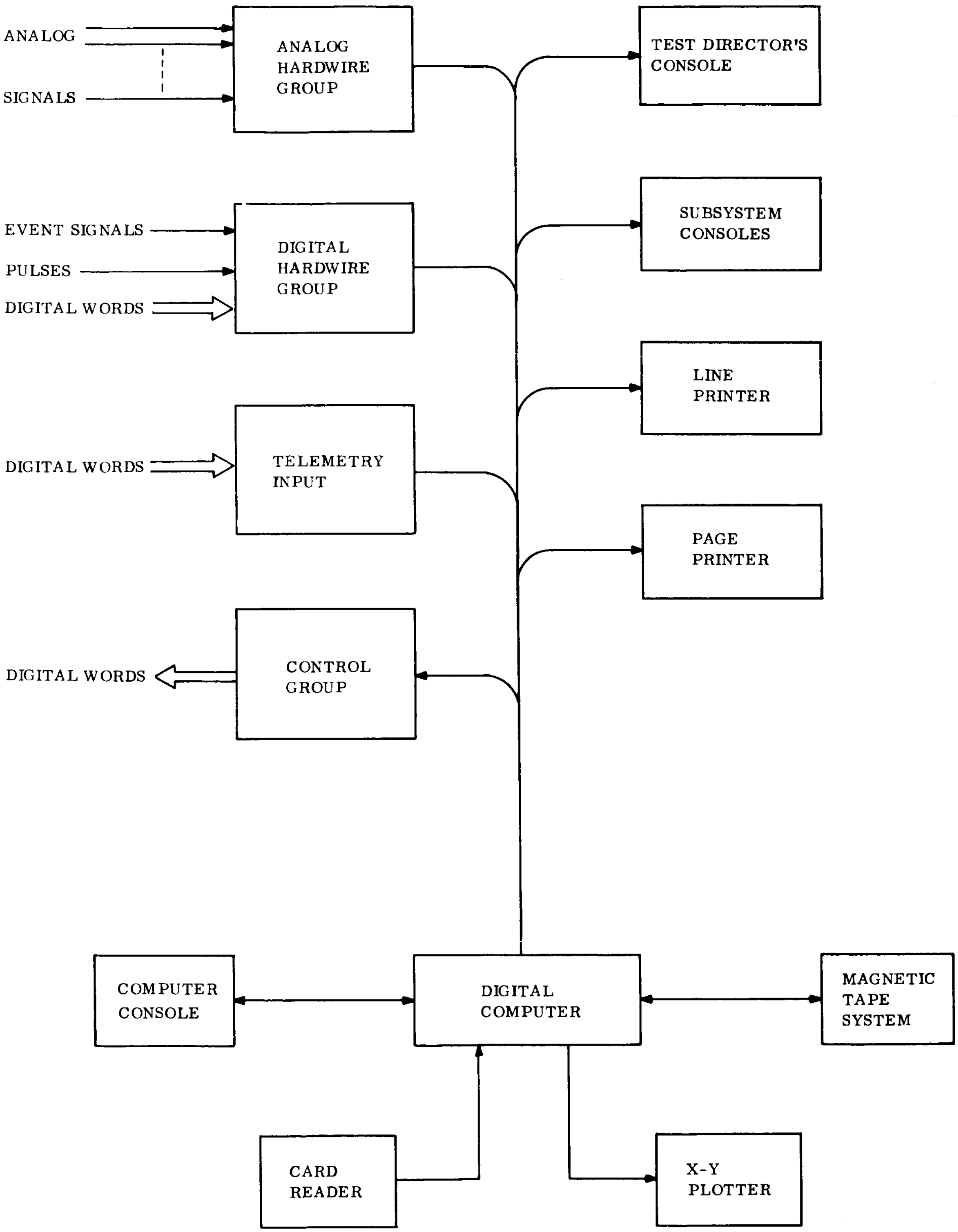


Figure 3-1. Computer Data System for the System Test Complex, Block Diagram

3.3 FUNCTIONS

3.3.1 DATA ACQUISITION

The CDS acquires data from various sources as indicated below.

a. PCM Telemetry — The CDS acquires all telemetry data from the Ground Telemetry System or the Flight Telemetry System OSE. The data will be provided in parallel with all bits of one telemetry word being presented in parallel together with an identification word and a word sync pulse. The word sync pulse will notify the CDS that the data and identifier is available at the input lines.

b. Analog Hardwire Signals — The data system scans a series of analog signal voltage lines with a computer controlled data multiplexer. The samples are converted to digital form and entered into the computer for processing.

c. Digital Hardwire Signals — The data system scans a series of digital signal voltage lines with a computer controlled multiplexer. The digital signals include the following types of data:

   1. Control Switch Status Signals indicative of the status of control switches and spacecraft or OSE control voltages.

   2. Event Signals indicative of control pulses or transient voltages.

   3. Counting Registers for pulse or sinewave signal cycles.

   4. Real Time Clock Register used to accumulate real time pulses from the central timing system for entering time information into the computer.

   5. Spacecraft Clock Register used to accumulate spacecraft time pulses originating in the C&S OSE.

d. Internal Data Acquisition — The Computer Data System acquires information from the following additional data system sources for control purposes:

   1. Test Director's Control Console

   2. Subsystem OSE Consoles

   3. CDS Operator's Console Keyboard

   4. Digital Magnetic Tape Reproducers

   5. Punched Card Readers

6. Punched Paper Tape Readers

7. Printer Control Panel

3.3.2 DATA RECORDING

All data acquired is recorded in digital form on magnetic tape for subsequent detailed analysis. In the case where the data is recorded on tape prior to entry into the CDS, recording will not be duplicated. However, all other data will be recorded by the CDS. This includes event occurrences, changes of state of status channels, counting register reading and time; and information for test identification, spacecraft subassembly identification, definition of data system operating characteristics, etc.

3.3.3 ON-LINE DATA PROCESSING

The real time program acquires data from the system test complex and stores it in memory. It records this data on digital magnetic tape for later analysis and it performs computations on the data as required for the test in progress.

The program has the capability of displaying the status of selected data at any time or periodically. It controls the sequence of the test, depending on previously acquired data, by directing a sequence of commands to the peripheral equipment and the subsystem OSE. Time correlation of certain OSE stimulation and spacecraft function responses are provided.

The computer programs associated with the real time system are classified, according to the function they perform, as Data Processors and Control Processors.

3.3.3.1 DATA PROCESSORS

a. The Interrupt Routine is entered upon receipt of an interrupt from an external device. When a computer senses an interrupt it immediately transfers control to the interrupt routine associated with the external device that caused the interrupt. The responsibilities of the interrupt routine are to transfer the data into the computer, identify it, and indicate to the other programs the need for further processing.

   There are two classes of interrupt routines, buffered transfers and normal transfers. The buffered transfers are accomplished with blocks of data with an interrupt indicating the end of a block, whereas the normal transfers process one word at a time. There is an interrupt routine for each type of data input, i.e., Analog, Digital, Telemetry, Control Information, etc.

b. The Analog Data Processor controls the operation of the analog hardwire equipment group and processes the data obtained from it. It identifies and associates time with the analog data and performs an alarm limit test. In addition to providing data for the display and print routines, this routine also provides information to the control routine for use in determining the test sequence.

c. The Digital Data Processor functions in a manner similar to the analog data processor except that it controls the digital hardwire equipment group and receives data from it.

d. The Control Group Processor receives its input from the control processors and formats and sends control words to the subsystem OSE to provide real time control of the test sequence.

e. The Engineering Unit Conversion Routine is used to convert the telemetry signals and analog voltages to appropriate engineering units. It is also used to counteract the effect of nonlinearities in the measuring circuitry. For this function, it utilizes either a table lookup or a polynomial solution technique.

f. The Binary to BCD conversion routine makes the necessary calculations to present a binary number in decimal notation. This routine is entered whenever it is necessary for the computer to communicate decimal data to an operator.

g. The Print Format routines arrange binary information in the proper output format. These routines utilize the Engineering Unit Conversion and Binary-BCD subroutines. Each different print format requires a separate format routine. The output from these routines is in condensed BCD format which subsequently is sent to the Output Processors for further formatting and dissemination to the specific output devices.

h. The Tape Write routine records all incoming information on the digital magnetic tape. Each variable length record contains a particular type of data with the necessary identification included for each one.

i. The Output Processors receive BCD information from many buffers since there is a buffer for each output device and for each different output format. The output processors gather one line of print information at a time and transfer this information directly to the output device for which it is intended. The line printer receives its information a line at a time, whereas the character printers receive their information one character at a time.

j. The Alpha Numeric Display routine will update the Test Conductor's A/N Display once per second. It will establish what block of data is requested and will then output that block in a manner similar to the line printer.

3.3.3.2 CONTROL PROCESSORS

a. The Executive routine controls the order in which routines are selected for execution. The determining factors are the speed at which data is being transferred to the computer, the length of time that the routine takes to perform its task, and the importance of the routine's task in relation to other tasks. This routine searches through the list of flagged program names to determine which program requires attention. A test is performed which determines whether the program has been interrupted so that control can be returned to the proper location.

b. The Test Sequence Control Processor reads one record containing control and parameter information into memory from a control tape. The routine distributes modifications to the control and parameter buffers associated with the different types of input data. It also can be actuated upon the occurrence of a unique step number or a unique event. The capability of repeating portions of the system test by searching the tape for the proper sequence and rereading the information also exists.

c. The Request Processors determine which information required in a block printout is directed to which print device at a given time during the system test.

   When a print request is recieved from a particular printer, this routine utilizes a list of preselected measurement identifications that are associated with the printer in order to find the appropriate data in the Latest Available Data Buffer. This binary information is collected and transmitted to the format processors for output.

3.3.4 NON-REAL TIME DATA PROCESSING

The primary purposes of these programs are to display portions of the system test data in greater detail than that provided during the test and to analyze the data collected during several tests in order to detect trends in the performance of the vehicle.

The input for this program will be the digital data tapes written during the system tests. The program will also utilize many of the routines written for the Real Time Program.

Because of the loading problem on the CDS which may occur when several spacecrafts are being tested at the STC, much of the non-real time data processing should be done on an off-line processor such as the IBM 7094. In this case, the programs used will be made up largely of the standard programs available as part of the data processing operating system.

The non-real time programs will be capable of presenting selected measurements and derived data in printed and/or plotted form.

3.3.5 REAL TIME DATA PRINTOUT AND PLOTTING

The data system provides printout in engineering units of telemetry data and analog hardwire data or of variables which are derived from these data. The occurrence of events and alarm information is also indicated on the system printers. Time identification is also provided.

The data system also provides the capability to plot data in the form of X versus Y and X versus time charts.

The equipment used for this purpose is described in the Functional Description of the Systems Test Complex, Printer and Displays.

3.4 MAJOR DATA SYSTEM ELEMENTS

3.4.1 COMPUTER SUBSYSTEM

a. Computer — The data system computer is a Sperry-Rand Univac 1218 with 16,384 words of core storage and eight input/output channels. Word size is 18 binary digits (bits). Memory cycle time is four microseconds.

b. Computer Control Console — The control console is a Univac Model 1233 containing an alphanumeric keyboard and character printer, a 300 character/second paper tape reader and a 100 character/second paper tape punch.

c. Magnetic Tape System — The magnetic tape system is a Univac Model 1240 containing four transports capable of dual density recording, 200 bits per inch and 556 bits per inch. The tape system writes in IBM compatible format to permit off-line data processing of CDS tapes.

d. Punched Card Reader — The data system contains a punched card reader for use in setting up special programs, changing parameter values and data reduction constants, etc.

e. Data Plotter — The data system contains an X versus Y or time plotter. The plotter is an incremental type, 0.01 inch/increment (X or Y axis) with a maximum rate of 200 increments/second.

3.4.2 DATA INPUT SUBSYSTEM

One Data Input Subsystem is used with each System Test Complex.

a. Telemetry Input — The DIS receives telemetry from the Telemetry and Command Data Handling Subsystem (part of the Telemetry Subsystem OSE). The input data consists of an 18-bit word containing one telemetry word, a word identification code, and a data limit flag bit. A word sync pulse will be provided to indicate the presence of the input data word.

c. High-Speed Line Printer

d. Four Low-Speed Page Printers

The physical characteristics of this equipment will be supplied later.

7.0 SAFETY CONSIDERATIONS

There are no special requirements applicable to this equipment for providing features or procedures to avoid damage to flight equipment or to avoid hazard to mission characteristics. All such special requirements applicable to the STC will be implemented in the various subsystem OSE sets.

b. Analog Hardwire Group — The analog hardwire group consists of a patch cord programming system, signal conditioning circuits, an analog signal multiplex system, a group of analog signal amplifiers and an analog-to-digital converter.

c. Digital Hardwire Group — The digital hardwire group consists of the circuitry required to enter digital signals into the computer. It includes the following types of circuits:

   1. Status Registers (18 bits) for sensing status signals from isolated contacts in the subsystem OSE or the spacecraft.

   2. Counters (18 bits) to totalize pulses or sinewave signal cycles.

   3. Parallel Registers (18 bits) for entering digital words generated in the OSE or spacecraft.

d. Control Group — The control group provides for control words going out of the computer to the subsystem OSE consoles. It provides the means by which the CDS exercises control over the test sequence and indicates to the various consoles the status of the test. The output signal will be an 18-bit parallel word containing data and destination information.

3.4.3 PRINTER AND DISPLAY SUBSYSTEM

This subsystem provides the primary output from the CDS. It is described in VB260FD105 Functional Description, STC Printers and Displays.

4.0 INTERFACE DEFINITION

The CDS has interfaces with the following pieces of OSE:

a. The Ground Telemetry Subsystem

b. The Ground Command Subsystem

c. The Central Timing System

d. The OSE for each flight subsystem

e. The Test Director's Console

4.1 CDS/GROUND TELEMETRY SUBSYSTEM INTERFACE

The Ground Telemetry Subsystem is, functionally, the JPL Telemetry and Command Data Handling System (TCD). With respect to telemetry, it can be characterized as a general purpose digital computer with some special purpose input/output equipment.

All telemetry data will be preprocessed by the TCD and transmitted to the CDS. The interface consists of the following signals:

a. Data — 18-bit words in parallel consisting of the binary telemetry word, an identification word and several flag bits indicative of sync status and out-of-limit status.

b. A word sync pulse indicating the presence of the data word on the input lines.

4.2 CDS/GROUND COMMAND SUBSYSTEM INTERFACE

The Ground Command Subsystem consists of that portion of the Telemetry and Command Data Handling System known as the Command Verification Equipment (CVE). The CVE is a general purpose digital computer with special input/output equipment. The CDS will interface with the input normally used for TTY input from SFOF. This input is five level characters in format described in JPL Specifications BCC-50305-GEN and GMG-50109-DSN-A.

4.3 CDS/CENTRAL TIMING SYSTEM INTERFACE

A register is used to accumulate real time pulses from the central timing system for entering time information with the computer. Clock register readings are required for time tagging data, status changes and event occurrences and for control purposes.

4.4 CDS/SUBSYSTEM OSE INTERFACES

The Computer Data System has an interface with each set of subsystem OSE used in the System Test Complex. The CDS receives analog and discrete signals from the OSE. The CDS also provides certain control signals to the subsystem OSE for control of stimuli and spacecraft not practicable to implement by radio or by the subsystem engineer using manual control.

These interfaces are described in the subsystem OSE functional descriptions, as follows:

VB263FD101	Radio Subsystem OSE
VB263FD103	Command Subsystem OSE
VB263FD104	Relay Radio Subsystem OSE
VB263FD105	Bulk Storage
VB263FD106	Data Encoder
VB264FD105	Controller and Sequencer
VB266FD101	Power Subsystem

VB264FD101	Vehicle Control and Planet Sensor Subsystem OSE
VB264FD102	Autopilot Subsystem OSE
VB264FD103	Attitude Control Subsystem OSE
VB265FD101	Pyrotechnics OSE

4.5 CDS/TEST CONDUCTOR'S CONSOLE INTERFACE

The interface between the CDS and the Test Conductor's Console consists entirely of digital signals used to control the test sequence and to provide display information to the test conductor.

The following signals originate in the Test Conductor's Console:

a. Test Start (1 Bit)

b. Test Hold (1 Bit)

c. Test Resume (1 Bit)

d. Test Mode (2 Bits)

e. Command Mode (2 Bits)

f. Test No. Select (8 Bits)

g. Alphanumeric Page Select (16 Bits)

The following signals originate in the CDS:

a. Alphanumeric Data (18 Bits)

b. Control (Display) Data (18 Bits)

5.0 PERFORMANCE CAPABILITIES

5.1 GENERAL

The computer subsystem of the CDS is a permanently located facility. Support of systems test in different test complexes requires moving the analog hardwire group, the digital hardwire group, the control group, the test conductor's console, the subsystem engineer's control equipment and the printer subsystem as necessary. The data system can operate with a maximum of 3700 feet between the computer subsystem and its associated test complexes. A digital data communication channel is required between the computer subsystem and the test complexes.

5.2 SYSTEM TEST

The CDS is designed to completely support one system test, including acquisition, processing, display and control of data from the telemetry channel and hardwire channels associated with one test complex. The computer subsystem can be configured such that it can support up to three test complexes if each test complex has its own set of computer peripheral equipment. In this expanded configuration, the computer subsystem can support systems testing on the three STC's simultaneously.

5.3 SUBSYSTEM TESTS AND CALIBRATION

Support of subsystem tests and calibration can be provided as required.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 CENTRAL COMPUTER AREA

This area houses the following equipment:

a. Central Computer (Univac 1218 or 1219)

b. Computer Control Console (Univac 1232)

c. Digital Magnetic Tape Recorders (Univac 1240)

d. Punched Card Reader

e. X-Y Plotter

f. Tab Card Keypunch

g. Input/Output Adapter

Approximately 500 square feet of floor space is required for this equipment.

The CDS used by JPL for the Mariner C Program will satisfy the requirements for items a through f, above. The Input/Output Adapter (item g) will be new for the Voyager Program.

6.2 SYSTEM TEST COMPLEX AREA

This area will provide space for the following items of the CDS:

a. Analog Hardwire Equipment Group

b. Digital Hardwire Equipment Group

CII - VB260FD105

SYSTEM TEST COMPLEX

PRINTERS AND DISPLAYS

Index

1.0 SCOPE

This document contains a functional description of the data printers and plotters used by the Computer Data System (CDS) in the System Test Complex to display data collected during spacecraft system tests. It does not include those displays provided as part of the System Test Conductor's Station Equipment or the Subsystem Test Equipment. It does describe the Telemetry Data Conversion equipment, which converts the output of the telemetry demodulator.

2.0 APPLICABLE DOCUMENTS

The following documents pertain to this specification:

G E Publications

VB260SR101	STC Test Objectives and Design Criteria
VB260SR102	STC Design Characteristics and Restraints
VB260FD103	Voyager, Operational Support Equipment, Computer Data System Functional Description
VB260FD104	Voyager, Operational Support Equipment, Test Directors Console, Functional Description
VB263FD106	Voyager, Operational Support Equipment, Data Encoder Functional Description

3.0 DESCRIPTION

3.1 GENERAL

The Computer Data System has three types of devices for presenting large quantities of data to the system test conductor and subsystem engineers. The system test conductor will have a line printer capable of displaying large quantities of data at relatively high speed. The subsystem engineers will have access to low-speed page printers for displaying selected data. The data system plotter provides X vs Y or X vs T plots from system data during the test.

3.2 SYSTEM TEST CONDUCTOR'S PRINTER

This printer is a 1000-line-per-minute printer connected on-line with the CDS Computer. It can print 120 alphanumeric characters per line. Sixty-three printable characters plus space, selected by a six-bit binary code, are available in each character position. The unit consists of an electromechanical printer, print control unit, power supply and cooling system. The CDS computer program directs the format and printing of a line and page. This printer will be used to display all data from all Subsystems pertinent to the Spacecraft System Test currently in progress.

3.3 SUBSYSTEM ENGINEERS' PRINTERS

These printers are 100-word-per-minute printers connected on-line with the CDS Computer. They can print 85 alphanumeric characters per line. Sixty-three printable characters plus space, selected by a six-bit binary code, are available in each character position. The unit is essentially a computer controlled electric typewriter. They will be used to display selected subsystem parameters pertinent to the spacecraft test currently in progress.

3.4 DATA SYSTEM PLOTTER

The plotter is a digital incremental recorder for high-speed plotting of the CDS output. The plotting sheet is 11 inches wide and 120 feet long. The pen moves 1/100 inch per step at the rate of 200 steps per second. All recording (discrete points continuous curves or symbols) is accomplished by the incremental stepping action of the paper and the pen. The CDS Computer program directs the motion of the pen relative to the paper to produce the required displays. The plotter will be used to plot one spacecraft variable as a function of a second spacecraft variable or one spacecraft variable as a function of time.

3.5 DIGITAL TO ANALOG CONVERTERS

The converters consist of precision resistor networks with a reference supply voltage which convert the binary coded signals received from telemetry to a unipolar dc voltage. Each includes a six-bit storage register together with control logic for decommutating the desired telemetry word.

4.0 INTERFACE DEFINITION

The STC Printers and Plotter are connected to the CDS only. They do not have a functional interface with any other piece of equipment although they may be located in close proximity to other equipment. The interface with the CDS consists of digital data signals and digital control signals generated in the CDS Computer and in the Plotter/Printers. These signals are described in the CDS functional description. The digital-to-analog converters receive their input from the ground telemetry system. The input is an 18-bit digital word which contains the following data:

a. Data - 7 bits

b. Data Identification - 9 bits

c. Sync Status Flag - 1 bit

d. Out-of-Limits Flag - 1 bit

The output signal is used to drive display devices in the Test Director's Console and the Subsystem Consoles. It consists of the following signals:

a. Data - Analog Voltage (0 to 10 VDC)
Discrete Levels (Grd-Open)

b. Out-of-Limit Flag - Digital Signal (OV or 6VDC)

5.0 PERFORMANCE PARAMETERS

5.1 SYSTEM TEST CONDUCTOR PRINTER

Speed: 1000 lines per minute

Line Size: 120 characters

Spacing: 10 Characters per inch horizontally
6 lines per inch vertically

Paper Slew Rate: 25 inches per second

Vertical Paper Position Control: 8-track punched paper

Program Controlled Functions: Disable line feed
Enable line feed
Move paper to top of form

Manually Controlled Functions: Move paper to top of form
Move paper to load position

Printer Adjustments: Vertical line adjust 2.25 inches
Index Pulse Phase 15 degrees
Paper tension control
Penetration control
Density of print

Copies: One master and up to three carbons

Printer Computer Interface: 18-bit word, parallel

5.2 SUBSYSTEM ENGINEERS' PRINTERS

Speed: 100 words per minute

Line Size: 85 Characters

Spacing: 10 Characters per inch horizontally
6 Characters per inch vertically

Printer-Computer Interface: 6-bit word, parallel

5.3 DATA SYSTEM PLOTTER

Speed: X-Axis 200 steps/second
Y-Axis 200 steps/second
Pen 10 operations/second

Step Size: 1/100 inch

Chart Paper: Width 12 inches
Plotting width 11 inches
Length 120 feet
Sprocket Holes 0.130 inch diameter on 0.318 inch centers

Plotter-Computer Interface: 19-bit word, parallel

5.4 DIGITAL TO ANALOG CONVERTER

Input:	Digital Signals - 0 or -6VDC
Output:	0 to -10VDC at 6000 ohms
Resolution:	One part in 128
Accuracy:	0.1% of Reference Voltage
Bit Rate:	DC to 200KC
Settling Time	(Maximum) 5 microseconds
Power Description:	1 Watt (maximum)

Size: Each D/A converter consists of one digital logic module (Approx 5" x 7"). One reference voltage supply is required for every 10 D/A converters. Each reference supply consists of one digital logic module.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 SYSTEM TEST CONDUCTORS PRINTERS

Size: 60 inches high x 64 inches wide x 30 inches deep

Weight: 1500 pounds

Clearance Requirements: Front 4 feet
Rear 4 feet
Left Side 4 feet
Right Side 4 feet

Power Requirements: 115V, ± 10%; 400 cps, ± 5%, 3 phases; 250 watts
115V, ± 10%; 60 cps, ± 5%, 1 phase; 2KW

Environmental Characteristics: Forced Air Cooling, 500CFM 15.6 °C Min and 35 °C Max
Relative Humidity: 40 to 80%

6.2 SUBSYSTEM ENGINEERS PRINTERS

Size: 40 inches high x 20 inches wide x 18 inches deep

Weight: 120 pounds

Power Requirements: 115 V, 60 cps, 65 watts

Environmental Characteristics: None

6.3 DATA SYSTEM PLOTTER

California Computer Products model No. 560R, or equivalent.

7.0 SAFETY CONSIDERATIONS

There are no special requirements, applicable to this equipment, for providing features or procedures to avoid damage to flight equipment or to avoid hazard to mission characteristics.

CII - VB260FD104

SYSTEM TEST COMPLEX

TEST CONDUCTOR'S CONSOLE

Index

1.0 SCOPE

This document contains a functional description of the Test Conductor's Console of the Voyager System Test Complex (STC). It describes the functions, interfaces and characteristics, of the Test Conductor's Console used to coordinate and support tests of the spacecraft as a part of the System Test Complex.

2.0 APPLICABLE DOCUMENTS

VB260SR101 STC Test Objectives and Design Criteria

VB260SR102 STC Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

3.1 PURPOSE

The Test Conductor's Console shall provide the controls and displays required by the STC's Test Conductor to control, supervise and monitor the progress of the test sequences to the degree described below.

3.1.1 SYSTEM TESTS

These tests will be performed using one of two modes.

a. Test sequenced and monitored by the Computer Data System. In this mode the Test Conductor will select the tests to be run, the stop points (i.e. end of each test step, end of each sequence, end of each major test), control CDS as to when to restart after a programmed hold (to perform manual functions as in b, below), and monitor the data being obtained via TLM and hardwire and the analysis being made by the CDS and generated holds based on his or the subsystem engineer's analysis.

b. Test sequenced by the Subsystem OSE engineers and monitored by them and the CDS. In this mode the Test Conductor will exert supervisory control of the commands and stimulation being provided to the vehicle to ensure that it follows the agreed-upon sequence. He will monitor the TLM and hardwire data obtained and the analysis made by the CDS of that data and the monitors of the OSE control signals.

3.1.2 SUBSYSTEM TESTS

These tests will normally be performed essentially as in 3.1.1b, above. Subsystem type tests will also be required to be run during troubleshooting or during investigative types of testing. This mode will require the manual setting up of vehicle commands. These commands must be verified as being correct and appropriate by the Test Conductor.

3.1.3 SUPPORT OF LCE

The STC equipment is used to support the Launch Control Equipment (LCE) in the area of generating commands and in analyzing the Telemetry data manually and by the CDS. The Test Conductor shall coordinate and supervise these STC activities and serve as the main contact between the STC and the LCE.

3.2 DESCRIPTION

The Test Conductor's Console will consist of three bays of a sit down type console. The displays and controls contained are described as follows:

3.2.1 BAY NO. 1 - ALPHA NUMERIC DISPLAY

This display shall contain a Cathode Ray Tube (CRT), and its associated controls, on which groups of data and comments from the CDS will be presented to the Test Conductor. The display will consist of approximately 30 lines of data, each line being 40 spaces (characters) long, with any one of 60 characters being possible in each space. The data to be displayed will be divided into approximately 60 groups of 15 lines of data each (i.e. each group consist of 1/2 of a total display). The Test Conductor will be able to select any two of these groups, one to be displayed on the top half, the other on the bottom half. The data displayed will be updated once per second if new data has been received. The groups to be selected will consist of the following:

a. <u>Processed Data</u> - There will be approximately 40 groups (1/2 display) consisting of CDS processed and analyzed TLM and hardwire data. The values will be grouped by subsystem or by function as governed by a fixed CDS program. The display will be formatted by the CDS and will consist of:

   1. Values identified in English.

   2. Values in engineering units (or as desired).

   3. Limits in same units as 2, above.

   4. Out of limits indicated by a blinking asterisk preceeding the line.

The following is a typical display of this type of data.

23	POWER S.S.	LL	VAL	HL
	UNREG BUS	39.0	40.1	41.0
	REG BUS	27.5	28.3	28.5
*	INV φ1	110	109	112
	etc.			

b. CDS Data - This data will consist of one group (1/2 display) that will furnish pertinent data to the Test Conductor as governed by the CDS computer program. This would include the following:

1. Values of particular interest during existing test step and their limits. (Values selected from the 40 groups in a, above.)

2. OSE parameters being controlled.

3. Instructions or comments as appropriate.

It would be expected that the Test Conductor would select this display continuously during System Tests or test in which the CDS plays a major role. The other group to be displayed would be one of the groups in a, above.

c. Unprocessed TLM data - There will be approximately 20 groups (1/2 display) of unprocessed TLM data as obtained from the TLM decommutator. The data will be identified by a binary number (as supplied by the decommutator). This number will be used to separate the data into groups and to data values within those groups as follows:

Values within group

16 8 4 2 1 8 4 2 1 I. D. Number

Group

The display will consist of the following:

1. Group Number

2. Data Number

3. Data in Decimal Units

4. Data in Binary Bits

The following is a typical example of the resulting display.

18	1	017	0010001
	2	109	1101101
	3	067	1000011
	4	098	1100010

This display is essentially a backup display for a and b, above, and the data will, therefore, be handled independently of the CDS.

d. Summary Data - The top two lines of this display will be reserved for CDS controlled display. This will consist of displaying the group numbers that contain out-of-limits values.

3.2. BAY NO. 2 - CONTROL PANEL

The bay will contain the Test Conductor's operational controls and the displays intimately associated with them. This will include the following:

a. CDS Controls - These controls will be used to instruct the CDS as to what tests to run, when to stop for operator actions, when to resume after a program hold etc. This will include the following:

1. Stop/Pause Points Select

    (a) Test Step - (Stop after each test step - shortest test that can be performed without stopping.)

    (b) Test Sequence - (Stop after each test sequence - group of test steps that test a particular function.)

    (c) Test Program - (Stop after each Test Program - group of test sequences that test a subsystem or stage of a mission profile.)

2. Select Test Program - Display test program number.

3. Select Test Sequence - Display test sequence number.

4. Advance Test Step - Display test step number (Note: test steps within a test sequence must be performed sequentially).

5. Test Start switch

6. Resume Test switch

7. Test Hold switch - (Stop at next test step)

8. CDS status indications

    (a) Test in progress

    (b) Test finished - Go

(c) Test finished - No Go

(d) Program Hold

b. Command Mode Controls - These controls will set up the point from which Vehicle commands can be originated. This will include the following:

1. Command OSE enable

2. CVE enable

3. Subsystem OSE enable - (OSE consoles to CDS to CVE)

4. CDS sequence

5. Command Display/Send (Commands set up by 1-3 will require verification by the Test Conductor prior to transmitting the data to the vehicle. This will include the display of the command bits that establish the function to be accomplished (Note - time bits, preamble etc., not required to be displayed). After verification, the Test Conductor will enable the transmission by operating this switch.

c. Communication - The Test Conductor will have a communication center which will allow him to talk and listen to any area concerned with the STC. This will include:

1. All OSE Consoles

2. CDS Area

3. Vehicle Area

4. Environment Control Area

5. LCE

This area will also contain controls etc., for a loud speaker system in areas 1-4, above. All data into and out of this communication center will be recorded on the raw data recorder.

3.2.3 BAY NO. 3 - CONTINUOUS DISPLAY

This panel will allow the Test Conductor to obtain directly critical readings or those that summarize the general state of the system (vehicle and STC). These would include the following.

a. Continuous Meter Displays - The following analog signals would be displayed continuously:

   1. Vehicle main bus voltage

   2. Vehicle main bus current

   3. Ground cooling outlet temp.

b. Continuous Lamp Displays - The following discretes will be displayed by lights:

   1. OSE - "GO" (1/subsystem)

   2. CDS "GO"

   3. Cooling air pressure switches

   4. Etc.

c. Selected Meter Displays - Two meters and associated selector switches will allow the Test Conductor to select two out of approximately 20 analog signals for display

d. Time Displays - This panel will contain two time displays each to seconds. They will be STC station time as obtained from the Timing OSE and Vehicle time (binary code-displayed in octal) obtained from the C&S OSE. The vehicle time will follow the vehicle as to speed up-hold-set controls exercised through the C&S OSE.

e. Emergency Shut Down - This panel will contain an emergency power-off control. This switch when operated will remove power to all OSE consoles connected to the vehicle. This will not shut down the CDS - Ground Cooling or the Test Conductor's Console.

3.3 SELF TEST

The Test Conductor's Console will not require any special self-test capabilities. The only functions performed by this equipment are to transfer its incoming data into displays (which can be verified by having the interface equipment generate special formats or signals to verify this conversion) and to take switch operations and send them to other OSE (verified by operating switches and verify receipt at receiving end by observing its response to the switch operation). This equipment will, therefore, be verified by coordinating the interface equipment using special self-test procedures.

3.4 ANCILLIARY EQUIPMENT

3.4.1 HIGH SPEED LINE PRINTER

The Test Conductor's area will contain one high speed line printer that is controlled by and is considered a part of the CDS. This printer will be used to record in near real-time the results of the test sequences. This record will be used as a primary source of data for test approval and customer buy off. The format, data, etc., will be as controlled by the computer program. This printer will be positioned such that the Test Conductor will be able to read the printed matter without leaving his console.

3.4.2 CHARACTER PRINTER

The Test Conductor's area will contain one Teletype printer. This printer will be a part of the CDS. The data to be printed will be a copy of all of the data printed on the other subsystem teletype printers (into and out of the CDS). It can also be used by the computer programmer to maintain a scratch pad type interface with the Test Conductor.

The Test Conductor, by typing precise coded sequences, will be able to control the CDS to the degree programmed. This degree will probably not allow alteration of the basic test program, but could be used to alter the format of the Alphanumeric Display or other special messages, etc. The primary justification for the incorporation of this device is the flexibility it gives the Test Conductor in controlling the CDS by software techniques as opposed to hardware techniques.

This printer will be positioned such that the Test Conductor will be able to read and type without leaving his console.

4.0 INTERFACE CHARACTERISTICS

4.1 GENERAL

Figure 4-1 defines the interface between the Test Conductor's Console and the remainder of the STC.

4.2 CDS PROGRAM REQUIREMENTS

The Test Conductor's Console Alphanumeric Display places a requirement on the CDS program and memory. Paragraph 3.2.1 a and b indicate that the CDS computer must be capable of formatting 41 half display conditions. Each half display will require six bits per character x 40 characters per line x 15 lines per half display = 2600 bits per half display. For 41 half displays this requires a total storage of 41 x 2600 = 106,600 bits of buffer type storage. For an 18-bit word this implies approximately 6000 words of storage to satisfy the format function. This data is to be transferred to the Test Conductor's Console where it will be stored in the Alphanumeric Display

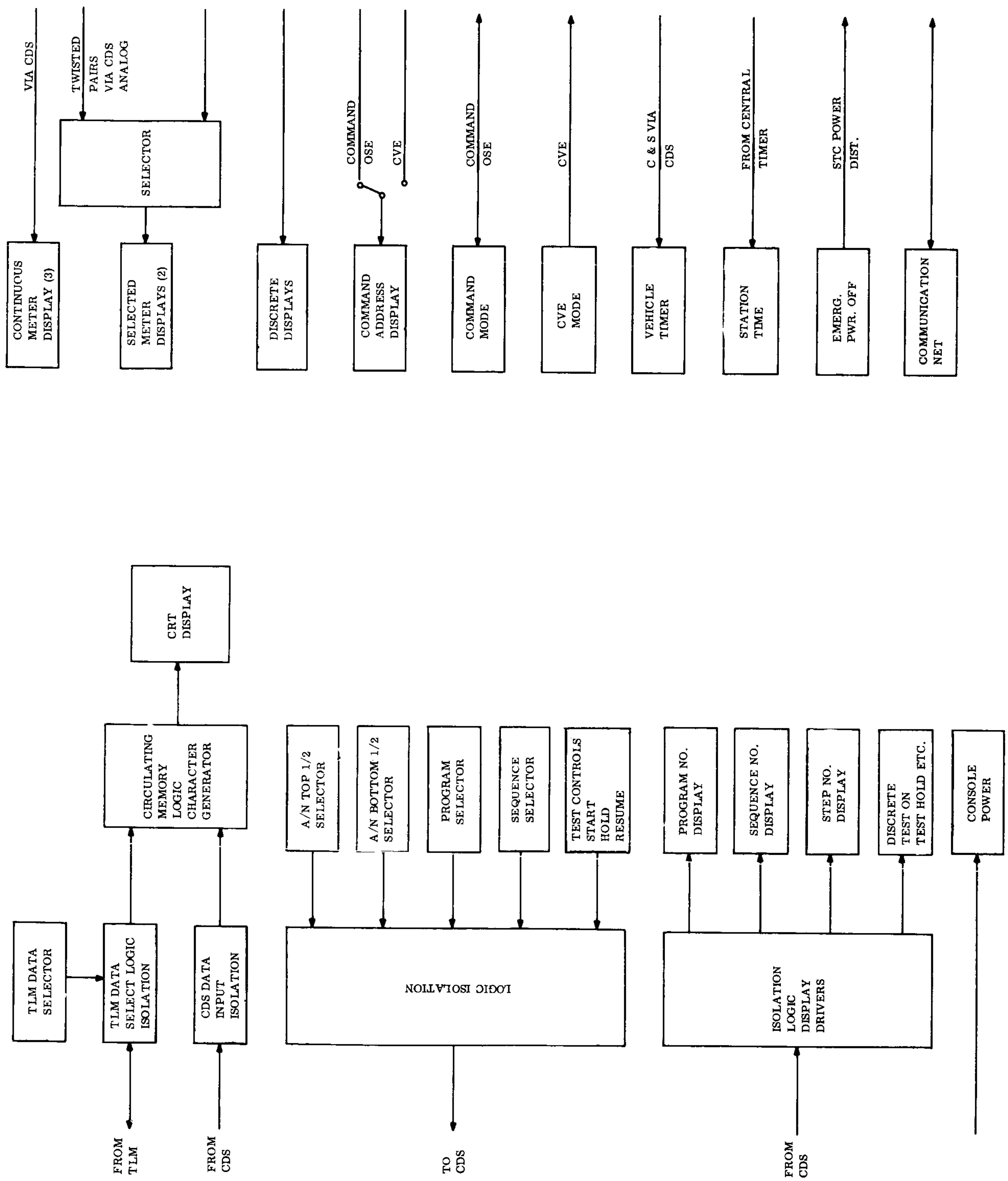


Figure 4-1. Test Conductors Console, Block Diagram

circulating memory. It will be stored in the sequence it is received starting at position x = 0 y = 0 (y being incremented for every 40 x characters). These format functions should be available with reasonable access time, i.e., two seconds after requesting a different display, it should be available.

Only the data and their limits that are to be outputted need be converted to engineering units. Only that data that has been changed since last outputted need by recalculated. An approach to programming this is that the incoming TLM and hardwire data can be grouped in the groupings to be used for the displays. Once every second the CDS can establish what two groups are requested by the Test Conductor by interrogating his selector switches. These two groups of data are then converted into the proper units and packed along with the other characters required (value identification-spacing etc.) in output buffer and shifted out. The entire display need not be updated all at once, but can be spread out over the entire one second, if desired, by the computer.

4.3 SIGNAL INTERFACE

4.3.1 ANALOG SIGNALS

The analog signals received by the Test Conductor's Console will be displayed on meters. Signal and return will be supplied by the sender.

The following three signals will be displayed continuously:

a. Main Bus Volts

b. Main Bus Amps

c. Ground Cooling Temperature

The following types of signals (20 estimated) will be displayed two at a time as selected by the Test Conductor by operation of a manual selector switch:

a. Subsystem Voltages

b. Subsystem Currents

c. Critical Subsystem points

4.3.2 DISCRETE SIGNALS (Low Frequency Digital Signals)

The discrete signals received by the Test Conductor's console will be in the form of contact closures. The "power" side of the contact will be supplied by the Test Conductor's console to the console supplying the contact. The following discretes will be received.

a. Vehicle Master Timer - 26 bits - displayed in octal code - from C&S OSE.

b. STC Time - 30 bits - displayed days - hours - minutes - seconds - from central timing.

c. Command Address - 8 bits - displayed in octal code - from CVE or command OSE when inserting manually formated commands into vehicle.

d. Subsystem Status Signals - One bit from each major OSE console - used to indicate satisfactory state. Displayed on individual lamps.

e. Cooling Pressure - Output pressure switches (in series) to indicate ground cooling flow is on. Displayed on individual lamp.

The discrete signals sent by the Test Conductor's console will be in the form of contact closures. The "power" side of the contact will be supplied by the user. The following discretes will be sent:

a. CVE - command mode (3) (CVE remote-CVE local - CVE off) and transmit enable.

b. Command OSE - command mode (2) - (Command OSE local - off) - and transmit enable.

c. CDS

1. Command via CVE (1 bit)
2. Commands from OSE consoles (1 bit)
3. Commands manual (1 bit)
4. Test mode (3) - test steps - test sequence - test program
5. Test N° select - (8 bits)
6. Test start (1 bit)
7. Test hold (1 bit)
8. Test resume (1 bit)
9. Alpha numeric top select - 8 bits
10. Alpha numeric bottom select - 8 bits

d. STC Power Distribution - Normally closed contact for emergency power off.

4.3.3 DIGITAL SIGNALS

The digital signals received by the Test Conductor's console will be transformer coupled. They will consist of the following signals'.

a. CDS

1. Alphanumeric Data - 18 bits - 6 bits/character.

2. Data - 18 bits used to drive displays - bits 1-6 used to indicate destination - bits 7-18 used for data. Displays include:

(a) Program No. (0-99)

(b) Sequence No. (0-99)

(c) Step No. (0-99)

(d) Test ready

(e) Test on

(f) Program hold

(g) Test finished go - no go

b. Telemetry Decommutation - 18 bits - bits 1-8 - data identification, bits 9-15 data (binary).

4.3.4 COMMUNICATION

The Test Conductor's console will be tied into the STC communication network as specified in paragraph 3.2.2e.

5.0 PERFORMANCE CHARACTERISTICS

The Test Conductor's console will perform the functions above. The meters used will be standard commercial grade meters with an accuracy of ± 2%. The meters that are selected to read one of many will be calibrated in 0-100% full scale units. The lights, switches, logic, etc. will all be standard components used, throughout the STC.

6.0 PHYSICAL CHARACTERISTICS

6.1 GENERAL

The Test Conductor's console will consist of three bays of an operator sit down type. The displays and controls as discussed in paragraph 3.2 will be grouped on the sloping panels. A writing surface across the entire width will be provided with drawers for storage of manuals, records, procedures, etc. being provided as a part of this shelf.

The line printer (part of the CDS) will be located such that the test conductor will be able to conviently read the printed matter while sitting at his console. The teletype (part of the CDS) will be located such that the Test Conductor will be able to conviently read the printed matter and be able to use its keyboard while sitting at his console.

6.2 HUMAN FACTORS

The Test Conductor will require the data to be presented to him in such a manner that he will be able to immediately evaluate it and issue any verbal instructions or operate his limited number of controls, as required. This capability will be achieved by applying good human factor techniques throughout the design of this console.

6.3 PHYSICAL

Weight - later

Size - later

Power - later

Connector and Location - later

7.0 SAFETY

7.1 VEHICLE SAFETY

The Test Conductor's primary task is to ensure the vehicle's safety during the system tests. His main control will be by discipline, ensuring that the subsystem engineers perform their functions following an agreed-upon procedure. The function of his console is to provide him with the data and indications to assure him that the tests are proceeding as planned. The CDS is his prime source of data and as such, his prime displays (alphanumeric and printers) are controlled by the CDS.

The Test Conductor also has control of inhibiting (by refusing to release) any manually generated vehicle command until he has seen the bit structure to be transmitted. This function provides a check on the fact that the command has been constructed properly as well as a check on the desirability of sending that command to the system. His other displays and controls are provided as gross vehicle safety type functions (i.e., ground cooling - power monitors - shut down, etc.).

7.2 PERSONNEL SAFETY

There will be no personnel safety hazzards beyond those associated with normal electronic circuits.

CII - VB260FD106

SYSTEM TEST COMPLEX

CENTRAL TIMING GENERATOR

Index

1.0 SCOPE

This document provides a functional description of the timing system required by the Voyager System Test Complex. The use of the Central Timing Generator (CTG), the Timing Isolation Unit (TIU), the Time Code Translator (TCT) and the off-line data playback mode is explained.

2.0 APPLICABLE DOCUMENTS

General Electric

VB260SR101	STC Test Objectives and Design Criteria
VB260SR102	STC Design Characteristics and Constraints
AMR (AFETR) Document 10459	Time Codes

Astro-data

6420-610-611	Time Code Translator

3.0 FUNCTIONAL DESCRIPTION

The Central Timing System (Figure 3-1) provides timing and synchronization signals for system and environmental tests at the factory, JPL and AMR. The timing signals are distributed to a TIU at each system test complex (STC) so that isolated outputs may be supplied to each OSE S/S console which uses the timing signals. The TIU's provide either parallel or serial outputs as defined in Figures 3-2 and 3-3.

Often tape recordings of test data are played back for a more thorough analysis. To allow for reduction of the recorded time code, the TIU's are compatible with the output of the time code translator (TCT) associated with the playback tape recorder. By changing the TIU from cables coming from the CTG to cables from the TCT, the timing on the complex will be slaved to the TCT, which derives its time reference from the recorded tape. For a detailed description of the TCT refer to Astro-data bulletin (204-607) entitled Model 6204 Time Code Translator.

4.0 INTERFACE DESCRIPTION

The CTG will provide the following outputs (refer to Figure 4-1):

Code	Description	To
Am	NASA 36 bit serial code modulating 1000 cps carrier	TIU
A	NASA 36 bit serial code 100 pps dc level shift	TIU
B	NASA 28 bit serial code 2 pps dc level shift	TIU

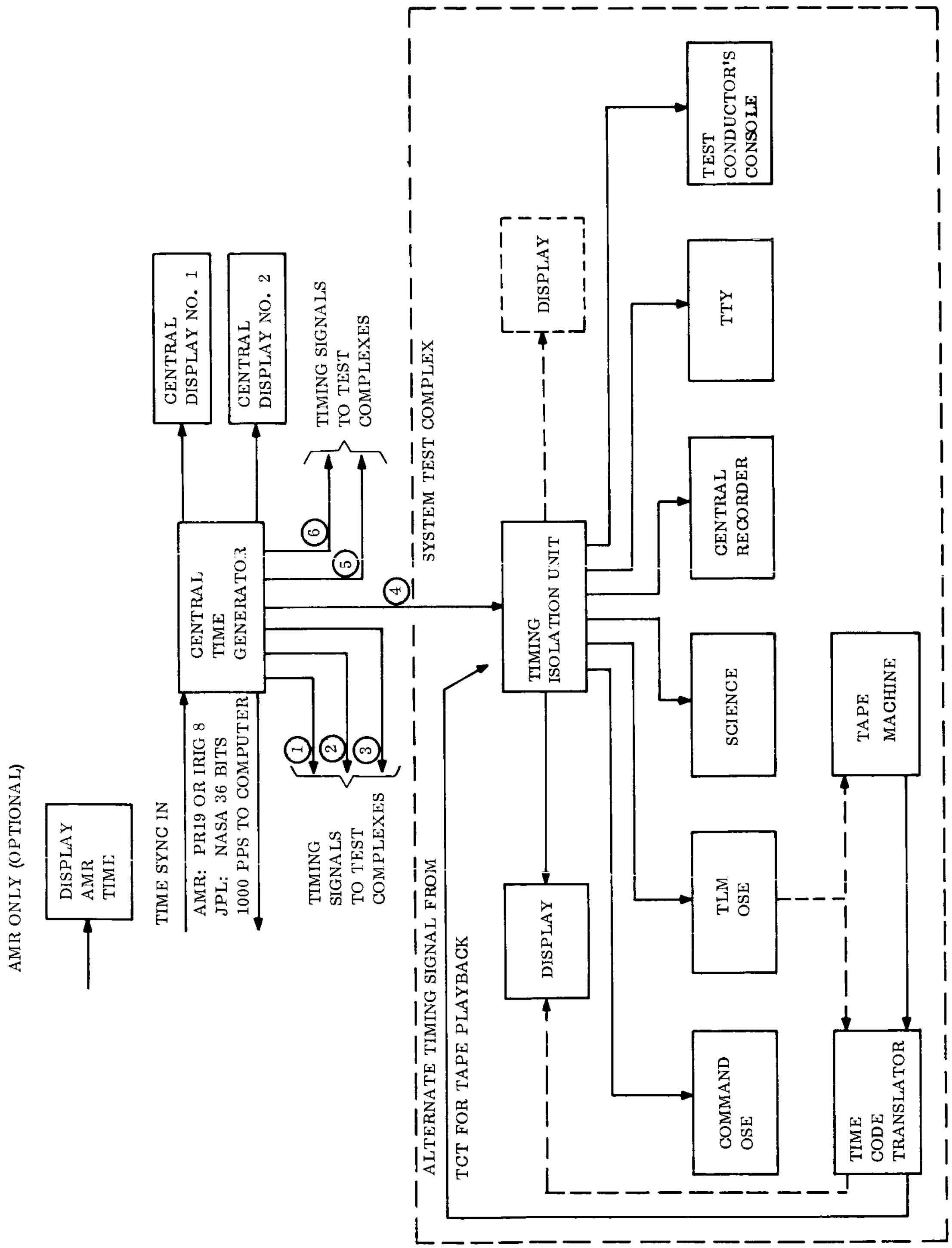


Figure 3-1. Overall Central Timing System

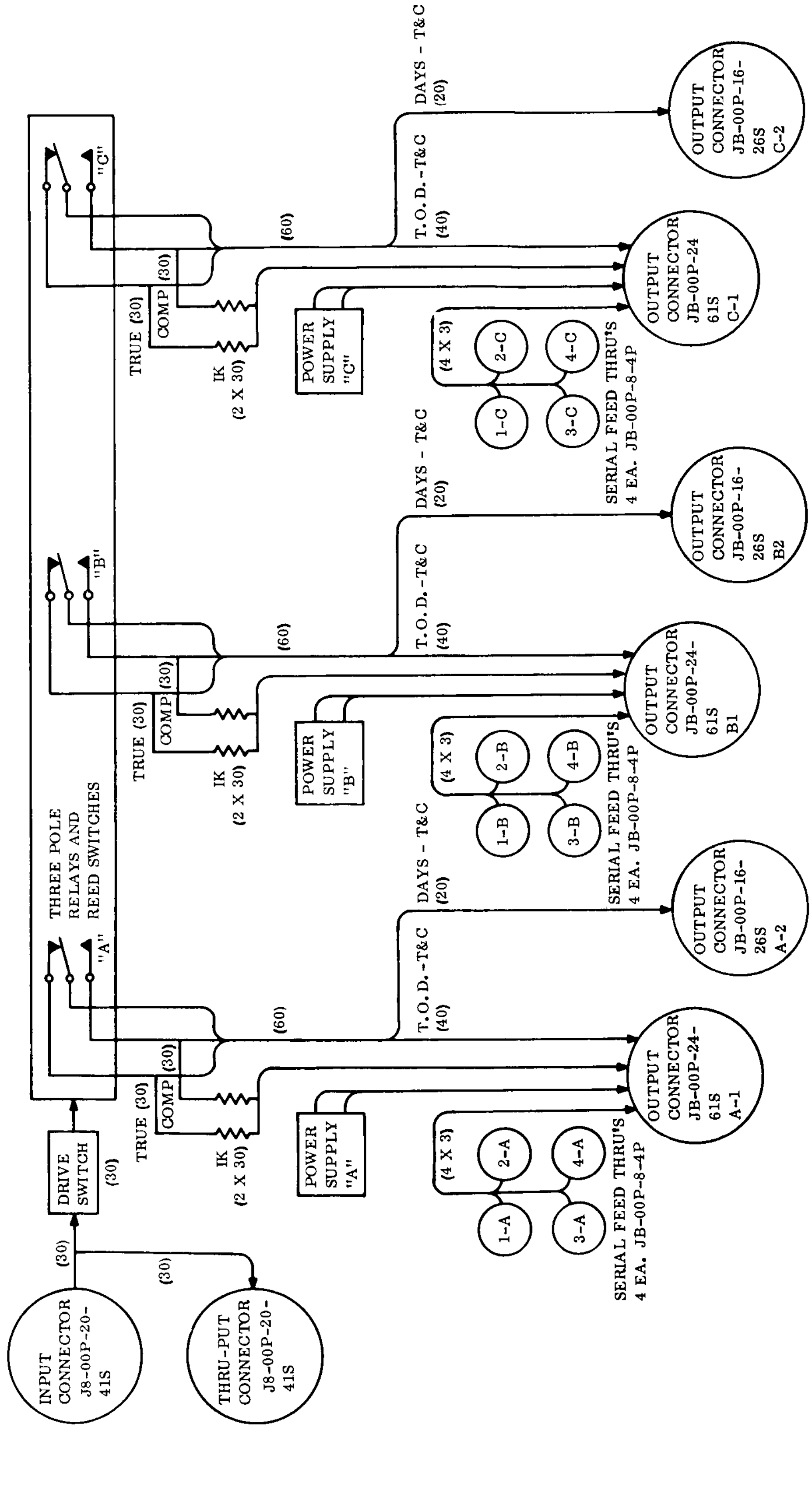
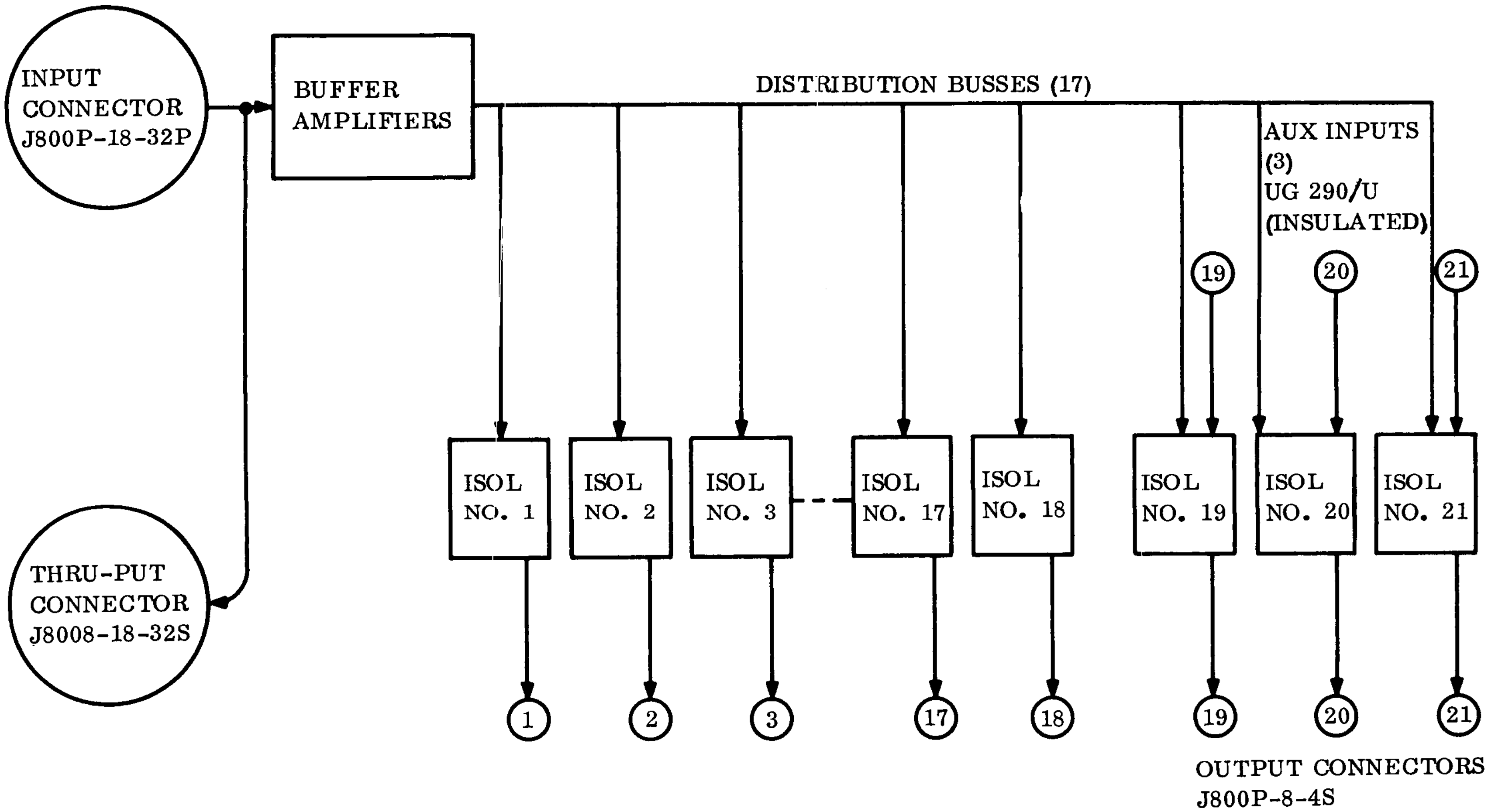


Figure 3-2. Timing Isolation Unit Interface — Parallel Time Code Isolator



NOTE:
ISOLATOR LOCATIONS NO. 1 THRU NO. 18 WILL ACCEPT PULSE FAST OR SLOW ISOLATORS.
LOCATIONS NO. 19, 20 & 21 WILL ACCEPT STANDARD OR SPECIAL PURPOSE ISOLATORS.

Figure 3-3. Timing Isolation Unit Interface — Serial Time Code Isolator

Code	Description	To
Bm	NASA 28 bit serial modulating 100 pps carrier	TIU
C	6.25 kc	TIU
Cp	Composite AM + BM + C mixed 3:3:1	TIU
E	1000 cps Sine Wave	TIU
F	100 pps 10% duty cycle	TIU
G	10 pps 10% duty cycle	TIU
H	1 pps 10% duty cycle	TIU
K	1 pp10s 20% duty cycle	TIU
M	1 ppm 33% duty cycle	TIU
N	1 pp10m 20% duty cycle	TIU
BCD	Parallel time code, days, hours, minutes, seconds	TIU
I	1000 pps 10% duty	Spare
R	10 k pps	Spare
S	100 k pps	Spare
M	1pp minute	SCF Computer
P	1000 pps 40% duty cycle	SCF Computer

Two outputs for central time displays will be provided by each CTG. Inputs to the CTG will be the following:

a. NASA 36-bit time sync — At JPL Lab.

b. AMR one-sec PR19 — At AMR

c. AMR one-RIG - B — At AMR

d. 120 Volts ± 10% 60-cycle, single phase, 180 watts

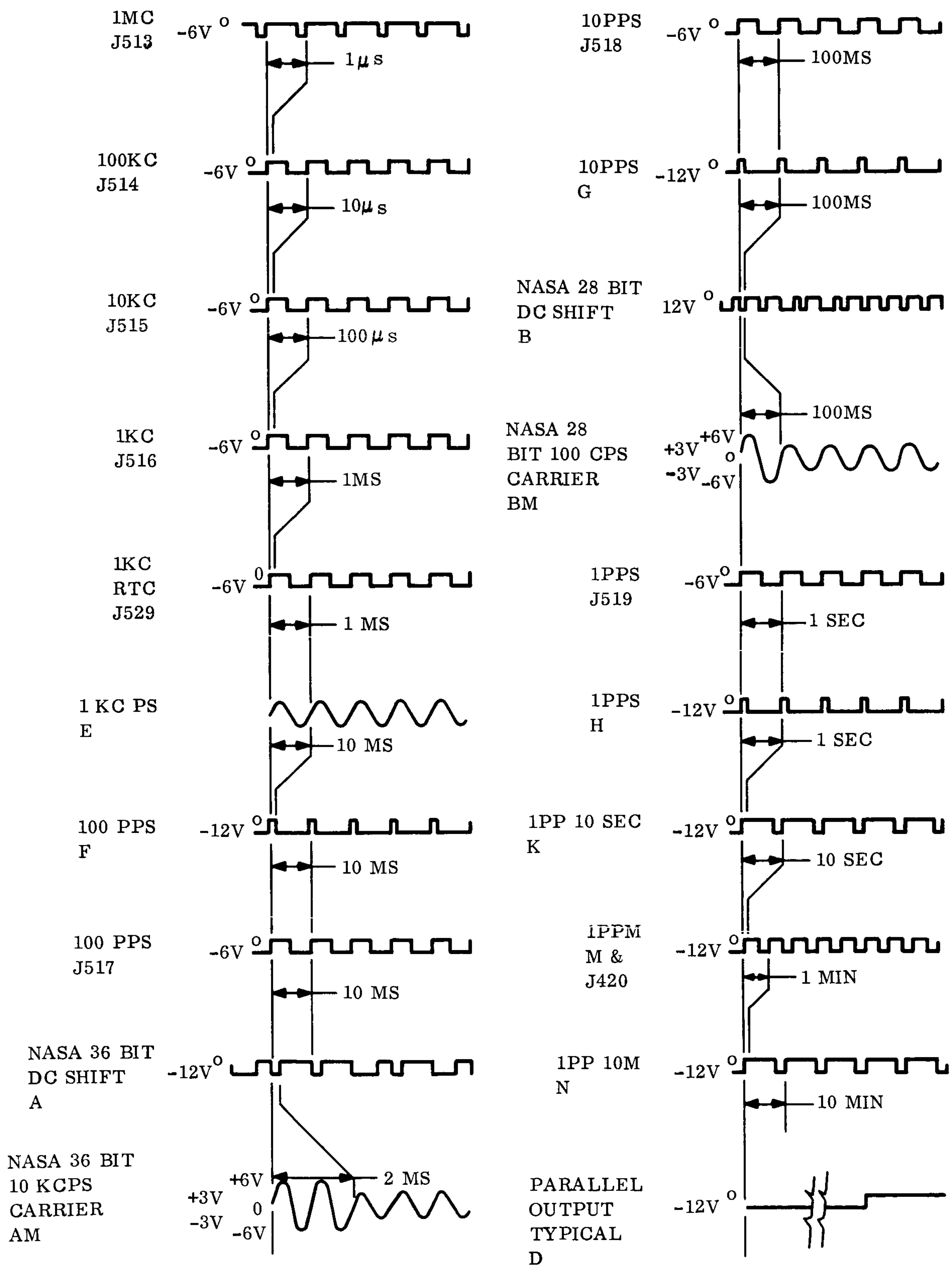


Figure 4-1. Time Code Generator Output Waveforms

The TIU inputs will be the following:

Code	Description
Am	NASA 36-bit modulating 1000 H Z
A	NASA 36-bit 100-pps d-c level shift
B	NASA 28-bit 2-pps d-c level shift
Bm	NASA 28-bit modulating 100 H Z
C	6.25 KC
Cp	Composite AM + BM + C mixed 3:3:1
E	1000 H Z sine wave
F	100-pps 10% duty cycle
G	10-pps 10% duty cycle
H	1-pps 10% duty cycle
K	1-pps 20% duty cycle
M	1-pps 10M 20% duty cycle
BCD	Parallel time code, days, hours, minutes, seconds.
Power	120 Volts ± 10%, 60-cycle, single phase, 300-Watt

Outputs of the TIU will interface with the following listed OSE S/S consoles, and defined by Figures 3-2 and 3-3.

a. Science OSE

b. Pyro OSE

c. Central records

d. Computer Data System

e. TCM OSE

f. Test Conductor's Console

5.0 PERFORMANCE PARAMETERS

The TIU's provide an isolated interface between the CTG and the users of the complex. Every effort to minimize capacitance coupling between system grounds is required.

In system tests the timing signals provide a uniform time reference between subsystems and the computer data system. However, the timing signals may be used by any subsystem console at any time without dependence upon other OSE consoles.

TIU power requirements:

120 Volts ± 10%, 60-cycle ± 2%, 300 Watt, single phase.

The TIU output is convertible from voltages to relay closures. The output voltage levels are as follows:

Output (Voltage)	Output Voltage	Output Imp.
Am	6 VPP	100 Ω
A	0-12V	1000 Ω
B	0-12V	1000 Ω
Bb	6 VPP	100 Ω
C	6 VPP	100 Ω
CP	6 VPP	100 Ω
E	6 VPP	1000 Ω
F	6-12V	Buffer Ω
H	0-12V	1000 Ω
G	0-12V	Buffer Ω
K	0-12V	1000 Ω
M	0-12V	1000 Ω
N	0-12V	1000 Ω
D	0-12V	1000 Ω
BCD		

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Central Timing System will consist of a CTG (Astro-data 6420) and a maximum of six TIU's. This is identical to the equipment which was used on the Mariner C program; it could be used on the Voyager program with no modification. The CTG must be centrally located in an air conditioned area.

7.0 SAFETY

Frame grounds will be installed on all chassis, and fuses must be provided on all incoming power leads for protection of operating personnel.

CII - VB260FD107

SYSTEM TEST COMPLEX

CENTRAL RECORDER

Index

1.0 SCOPE

This document provides a functional description of the Central Recorder, a part of the Voyager System Test Complex. The use of the Central Recorder to support Voyager system testing is described, and a description of its use in supporting subsystem test is also included.

2.0 APPLICABLE DOCUMENTS

GE

VB260SR101	1971 Voyager STC Test Objectives and Design Criteria
VB260SR102	1971 Voyager STC Design Characteristics and Constraints

3.0 FUNCTIONAL DESCRIPTION

The Central Recorder (CR) will be capable of monitoring a maximum of 50 analog functions simultaneously. The data will come from the vehicle via the subsystem OSE consoles, and approximately 300 functions will be available for monitoring. To evaluate the data in a logical manner a patch panel will be provided for each test so that individual subsystems can use the recorder to maximum advantage. For example, if a subsystem test is being conducted on the attitude control S/S, the appropriate patch panel will be installed and up to 50 functions which are particular to the test could be displayed and recorded for detailed analyses. This capability is extremely flexible because all functions are available to the CR and only a repatch is necessary to monitor any data point.

Variable gain amplifiers and calibration controls will be provided so that proper scale factors, source isolation and impedance matching can be accomplished.

Figure 3-1 is a functional block diagram of the Central Recorder.

4.0 INTERFACE

a. Since the test conductor has the capability to patch-in any function for recording, all analog vehicle data will be cabled to the CR. Functions available for monitoring by the CR are as follows:

SUBSYSTEM SOURCE	NUMBER OF FUNCTIONS
Power S/S	30
Science S/S	70
Capsule	Undetermined
Video Tape Rec	20
Radio S/S	20
Radio Relay S/S	15
TLM S/S	30

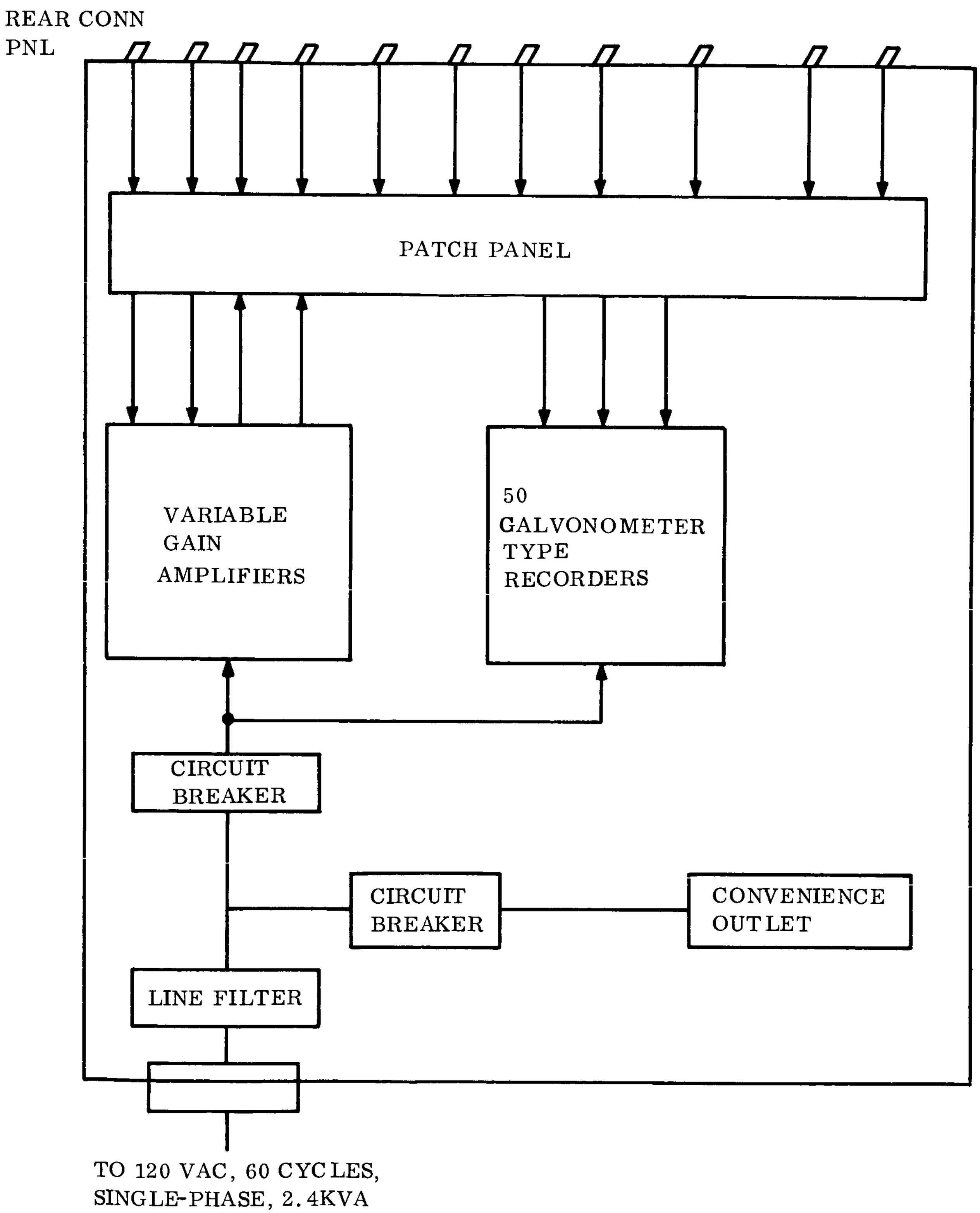


Figure 3-1. Central Recorder, Block Diagram

SUBSYSTEM SOURCE	NUMBER OF FUNCTIONS
Command SS	30
Pyro SS	80
Attitude Control	40
Controller and Sequencer	5
Propulsion	5
Control Timing	36 Bit serial time code

b. Miscellaneous Interfaces are as follows:

1. Power Distribution Console — 120 VAC, 60-cycle, single-phase, 2,4 KVA

2. Test Conductor's Console — Intercomm. network.

5.0 PERFORMANCE

The CR will be a 50-channel direct-write oscillograph that uses 12-inch wide paper with a maximum capacity of 350 feet. The record will be capable of being permanently fixed if desired.

The majority of the galvanometers will have an undamped natural frequency of 150 cps; the flat response (± 5%) will be 0-70 cps; the D-C sensitivity will be 0.021 ma/in. or 3.67 mv/in. with a nominal D-C coil resistance of 30 ohms.

Galvanometers with a higher frequency response will be required for some applications; these will have an undamped natural frequency of 1500 cps; the flat response (± 5%) will be 900 cps; the D-C sensitivity will be 10.7 ma/in. or 364 mv/in. with a nominal coil resistance of 17 ohms. Galvanometers with a higher frequency response can be obtained if required.

Forty-eight galvanometer controls will be provided which will handle from 150V down to the minimum voltage required for a particular galvanometer deflection. Additional wide-band (0-10kc) differential D-C amplifiers will be provided. They will have very good input-to-output and input/output-to-ground isolation and will provide 10 volts at 10 ma for voltage recording or 10 volts at 100 ma for the current necessary to drive the galvanometer.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The central recorder that was used on the Mariner C program could be used on the Voyager program. The addition of a patch panel and possibly a change in connector types would be required to make it compatible with the recommended Voyager System Test Complex.

The CR will be contained in one standard electronic equipment rack, and will weigh approximately 400 pounds. It will require power of 120 VAC, 60-cycle, single-phase at 2.4 KVA and will consist of the following items:

a. Midwestern Model 607F direct-write oscillograph or equivalent.

b. Approximately 40 Midwestern type 102-150 galvanometers.

c. 10 Midwestern type 102-1500 galvanometers.

d. Four 12-channel galvanometer control panels; Dynamic Instrumentation Model 6065 or equivalent.

e. Six wide-band differential amplifiers; dynamic Instrumentation Model 122 or equivalent.

f. A 480-point patch panel; AMP model P480D with removal programmable board Model 595534-2 or equivalent.

The incoming signal leads will be two-conductor twisted pair shielded. The shield will be connected together and grounded at the CR end only.

7.0 SAFETY PROVISIONS

No special provisions are required for the protection of personnel or the spacecraft. Normal good practice is required in patching to preclude damaging galvanometers with overloads and loss of data.

CII - VB260FD108

SYSTEM TEST COMPLEX AND LAUNCH COMPLEX EQUIPMENT

UMBILICAL SIMULATOR

Index

1.0 SCOPE

This document is a functional description of the simulator required to test both the System Test Complex and the Launch Complex Equipment. The simulator is interfaced with the OSE as a precautionary measure to verify the OSE condition of readiness prior to connecting to the Spacecraft umbilical.

2.0 APPLICABLE DOCUMENTS

VB260SR101 1971 Voyager STC	Test Objectives and Design Criteria
VB260SR102 1971 Voyager STC	Design Characteristics and Constraint
VB280SR101 1971 Voyager LCE	Test Objectives and Design Criteria
VB280SR102 1971 Voyager LCE	Design Characteristics and Constraint

3.0 FUNCTIONAL DESCRIPTION

The principle function of the Spacecraft Umbilical Simulator is to provide an approximate simulation of the loads, current, voltages etc. of a nominal spacecraft, to the Operational Support Equipment. This is a test or precaution which equally applicable to both the System Test Complex and to the Launch Complex Equipment.

The simulation of the spacecraft to the OSE is at the interface of the Voyager/Centaur in flight disconnect or the Voyager Umbilical. This is a more critical interface than the spacecraft Direct Access Test Point because:

a. The umbilical wires, especially those used for control purposes, do not necessarily have the built-in isolation that is characteristic of direct access test points.

b. The interfaces between spacecraft direct access test points and OSE will have had a greater history of pre-delivery validation than the umbilical to OSE interface because of flight subsystem testing operations. Any potential damages to the spacecraft are more readily fixed if incurred through the electronic bay direct access test points. Spacecraft simulation, insofar as direct access test points are concerned, is normally a self test feature of the OSE in the STC.

c. Potential damages which faulty LCE could inflict upon the Spacecraft, through the umbilical, are more prejudicial to the launch window than casualties which might occur in the SCF.

The simulation should be directional; spacecraft signals are generated in an approximate manner at the simulator and delivered to the OSE for analysis and evaluation, while OSE signals are received and displayed at the simulator. Figure 3-1 illustrates the nature of this simulation.

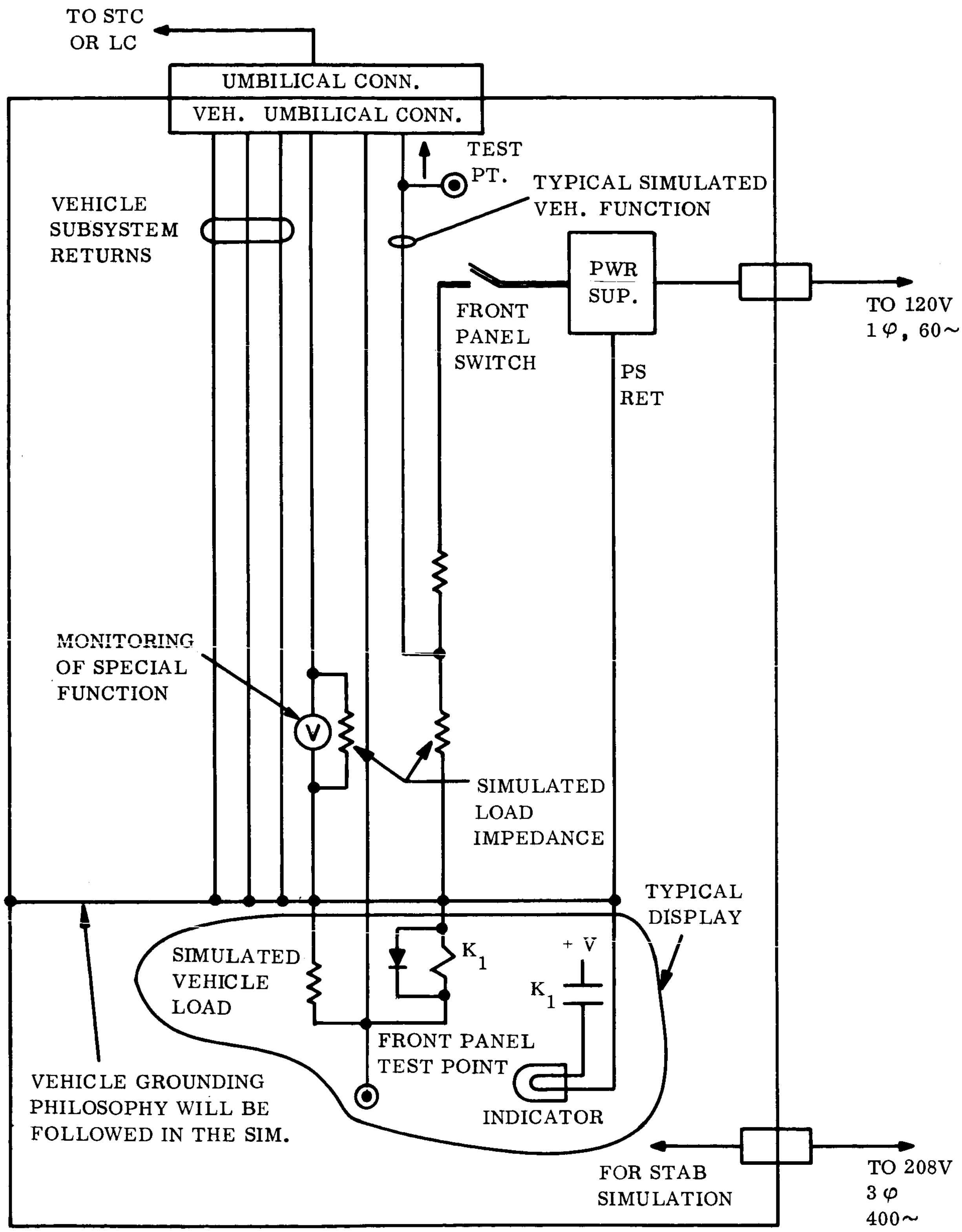


Figure 3-1. S/C Simulator

Cable validation, resolution of incompatabilities, and recognition of degraded transmission are objectives of the Spacecraft Umbilical Simulator. For these reasons, especially the recognition of degradation (a noisy connection as opposed to an open or shorted wire), the functions generated by the Umbilical Simulator are shown in Figure 3-1 as adjustable.

4.0 INTERFACE

In the STC the S/C simulator will mate to the OSE cable which contains a connector similar to inflight disconnect which is mounted on the Centaur. These signals will then be routed to the umbilical "J" Box, from which they will be distributed to the OSE S/S console.

Umbilical signals simulated by Simulator/Interfacing OSE Subsystem Console are given in Table 3-1.

Table 3-1. Simulated Umbilical Signals

RADIO SUBSYSTEM	
1. (Coax) High Gain Antenna Probe	4. (Coax) Secondary Low Gain Antenna Probe
2. (Coax) Medium Gain Antenna Probe	5. (Coax) Precision Link
3. (Coax) Primary Low Gain Antenna Probe	
RELAY RADIO SUBSYSTEM	
1. (Coax) Antenna Probe	
COMMAND SUBSYSTEM	
None	
POWER SUBSYSTEM	
1. Array/Battery Bus Voltage	7. External Power Return
2. Array Enable Switch SW-1 Monitor	8. Battery No. 1 Voltage
3. Turn-on Enable Switch, SW-1	9. Battery No. 2 Voltage
4. Turn-off Enable Switch, SW-1	10. Battery No. 3 Voltage
5. Enable Switch, SW-1, Return	11. Battery No. 1 Temperature
6. External Power, 44-55 VDC, 15 Amps	12. Battery No. 2 Temperature

Table 3-1. Simulated Umbilical Signals (Continued)

13. Battery No. 3 Temperature	17. 400 cps, Phase 3 Voltage
14. Battery Sensor Temperature Return	18. 2.4 KC Voltage
15. 400 cps, Phase 1 Voltage	19. 2.4 KC Voltage Return
16. 400 cps, Phase 2 Voltage	20. Raw Battery Bus Voltage
CONTROLLER AND SEQUENCER	
1. Alert Signal	6. Inhibit Master Timer
2. Sync Signal	7. Update Master Timer
3. Command Information	8. Clean Timer
4. Engine Burn Enable State	9. Inhibit Power Supply No. 1
5. Speed-Up Timing	10. Inhibit Power Supply No. 2
DATA ENCODER	
1. Modulated Subcarrier A	13. Telemetry Mode 5 Select
2. Modulated Subcarrier B	14. UHF Transmitter On/Off
3. Modulated Subcarrier C	15. UHF Power Amplifier On/Off
4. Word Sync	16. VHF Transmitter On/Off
5. Bit Sync	17. Sequencer Enable
6. Frame Sync	18. Sequencer Rate Select
7. Mode II Command	19. Ground Telemetry Clocks
8. Data Encoder Return	20. External/Internal Telemetry
9. Recorders to Launch Mode	21. Telemetry On/Off
10. Telemetry Mode 3 Select	22. Tape Recorder On/Off
11. Telemetry Mode 4 Select	23. Tape Recorder Record/Play-back Select
12. Telemetry Mode 5 Select	24. Telemetry Bit Rate Output

Table 3-1. Simulated Umbilical Signals (Continued)

25. Telemetry Frame Rate Output	28. Pyrotechnic Continuity
26. Telemetry Encoder Output	29. Telemetry Mode Monitor
27. Sensor Power Supply Monitor	30. Tape Recorder Reset
GUIDANCE AND CONTROL	
1. Gyro Heater Power	5. Common Return
2. Gyro Rebalance Amplifier 1	6. Shield Return
3. Gyro Rebalance Amplifier 2	7. Approach Guidance Update 1
4. Gyro Rebalance Amplifier 3	8. Approach Guidance Update 2
PYROTECHNIC SUBSYSTEM	
1. Continuity	3. Arm Signal 1
2. Continuity Return	4. Arm Signal 2
DATA STORAGE SUBSYSTEM	
None	
SCIENCE	
Undetermined. Ten umbilical lines are assumed for this subsystem.	
CAPSULE	
1. Ground Power Main Bus	6. Checkout Sequence Signal
2. Ground Power Main Bus Monitor	7. Spacecraft/Capsule External Clock Select
3. Internal/External Power Transfer	8. Telemetry Mode 1 Select
4. Internal/External Power Monitor	9. Telemetry Mode 2 Select
5. Barrier Pressure Monitor	

5.0 PERFORMANCE CHARACTERISTICS

The S/C simulator is primarily a quantitative instrument for determining if the cabling between the S/C inflight disconnect and the S/S OSE exists and performs as defined on the drawings. For most functions lamp indication will be the only response to an OSE S/S stimuli.

There will be some functions that will require monitoring with a meter which will have an accuracy of ± 1%; to do this appropriate meters will be built into the simulator so they can be patched or switched, for multiple purpose monitoring. Normally the meters will be used to calibrate the S/S OSE through the interconnecting cable.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Voyager simulator will use the Mariner C design philosophy as a guideline. It will be a self-contained portable unit weighing approximately 40 lbs; and requiring power of 120VAC, 60 cycle, single phase, 200W; and 208V, 400 cycle, three-phase, 30W.

It will be a weatherproof suitcase type box containing the following:

a. Control Panel
    1. Indicator light
    2. Meters
    3. Fuse
    4. Test Points
    5. Switches
b. Relays
c. Power Supply

7.0 SAFETY

The unit will be fused and grounded through the power cable for protection of personnel and equipment. Caution should be observed when working on the unit since potentially dangerous voltages will exist within the unit.

CII - VB263FD101

SYSTEM TEST COMPLEX

RADIO SUBSYSTEM OSE

Index

1.0 SCOPE

This document is a functional description of the Voyager Radio Subsystem Operational Support Equipment (OSE) which forms a part of the System Test Complex (STC) and is used to test and monitor the S-Band Radio Subsystem during system and subsystem tests.

2.0 APPLICABLE DOCUMENTS

Reference to the following documents is required for completeness in this document:

General Electric/Motorola Document

VB263FD103	Command Subsystem OSE; Functional Description
VB260SR102	System Test Complex Design Characteristics and Restraints
VB260SR101	System Test Complex Test Objectives and Design Criteria

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The Voyager Radio Subsystem OSE is used to test the subsystem from initial subsystem tests through STC tests and supports testing at AFETR.

3.2 BLOCK DIAGRAM

A block diagram of the Voyager Radio Subsystem OSE is shown in Figure 3-1.

3.3 REQUIRED FUNCTIONS

The Voyager Spacecraft Radio Subsystem OSE is capable of performing the following eleven functions:

a. Provide six stable transmitter signals for checking the three spacecraft (S/C) Receiver performances.

b. Provide a phase coherent receiver for checking the S/C transmitters characteristics.

c. Provide for measuring, monitoring, and recording of S/C r-f powers and frequencies, S/C receiver and transmitter functions, and S/C transformer rectifier (TR) and DC-DC converter voltages and currents; monitor the S/C radio subsystem operating mode and reaction to S/C commands and OSE simulated commands.

d. Provides the S/C with a pseudo-random ranging code via the r-f link.

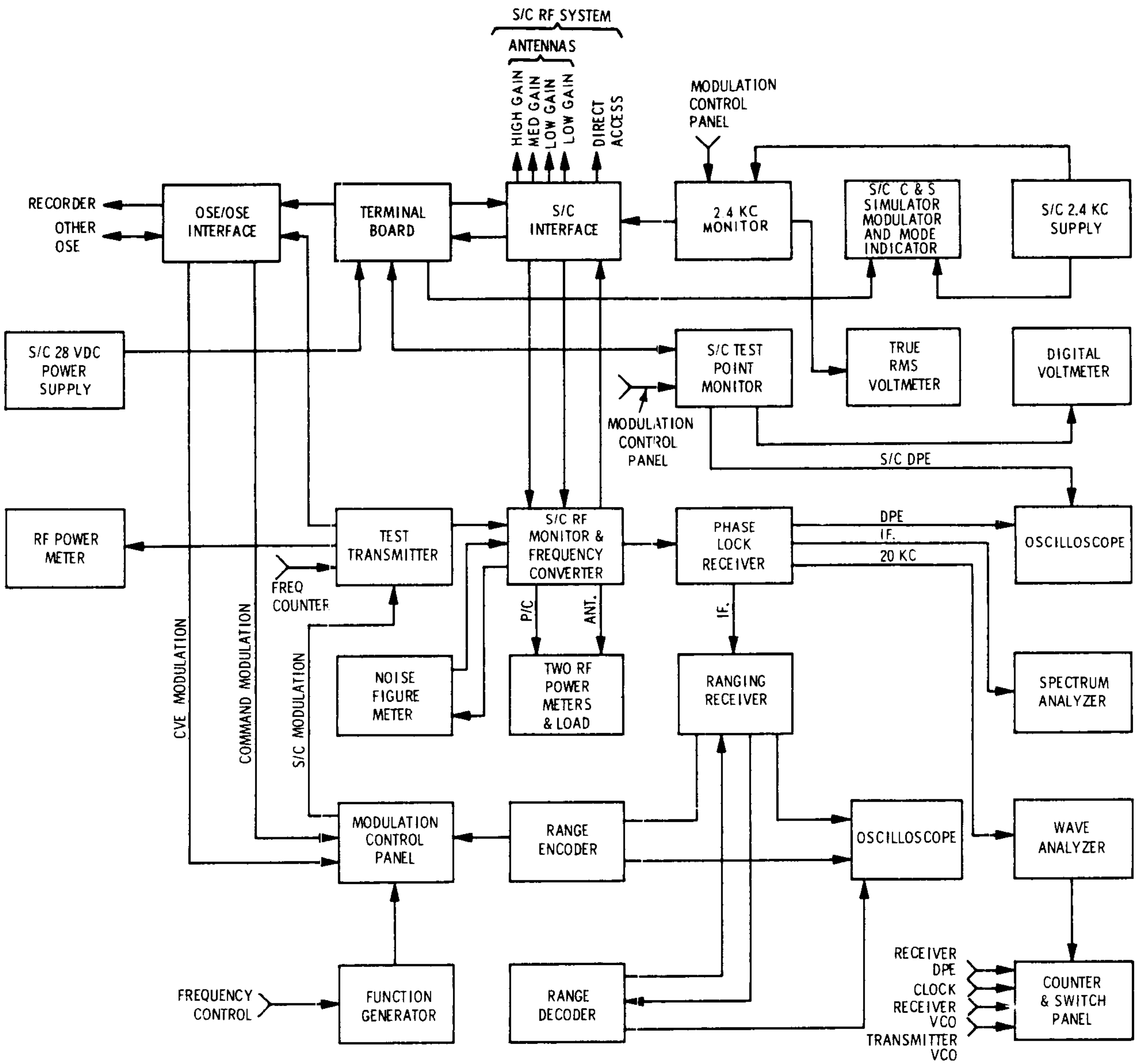


Figure 3-1. Voyager Radio Subsystem OSE

e. Demodulate the ranging code from the S/C turnaround ranging subsystem and examine the velocity and range determining characteristics.

f. Provide an alternate source of power to the DC-DC converter and TR's in the S/C r-f package when S/C power is not available and provide loads for the S/C power simultaneously.

g. Monitor and record operating times of the S/C radio subsystem.

h. Provide the necessary OSE/OSE interfaces.

i. Provide power and signal isolations as described in VB260SR101 and VB260SR102.

j. Provide a self test capability independent of external testing.

k. Receive the spacecraft RF telemetry signal and provide a demodulated telemetry signal to the Data Encoder OSE.

3.4 GENERAL DESCRIPTION

The following paragraphs contain general descriptions of the units contained within the OSE. The paragraphs are arranged in the order the equipment appears in the rack enclosures. Refer to Figure 3-2 for a drawing showing the four basic racks with equipment location. All four racks are used both in system and subsystem tests. The case and cable storage console and the S/C power supply rack are used exclusively for subsystem tests.

3.4.1 OSE RACK I

OSE Rack I contains the following equipment:

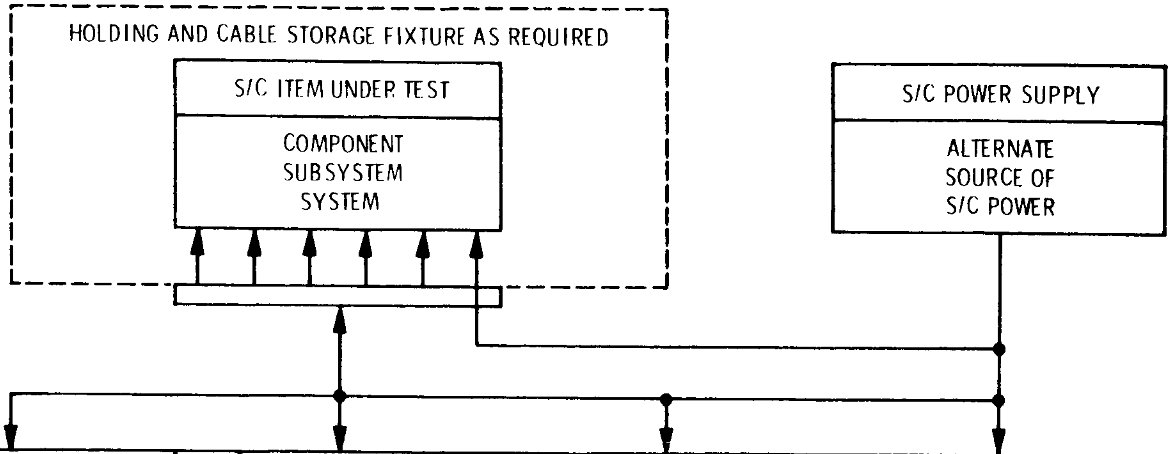
- Patch and Breaker Panel — Master circuit breaker for OSE Rack I, a running time meter for recording operating time of Rack I, a 115 vac convenience outlet, and connectors for interconnections between OSE Rack I and II.

- Range Code Control Panel — Functions as the control and monitor panel for the PN Code Generators.

- Oscilloscope — Monitors outputs of Ranging Receiver and PN Code Generators.

- Spectrum Analyzer — Monitors frequency spectrum.

- Wave Analyzer — Measures the relative amplitude of the Test Receiver i-f spectral components.



NO.	RACK I	NO.	RACK II	NO.	RACK III	NO.	RACK IV
1A1	PATCH & BREAKER PANEL	2A1	PATCH & BREAKER PANEL	3A1	PATCH & BREAKER PANEL	4A1	PATCH PANEL & S/C RUNNING TIME
1A2	RANGE CODE CONTROL PANEL	2A2	BLANK PANEL	3A2	ISOLATION AMPLIFIER UNIT	4A2	FUNCTION GENERATOR
1A3	4 TRACE OSCILLOSCOPE	2A3	RANGING RECEIVER	3A3	MODE INDICATOR	4A3	DC MICROVOLTMETER
1A4	SPECTRUM ANALYZER	2A4	BLANK PANEL	3A4	DIGITAL VOLTMETER	4A4	S/C RF POWER METER ASSEMBLY
1A5	WAVE ANALYSIS	2A5	TEST RECEIVER	3A5	S/C TEST POINT MONITOR PANEL	4A5A1	POWER METER
1A6	MOPS	2A6	DUAL OSCILLOSCOPE	3A6	S/C RF MONITOR - RF CONVERTER	4A5A2	POWER METER CALIBRATOR
1A7	NOISE FIGURE METER	2A7	AC VTVM	3A7	ELECTRONIC COUNTER	4A6	TEST TRANSMITTER
1A12	CONNECTOR PANEL •	2A8	POWER SUPPLY (±300 VDC)	3A8	COUNTER SWITCH PANEL	4A7	MODULATION CONTROL PANEL
		2A9	POWER SUPPLY (±15 VDC)	3A9	S-BAND SIGNAL GENERATOR	4A8	AUDIO OSCILLATOR
		2A10	POWER SUPPLY (+150 VDC)	3A10	S/C POWER SUPPLY (28 VDC)	4A9	2.4 KC S/C MONITOR
		2A11	POWER SUPPLY (+7 VDC)	3A11	2.4 KC S/C POWER SUPPLY	4A10	TRUE RMS VOLTMETER
		2A12	CONNECTOR PANEL •	3A12	TERMINAL BOARD ASSEMBLY •	4A11	POWER SUPPLY (±15 VDC)
		2A13	DECODE GENERATOR •	3A13	S/C CONNECTOR PANEL*	4A12	OSE/OSE INTERFACE PANEL •
		2A14	ENCODE GENERATOR •				
RADIO SUBSYSTEM OSE							

• LOCATED IN REAR

Figure 3-2. Equipment Location, Radio Subsystem OSE

- System Tests Intercommunication Unit — Provides voice communication within the System Test Complex.

- Noise Figure Meter — Provides for automatic noise figure measurements.

3.4.2 OSE RACK II

OSE Rack II houses the following equipments:

- Patch and Breaker Panel — Provides a master circuit breaker for Rack II, connectors for interconnection between racks, and a running time meter for recording Rack II running time.

- Ranging Receiver — Provides for testing transponder turnaround ranging.

- Test Receiver — Phase coherent, double conversion superhetrodyne receiver for testing the S/C transmitter.

- Oscilloscope — Monitors the DPE outputs of the S/C and test receivers.

- Vacuum Tube Voltmeter — Measures various OSE and/or S/C signal and power rms levels.

- Power Supply — Provides ± 300 vdc to the Test Receiver.

- Power Supply — Provides regulated ± 15 vdc to assemblies in OSE Racks I and II.

- Power Supply — Provides + 150 vdc and 6.3 vac to the Test Receiver.

- Power Supply — Provides regulated 7 vdc to the Test Receiver.

- PN Code Generators — Provides pseudo-random range codes for modulating the test transmitter and demodulating the code received by the Test Receiver.

3.4.3 OSE RACK III

The following equipment is located in OSE Rack III

- Patch and Breaker Panel — Contains master circuit breakers for OSE Racks III and IV, connectors for interconnections between racks, a running time meter for recording Racks III and IV operating times, and a 115 vac convenience outlet.

- Isolation Amplifier — Provides isolation of the spacecraft and OSE functions to prevent loading by other subsystems.

- Mode Indicator Panel — Used to simulate S/C Commands and C and S control and indicate the operating mode of the S/C radio subsystem.

- Digital Voltmeter — Measures various S/C and OSE voltage levels and used in conjunction with the temperature sensitive devices.

- S/C Test Point Monitor — Used to select specific test points in the S/C for making various measurements using OSE test equipment.

- S/C RF Monitor and RF Converter — Monitors received and transmitted signal levels and provides for distributing r-f power to and from the S/C and frequency converter. The frequency converter coherently translates the test transmitter output frequency to the test receiver input frequency.

- Frequency Counter — Monitors OSE and S/C operating frequencies.

- Counter Switch Panel - Functions as the input selector for the frequency counter.

- S-Band Signal Generator — Provides for measuring receiver spurious response and image rejection.

- 25-50 VDC Power to S/C — Supplies 28 vdc nominal to the S/C DC-DC Converter in the radio subsystem.

- 2.4 KC S/C Power Supply — Supplies 50 vrms at 2.4 kc to the S/C radio subsystem TR units.

- Terminal Board Assembly — Provides primary interconnection for OSE-to-OSE.

- S/C Connector Panel — Used for all interconnections between OSE and S/C or cable storage console.

3.4.4 OSE RACK IV

The following equipment is contained within OSE Rack IV:

- Breaker, Patch and S/C Running Time Panel — Contains a circuit breaker and running time meter for recording S/C radio subsystem running times.

- Function Generator — Used to generate output square waves, triangular waves and sine waves to modulate the OSE and S/C transmitters and other uses.

- DC Microvoltmeter — Measures various OSE and/or S/C signal levels.

- S/C RF Power Meter — Measures the RF output power from the S/C.

- Transmitter r-f Power Meter Panel — Monitors the r-f output of the test transmitter.

- Test Transmitter — Used to provide the S/C transponder receivers with stable S-band signals.

- Modulation Control Panel — Used to control the test transmitter phase modulator input. It provides for selecting different input devices and for controlling the phase modulator input level.

- Audio Oscillator — Used to generate output frequencies for modulating the test transmitter and S/C modulators and other uses.

- 2.4 Kc S/C Monitor Panel — Connects the output of the 2.4-kc S/C power supply to the S/C Interface Panel and provides selective switching of the 2.4-kc output voltages for application to the true rms voltmeter.

- True RMS Voltmeter — Monitors the 2.4-kc power to the S/C and other similar functions.

- Power Supply — Supplies regulated ± 15 vdc to the Test Transmitter.

- OSE-to-OSE Interface Panel — Connects the OSE and S/C functions to external systems.

3.4.5 S/C POWER SUPPLY RACK

The S/C Power Supply Rack may be used during subsystem tests to replace the radio subsystem transformer/rectifier and DC/DC converter units.

3.4.6 HOLDING AND CABLE STORAGE FIXTURE

The Holding and Cable Storage Fixture is used during subsystem tests to house and provide connectors to the subsystem and provide cabling to the OSE.

3.5 SELF TEST AND CALIBRATION

3.5.1 GENERAL

The Voyager Radio Subsystem OSE has self-test and calibration capabilities usable prior to and during external testing without external test interference. These functions are performed by equipments internal to the four basic OSE racks.

3.5.2 BLOCK DIAGRAM

Table 3-1 is a list of the self-test capabilities provided by the OSE. A functional block diagram of this mode is shown in Figure 3-3.

3.5.3 TEST DEFINITIONS

The following paragraphs present a brief discussion of the tests listed in Table 3-1:

a. Power Supplies — OSE and OSE-contained S/C DC power supply voltages and currents are monitored with meters mounted on the power supply faces and with the Digital D-C voltmeter. The 2.4 KC-SC Monitor Panel provides for selecting any 2.4 kc power supply output voltage or current to the S/C for monitoring by the true rms voltmeter. The 2.4 kc power supply output frequency is also monitored by the electronic frequency counter.

b. Running Time Indicators — Elapsed time indicators are supplied on all racks and record accumulated operating times.

c. Test Transmitter — Test Transmitter output power, VCO frequency and modulated spectrum are monitored. Output power is measured with the r-f power meter and Bolometer. A power meter calibrator is provided to calibrate these measurements. Test transmitter VCO frequency is monitored with the frequency counter by selecting the VCO from the counter function select panel. The modulated spectrum is monitored by the Spectrum Analyzer tuned to the S-Band carrier frequency. Modulation sensitivity and carrier suppression are determined using this test set up. Transmitter/Receiver Phase Jitter is measured using the same basic test set up with the Receiver DPE monitored by the oscilloscope. The residual rms noise with the loop locked is a measure of the Phase Jitter. This same basic test set up is utilized to measure and monitor other similar test transmitter operations and functions.

d. Test Receiver — Test Receiver AGC, SPE, DPE, IF Spectrum, VCO Frequency and Phase Jitter are measured or monitored by equipment internal to the basic OSE racks. The AGC, SPE and DPE are monitored by either the rms voltmeter, Digital D-C voltmeter or oscilloscope. The IF spectrum is monitored directly by the spectrum analyzer connected to the receiver IF and the VCO frequency measurements are selectable from the counter control panel. Receiver/Transmitter Phase Jitter are measurable as in the previous paragraph for transmitter self tests.

e. Ranging Receiver — The Ranging Receiver has a self test mode utilizing the test transmitter/r-f Converter/test receiver S-band link. Correlation voltage, lock indicators, clock frequency, and other similar functions are monitored to determine proper operating characteristics.

Table 3-1. Self Test Capability

Functional Equipment	Monitor/Measure
1. OSE & OSE-S/C Power Supplies	Voltages/Currents, frequency
2. Running Time Indicators	OSE, all racks
3. Test Transmitter	a. Output power b. VCO frequency c. Modulated spectrum d. Phase Jitter
4. Test Receiver	a. (AGC) Automatic Gain Control Voltage b. (SPE) Static Phase Error Voltage c. (DPE) Dynamic Phase Error Voltage d. I-F Spectrum e. VCO frequency f. Phase Jitter
5. Ranging Receiver	a. PN code correlation voltage b. Lock indications c. Clock frequency

3.6 SUBSYSTEM TESTS

3.6.1 GENERAL

These tests are performed with the basic Radio Subsystem OSE racks, the power supply rack and the holding and cable storage console. The tests evaluate and confirm the adequacy of the subsystem design, workmanship, and electrical performance to meet mission requirements. The OSE is capable of measuring or monitoring the functions listed in Table 3-2. These capabilities are not all required for testing on the subsystem level but provide the OSE with the added flexibility of testing on the assembly level.

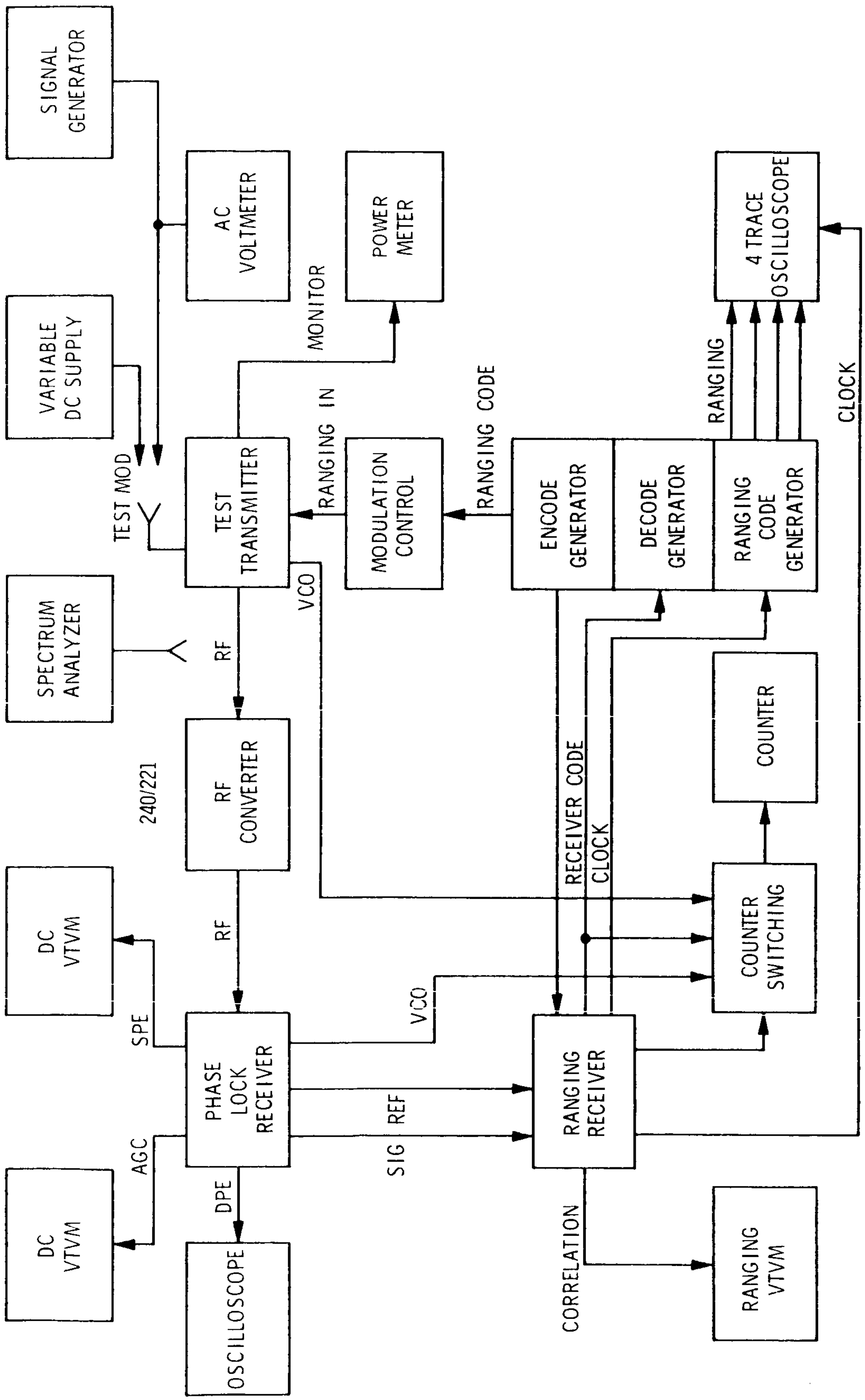


Figure 3-3. Self Test and Calibration

Table 3-2

Assembly	Tests/Measurements
Transponder (RCVR)	1. Power Supply Voltages/Currents 2. Command and Verify 3. Lock Indications 4. AGC Characteristics 5. Threshold 6. R-F Loop BW 7. AGC Loop BW 8. Phase Jitter 9. Amplitude Jitter 10. Image rejection 11. Spurious Response 12. Inter Channel Cross Coupling 13. Doppler Tracking Rate 14. Telemetry Channel, Output Signal Levels and Signal-to-Noise Ratios 15. Coherent Interference
Transponder (Exciter)	1. Power Supply Voltages/Currents 2. Command/Verify 3. Output Power 4. Spurious Output 5. Output Bandwidth 6. Modulation Characteristics 7. VCO Frequency and Range 8. Test Transmitter/Radio Receiver Phase Jitter
Transponder (Ranging Characteristics)	1. Delay 2. Delay Variations 3. Ranging Spectrum

Table 3-2 (Continued)

Assembly	Tests/Measurements
Power Amplifier	1. Power Supply Voltage/Currents 2. Output Power 3. Bandwidth 4. Spurious Output 5. Delay 6. Command/Verify 7. Transfer Characteristics

3.6.2 ASSEMBLY MODE

The Radio Subsystem OSE performs the tests defined in Table 3-2 on an assembly level using equipment internal to the four basic racks and the holding and cable storage console. The tests are performed using accepted test procedures defined by test specifications where applicable. The assemblies to be tested will be connected within the holding and cable storage console. Auxiliary cables are provided to operate one assembly independent of the others.

3.6.3 SUBSYSTEM MODE

A functional block diagram showing principle signal flow through subassemblies is shown in Figure 3-4. The following paragraphs augment the illustrated definition with brief definitions of the test to be performed.

Subsystem tests include assembly testing to verify that each assembly is functioning properly, plus sufficient engineering tests to ensure that the assemblies are interconnected properly and will meet required performance levels.

The basic subsystem level tests include the following:

a. Measuring turnaround delay and delay variation with the test Ranging Receiver.

b. Tests to insure that the subsystem will interface with other S/C subsystems and perform adequately as a system. This will be accomplished by simulating external interfaces within the OSE.

c. Tests to verify that all commanded functions respond properly to commands generated and verified by the Command Simulator and Mode Indicator unit.

d. Each redundant signal path is subjected to sufficient tests to establish performance levels.

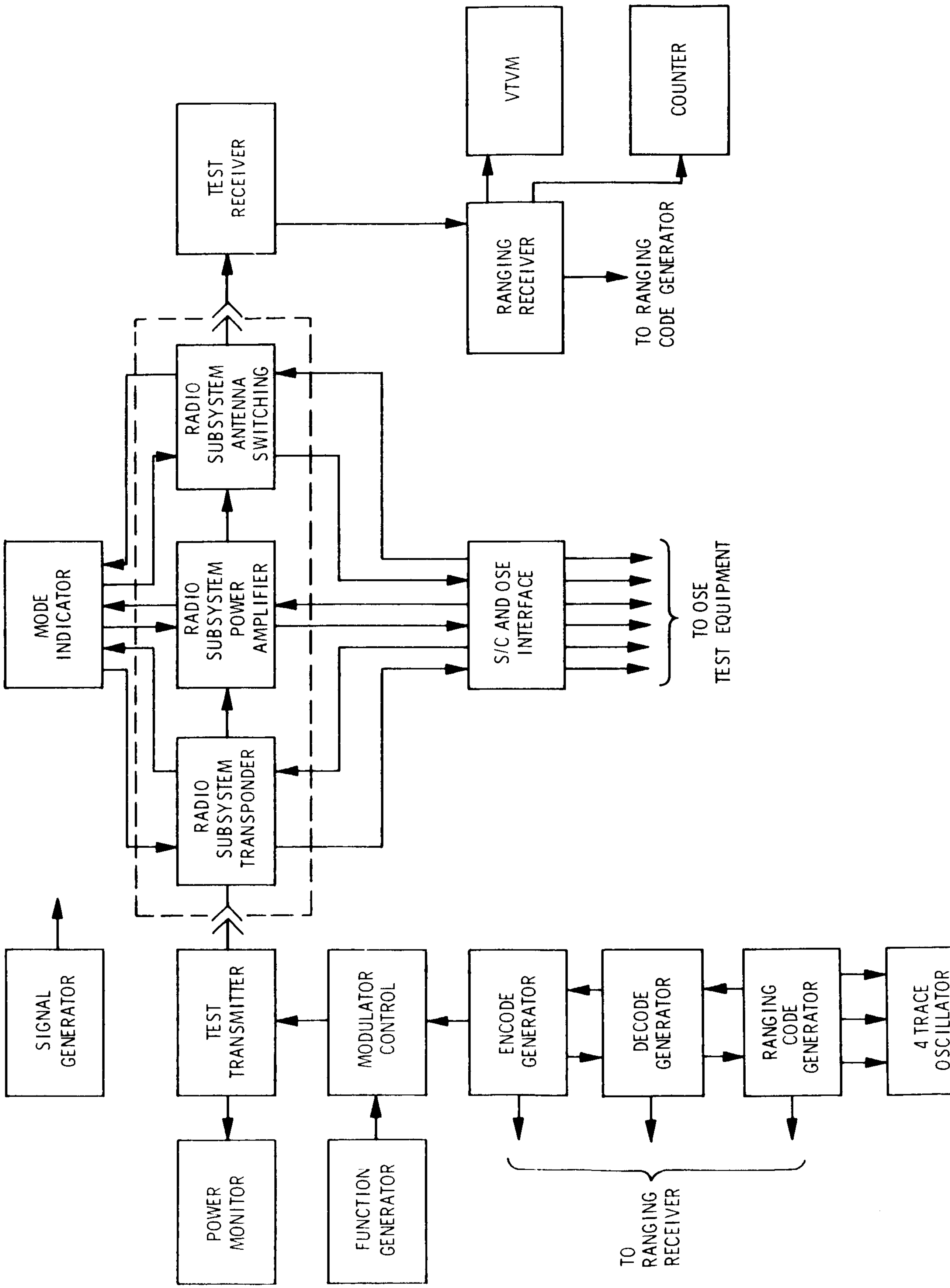


Figure 3-4. Radio Subsystem OSE Subsystem Tests

3.7 SYSTEM TEST

The Voyager radio subsystem OSE four basic racks are interfaced with the command subsystem OSE, data storage and display equipments, and recorders in the system test complex. This equipment is capable of the following:

a. Permit complete exercising of all spacecraft mechanical and electrical functions.

b. Afford monitoring in real-time of the spacecraft radio subsystem behavior.

c. Afford test repeatability.

d. Have self-check capability prior to and during test without test interruption.

e. Provide real-time test records with the following information:

   1. Name of test

   2. Function exercised

   3. Time

   4. Edited data with indications of questionable data.

f. Provide for detection of failures. The fault isolation need only to be a subsystem level.

g. Provide a record of accumulated test time on overall spacecraft equipments.

A functional block diagram of the Radio Subsystem OSE and interfaces with the spacecraft Radio Subsystem and other OSE is shown in Figure 3-5. The Radio-Subsystem OSE forms an integral part of the STC and provides measurement and monitor capabilities including, but not limited to, those for Subsystem and Assembly tests. The STC provides for integrated test control using the CDS and Command Verification Equipment. The Radio Subsystem OSE will function in this automated testing mode.

The Radio Subsystem OSE in conjunction with the other OSE within the STC supports the operations of the LCE at the Launch Complex and explosive safe areas.

4.0 INTERFACES

OSE-to-OSE Interfaces are shown in Figure 3-5.

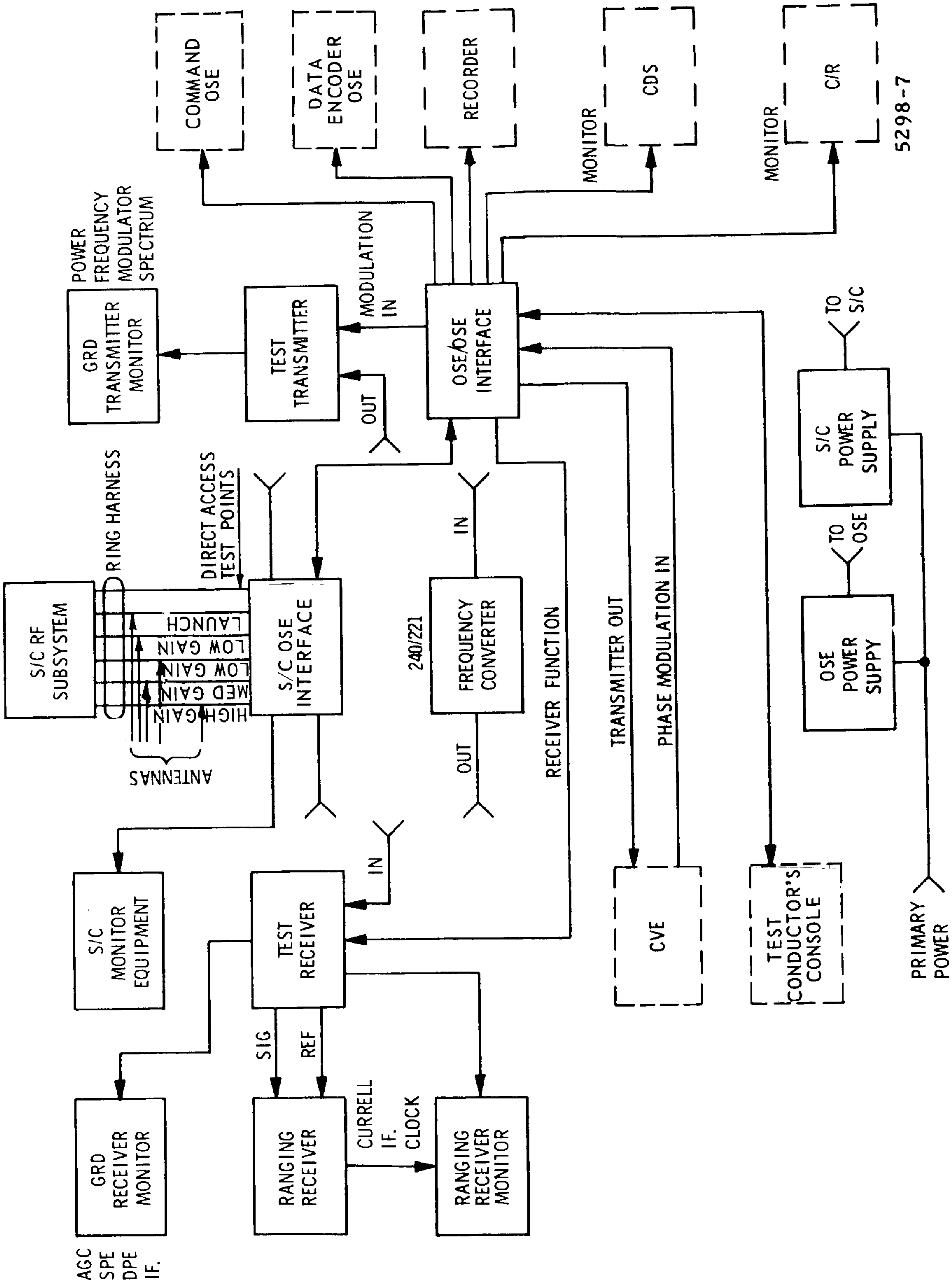


Figure 3-5. System Tests in STC

The following tables delineate the interfaces:

a. Table 4-1 shows the Radio OSE and Computer Data System interfaces.

b. Table 4-2 shows Radio OSE and Central Recorder Interfaces.

c. Table 4-3 shows Radio OSE and Test Conductors Console Interfaces.

d. Table 4-4 shows Radio OSE and CVE Interfaces.

e. Table 4-5 shows Radio OSE and Recorder Interface.

f. Table 4-6 shows Radio OSE and Spacecraft Interfaces.

g. Table 4-7 shows Radio OSE and Command OSE Interface.

h. Table 4-8 shows Radio OSE and Data Encoder OSE Interface.

Table 4-1. Radio OSE and Computer Data System Interfaces

Function	Voltage	Impedance (ohms)	Comments
ANALOG			
1. S/C RCVR No. 1 AGC	-1.5 to + 5.5 vdc	10k	Alarm Limits
2. S/C RCVR No. 2 AGC	-1.5 to +5.5 vdc	10k	Alarm Limits
3. S/C RCVR No. 3 AGC	-1.5 to +5.5 vdc	10k	Alarm Limits
4. S/C RCVR No. 1 SPE	-3.0 vdc to +3.0 vdc	1	D-C Iso Amp
5. S/C RCVR No. 2 SPE	-3.0 vdc to +3.0 vdc	1	D-C Iso Amp
6. S/C RCVR No. 3 SPE	-3.0 vdc to +3.0 vdc	1	D-C Iso Amp
7. S/C RCVR No. 1 TR	+15 vdc ±1%	15k	Alarm for 1% Change
8. S/C RCVR No. 1 TR	-15 vdc ±1%	15k	Alarm for 1% Change
9. S/C RCVR No. 2 TR	+15 vdc ±1%	15k	Alarm for 1% Change
10. S/C RCVR No. 2 TR	-15 vdc ±1%	15k	Alarm for 1% Change
11. S/C RCVR No. 3 TR	+15 vdc ±1%	15k	Alarm for 1% Change

Table 4-1. Radio OSE and Computer Data System Interfaces (Continued)

Function	Voltage	Impedance (ohms)	Comments
12. S/C RCVR No. 3 TR	-15 vdc ±1%	15k	Alarm for 1% Change
13. S/C Ex. No. 1 TR	-25 vdc ±1%	27k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.
14. S/C Ex. No. 2 TR	-25 vdc ±1%	27k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.
15. S/C Ex. No. 3 TR	-25 vdc ±1%	27k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.
16. S/C Ex. No. 1 TR	-15 vdc ±1%	15k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.
17. S/C Ex. No. 2 TR voltage	-15 vdc ±1%	15k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.
18. S/C Ex. No. 3 TR voltage	-15 vdc ±1%	15k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.
19. S/C Pow. Amp. No. 1 current Helix	0-3 vdc ±1%	1k	Alarm 1% change and Monitor during command and CC&S R-F Subsystem commands.

Table 4-1. Radio OSE and Computer Data System Interfaces (Continued)

Function	Voltage	Impedance (ohms)	Comments
20. S/C Pow. Amp. No. 2 Helix current	0-3 vdc ± 1%	1k	Alarm 1% change and Monitor during command & CC&S R-F subsystem commands.
21. S/C Pow. Amp. No. 3 Helix current	0-3 vdc ± 1%	1k	Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.
22. S/C Pow. Amp. No. 1 Anode Voltage		1 meg	Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.
23. S/C Pow. Amp. No. 2 Anode Voltage		1 meg	Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.
24. S/C Pow. Amp. No. 3 Anode Voltage	0/+125/+400 vdc/± 1%	1 meg	Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.
25. S/C Pow. Amp. No. 1 Helix voltage			Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.
26. S/C Pow. Amp. No. 2 Helix voltage			Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.

Table 4-1. Radio OSE and Computer Data System Interfaces (Continued)

Function	Voltage	Impedance (ohms)	Comments
27. S/C Pow. Amp. No. 3 Helix voltage			Alarm 1% change and Monitor during command & CC&S R-F Subsystem commands.
28. S/C Control unit Voltage	+10 vdc ± 1%	10k	Alarm 1% change and Monitor during command.
29. S/C Power Amp. and Exciter No. 1	0 to 1 vdc ± 2%	1k	Monitor from Power Meter output.
30. S/C Power Amp. and Exciter No. 2	0 to 1 vdc ± 2%	1k	Monitor from Power Meter output.
31. S/C Power Amp. and Exciter No. 3	0 to 1 vdc ± 2%	1k	Monitor from Power Meter output.
32. GRD RCVR SPE	± 5.0 vdc	1	DIC 6122 ISO Amp. output.
33. Test XMTR Frequency	"0" State Level -8 vdc "1" State Level +18 vdc	100k	From Frequency Counter, 4 line BCD (1-2-2-4), Print Output, (print command 13V to 0V step dc coupled.)
34. Command Requests			Switch Closures.

Table 4-2. Radio OSE and Central Recorder Interfaces

Function	Voltage	Impedance (ohms)	Comments
1. S/C AGC	-1.5 to 5.5 vdc	10k	Alarm limits
2. S/C SPE	±3.0 vdc	1	Output from DIC 6122
3. S/C DPE	±4.0 v (mixed signal)	10k	(S/C Phase detector output)
4. S/C R-F Power Output	0 to -1 vdc ± 2%	1k	
5. GRD RCVR SPE	±5.0 vdc	1	DIC 6122 output

Table 4-3. Radio OSE and Test Conductors Console Interface

Function	Voltage	Impedance (ohms)	Comments
Radio OSE Status as required.			Switch closures

Table 4-4. Radio OSE and CVE Interface

Title	Voltage	Impedance (ohms)	Comments
1. Phase Mode. Input	3 rad/volt	50	Command Mod.
2. Test XMTR output	-60 dbm ±3 dbm	50	Test XMTR R-F output.
3. Test XMTR Frequency	"0" State Level -8 vdc "1" State Level +18 vdc	100k	From Frequency Counter, 4 line BCD (1-2-2-4), Print output, (print command 13V to 0 V step dc coupled.)

Table 4-5. Radio OSE and Recorder Interfaces

Function	Voltage	Impedance (ohms)	Comments
1. TLM SPE	-1.5 to + 1.5 vdc	1k	
2. S/C Command DPE	Command Signal from Phase detector	10k	
3. GRD RCVR DPE	0-6 v	1	
4. GRD RCVR SPE	0-6 vdc	1	
5. TLM AGC (course)	0-0.1 vdc	1k	
6. TLM AGC (fine)	0-3 vdc	1k	
7. TLM LO Drive	0-0.1 vdc	1k	
8. S/C RF Powers (HP 431R Recorder Output)	0-1 vdc	1k	
9. TLM Power Ampl.'s Cathode Currents	0-3 vdc	10k	
10. TLM Hi Gain Antenna Drive Level	0-0.1 vdc	10k	
11. TLM LO Gain Antenna Drive Level	0-0.1 vdc	10k	
12. TLM Omni Antennas Drive Levels	0-0.1 vdc	10k	

Table 4-6. Radio OSE and Radio Interface

Function	Voltage	Impedance (ohms)	Comments
DIRECT ACCESS TEST POINTS			
1. AGC RCVR No. 1	-1.5 to 5.5 vdc	10k	ISO-amp out.
2. AGC RCVR No. 2	-1.5 to 5.5 vdc	10k	ISO-amp out.
3. AGC RCVR No. 3	-1.5 to 5.5 vdc	10k	ISO-amp out.
4. SPE RCVR No. 1	±3 vdc	1	ISO-amp out.
5. SPE RCVR No. 2	±3 vdc	1	ISO-amp out.
6. SPE RCVR No. 3	±3 vdc	1	ISO-amp out.
7. DPE RCVR No. 1	±4v (mixed)	10k	Phase det out
8. DPE RCVR No. 2	±4v (mixed)	10k	Phase det out
9. DPE RCVR No. 3	±4v (mixed)	10k	Phase det out
10. Radio RCVR No. 1 T/R voltage	±15 vdc	15k	
11. Radio RCVR No. 2 T/R voltage	±15 vdc	15k	
12. Radio RCVR No. 3 T/R voltage	±15 vdc	15k	
13. Exciter No. 1 T/R voltage	-25, -15 vdc	27k	
14. Exciter No. 2 T/R voltage	-25, -15 vdc	27k	
15. Exciter No. 3 T/R voltage	-25, -15 vdc	27k	
16. Pow Amp No. 1 voltage Cath, Anode, Collector		1 meg	
17. Pow Amp No. 2 voltage Cath, Anode, Collector		1 meg	
18. Pow Amp No. 2 voltage Cath, Anode, Collector		1 meg	

Table 4-6. Radio OSE and Radio Interface (Continued)

Function	Voltage	Impedance (ohms)	Comments
19. Ranging	+15 vdc or 0 vdc		Monitor on/off
20. Exciter No. 1 Pwr. Out. Mon.	DC	1	
21. Exciter No. 2 Pwr. Out. Mon.	DC	1	
22. Exciter No. 3 Pwr. Out. Mon.	DC	1	
23. Pow Amp No. 1 Pwr. Out. Mon.	DC	1	
24. Pow Amp No. 2 Pwr. Out. Mon.	DC	1	
25. Pow Amp No. 3 Pwr. Out. Mon.	DC	1	
RF Inputs and Outputs			
26. Launch Antenna		50	Provide loads for, measure power, also provide input to receivers.
27. Pri. Low Gain Antenna*		50	
28. Sec. Low Gain Antenna*		50	
29. Med. Gain Antenna*		50	
30. High Gain Antenna*		50	

*When antennas are installed and connected, a probe at each provides for RF coupling. While probes are indicated here as the means for coupling, a precision coupler may be required to effect accurate receiver sensitivity measurements.

Umbilical

There is an umbilical cable for each of the four antenna probes. There is no umbilical cable for the Launch Antenna.

Table 4-7. Radio OSE and Command OSE Interfaces

Function	Voltage	Impedance (ohms)	Comments
1. Command Subcarrier	0-3.3 V rms	600	
2. Synchronization Signal	0-3.3 V rms	600	
3. Combined Signals	0-4.0 V rms	600	

Table 4-8. Radio OSE and Data Encoder OSE Interface

Function	Voltage	Impedance (ohms)	Comments
Radio Subsystem OSE Receiver telemetry output	Demodulated S/C Subcarrier Output	-	Isolated Output

5.0 PERFORMANCE CHARACTERISTICS

The Radio Subsystem OSE provides the measurement/control capability listed below, with the accuracy indicated for each parameter. This level of performance is required for all modes of use; assembly, subsystem, and system testing.

5.1 PARAMETERS, VALUES, RANGES, AND ACCURACIES

a. D-C Voltages - The OSE has the capability of measuring voltages to 1 part in 10^5 or voltages down to ± 1 microvolt on the most sensitive scale. The meter has a 20 db noise rejection capability and gives very accurate d-c measurements even under noisy signal conditions. (Where this type of accuracy is not needed a higher input impedance meter is available with 2 per cent accuracy).

b. A-C Voltages - A-C voltage can be read with 2 per cent accuracy to 1 Mc; 3 per cent to 2 Mc; and 5 per cent to 4 Mc. A-C voltage can be read as true rms with the Systron-Donner 1240 with 1/4 per cent accuracy from 50 cps to 10 kc.

c. R-F Measurements - All r-f interconnections such as cables, attenuators, and directional couplers are calibrated to ±0.1 db. All r-f power readings are accurate to ± 1/2 db on both transmitted and received r-f signals to -100 dbm and ± 1 db to threshold.

   The OSE also has the capability of measuring S-band power to a level of 100 watts within ±1 db.

d. Spectrum Analysis - Frequency spectrum relative power measurements are accurate to ± 1.5 db with a frequency resolution down to 1 kc.

   Modulation side band analysis with the wave analyzer provides frequency resolution equal to that of the Frequency Counter, and a relative power level within ± 1/2 db.

e. Frequency Measurements - All frequency measurements up to 100 Mc are accurate to 1 part in 10^8 (or ± 1 cps up to 100 Mc) at signal levels of -25 dbm or greater.

f. Receiver Phase Stability - Residual phase modulation are measurable to an accuracy better than 1/2 degrees peak.

g. Exciter Phase Stability - Residual phase modulation of the exciter is measurable to an accuracy better than 1 degree peak.

h. Static Phase Error (SPE) - The OSE monitors the S/C receivers SPE and VCO frequencies.

i. AGC, S/C Receiver - The OSE monitors the S/C receivers coarse and fine AGC.

j. Local Oscillator Drive - The OSE monitors the S/C receivers local oscillator drives.

k. Reference Oscillator Frequency - The OSE monitors the reference oscillator frequencies.

l. Receiver Frequency Response - The Amplitude versus frequency characteristics of the S/C receivers is measured at the output of the phase detectors.

m. Modulation Characteristics - Phase modulation linearity and sensitivity is measurable by the OSE with both square wave and sine wave modulation.

n. Power Amplifier - The OSE monitors the S-band power amplifier operating currents/voltages and provides an alarm/fault indication.

o. Ranging Signal Detection and Modulation - The OSE will test and verify that the ranging function performs properly.

p. Temperature Transducers - The OSE will provide temperature measurement within an accuracy of 1 degree centigrade.

q. Coherent Interference - The OSE is capable of measuring receiver Coherent Interference using the Digital D-C Voltmeter.

r. Amplitude Jitter - The OSE will measure Amplitude Jitter of the AGC Loop.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 MARINER C OSE UTILIZATION

The existing Mariner C OSE contains the basic instrumentation necessary to perform the required functions for Voyager with modification and additions to the design. The required modifications include the following:

a. Additional crystal switching to provide for selecting six transmitting and receiving channels.

b. Additional test point monitoring capability as required by the Voyager S/C.

c. Provide for Voyager S/C radio system interfaces.

d. Provide for additional OSE/OSE interfaces as required.

e. Update the Mariner C Radio System OSE to improve performance.

f. Additional commercial equipment:

1.	Noise figure meter and noise source	HP - 340B (modified) HP - 349A
2.	Function Generator, Exact Electronics - 250 RM	
3.	S-Band Termination (100 w), Bird - 8130	
4.	S-Band Signal Generator	HP - 8614A
5.	Spectrum Analyzer	HP - 851A w/HP 8551A

g. Use of integrated circuitry in performing logic functions.

6.2 PHYSICAL CHARACTERISTICS

6.2.1 RADIO SUBSYSTEM OSE

The Voyager Radio Subsystem OSE consists of two pairs of integrally connected standard 75-inch high, NASA type, cabinet rack enclosures. Racks I and II are physically connected and house the Ranging Receiver, Test Receiver, Encode and Decode Generators and commercial equipment. Racks III and IV are physically connected and house the transmitting and S/C monitor/control equipment. All racks are designed to mount standard 19-inch wide panel assemblies.

a. Weight - The approximate weight of the OSE is 1600 pounds for Racks I and II and 1500 pounds for Racks III and IV.

b. Power - The power requirements are approximately 2200 watts for Racks I and II and 1700 watts for Racks III and IV. The required supply is 105-125 volts 46-65 cps, single phase.

c. Commercial and Special Equipment - The following commercial and special equipment or its equivalent is housed in these racks and constitutes the radio subsystem OSE:

1. Special Equipment

   (a) Test transmitter

   (b) Test phase coherent receiver

   (c) Ranging receiver

(d) Frequency converter (240/221)

(e) Wave analyzer reference oscillator and mixer

(f) Psuedo-random range code generators

(g) PN ranging code sync and switching control panel

(h) S/C command-Command simulator and mode indicator panel

(i) S/C voltages/currents and function tests point monitor panel

(j) S/C power monitor

(k) Intercommunication equipment

2. Commercial Equipment

Quantity	Equipment	Mfr.	No.
4	Amplifiers, Isolation	DIC	6122
1	Digital Integrating D-C Voltmeter	Dymec	2401
1	Frequency Counter w/HP 525A 100 Mc plug-in	HP	524C
1	S/C Radio Subsystem DC/DC Converter Power Supply	NJE	QR-60-25
1	2.4 kc square wave S/C radio Subsystem T/R Power Supply	IDS	
1	Microwave Power Meter	HP	430C
2	Microwave Power Meter	HP	431C
1	Wide Range Oscillator	HP	200CDR
1	Oscilloscope w/type 67 time base and type 72 dual trace plug-ins	Tektronix	RM561

Quantity	Equipment	Mfr.	No.
1	Oscilloscope w/type M four trace plug-in	Tektronix	RM45A
1	Spectrum Analyzer w/RF Unit	HP HP	851A 8551A
1	Wave Analyzer	HP	302A
1	VTVM	HP	400DR
1	Data Printer	Supplied by Data Display	
-	Associated Power Supplies for powering special equipment.		
1	True RMS Voltmeter Systron Donner		1240
1	Noise Figure Meter	HP	340B
	Noise Source	HP	349A
1	Function Generator	Exact Electronics	250RM
1	Termination (S-band, 100w)	Bird	8130
1	S-Band Signal Generator	HP	8614A

6.2.2 SPACECRAFT POWER SIMULATOR RACK

The spacecraft power simulator unit is used to supply power to the S/C radio subsystem when its DC-DC converters and transformers-rectifiers are not being used during testing, trouble shooting or failure analysis. The unit is housed in a standard 75-inch high NASA cabinet rack.

a. Weight - This unit weighs approximately 600 pounds.

b. Power - Approximately 1500 watts of 105-125 volts 45-65 cps single phase power are required.

6.2.3 HOLDING AND TEST FIXTURE

A holding and test fixture for use in subsystem tests is provided. This fixture mates with extender connectors to simulate the ring harness and antenna interfaces.

a. Weight - The weight of this unit is approximately 650 pounds.

b. Power - There is no power to this console.

c. Equipment - The following will be housed within this console:

   1. Top mounting pad with dust proof lid for S/C r-f packages.

   2. Cables on spool (with ring harness, direct access, and r-f cables making one composite cable).

   3. Vacuum and temperature adaptor cables and miscellaneous adaptor cables.

7.0 SAFETY

7.1 GENERAL

The Radio Subsystem OSE conforms to good engineering practice providing circuit protection and safety devices such as current overload relays and voltage stabilization devices. The OSE presents no hazards to personnel. Safety features such as cabinet interlocks for cabinets operating with greater than 120 volts, high voltage point isolations and warning, and proper weight distribution to prevent tipping resulting in damage to equipment and possible injury to personnel are utilized to the fullest extent.

7.2 SPACECRAFT INTEGRITY

The OSE is designed to interface safely with any overall flight S/C and to eliminate any possibility of damage to the S/C radio subsystem and other OSE. The OSE provides self-check capabilities on all "launch hold" criteria included in the LCE and STC. Self checks are on a non-interfering test basis.

7.3 FACILITIES

Test Procedures are written to insure the proper installation and operation of the equipment and to insure over-all safety of the S/C and OSE equipment.

CII - VB263FD102

SYSTEM TEST COMPLEX

COMMAND VERIFICATION EQUIPMENT

Index

1.0 SCOPE

This document is a functional description of the Command Verification Equipment used in the STC for system testing of the 1971 Voyager spacecraft.

2.0 APPLICABLE DOCUMENTS

JPL

GMG-50109-DSN-A Design Specification, Telecommunications Development, GSDS Command System, Ground Subsystem (Command Verification Equipment).

General Electric

VB260SR102 Voyager 1971 System Test Complex Design Characteristics and Restraints.

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The use of the Command Verification Equipment (CVE) in the STC will assure accurate and reliable command information transfer between a command originator and the Radio Subsystem OSE transmitter, for system tests of the Voyager.

3.2 BLOCK DIAGRAM

A block diagram of the CVE is shown in Figures 1 and 2 of GMG-50109-DSN-A. For Voyager system testing, the block diagram is similar. The exceptions for the system test application, as the part of the System Test Complex which generates the command word stream, are that the Radio Subsystem OSE replaces the DSIF Transmitter, and the Data Encoder Subsystem OSE replaces the DSIF Telemetry. This is iillustrated by Figure 3-1 the block diagram of the CVE configuration within the STC. In addition, the CVE receives command messages originated by the CDS, and inputs from teletype, paper punch, typewriter, and manual inputs.

3.3 COMPATIBILITY WITH DSIF/SFOF

The use of CVE in a system test performed by the STC serves to establish an early confidence in SFOF/CVE/Spacecraft/DSIF compatibility. It also will meet the intent of maximizing compatibility between operating and testing procedures.

4.0 INTERFACES

4.1 CVE-TO-RADIO OSE INTERFACE IS SHOWN IN TABLE 4-1.

4.2 CVE-TO-COMMAND OSE INTERFACE IS SHOWN IN TABLE 4-2.

4.3 CVE-TO-DATA ENCODER OSE INTERFACE IS SHOWN IN TABLE 4-3.

4.4 CVE-TO-CENTRAL TIMING INTERFACE IS SHOWN IN TABLE 4-4.

4.5 CVE-TO-CDS INTERFACE IS SHOWN IN TABLE 4-5.

4.6 CVE-TO-C&S OSE INTERFACE IS SHOWN IN TABLE 4-6.

5.0 PERFORMANCE PARAMETERS

Performance parameters are equivalent to those given in GMG-50109-DSN-A.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

Physical characteristics and constraints are equivalent to those given in GMG-50109-DSN-A.

7.0 SAFETY CONSIDERATIONS

There are no specific safety considerations beyond those provided by GMG-50109-DSN-A for accurate and reliable command.

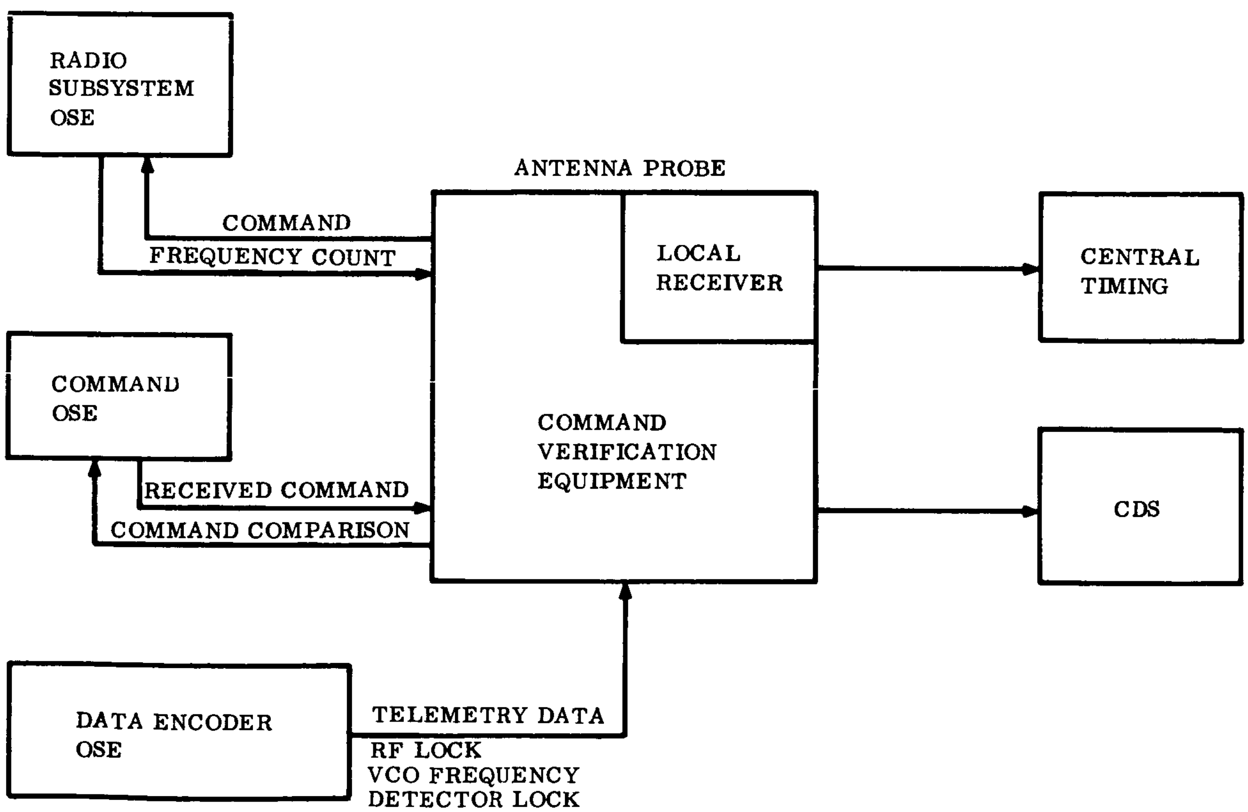


Figure 3-1. Command Verification Equipment, Functional Block Diagram

Table 4-1. CVE-to-Radio OSE Interface

Function	I/0	Comment
1. Modulation	0	3.7.2.1* Bipolar, Compatible with DSIF transmitter input.
2. Frequency Count	I	3.7.1.4* BCD
3. Antenna Probe	I	3.7.1.6*

*Appropriate section of GMG-50109-DSN-A.

Table 4-2. CVE-to-Command OSE Interface

Function	I/0	Comment
1. Received Command	I	3.7.1.5* Data, word sync, and bit sync.
2. Command Comparison	0	3.7.2.3*

*Appropriate section of GMG-50109-DSN-A.

Table 4-3. CVE-to-Data Encoder OSE Interface

Function	I/0	Comment
Telemetry data RF lock VCO frequency Detector lock.	I	3.7.1.7* Binary, 7 bit parallel, plus read commands.

*Appropriate section of GMG-50109-DSN-A.

Table 4-4. CVE-to-Central Timing Interface

Function	I/0	Comment
Timing	I	Days, hours, minutes and seconds in BCD format as described in VB260FD106.

Table 4-5. CVE-to-CDS Interface

Function	I/0	Comment
1. Coded Command Preamble and Messages	I	Five level characters 3.7.1.1 (GMG-50109-DSN-A)
2. Verification Signal	0	As above
3. Command Initiate Signal	0	per 3.7.2.5 (GMG-50109-DSN-A)
4. Bit Synch Pulses	0	As above
5. Command Data (to modulator)	0	As above
6. Command Data (from detector)	0	As above

Table 4-6.

Function	I/O	Comment
Received Commands	I	C&S commands received; not Manchester

CII - VB263FD103

SYSTEM TEST COMPLEX

COMMAND SUBSYSTEM OSE

Index

1.0 SCOPE

This document describes the functions of the Voyager Operational Support Equipment (OSE) used to monitor and test the spacecraft Flight Command Subsystem.

2.0 APPLICABLE DOCUMENTS

Reference to the following documents is required for completeness in this description.

JPL

GMG-50109-DSN-A
Design Specification, Telecommunications Development, GSDS Command System, Ground Subsystem (Command Verification Equipment)

GE/Motorola

VB263FD101	Functional Description Radio OSE
VB233FD103	Functional Description Flight Command Subsystem
VB260SR102	System Test Complex Design Characteristics and Restraints
VB260SR101	System Test Complex Test Objectives and Design Criteria

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The Voyager Flight Command Subsystem OSE will be used to test the subsystem from initial subsystem acceptance tests through STC tests and prelaunch system tests at AFETR.

3.2 BLOCK DIAGRAM

A block diagram of the Voyager Flight Command Subsystem OSE, including interfaces, is shown in Figure 3-1.

3.3. REQUIRED FUNCTIONS

The Voyager Flight Command Subsystem OSE will perform the following Functions:

a. Generate Command Words in the Format shown in VB233FD103

b. Modulate the appropriate subcarrier with command and sync data.

c. Monitor the Flight Command Subsystem direct access test points.

d. Display the status of the Flight Command Subsystem Operation.

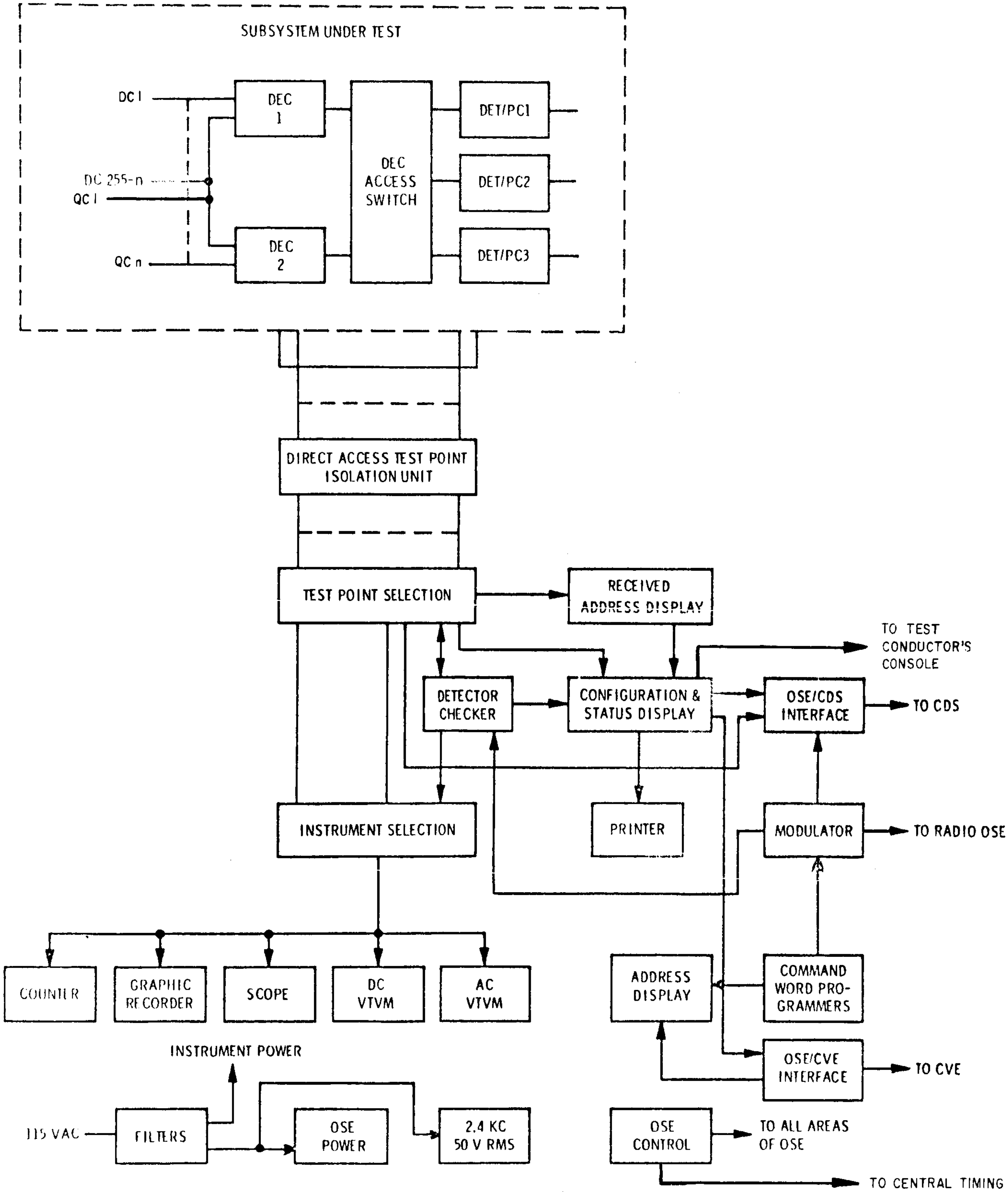


Figure 3-1. Command OSE In System Test Configuration

e. Record any incorrect Flight Subsystem operation.

f. Perform special Flight Subsystem tests:

   1. Command word bit error tests.

   2. Detector phase locked loop noise bandwidth measurements.

   3. Detector phase locked loop error signal tests.

   4. Quantitative measurement of command decoder isolated switch characteristics (not available during system tests).

   5. Individual testing of each of the Flight Command Subsystem functional units without need for the other functional units (not available during system tests).

   6. Measurement of the power consumption of each of the Flight Subsystem functional units (not available during system tests).

g. Provide an alternate source of power for the subsystem or any of its functional units.

h. Monitor and record operating time for both OSE and systems under test.

i. Provide the necessary interfaces with other OSE as described below.

j. Provide power and signal isolation as described in VB260SR101 and VB260SR102.

k. Provide the capability of Self Test.

3.4 SELF TEST AND CALIBRATION

3.4.1 SELF TEST

The Voyager Command Subsystem OSE will have the capability for self test (confidence check) both before connection to a system, subsystem or assembly and during test.

a. Word Programmer/Modulator Tests - These functions are checked by monitoring the modulation signal with the oscilloscope while continuous "ONES", "ZEROS", "ONE-ZERO" combinations and word start signals are set on the programmer. These signals do not result in action by the Flight Subsystem. Print out of pseudo-errors confirm and record the test results as observed on the oscilloscope.

b. Logic Function Tests - These tests are performed by grounding selected direct access test points to induce known error responses in the OSE. The error type is printed for correllation with expected error. No anamolous response is shown by the flight subsystem. For example, grounding the isolated Direct Test Point (DTP)_corrrsponding to Bit-Sync would result in print out of Bit-Sync errors once each bit time. Grounding of the matrix interrogate DTP results in an error print at the end of each word.

3.4.2 CALIBRATION

The commercial test equipment requires periodic calibration utilizing standard test equipment laboratory facilities and secondary standards.

The OSE frequency generators, filters, power supplies, and modulator can be calibrated using the test equipment in the OSE.

None of the above require periodic calibration. However, calibration should be checked after each move of the OSE or after any failure.

3.5 SUBSYSTEM TESTS

3.5.1 GENERAL

The command subsystem consists of the following functional assemblies:

a. Detector/program controller unit (three per subsystem).

b. Decoder access unit/decoder (one access unit per subsystem two decoders per subsystem).

c. Transformer-Rectifiers (two per subsystem).

A functional assembly may consist of one or more physical assemblies and one physical assembly may contain more than one functional assembly.

The subsystem redundancy makes independent evaluation of the performance of each functional assembly necessary for complete subsystem test. The following paragraphs describe the independent tests which can be performed on each functional assembly and their relationship to the subsystem performance test.

3.5.2 DETECTOR/PROGRAM CONTROL UNIT TESTS

With a detector/program controller functional element and the OSE in the configuration shown in Figure 3-2, the following tests can be made.

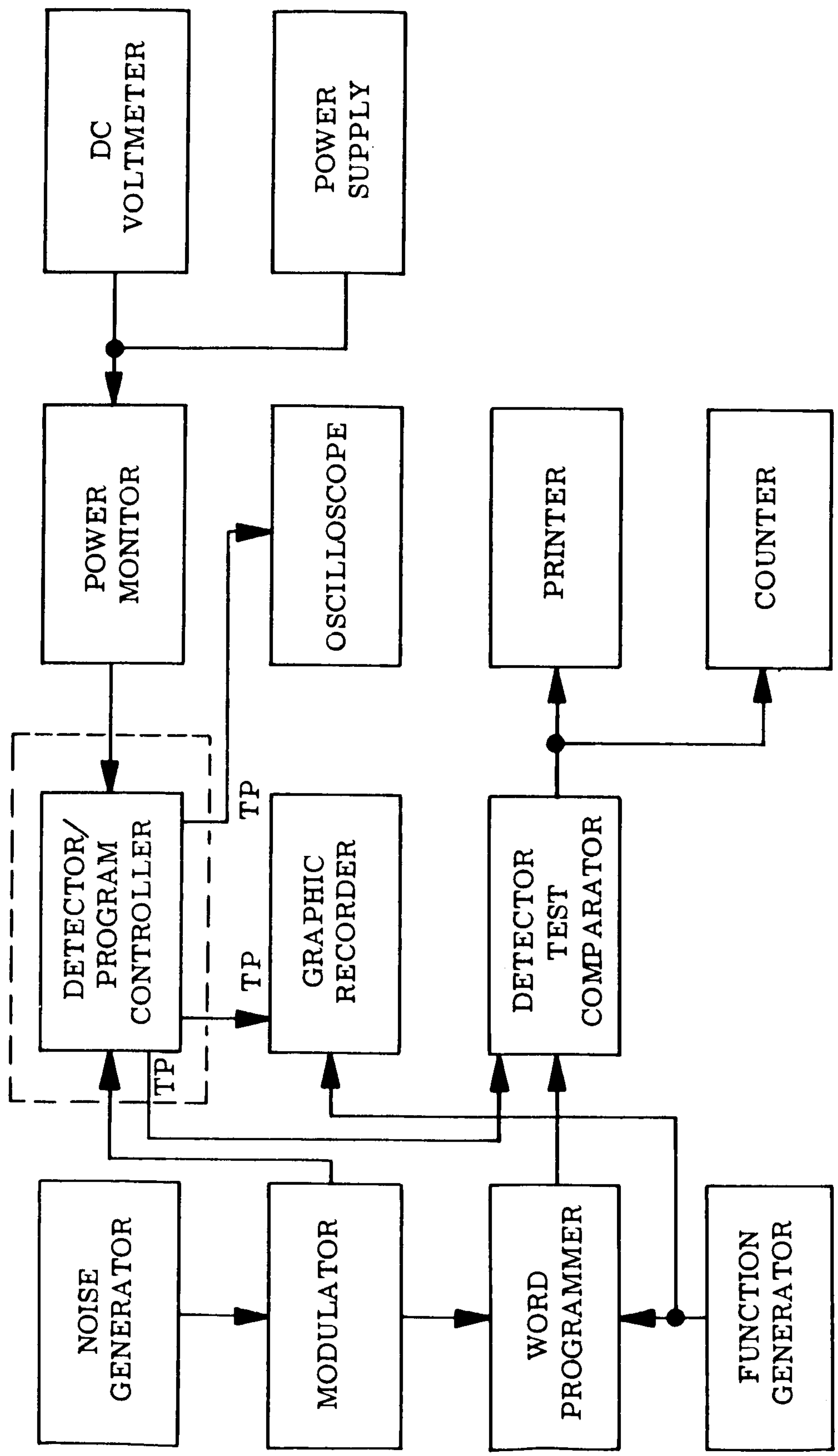


Figure 3-2. Detector/Program Controller, Test Setup

a. Detector Bandwidth Tests - This test uses the graphic recorder to plot the phase detector output on one channel and the function generator output on the other. Variation of the sine wave output frequency of the function generator permits determination of the frequency for peak response. This frequency is proportional to the loop bandwidth.

b. Sub-bit Error Test - This test uses the correct word counter, the general purpose counter, and the printer, in conjunction with the word programmer, modulator, and noise generator to record and display the results of long term tests of sub-bit error rate as a function of signal-to-noise ratio. There is sufficient data in the Detector Test Comparator to show correlations with out-of-lock conditions and inter-symbol influence.

c. Power Monitor - This test uses a press-to-test conversion to permit d-c current measurements with the d-c voltmeter. These, with normal voltage readings, permit assembly power drain determination.

d. Since all tests requiring comparison of output data bits with programmed data require proper operation of the program controller this function is checked by the bit error test set up.

3.5.3 DECODER ACCESS UNIT/DECODER TESTS

Since the Decoder Access Unit and Decoders are physically inseparable they are tested simultaneiously. However, the address structure and unit switch logic are used to demonstrate the integrity of the redundant decoders. Figure 3-3 shows the active poritions of the OSE during these tests.

a. Qualitative Tests - The OSE main consoles are capable of performing qualitative tests of the access switch and decoders. Detector/program controller outputs are sent to the functional assembly. These inputs may be sent through one of the detector/program controllers and are available as simulated outputs from the OSE. The OSE verifies that the correct switch closed at the correct time and that neither it nor any other switch closed when not expected.

b. Quantitative Tests - Quantitative measurements of the switch closure parameters; rise time, fall time, dwell time, leakage, and saturation resistance can be made using the data logger to control the OSE main console and special measuring instruments. Results for each switch are printed as permanent records of each test. Figure 3-3 is a block diagram of this mode of operation.

c. Power Consumption Test - A similar test of the decoder to that described in paragraph 3.5.2 c can be made.

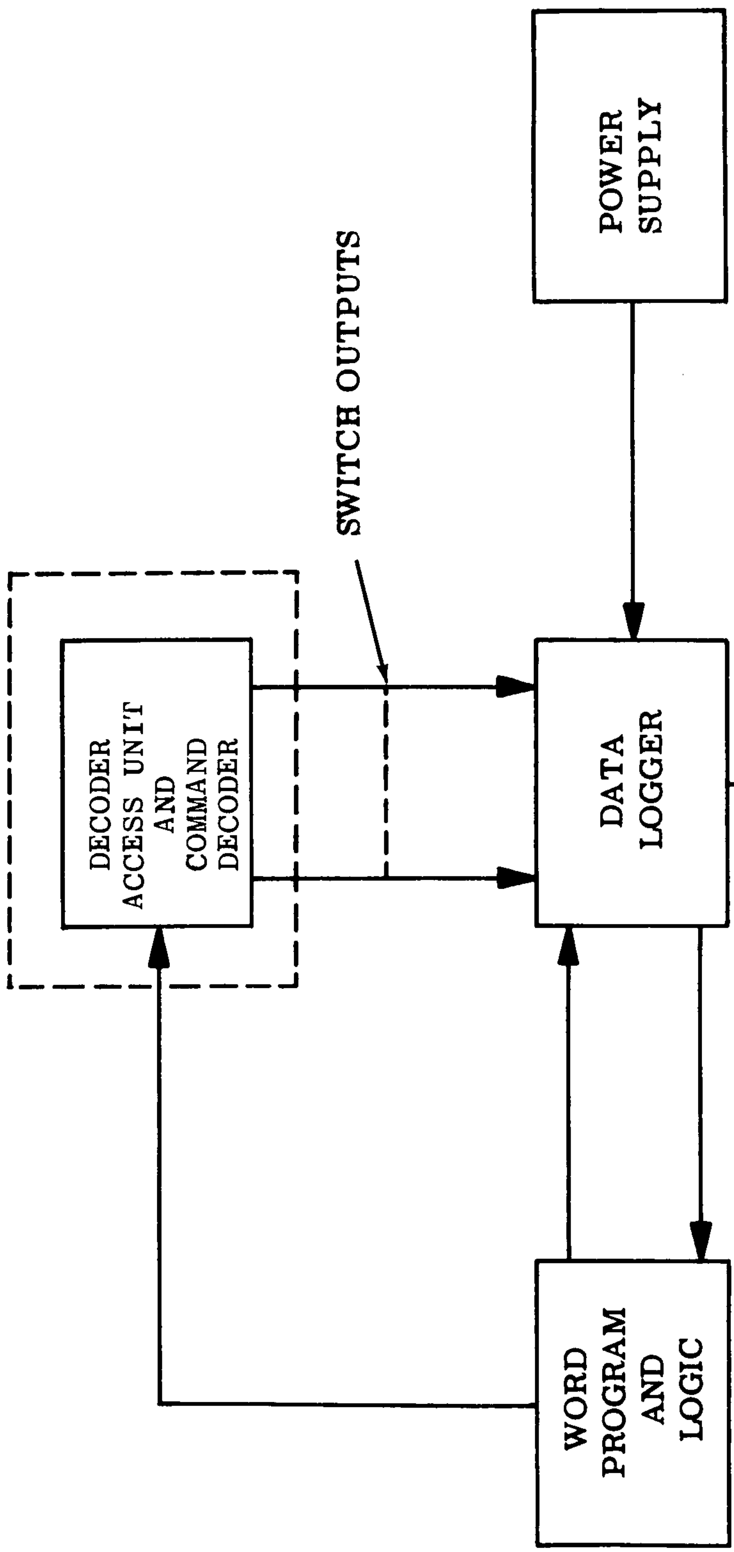


Figure 3-3. Isolated Switch Tests

3.5.4 TRANSFORMER-RECTIFIER TESTS

Through the use of dummy loads the performance of the transformer rectifier can be checked in both the redundant mode and under conditions of simulated failure either open or short for either of the redundant supplies.

3.5.5 SUBSYSTEM TESTS

Tests of the subsystem as a complete unit are the most revealing and comprehensive of all. All of the subassembly tests except transformer-rectifier failure mode tests can be made on the complete subsystem. The block diagram for this mode is the same as that shown in Figure 3-1.

In the preceding discussion the subsystem tests have been developed from the thesis that the subsystem is the sum of its assemblies and if they are adequately tested separately and their performance together compared with their individual performances a sharper picture of subsystem operation is obtained.

An added advantage of separable testing comes from the ability to perform certain long tests simultaneously. For instance, bit error tests and data logging of the same subsystem in the same time span using two OSE. Time can also be saved by data logging the switches at accelerated rates and sampling a few outputs at normal complete system rates.

3.6 SYSTEM TESTS

In tests conducted on the assembled spacecraft the OSE main console only is used. Although the primary mode described in paragraph 3.6.1 is normal, sufficient flexibility will be incorporated in the OSE to permit operation in two other modes.

All error print outs are referenced to Central Timing and Synchronization time. All operations, sequence initiations and monitoring functions are referenced to the bit sync of the active modulator/detector combination.

3.6.1 SYSTEM TEST CVE GENERATING COMMANDS

In this mode the OSE modulator is disabled and its oscillator and PN Generator are phase locked to the Command Detector in lock with the CVE. The OSE then monitors and prints all received addresses and command events. These can then be compared with the intended addresses, sent to the OSE from the CVE. Aggreement of these checks inhibits error print out. At the same time the received address is sent to the CVE for its master checks.

3.6.2 SYSTEM TEST-OSE GENERATING COMMANDS

In this mode the OSE Modulator output is fed to the test transmitter of the Radio Subsystem OSE and the word programmer acts as a manual command generator.

The received address is checked against the programmed address and only errors are printed. If a permanent record of all commands is desired, the received address can be printed in black for agreement with that transmitted and red for error.

3.6.3 SYSTEM TESTS WITHOUT RADIO

If for any reason a system test without radio is required the output of the OSE command modulator can be wired directly to a command detector input and tests run, similar to those in paragraph 3.6.2.

4.0 INTERFACE CHARACTERISTICS

4.1 GENERAL

The Command Subsystem operational support equipment in the STC has interfaces with the following areas (see Figure 4-1).

a. Command subsystem direct access test points

b. Radio subsystem operational support equipment

c. Central recorder

d. Computer data system

e. Command verification equipment

f. Test conductor's console

g. Central timing and synchronization

4.2 FLIGHT COMMAND SUBSYSTEM INTERFACE WITH COMMAND OSE

The command OSE will have the capability of monitoring the direct access test point subsystem signals when in the system test configuration. The command subsystem direct access test points are listed in Table 4-1. The d-c isolation of the test points is accomplished by differential line amplifiers and transformers.

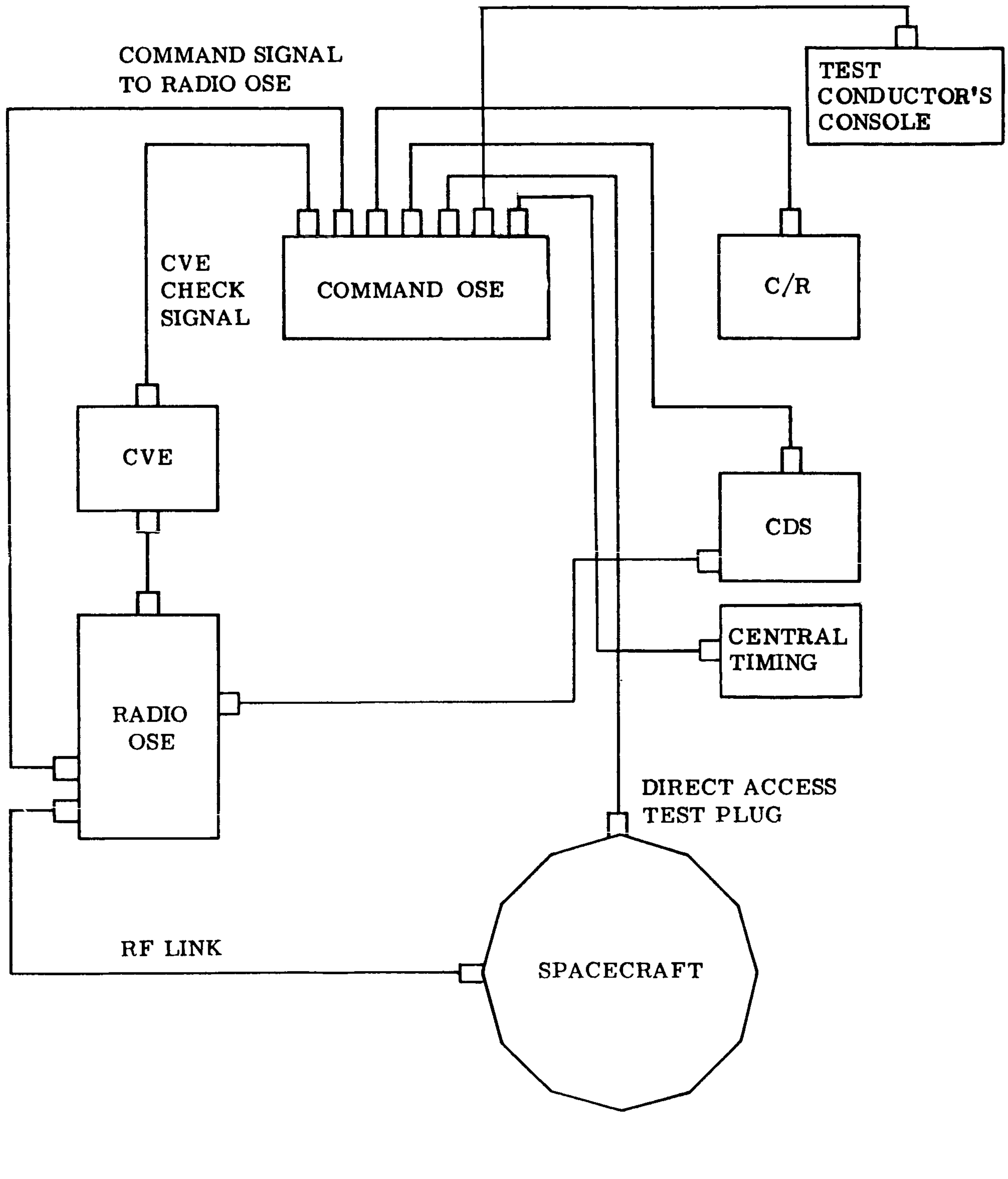
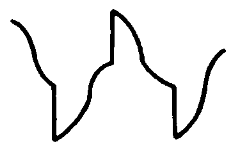
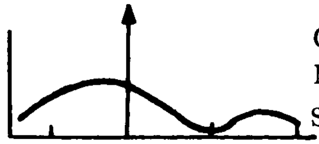


Figure 4-1. Command Interface in STC

Table 4-1. Direct Access Test Points

Signal	Waveform	Characteristics
Detector*		
1. Flight Subsystem Direct Access Test Point Composite Command Signal		FREQUENCY SPECTRUM  The ratio of sync information signal to Command word info subcarrier is $Ps/P_c = 2$.
2. Detector 4 F_s	1v 0v 	Pulse width: 1/8 F_s. Repetition Rate: 4 F_s.
3. Sub-bit Sync	1v 0v 	Pulse width: 2 milleseconds when directed from flight subsystem to command OSE.
	1v 0v 	Repetition Rate: One per second Sync pulse coincident with command word sub-bits.
4. Delayed Sync	1v 0v 	Same as 3 except that delayed sync pulse trails the sync pulse by 100 milliseconds.

*The detector waveforms and characteristics are for the one sub-bit per second detectors. Suitable multiplying factors should be applied for 30 sub-bit per second.

Table 4-1. Direct Access Test Points (Continued)

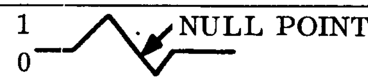
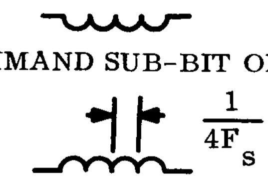
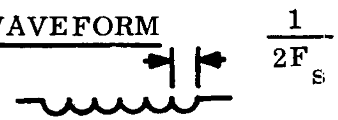
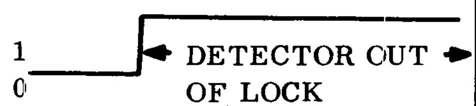
Signal	Waveform	Characteristics
5. Command Sub Bits		Pulse width: One second. Repetition Rate: One per second.
6. Limiter Emitter Follower	Same as 2. Amplitude: 2 v peak-to-peak	Pulse Width: 1/2 F_s. Pulse rate: F_s.
7. Loop Phase Detector	NULL POINT	At null point when detector is in lock.
8. Command Phase Detector		Full wave rectified sinusoidal wave. T = 1/4 F_s.
9. Detector Lock Signal Phase Detector		Full wave rectified sinusoidal wave when detector is in lock. Random noise otherwise. T = 1/2 F_s.
10. Detector Lock Signal		DC level when detector goes out of lock.

Table 4-1. Direct Access Test Points (Continued)

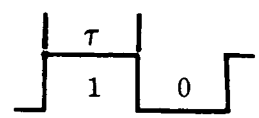
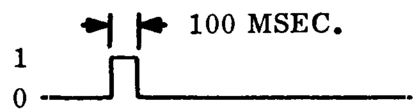
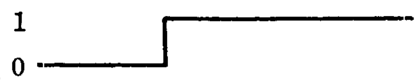
Signal	Waveform	Characteristics
Program Control		
1. Program Counter Set Pulses	Same as 3.	Pulse Width: Two milliseconds. Repetition rate: One per second. Pulses occur when decoder is not processing a command word, when the detector is out of lock.
Decoder Access Unit/ Decoders		
1. DC X (spare)	See Is switch characteristics.	See Is switch characteristics.
2. Event Pulse to Data Encoder	Same as 3. (detector)	Pulse width: 100 milliseconds Pulses occur at decoder matrix interrogation time, and after the last bit of a C&S word.
3. Command Bits	$\tau \approx$ 2 sec. 	Command word bits appear here.
4. QC Bit Sync	Same as 3 (detector)	Same as 2, except that QC sync pulses are coincident with QC command bits directed to CC&S.
5. Decoder Bit Sync.	Same as 4	
6. QC Alert Pulse	Same as 3 (detector)	

Table 4-1. Direct Access Test Points (Continued)

Signal	Waveform	Characteristics
7. Detector Select A		Voltage level
8. Detector Select B		Voltage level
9. Detector Select C		Voltage level
10. End of C&S Word		Pulse occurs at last bit of C&S word.
11. Decoder Inhibit		DC Level when detector goes out of lock.
Power Supply		
1. Test Point No. 20: -v DC level		
2. Test Point No. 21: +v DC level		
3. Test Point No. 22: +28v DC level		

4.3 RADIO OSE INTERFACE WITH COMMAND OSE

The command OSE directs to the radio OSE the command signal through an isolating transformer. The interface is defined in Figure 4-2. The interface characteristics are given below:

a. Command Subcarrier 0-3.3 volts rms (no load).

b. Synchronization signal 0-3.3 volts rms (no load).

c. Combined signals 0-4.0 volts rms (no load).

d. Z_1 minimum 600 ohms ± 100 ohms (resistive).

4.4 CENTRAL RECORDER INTERFACE WITH COMMAND OSE

The command OSE will direct to the central recorder the following signals:

a. The OSE transmitted command word bits.

b. The transmitted command word sub-bits recovered by the detector.

c. The detector lock signal.

Each signal above is directed through digital drivers to the recorder with binary levels at 0 v and +6 vdc. The command OSE output impedance for each signal above is less than 1.5 k ohms.

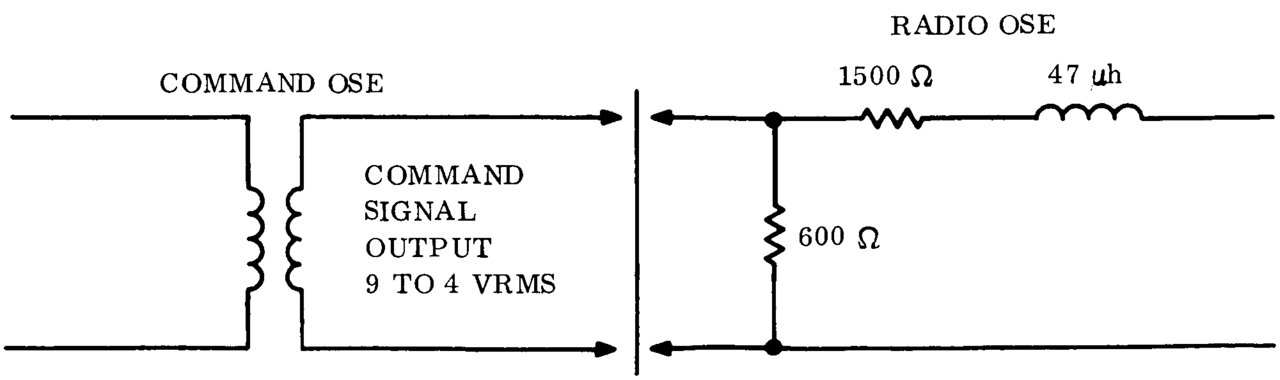


Figure 4-2. Command OSE/Radio OSE Interface

4.5 COMMAND OSE INTERFACE WITH CDS

The command OSE directs to the CDS the signals listed in Table 4-2. The digital signals are directed through digital drivers at 0 and binary levels. The pulse rise times are less than 2 milliseconds. The +6, -6, and +28 vdc voltages are directed to the CDS directly from the flight command subsystems. The digital signals from the flight command subsystem are shaped by the OSE before being directed to the computer. The output impedance of the digital signals is less than 1.5 k ohms.

Table 4-2. Command OSE - CDS Interface

1. OSE command word bits
2. Detector recovered command word bits
3. QC word bits complement
4. OSE 4 F_s frequency 1 bit per second modulator
5. OSE 4 F_2 frequency 30 bit per second modulator
6. Detector 4 F_s frequency
7. Detector lock signals for detectors 1, 2, and 3
8. Decoder inhibit flip-flop
9. Decoder event pulses to TLM
10. Decoder program flip-flop
11. +6 vdc TR supply 1 and 2
12. -6 vdc TR supply 1 and 2
13. +28 vdc TR supply 1 and 2
14. OSE bit sync
15. Detector bit sync 1, 2, and 3
16. QC bit sync
17. QC alert

4.6 CVE INTERFACE WITH COMMAND OSE

The command OSE will interface with the CVE in accordance with GMG-50109-DSN-A and as shown in Table 4-3.

Table 4-3. CVE - Command OSE Interface

1.	Input from CVE	-	command bits.
2.	Output to CVE	-	spacecraft interpretation of command.
3.	Input from CVE	-	comparison of CVE command bits with spacecraft interpretation, positive or negative.

4.7 TEST CONDUCTOR'S CONSOLE INTERFACE WITH COMMAND OSE

The interface is shown in Table 4-4.

Table 4-4. Command OSE - Test Conductor's Interface

1.	Address command bits to Test Conductor prior to transmission. (Switch closures).
2.	Command release from Test Conductor.
3.	Command OSE status and mode indicators (Switch closures).

4.8 CENTRAL TIMING INTERFACE WITH COMMAND OSE

The interface is shown in Table 4-5.

Table 4-5. Command OSE - Central Timing Interface

1.	Digital timing signal from Central Timing

5.0 PERFORMANCE CHARACTERISTICS

The command subsystem OSE must monitor all spacecraft command subsystem functions and supply all stimuli listed in Section 4. The performance parameters associated with these functions are described in Table 4-1.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 MARINER C UTILIZATION

The existing Mariner C Flight Command OSE contains the basic instrumentation necessary to perform the required functions for Voyager. Design modifications necessary or desirable for Voyager are described below:

a. The required modifications include the following:

   1. Provision of a 30 bit/second modulator.

   2. Conversion to the Voyager word format.

   3. Addition of Mariner B type volunteer command error checking.

   4. Expansion of data logging capability.

   5. Addition of multiple detector input simulation of decoder access unit checking.

b. The desirable modifications include the following:

   1. Implementation of logic with integrated circuits to:

      (a) Provide additional test data

      (b) Conserve volume in STC

      (c) Reduce costs.

   2. Redesign programmer to provide decimal reading for uniformity throughout STC.

6.2 PHYSICAL DESCRIPTION

The OSE is housed in a pair of integrally connected standard 76-inch NASA cabinet rack enclosures. The approximate weight of the OSE will be 1700 pounds. The power requirement is approximately 220 watts. The required supply will be 105-125 volts 46-65 cps single phase.

The following commercial and special equipment are housed in these racks:

a. Commercial Equipment

   1. 5" rack-mount Tektronix oscilloscope

   2. 2-channel Brush graphic recorder

3. D-C vacuum-tube voltmeter

4. A-C vacuum-tube voltmeter

5. Frequency counter

6. Random noise generator

7. 2.4 kc, 50 v peak square wave supply

8. Detector function generator

9. Digital printer

b. Special Equipment

1. Command modulators

2. Command word programmer

3. Transmitted address display

4. Central control panel and OSE timing circuitry

5. Detector special test

6. Received address display unit

7. Direct access isolation amplifiers

8. Test point and instrument selectors

9. Blank panel for accepting intercommunication equipment.

c. Holding and Test Fixture - A holding and test fixture for use in environmental testing is provided. This fixture will mate with extender connectors to the Flight Command Subsystem to simulate the ring harness interfaces.

1. Size 20" x 12" x 12"

2. Weight 10 pounds

3. There is no power in this console

4. The following is housed in this console:

   (a) Case harness simulator

   (b) Test cables

d. Quantitative Data Logging - This unit provides the capability for measuring and recording the quantitative characteristics of the isolated switch outputs of the command decoder.

1. This function is housed in a standard 75-inch NASA cabinet rack.

2. This unit weighs approximately 1000 lbs.

3. Approximately 2200 watts of 105-125 volt, 46-65 cps single phase power is required.

4. The following commercial and special equipment are housed in this rack:

   (a) Commercial Equipment

      (1) Digital voltmeter

      (2) Cross bar switches

   (b) Special Equipment

      (1) Input selector

      (2) Timer and control unit

      (3) Power supply

7.0 SAFETY PROVISIONS

7.1 SPACECRAFT INTEGRITY

The OSE will provide at least one megohm of d-c isolation to all direct access test points during system test. Compatible logic is used in subsystem and assembly tests to assure that damaging currents or voltages cannot be applied to the spacecraft subsystem. All cables to the holding fixture, direct access test points or harness are keyed to prevent application of the wrong signal to a given point.

7.2 FACILITIES

The OSE is designed to prevent interference with the proper operation of other equipments in the STC. Radiated and conducted EMI will be within current JPL specs.

7.3 PERSONNEL

No voltages are present within the OSE dangerous to personnel.

CII - VB263FD104

SYSTEM TEST COMPLEX

RELAY RADIO SUBSYSTEM OSE

Index

1.0 SCOPE

This document is a functional description of the Voyager relay radio operational support equipment (OSE) which forms a part of the system test complex (STC) and is used to test the relay radio subsystem during system and subsystem tests.

2.0 APPLICABLE DOCUMENTS

Reference to the following documents is required for completeness in this document:

GE/Motorola

VB233FD104	Functional Description, Relay Radio Subsystem
VB260SR102	System Test Complex Design Characteristics and Restraints
VB260SR101	System Test Complex Test Objectives and Design Criteria

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The Voyager relay radio subsystem OSE will be used to test the relay subsystem from initial subsystem tests through STC tests and will support testing at AFETR.

3.2 BLOCK DIAGRAM

A functional block diagram of the Voyager relay radio subsystem OSE is shown in Figure 3-1.

3.3 REQUIRED FUNCTIONS

The Voyager spacecraft relay radio subsystem OSE is capable of performing the following nine functions:

a. Provide a stable transmitter signal for checking the S/C relay radio receiver performance.

b. Provide for measuring and monitoring r-f powers and frequencies, relay radio subsystem functions, and transformer rectifier (TR) voltages and currents; monitor the relay radio subsystem operating mode and reaction to S/C commands and OSE simulated commands.

c. Provide the relay radio subsystem with a telemetry signal via the r-f link.

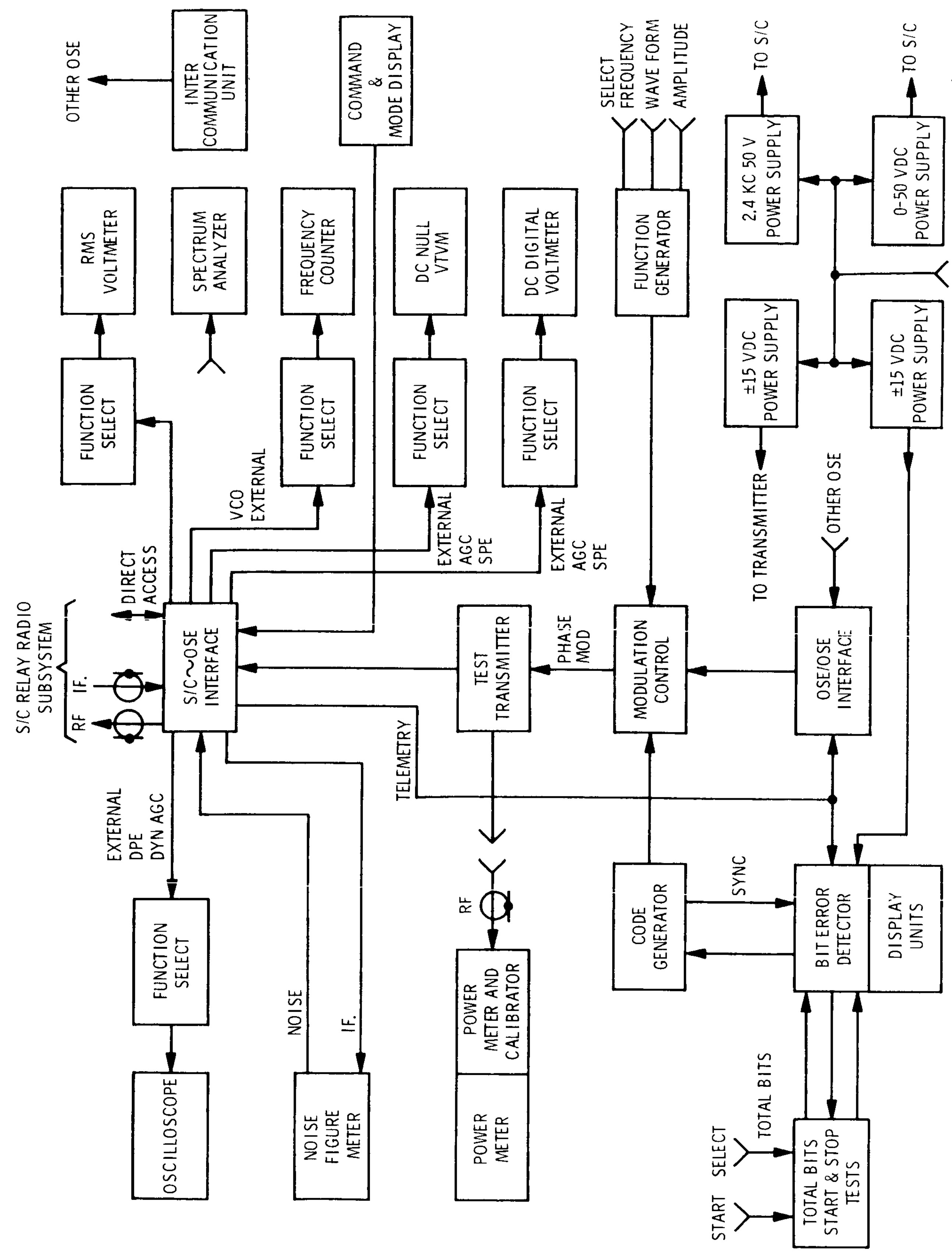


Figure 3-1. Relay Radio OSE Functional Diagram

d. Measure the bit error rate of the relay radio subsystem at baseband and via the r-f link.

e. Provide an alternate source of power to the TR's in the relay radio package when S/C power is not available and provide loads for the S/C power simultaneously.

f. Monitor operating times for the relay radio subsystem and OSE.

g. Provide necessary interfaces.

h. Provide power and signal isolations as described in VB260SR101 and VB260SR102.

i. Provide a self-test capability independent of external equipment and testing.

3.4 GENERAL DESCRIPTION

The following paragraphs contain general descriptions of the equipment contained within the relay radio OSE. The paragraphs are arranged in the order the equipment appears in the rack enclosures. Figures 3-2 is a drawing showing the equipment location within the OSE racks. Both of the OSE racks are used in subsystem and system tests. The case and cable storage console and the S/C power supply rack are used exclusively for subsystem and assembly tests.

3.4.1 RACK I

OSE Rack I will contain the following equipment:

- Breaker, Patch, and OSE timing panel - contains a master circuit breaker for racks I and II a running time meter for racks I and II, a 115-vac convenience outlet, and connectors for interconnections between OSE racks I and II.
- Frequency Counter - monitors OSE and S/C operating frequencies.
- Counter Function Select Panel - functions as the input selector for the frequency counter.
- Oscilloscope - monitors the dynamic outputs of the S/C receiver and TLM demodulator and OSE TLM equipment.
- Oscilloscope Function Select Panel - functions as the input selector for the oscilloscope.

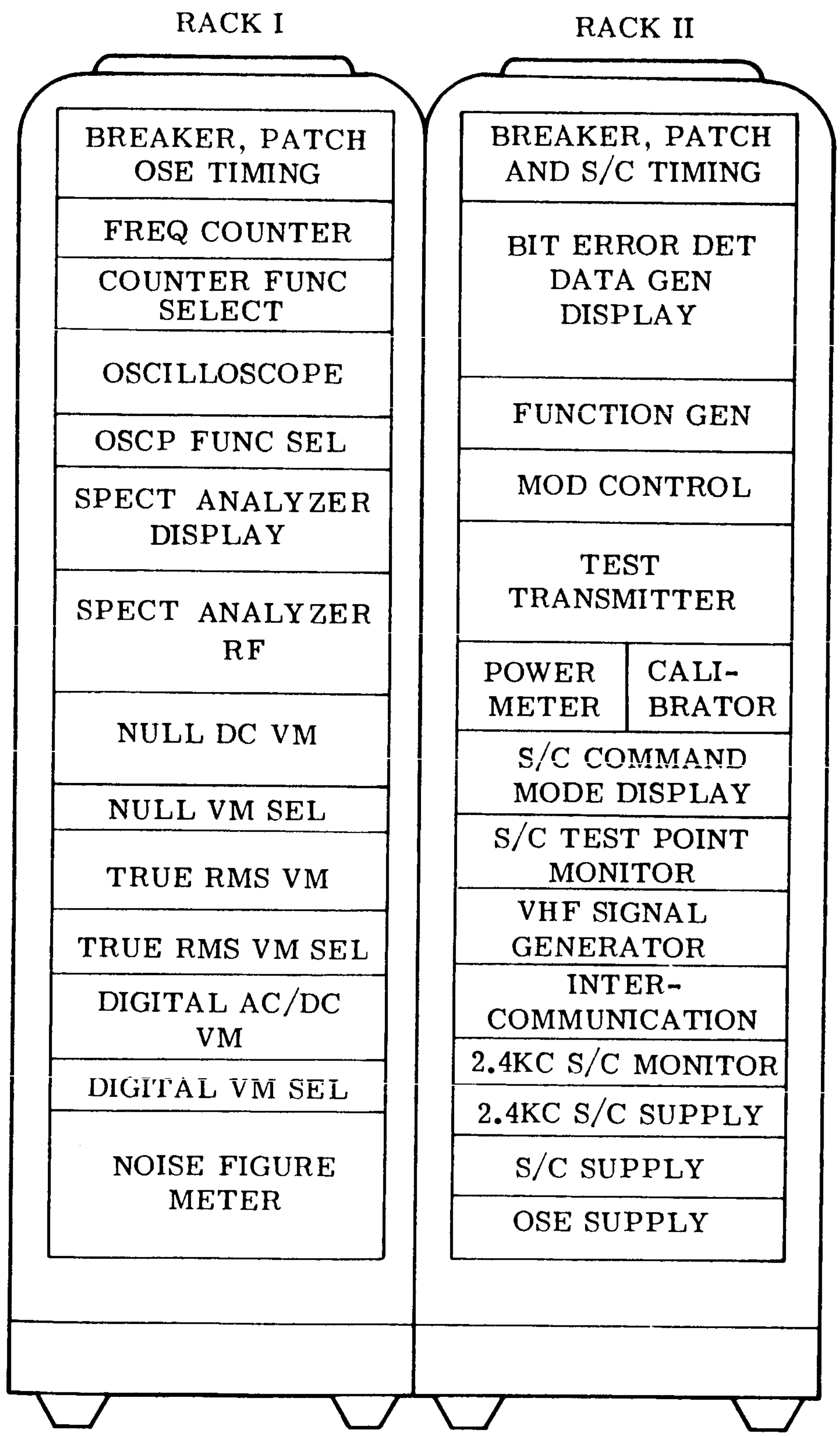


Figure 3-2. Relay Radio OSE

- Spectrum Analyzer - used to monitor frequency spectrums and measure spurious interference, carrier suppression, and other similar functions.
- Null DC Voltmeter - measures various OSE and S/C signal levels.
- Null DC Voltmeter Function Select - functions as an input selector for the null d-c voltmeter.
- True RMS Voltmeter - measures various OSE and S/C signal levels.
- True RMS Voltmeter Function Select - functions as an input selector for the null DC Voltmeter.
- Digital Voltmeter - measures various OSE and S/C signal levels.
- Digital Voltmeter Function Select - functions as an input selector for the Digital Voltmeter.
- Noise Figure Meter - provides for automatic noise figure measurements.

3.4.2 RACK II

OSE Rack II houses the following equipment:

- Breaker, Patch, and S/C Timing Panel - contains a circuit breaker and running time meter for the S/C-relay radio subsystem, a 115-vac convenience outlet, and connectors for interconnection of racks I and II.
- Bit Error Detector, Data Generator, and Display Units - determines the bit error rate of the relay radio TLM subsystem. Figure 3-3 is a simplified block diagram of this unit.
- Function Generator - used to generate square waves, triangular waves and sine waves to modulate the test transmitter and for general testing.
- Modulation Control Panel - used to control the test transmitter modulation input. Selects different input devices and provides amplitude adjustment.
- Test Transmitter - used to test the Relay Radio Receiver. A simplified block diagram of the test transmitter is shown in Figure 3-4.

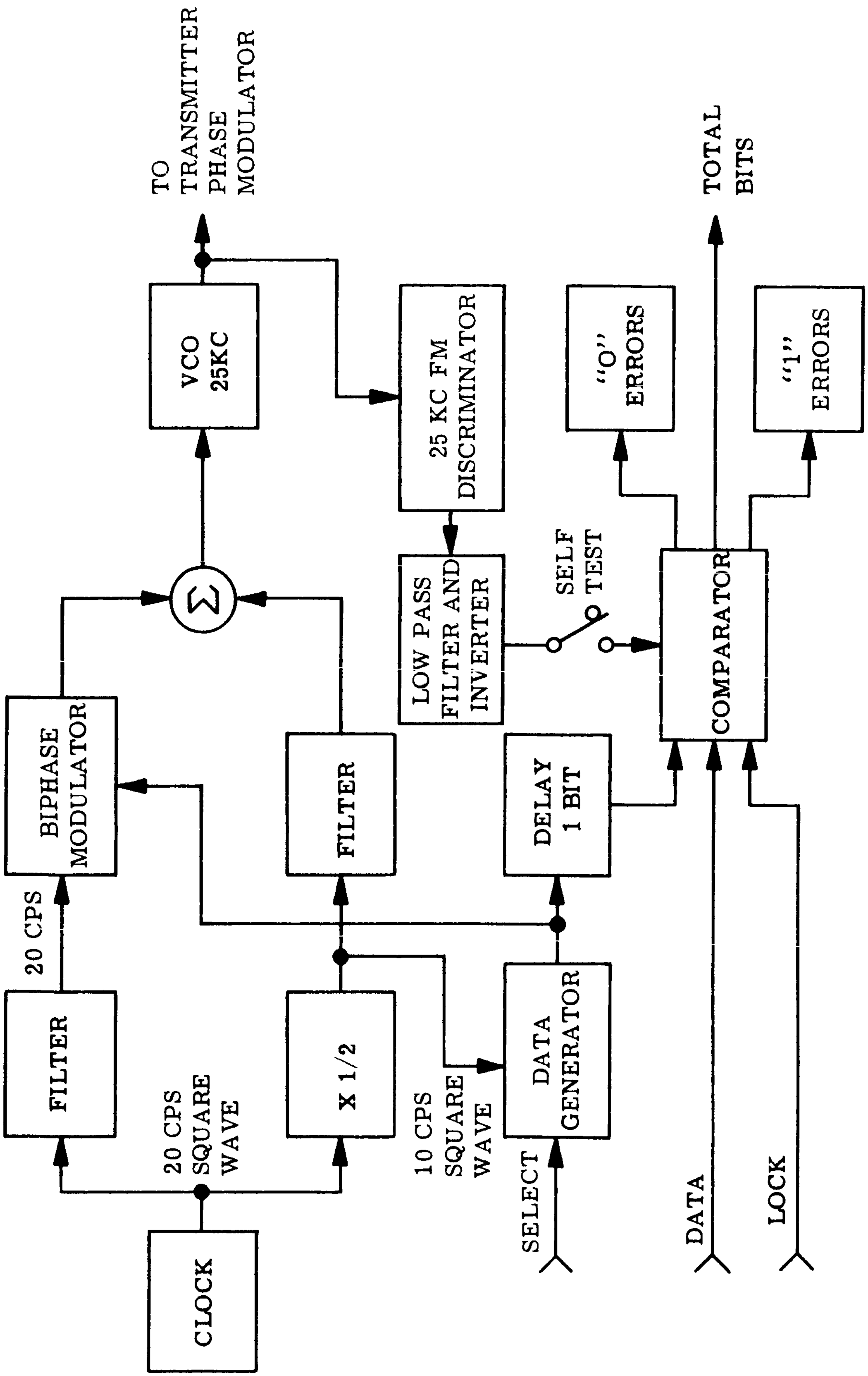


Figure 3-3. Relay Radio OSE Bit Error Detector

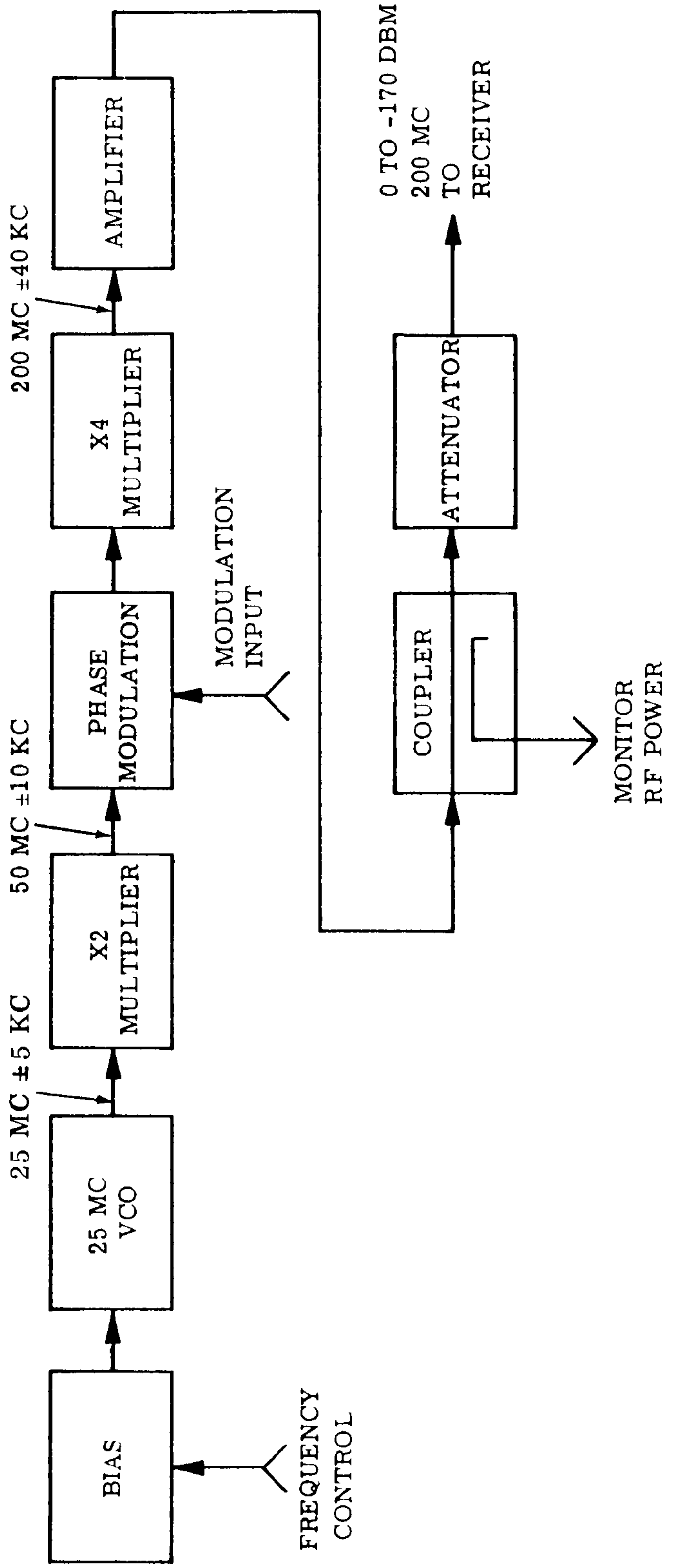


Figure 3-4. Relay Radio OSE Test Transmitter

- R-F Power Meter - monitors the test transmitter output.
- Power Meter Calibrator - will be used to calibrate the R-F Power Meter.
- S/C Command and Mode Display Unit - provides for simulating S/C commands and displaying the relay radio subsystem mode of operation.
- S/C Test Point Monitor - used to select specific test points in the S/C for making various measurements with OSE test equipment.
- VHF Signal Generator - used to test the relay radio receiver spurious responses and image rejection.
- Intercommunication Unit - provides voice communications within the System Test Complex.
- 2.4-kc S/C Monitor Panel - connects the 2.4-kc S/C power supply to the S/C Interface panel and provides selection of the 2.4-kc output voltages for application to the digital ac/dc voltmeter.
- 2.4-kc S/C Power Supply - will supply 50 v rms at 2.4 kc to the S/C relay radio subsystem.
- 50-vdc S/C Power Supply - used to supply 28 vdc nominal to the S/C dc-dc converters in the relay radio subsystem.
- OSE Power Supply - will supply regulated ±15 vdc to the test transmitter and modulator control unit.
- OSE Power Supply - will supply regulated ±15 vdc to the bit error detector and data generator.
- Isolation Amplifier Unit - provides circuit isolations for S/C monitored functions.
- OSE to OSE interface (rear) Panel - connects the relay radio OSE contained functions and S/C functions to other OSE.
- S/C Connector Panel (rear) - used for all interconnections between OSE and S/C.

3.4.3 S/C POWER SUPPLY RACK

This rack may be used during subsystem tests to replace the radio subsystem transformer/rectifier and dc-dc converter units.

3.4.4 HOLDING AND CABLE STORAGE CONSOLE

This unit is used during subsystem tests to house and provide connectors and cabling to the subsystem under test.

3.5 SELF TEST AND CALIBRATION

3.5.1 GENERAL

The Voyager Relay Radio Subsystem OSE has self-test and calibration capibilities prior to and during external testing without test interference. These functions are performed by equipments internal to the Basic OSE racks.

3.5.2 BLOCK DIAGRAM

A functional block diagram of the OSE self-test mode is shown in Figure 3-5. Table 3-1 is a list of the self-test capabilities to be provided by the OSE.

Table 3-1. Self-Test Capability

Functional Equipment	Monitor/Measure
1. OSE/OSE-S/C Power Supplies	Voltages/Current, Frequencies
2. Running Time Indicators	OSE and S/C
3. Test Transmitter	a. Output power b. Frequency c. Modulation d. Modulated Spectrum
4. Bit Error Detector	a. Code b. Decision Process c. Total bits d. Errors e. Sync

3.5.3 TEST DEFINITIONS

The following paragraphs provide a brief description of the self test capabilities presented in Table 3-1.

a. Power Supplies - OSE and OSE contained S/C d-c power supply voltages and currents are monitored with meters mounted on the power supply faces and with a digital voltmeter. The 2.4-kc S/C monitor panel provides for selecting any 2.4-kc output voltages or currents to the S/C for monitoring by the digital ac-dc voltmeter. The output frequency is also monitored by the electronic frequency counter.

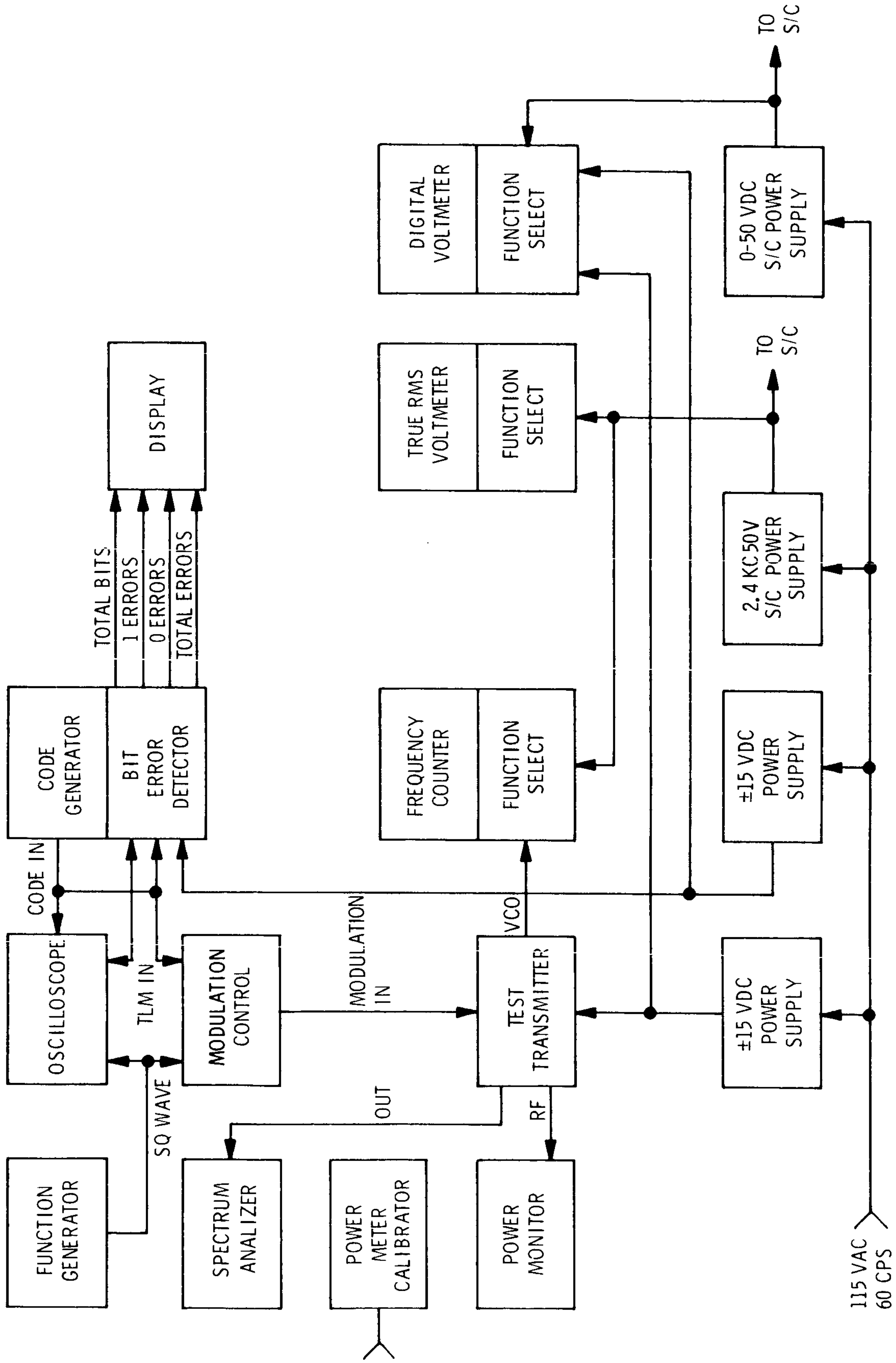


Figure 3-5. Relay Radio OSE Self Test Mode

b. Running Time Indicators - Elapsed Time indicators are supplied with both racks and record S/C and OSE operating times.

c. Test Transmitter - Output power, vco frequency, and output spectrum are monitored. Output power is measured with the r-f power meter and bolometer. A power meter calibrator is provided to ensure the accuracy of these measurements. Test transmitter vco frequency is monitored with the frequency counter by selecting from the counter function select panel. Modulated spectrum is monitored by the spectrum analyzer tuned to the 200 MC/S carrier frequency. Modulation sensitivity and carrier suppression are measured in the same manner. This same basic test setup is utilized to measure and monitor other similar test transmitter operations and functions.

d. Bit Error Detector - The bit error detector will function in a direct test mode to verify proper operation. In addition, internal test points will be monitored by the oscilloscope to verify proper operation during external testing.

3.6 SUBSYSTEM TESTS

3.6.1 GENERAL

Tests will be performed with the two basic relay radio subsystem OSE racks, the power supply rack, and the case and cable storage console. The tests will confirm the adequacy of the subsystem design, workmanship, and electrical characteristics to meet system requirements. Table 3-2 lists the tests to be performed on each assembly within the relay radio subsystem.

3.6.2 ASSEMBLY MODE

It is noted that the tests of Table 3-2 will be performed both on a subsystem and assembly level. The assembly level tests are conducted with the assembly connected within the holding and cable storage console.

Connectors and cables are provided with this unit and additional ancillary connectors and cables are used where necessary to operate one assembly independent of the others.

3.6.3 TEST DESCRIPTIONS

The following paragraphs present brief descriptions of the test capabilities listed in Table 3-2.

a. Receiver Tests - Figure 3-6 shows the relay receiver assembly test setup. For subsystem level testing the complete relay radio subsystem is included

Table 3-2. Test Capabilities

Assembly	Test/Measurements
Receiver	a. Power supply voltages/currents
	b. Command Responses
	c. Preamplifier Noise Figure
	d. Threshold
	e. R-F Loop Bandwidth
	f. Doppler Tracking Rate
	g. AGC Characteristics
	h. Image Rejection
	i. Spurious Responses
	j. AGC Loop Bandwidth
	k. Phase Jitter
	l. Amplitude Jitter
	m. Coherent Interference
	n. Inter-Channel Cross Coupling
	o. TLM Output Levels and Signal-to-Noise Ratios
	p. Sweep Acquisition
	q. Monitor Various Receiver Functions and Test Points
TLM Demodulator	a. Power Supply Voltages/Currents
	b. Command Responses
	c. Bit Errors
	d. Monitor Decision Process and Various Test Points and Functions

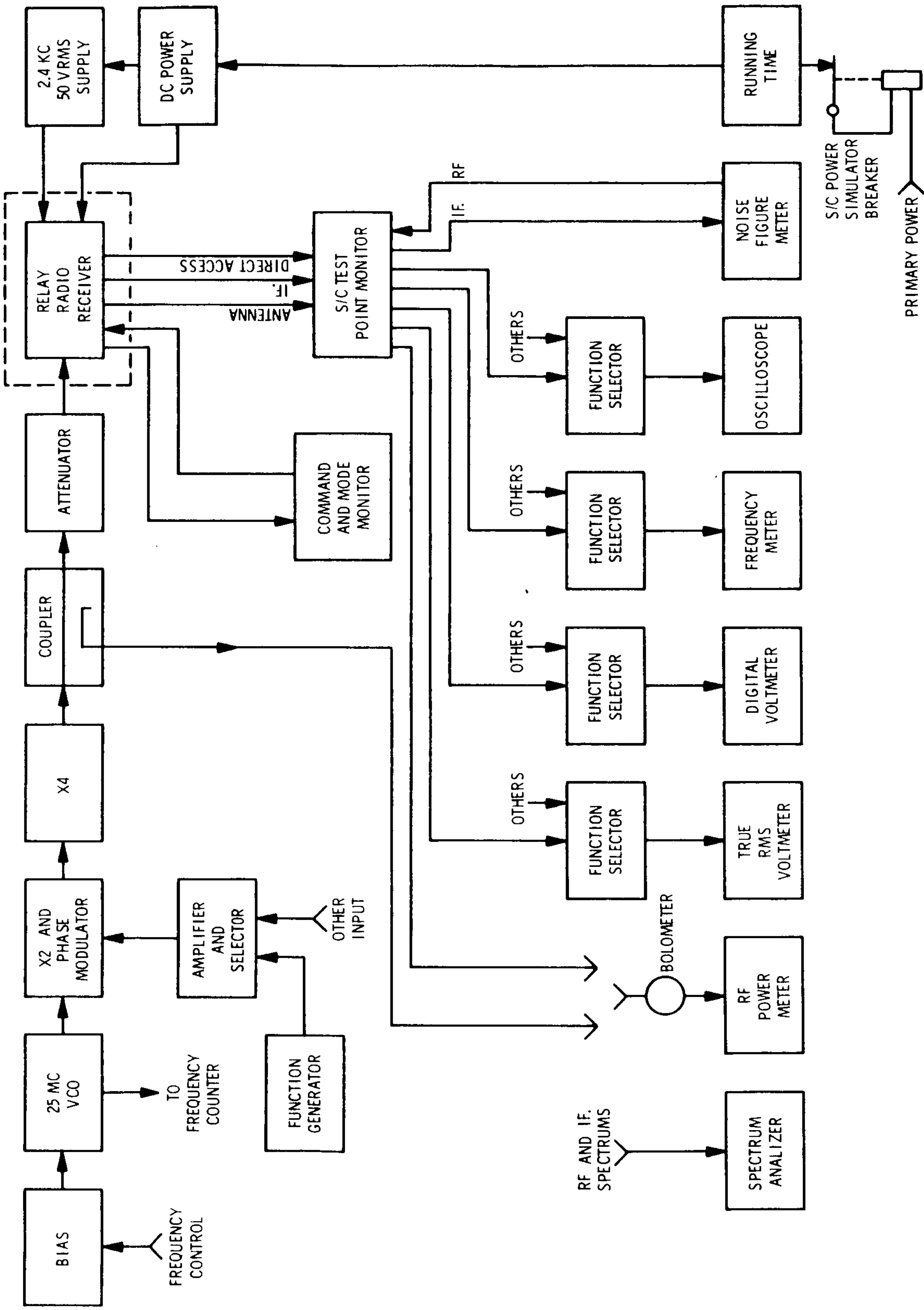


Figure 3-6. Relay Radio Receiver Assembly Tests

in the test setup. However, the receiver tests are essentially the same for both assembly and subsystem testing. The following paragraphs define the receiver tests of Table 3-2.

1. Power supply voltages/current are measured using the digital voltmeter. The test points are available from the Digital Voltmeter Function Select panel.

2. Commands are simulated and reactions are monitored from the Command/Verify and Mode Indicator Panel.

3. Preamplifier Noise Figure is measured using the automatic noise figure meter.

4. Receiver Threshold is measured using the OSE test transmitter and monitoring the relay receiver RF lock.

5. R-F Loop Bandwidth is measured by modulating the test transmitter with an appropriate signal and monitoring the receiver loop response at the phase detector output with the oscilloscope.

6. Maximum Doppler Tracking Rate is determined by modulating the test transmitter VCO with a triangular wave from the function generator of the appropriate amplitude and monitoring the receiver DPE with an oscilloscope. Maximum Doppler rate will be indicated by a loss of lock.

7. Relay Receiver AGC Characteristics are determined using the test transmitter as a calibrated source and monitoring AGC voltage with the digital voltmeter.

8. Image Rejection is measured using the VHF signal Generator provided with the OSE and monitoring the receiver intermediate frequency with the spectrum analyzer.

9. Spurious Responses are measured in the same manner as Image Rejection.

10. AGC Loop Bandwidth is measured by modulating the test transmitter with an appropriate low frequency and with the relay receiver r-f loop locked monitoring the dynamic AGC with the oscilloscope. The response versus modulating frequency describes the loop bandwidth.

11. Receiver/Test Transmitter Phase Jitter is measured by monitoring the receiver rms noise in the loop phase detector output with the general purpose oscilloscope.

12. Amplitude Jitter is measured by monitoring the receiver rms noise in the AGC loop phase detector output with the oscilloscope.

13. Coherent Interference is measured with the relay receivers r-f input terminated in 50 ohms and monitoring the open loop SPE with the d-c digital voltmeter. Coherent interference is indicated by obtaining an "S" curve when plotting SPE output versus phase shift of the loop phase detector reference.

14. Inter-Channel Cross Coupling between the two relay receivers is measured by exciting a channel with an RF signal and monitoring the signal output from the second channel.

15. TLM Output Levels and Signal-to-Noise Ratios are measured using the equipment internal to the OSE.

16. The relay receiver Sweep Acquisition is investigated and acquisition characteristics determined using the test transmitter and monitor capabilities of the OSE.

b. TLM Demodulator Tests - Figure 3-7 shows the fundamental assembly test concept for testing the relay TLM demodulators. For subsystem level testing the complete relay radio subsystem is included in the test setup. However, the same tests are conducted for both levels. The following paragraphs define the TLM Demodulator tests of Table 3-2.

   1. Power Supply Voltages/Currents are measured using the digital voltmeter. The desired test points are available from the digital voltmeter function select panel.

   2. Commands will be simulated and reactions are monitored from the Command/Verify and Mode Indicator Panel.

   3. Bit Errors are measured directly using the bit error detector and data generator.

   4. Various Test Points are monitored to establish performance levels and ensure proper operation. The S/C monitor unit will provide this monitor capability.

3.6.4 SUBSYSTEM MODE

A functional block diagram showing the principal signal flow through the Relay Radio OSE is shown in Figure 3-8. Subsystem testing will subject the relay radio subsystem to a series of formal tests verifying the overall subsystem performance. Quantities to be measured will include sufficient assembly testing to ensure their individual performances.

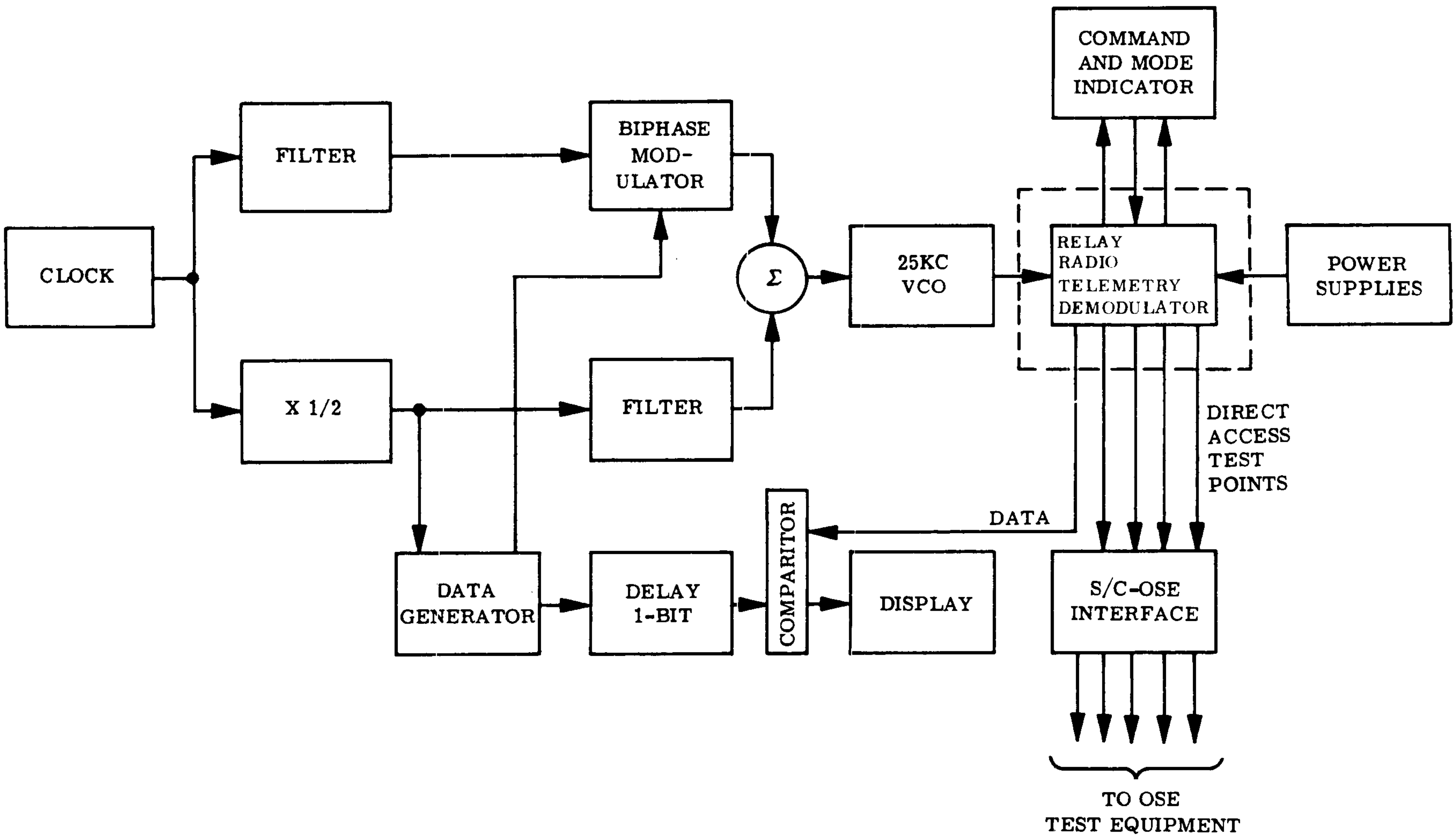


Figure 3-7. Relay Radio OSE Telemetry Demodulation Tests

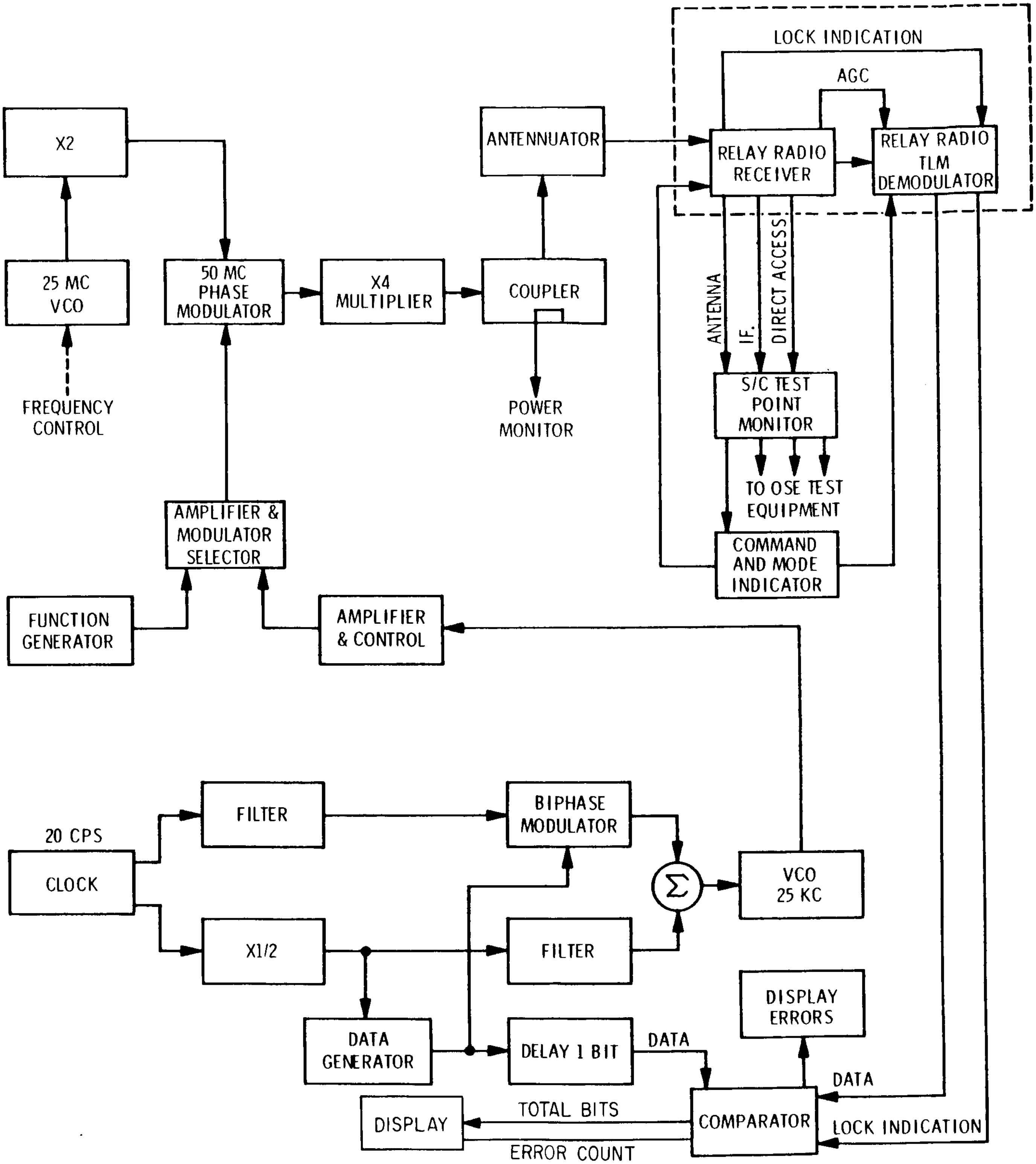


Figure 3-8. Relay Radio OSE Subsystem Tests

a. Test Descriptions - Subsystem level tests include Bit Error Rates via the r-f link. This test setup is illustrated in Figure 3-8. A brief description of subsystem level tests is given by the following:

1. The Bit Error Detector/Data Generator unit phase modulates the test transmitter with a 25-kc subcarrier FM modulated by a bi-phase modulated 10 cps signal summed with 20 cps sync. The data generator provides a random data stream to the bi-phase modulator and comparator.

2. The Modulation Control Unit is used to adjust the deviation of the phase modulator and select the desired input function.

3. The Test Transmitter provides an output signal at the receiving frequency tunable ±40 kc with an adjustable level from 0 to -170 dbm.

4. With the relay radio receiver locked to the test transmitter and the relay radio TLM demodulator interconnecting with the receiver, the bit error rate is measured by comparing the data from the demodulator with the generated data stream.

5. Additionally, sufficient investigations will be performed to ensure the harmonious integration of the various assemblies and subsystem performance proficiency.

3.7 SYSTEM TESTS

These tests are performed on the complete S/C system. The test complex utilized to perform system tests will be capable of the following:

a. Permit complete exercising of all spacecraft mechanical and electrical functions.

b. Afford monitoring in real time of the spacecraft subsystem behavior.

c. Afford test repeatability.

d. Have self-check capability prior to and during test without test interruption.

e. The CDS will provide real-time test records with the following information:

1. Name of test

2. Function exercised

3. Time

4. Edited data with indications of questionable data.

f. Provide for detection of failures. The fault isolation need only to be a subsystem level.

g. Provide a record of accumulated test time on over-all spacecraft equipments.

A functional block diagram of the relay radio subsystem OSE and interfaces with spacecraft relay radio subsystem and other OSE is shown in Figure 3-9. The Relay Radio Subsystem OSE will form an integral part of the STC and will provide measurement and monitor capabilities including, but not limited to those listed for self tests, assembly tests, and subsystem tests.

The STC will provide for integrated test control using the CDS and command verification equipment. The Relay Radio Subsystem OSE will function in this mode.

The Relay Radio Subsystem OSE in conjunction with other OSE within the STC supports the operations of the LCE at the launch complex and explosive safe area.

4.0 INTERFACE DEFINITIONS

Relay Radio OSE to OSE Interfaces are shown in Figure 3-9. The following paragraphs delineate the interfaces:

Table 4-1 shows the Relay Radio OSE and Computer Data System Interface.

Table 4-2 shows the Relay Radio OSE and Central Recorder Interfaces.

Table 4-3 Relay Radio OSE and External Recorder Interfaces.

Table 4-4 shows Relay Radio OSE and Capsule Radio OSE Interfaces.

Table 4-5 shows Relay Radio OSE and Relay Radio subsystem interfaces.

Table 4-6 shows Relay Radio OSE and Test Conductor's console interface.

5.0 PERFORMANCE CHARACTERISTICS

The Relay Radio Subsystem OSE provides the measurement/control capability listed below with the accuracy indicated for each parameter. This performance proficiency is provided for all modes of use: assembly, subsystem and system testing.

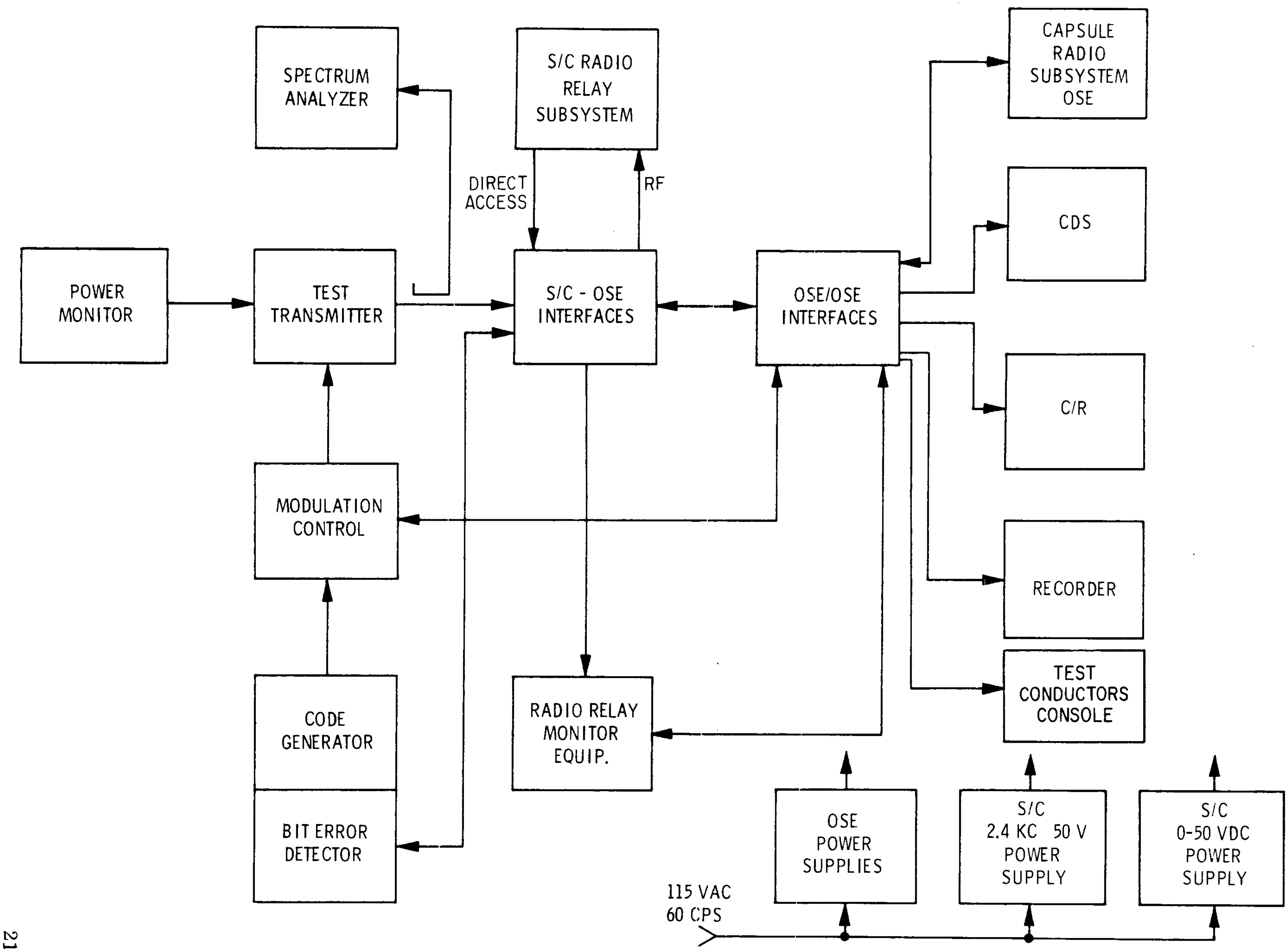


Figure 3-9. Relay Radio System Tests

Table 4-1. Relay Radio OSE and CDS Interface

Function	Voltage	Impedance ohms	Comments
1. Relay RCVR 1 AGC	-1.5 to +5.5 vdc	10 K	Alarm limits
2. Relay RCVR 2 AGC	-1.5 to +5.5 vdc	10 K	Alarm limits
3. Relay RCVR 1 SPE	±3.0 vdc	1	DC Iso-Amp Output
4. Relay RCVR 2 SPE	±3.0 vdc	1	DC Iso-Amp Output
5. Relay RCVR's TR Voltage	±15 vdc	15 K	Alarm 1% Change
6. Relay TLM Demodulator TR Voltage	±15 vdc	15 K	Alarm 1% Change
7. GRD Test Transmitter RF Power Out	0 to +1 vdc ±2%	1 K	Monitor from Power Meter Output
8. Command Requests	—	—	Switch Closures

Table 4-2. Relay Radio OSE and Central Recorder Interfaces

Function	Voltage	Impedance ohms	Comments
1. Relay RCVR 1 AGC	-1.5 to 5.5 vdc	10 K	
2. Relay RCVR 2 AGC	-1.5 to 5.5 vdc	10 K	
3. Relay RCVR 1 DPE	±4.0 vac	10 K	Phase Det. Output
4. Relay RCVR 2 DPE	±4.0 vac	10 K	Phase Det. Output
5. Relay RCVR 1 SPE	±3.0 vdc	1	Iso-Amp Output
6. Relay RCVR 2 SPE	±3.0 vdc	1	Iso-Amp Output
7. Test Transmitter RF Power Output	0 to 1 vdc ±2%	1	Power Meter Monitor

Table 4-3. Relay Radio and External Recorder Interfaces

Function	Voltage	Impedance ohms	Comments
1. RCVR 1 AGC	0-3 vdc	10 K	
2. RCVR 2 AGC	0-3 vdc	10 K	
3. RCVR 1 SPE	±1.5 vdc	1 K	
4. RCVR 2 SPE	±1.5 vdc		
5. RCVR 1 DPE	±4 vac	10 K	
6. RCVR 2 DPE	±4 vac	10 K	
7. Test Transmitter RF Output	0 to 1 vdc	1	
8. TLM Demod. Output	Bit Stream		

Table 4-4. Relay Radio OSE and Capsule Radio OSE Interface

Function	Voltage	Impedance	Comments
To be defined following definition of Capsule Radio OSE.			

Table 4-5. Relay Radio OSE and Relay Radio Subsystem Interfaces

Function	Voltage	Impedance ohms	Comments
Direct Access Test Points			
AGC RCVR 1	-1.5 to 5.5 vdc	10 K	
AGC RCVR 2	-1.5 to 5.5 vdc	10 K	
SPE RCVR 1	±3 vdc	1	
SPE RCVR 2	±3 vdc	1	
DPE RCVR 1	±4 vac	10 K	
DPE RCVR 2	±4 vac	10 K	
T/R Voltages	+15 vdc	15 K	
(RCVR 1)	-15 vdc	15 K	
T/R Voltages	+15 vdc	15 K	
(RCVR 2)	-15 vdc	15 K	
T/R Voltages	+6 vdc	15 K	
(TLM Demodulator 1)	-6 vdc	15 K	
T/R Voltages	+6 vdc	15 K	
(TLM Demodulator 2)	-6 vdc	15 K	

Table 4-5. Relay Radio OSE and Relay Radio Subsystem Interfaces (Continued)

Function	Voltage	Impedance ohms	Comments
Direct Access Test Points			
Output Selector	+6 vdc		
Output Selector	-6 vdc		
Lock Indication			
Data			
Sync			
RF Input			
Receiver input*		50	

*When the antenna is installed and connected, a probe provides for RF coupling through the umbilical.

Table 4-6. Relay Radio OSE and Test Conductor's Console Interface

Function	Voltage	Impedance Ohms	Comments
Relay Radio OSE Status and Mode Indicators			Switch closures

5.1 PARAMETERS, VALUES, RANGES AND ACCURACIES

a. D-C Voltages - The OSE will have the capability of measuring voltages to ±1 microvolt. The meter has a 20 db noise rejection capability that will enable accurate measurements even under noisy signal conditions (where this accuracy is not required a higher input impedance voltmeter will be available).

b. A-C Voltages - Average-responding rms calibrated measurements are provided to an accuracy of 1% of full scale from 50 cps to 1 mc with a resolution of 1 mv.

c. Receiver Phase Stability - Residual phase modulation is measureable to an accuracy better than 1/2 degrees peak.

d. R-F Measurements - All r-f interconnections such as cables, attenuators, and directional couplers are calibrated to ±0.1 db. All r-f power readings are accurate to ±1/2 db on both transmitted and received r-f signals to -100 dbm and ±1 db to threshold.

e. Frequency Measurements - All frequency measurements up to 350 Mc are accurate to 1 part in 10^8 (or ±1 cps up to 350 mc) at signal levels of -25 dbm or greater.

f. Spectral Analysis - Frequency spectrum relative power measurements are accurate to ±1.5 db with a frequency resolution down to 1 kc.

g. Static Phase Error (SPE) - The OSE monitors the relay receiver SPE and VCO frequency.

h. AGC, S/C Receiver - The OSE monitors the S/C receiver coarse and fine AGC.

i. Local Oscillator Drive - The OSE monitors the S/C receiver local oscillator drive.

j. Reference Oscillator Frequency - The OSE will monitor the reference oscillator frequency.

k. Receiver Frequency Response - The amplitude versus frequency characteristics of the relay receiver will be measured at the output of the phase detector.

l. Modulation Characteristics - Phase detector linearity and sensitivity are measurable by the OSE.

m. Power Supplies - The OSE monitors the power supply operating currents/voltages and provides an alarm/fault indication.

n. TLM Performance - The OSE will test and verify the S/C Telemetry Demodulator performance. Bit error rate is measured and test points are monitored.

o. Coherent Interference - Will be determined using the OSE.

p. Amplitude Jitter - Will be determined using the OSE.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 RELAY RADIO SUBSYSTEM OSE

The Voyager Relay Radio Subsystem OSE consists of two integrally connected standard 75-inch high cabinet rack enclosures. Rack I houses commercial test equipment, and Rack II houses the test transmitter, TLM test equipment, S/C monitor and command equipment and power supplies. The racks accept standard 19-inch wide panels. Figure 3-2 shows the two racks with the recommended equipment location.

6.1.1 COOLING

Cooling is provided for each rack by two fan assemblies, one mounted on rack top and the second on the rear door. The combination of these fans provides an internal positive pressure cooling system.

6.1.2 WEIGHT

The approximate weight of the OSE is 1900 lbs; 800 lbs for Rack I and 1100 lbs for Rack II.

6.1.3 POWER

The power requirements are approximately 2400 watts: 1100 watts for Rack I and 1300 watts for Rack II. The primary supply 105-125 volts, 46-65 cps, single phase. The equipment of each enclosure is isolated from the primary source by quadruple shielded isolation transformers.

6.1.4 COMMERCIAL AND SPECIAL EQUIPMENT

Relay Radio OSE Rack I contains the equipment listed in Table 6-1 or its equivalent.

Relay Radio OSE Rack II contains the equipment listed in Table 6-2.

6.2 SPACECRAFT POWER SIMULATOR RACK

This rack is used to supply power to the relay radio subsystem when its transformer/rectifier units are not in use during testing, trouble shooting, or failure analysis. The unit is housed in a standard 75-inch high NASA cabinet rack.

6.2.1 WEIGHT

This unit weighs approximately 600 pounds.

Table 6-1. OSE Rack I Assemblies

Title	Mfr.	Part No.
Breaker, Patch OSE Timing	Motorola	
Frequency Counter	HP	5245L
w/Plug-In Unit	HP	5252A
Frequency Counter Function Select	Motorola	
Oscilloscope	HP	130C
Oscilloscope Function Select	Motorola	
Spectrum Analyzer Display	HP	851A
Spectrum Analyzer RF Unit	HP	8551A
Null D-C Voltmeter	HP	413A
Null D-C Voltmeter Select	Motorola	
True RMS Voltmeter	HP	3400A
True RMS Voltmeter Function Select	Motorola	
Digital Voltmeter	HP	3440A
w/Plug-In Unit	HP	3445A
Digital Voltmeter Function Select	Motorola	
Noise Figure Meter	HP	340B (modified)
w/Noise Source	HP	343A

Table 6-2. OSE Rack II Assemblies

Title	Mfr.	Part No.
Breaker, Patch and S/C Timing	Motorola	
Bit Error Detector, Data Generator and Display	Motorola	
Function Generator	Exact Electronics	250RM
Modulation Control Unit	Motorola	
Test Transmitter	Motorola	
Power Meter w/Bolometer	HP HP	431B 478A
Power Meter Cal.	HP	8402A
S/C Command and Mode Display	Motorola	
S/C Test Point Monitor	Motorola	
VHF Signal Generator	HP	3200A
Intercommunication Unit	Motorola	
2.4-kc Power Supply Monitor Unit	Motorola	
2.4-kc Power Supply	IDS	
Two S/C DC Power Supplies	Lambda	LH 128FM
Four OSE DC Power Supplies	Lambda	LH 124FM
Isolation Amplifiers	DIC	6122
*S/C to OSE & Primary Power *Connectors (Rear)	Motorola	
*Ose to OSE & Primary Power Connectors (Rear)	Motorola	

*Located in rear of rack.

6.2.2 POWER

Approximately 1500 watts of 105-125V, 45-65 cps, single-phase power will be required.

6.3 HOLDING AND TEST FIXTURE

A holding and test fixture for use in subsystem tests will be provided. This fixture will mate with extender connectors to simulate the ring harness and antenna interfaces.

6.3.1 WEIGHT

The weight of this unit will be approximately 650 pounds.

6.3.2 POWER

There is no power to this console.

6.3.3 EQUIPMENT

The following will be housed within this console:

a. Top mounting pad with cover

b. Cables on spool (with ring harness, direct access, and r-f cables making one composite cable).

c. Vacuum and temperature adapter cables and miscellaneous adapter cables.

7.0 SAFETY CONSIDERATIONS

7.1 GENERAL

The relay radio subsystem OSE conforms to good engineering practice providing circuit protection and safety devices such as current overload relays and voltage stabilization devices. The OSE will present no hazards to personnel. Safety features such as cabinet interlocks for cabinets operating with greater than 120 volts, high voltage point isolations and warnings, and proper weight distribution to prevent tipping resulting in damage to equipment and possible injury to personnel will be utilized to the fullest extent.

7.2 SPACECRAFT INTEGRITY

The OSE is designed to interface safely with any Over-All Flight S/C and to eliminate any possibility of damage to the S/C radio subsystem and other OSE. The OSE will provide self-check capabilities on all "launch hold" criteria included in the STC. Self-checks will be on a noninterfering test basis.

CII - VB263FD105

SYSTEM TEST COMPLEX

DATA STORAGE SUBSYSTEM

Index

1.0 SCOPE

This document contains a functional description of the Voyager STC Data Storage OSE. The OSE is for use in systems and subsystems testing.

2.0 APPLICABLE DOCUMENTS

GE

VB 260 SR 101 1971 Voyage Spacecraft STC Test Objectives and Design Criteria

VB 260 SR 102 1971 Voyage Spacecraft STC Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The purpose of the Data Storage OSE (DS OSE) is to control and monitor the S/C DS subsystem during subsystem and system tests. The S/C DS equipment consists of the magnetic core memories (MCM), the magnetic tape recorders (MTR), the capsule relay buffers (CRB) and their associated power and control circuitry. The DS OSE performs its function by having the capability of simulating all inputs, output loads, and control functions and monitors and displays the various outputs to determine proper system operation. The OSE has the capability of performing bit error checks on the output data to determine that the S/C DS subsystems are operating within specification. Figure 3-1 is a block diagram of the DS OSE; its functional blocks are described below.

3.1.1 DATA SIMULATOR

The data input simulator duplicates all input interfaces to the DS system and generates a simulated data stream to be stored in either the MCM's, MTR's, or CRB's. The interfaces simulated are:

a. Bus Engineering data

b. Capsule data

c. Non-scanned science data

d. Scanned science data

The unit is designed so that the data transmitted to the spacecraft DS subsystem may be generated again and routed to the bit error checker for comparison with the stored data as it is recovered. Operation of this unit is controlled by the control unit.

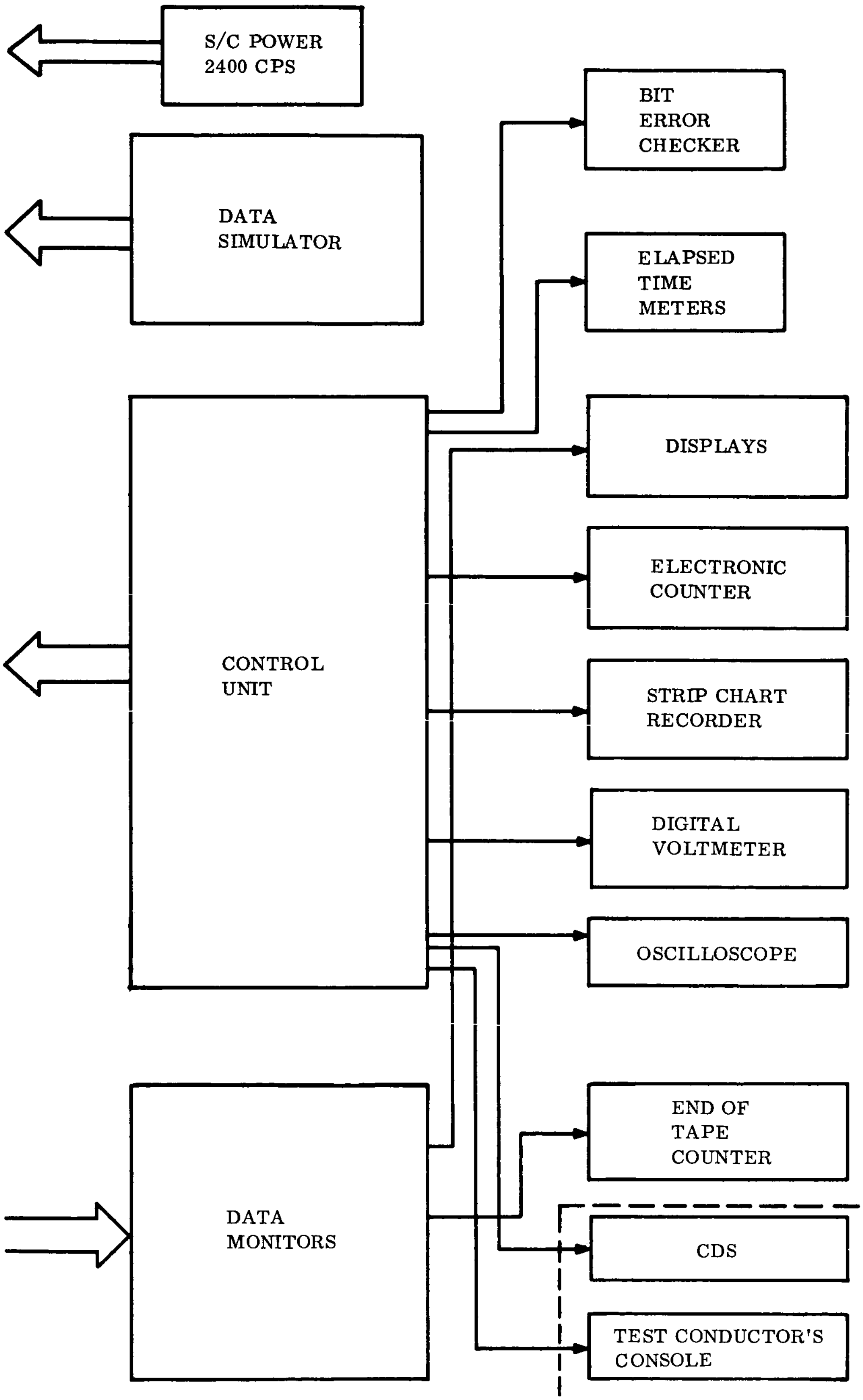


Figure 3-1. Data Storage OSE

3.1.2 CONTROL UNIT

The control unit controls the DS OSE in both the subsystems and systems test modes of operation. Basically, it will select modes of operation and will route data from the Data Monitor to the various display units and the bit error checker. It also contains all control circuitry required for testing the S/C DS subsystem control units through the various operational modes.

3.1.3 DATA MONITORS

This unit simulates all system output loads for the flight DS system. It also contains a selection capability to look at hardwire test points brought out during the subsystem and systems tests. Data conditioning necessary for data display and recognition of the preamble code is done in this unit. Operational control of the Data Monitor unit is maintained by the DS OSE control unit.

3.1.4 BIT ERROR CHECKER

The bit error checker provides a means of comparing data received from the S/C DS subsystem with a reconstructed data word identical to that transmitted to the system from the DS OSE during subsystem test. This allows verification of proper operation of the storage devices as well as establishing the bit error rates.

3.1.5 DATA DISPLAY

The data display consists of two types of equipment. These are commercial test equipment and status monitors. The commercial test equipment is used primarily for accurate measurements during the subsystem test phase and some monitoring during the systems test phase. The status displays are used primarily during the systems test as GO-NO-GO indications of the S/C DS subsystem condition. The display unit displays at least the measurement points of the following functions:

a. <u>Magnetic Tape Recording (MTR) Section</u>

Start/Stop (Each MTR)
Record/Playback (Each MTR)
Data Input
Data Output (Each MTR)
PLL Error (Each MTR)
Selected Channel Record Waveform (Each MTR)
Selected Channel Playback Waveform (Each MTR)
2 Motor Currents

b. Magnetic Core Memories (MCM) Section

Dump Commands
Read Command
All Data Input Points (3)
SOC Control
3 Drive Current Waveforms (Each MCM)
Data Output

c. Capsule Relay Buffer (CRB) Section

Data Input
Data Output (Each CRB)
Shift Pulse Waveform

d. Motor Timers

Motor Timers are provided for measuring start and stop time of the MTR.

e. End-of-Tape Counter

An end of tape counter will be provided for the MTR.

3.2 SUBSYSTEM TEST

The purpose of the subsystem test is to test and evaluate the operation of the spacecraft data storage subsystem. The DS OSE is capable of testing the DS subsystem in the spacecraft, when electrically isolated from it, or when removed from the S/C. The DS OSE, through testing, is capable of isolating faulty assemblies, though not intended to perform bench tests below the subassembly level. The S/C DS subsystem consists of five basic assemblies: (1) magnetic tape recorders (MTR), (2) magnetic core memories (MCM), (3) capsule relay buffers (CRB), control units (CU) and, (5) power supplies.

The S/C subsystem is powered and controlled by the DS OSE.

3.2.1 MTR ASSEMBLY

Figure 3-2 is a block diagram showing the DS OSE and S/C DS-MTR subsystem test configuration. The power source will power the S/C T/R unit. The control unit controls the Data Simulator, Data Monitor, Bit Error Checker units, and the S/C DS Control Unit. The tests are designed to exercise the assembly through at least the following functions:

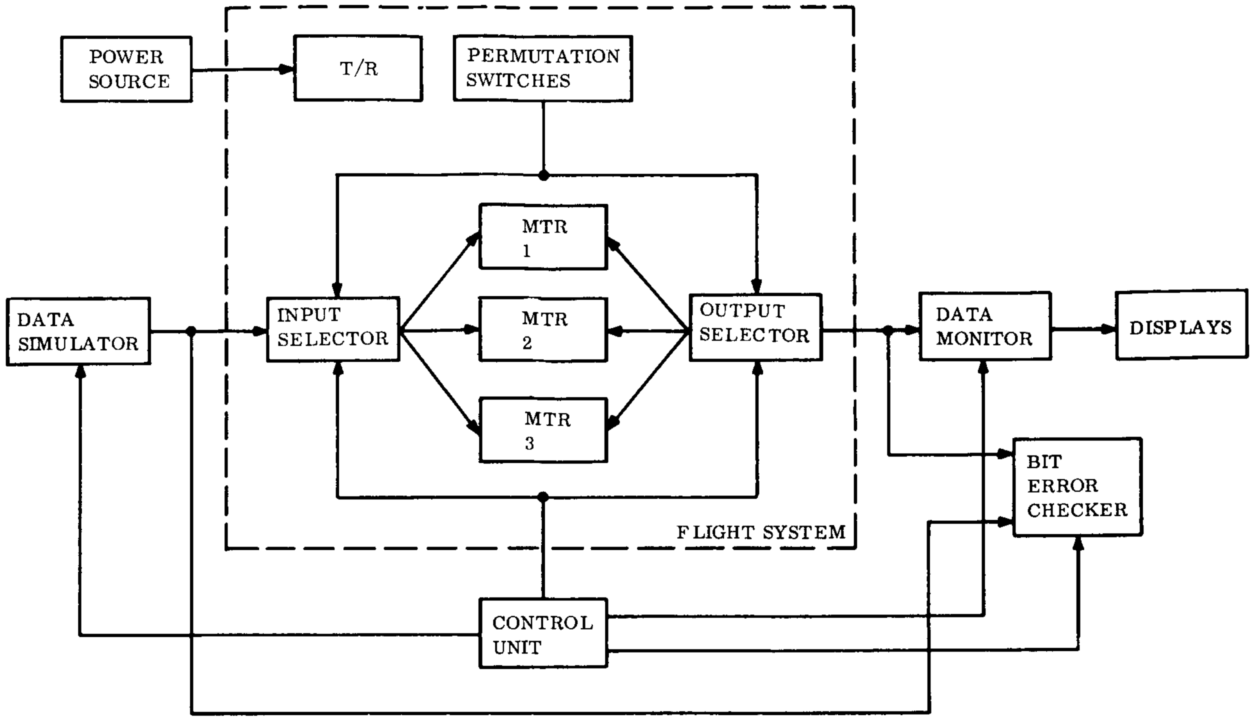


Figure 3-2. Subsystem Test, MTR Assembly

a. Turn-on and turn-off (each MTR)

b. Tape speeds (each MTR)

   1. Record

   2. Playback

c. Exchange Recorders

d. Recorder input selection

e. Recorder output selection

f. Bit error checks (each MTR)

Proper operation of the assembly is determined through data collected from hardwire testpoints, correct functional operation, and correlation of input data and recorded data read out of the MTR's.

3.2.2 MCM ASSEMBLY

Figure 3-3 is a block diagram of the DS OSE and the SC DS MCM assembly. The simulated power source supplies power to operate the DS subsystems T/R unit. The control unit controls the data input, data output, input selector and output selector units. The tests are designed to exercise the MCM subsystem through at least the following functions.

a. Power on

b. Exchange MCM's

c. Bit error checks with worst case memory formats

Proper operation of the subsystem is determined from data collected from hardwire testpoints, correct functional operation, and bit error checks.

3.2.3 CRB ASSEMBLY

Figure 3-4 is a block diagram showing the DS OSE and the S/C DS CRB assembly in a test configuration. The control unit controls the data input, data output, input selector, and the output selector units. The tests are designed to exercise at least the following functions:

a. Power on

b. Exchange CRB's

The proper operation of the assembly is determined through proper data retrieval and proper functional operation.

3.2.4 SUBSYSTEM OPERATION

The flight DS subsystem is operated after the assembly testing is complete. This guarantees the compatability and proper operation of the entire subsystem as a single unit. The unit is operated in the various configurations that are required of it during the mission, and proper operation is determined through monitoring of hard line data and correctness of data.

3.3 SYSTEMS TEST

During systems test, the DS OSE is primarily a monitoring unit with GO-NO-GO displays of system operating functions. In the systems test mode of operation all power, data, and commands are transmitted to the S/C DS subsystem through other spacecraft subsystems. The DS OSE has the capability of supplying data to the flight system. It is also capable of backing-up the flight systems command and control capabilities of the DS system. The functions monitored are listed in the interface section, paragraph 4.0. The OSE is ground isolated from the spacecraft at all times in the STC mode of operation.

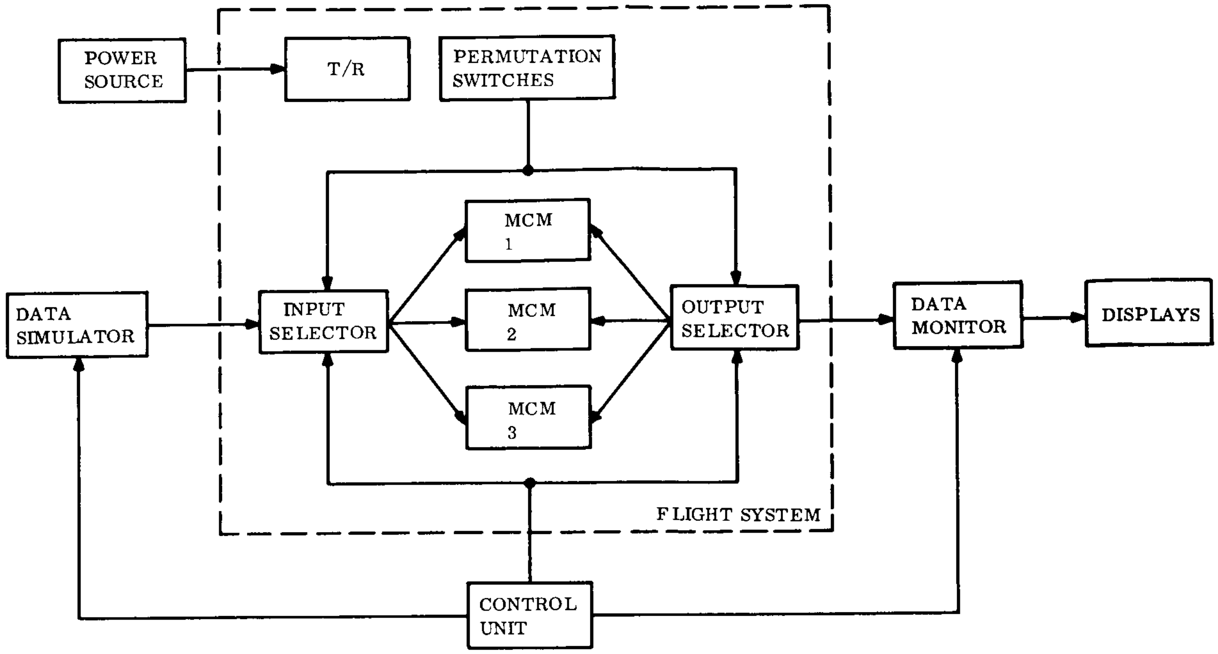


Figure 3-3. Subsystem Test, MCM Subsystem

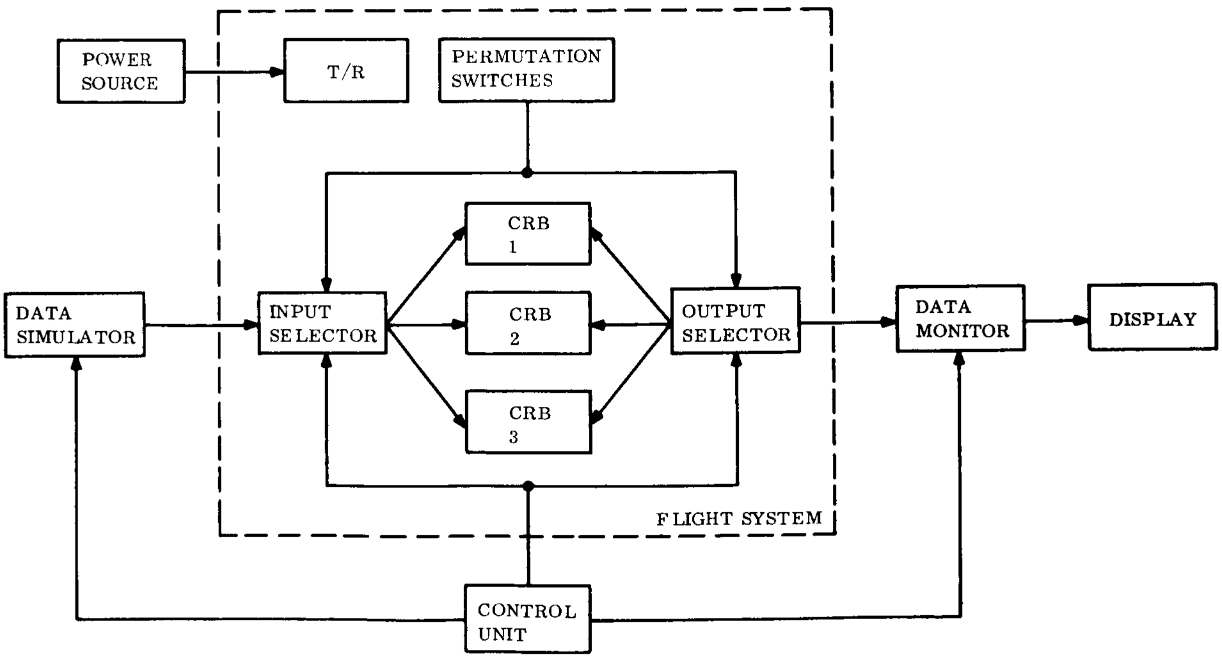


Figure 3-4. Subsystem Test, CRB Subsystem

3.4 DS OSE SELF CHECK

The Data Storage OSE is capable of performing the self check tests described below.

a. All OSE dc voltages can be measured with the Digital Voltmeter.

b. The bit error checker can be tested by entering the same digital data from the Data Simulator into both channels simultaneously.

c. The oscilloscope can be used to observe the amplitude and shape of the data simulator output and of the command pulses.

4.0 INTERFACE DEFINITION

The DS subsystem OSE interfaces with the S/C DS subsystem, the Test Conductor's Console, and the Computer Data System. The interfaces are listed in tables as follows:

a. Table 4-1 - DS OSE and CDS Interface

b. Table 4-2 - DS OSE and Test Conductor's Console Interface

c. Table 4-3 - DS OSE and S/C DS Subsystem Direct Access Test Points, Inputs

d. Umbilical - There is one umbilical function: MTR to launch mode.

Table 4-1. DS OSE and CDS Interfaces

Function	Characteristics	Comments
Command Requests	Switch Closures	Request for S/C commands
S/C DS Subsystem Performance	Analog inputs	Telemetry data from Telemetry Data Converter
Power Supply Voltages	Analog voltages	Alarm
Motor Temperatures	Analog voltage	Alarm
Loop Error Signal	Analog voltage	Alarm

Table 4-2. DS OSE and Test Conductor's Console Interface

Function	Characteristics	Comments
S/C Status	Switch closures	Go-No-Go Indicators
S/C Mode	Switch closures	S/C Mode of Operation

Table 4-3. DS OSE and S/C DS S/S Direct Access Interfaces - Inputs

Function	Comments	Signal Characteristic Notes
+3.5 volt	D/S T/R	Voltage Level
+28 volt	D/S T/R	Voltage Level
-3 volt	D/S T/R	Voltage Level
2 Motor Temperatures	Each MTR	Analog Signal 0 - 5-volt
Tape Motion Indicator	Each MTR	Voltage Waveform
Power Amplifier Output	Each MTR	Voltage Waveform
Loop Error Signal	Each MTR	Voltage Waveform
Scan Data Output	MTR Control	3.5-volt NRZ Data
Scan Data Gate	MTR Control	3.5-volt Logic Level
Dump Data Output	MCM Control	3.5-volt NRZ Data
Driver Current Waveform	Each MCM	Voltage Waveform
CRB Data Output	CRB Control	3.5-volt NRZ Data
CRB Filled Signal	CRB Control	3.5-volt Logic Level

5.0 PERFORMANCE PARAMETERS

The Data Storage OSE provides the measurement/control capability listed below.

a. D-C Voltages - The DS OSE has the capability of measuring D-C voltages to ±1 microvolt on the most sensitive scale.

b. Elapsed time is measured, with accuracy limited by the accuracy of the frequency of the STC power.

c. Simulated power of 2400 cps ±0.01%, 50v rms ±2% is generated.

d. Frequencies can be measured to an accuracy of greater than 0.01%.

e. Counters are provided to count events.

f. Data and control signals are simulated by the DS OSE.

The subsystem and system test operations will differ by the interfaces used. In subsystem test the DS OSE will have complete operational and monitoring capability. In systems test the OSE will be primarily a monitoring device with back-up control capability. During all testing the DS OSE will be ground isolated from the flight sistem.

6.0 PHYSICAL REQUIREMENTS AND CONSTRAINTS

6.1 POWER

The Data Storage OSE requires 105- to 125-volt, 60-cps, single-phase power. The maximum current not to be in excess of 50 amperes.

6.2 WEIGHT

The weight of the Data Storage OSE is less than 2000 pounds.

6.3 SIZE

The Data Storage OSE occupies two standard 75-inch NASA racks or less.

6.4 COMMERICAL TEST EQUIPMENT

The DS OSE has the following commercial equipment.

a. Electronic counter

b. Strip chart recorder

c. Digital voltmeter

d. Oscilloscope

e. Elapsed time meters

f. Flight power supplies

7.0 SAFETY

A feature of the DS OSE is the protection of the flight DS system from either operator error or electrical malfunctions. This is accomplished through minimizing operator requirements by automatic features within the OSE, interlocks between functions where improper operator operation could cause damage and protecting all interfaces from accidental or unintentional shorts by limiting access to these points. Care is also taken in the design of the unit to protect the operator from areas where high voltages or other functions which could cause bodily harm could exist.

SYSTEM TEST COMPLEX

DATA ENCODER OSE

Index

1.0 SCOPE

This document describes the Data Encoder Operational Support Equipment necessary for the Systems Test Complex (STC) to checkout test and demonstrate the S/C DE operation during systems tests. This document will also include the equipment necessary to test and calibrate the Data Encoder as an electrically isolated subsystem during subsystem tests, while installed in the spacecraft.

2.0 APPLICABLE DOCUMENTS

The following documents form a part of this specification:

VB260SR101	STC Test Objectives and Design Criteria
VB260SR102	STC Design Characteristics and Restraints
VB260FD106	STC Central Timing and Synch
VB260FD107	STC Central Recorder
VB260FD103	STC Computer Data System
VB260FD102	STC Ground Power Distribution
VB260FD105	STC Printers and Displays

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The data encoder operational support equipment is required to support system and subsystem tests in the system test complex by stimulating, controlling and monitoring the spacecraft data encoder subsystem. The DE OSE, through umbilical and/or direct access, will generate DE commands, stimulate the individual channels for calibration, decommutate the commutated data, and monitor and display the results. Stringent isolation requirements of the STC dictate use of isolation amplifiers and isolation switches between the DE OSE, the spacecraft and other ground equipment.

The DE OSE, shown in Figure 3-1, consists of the following major functional blocks:

a. Input Switching

b. Telemetry Demodulator

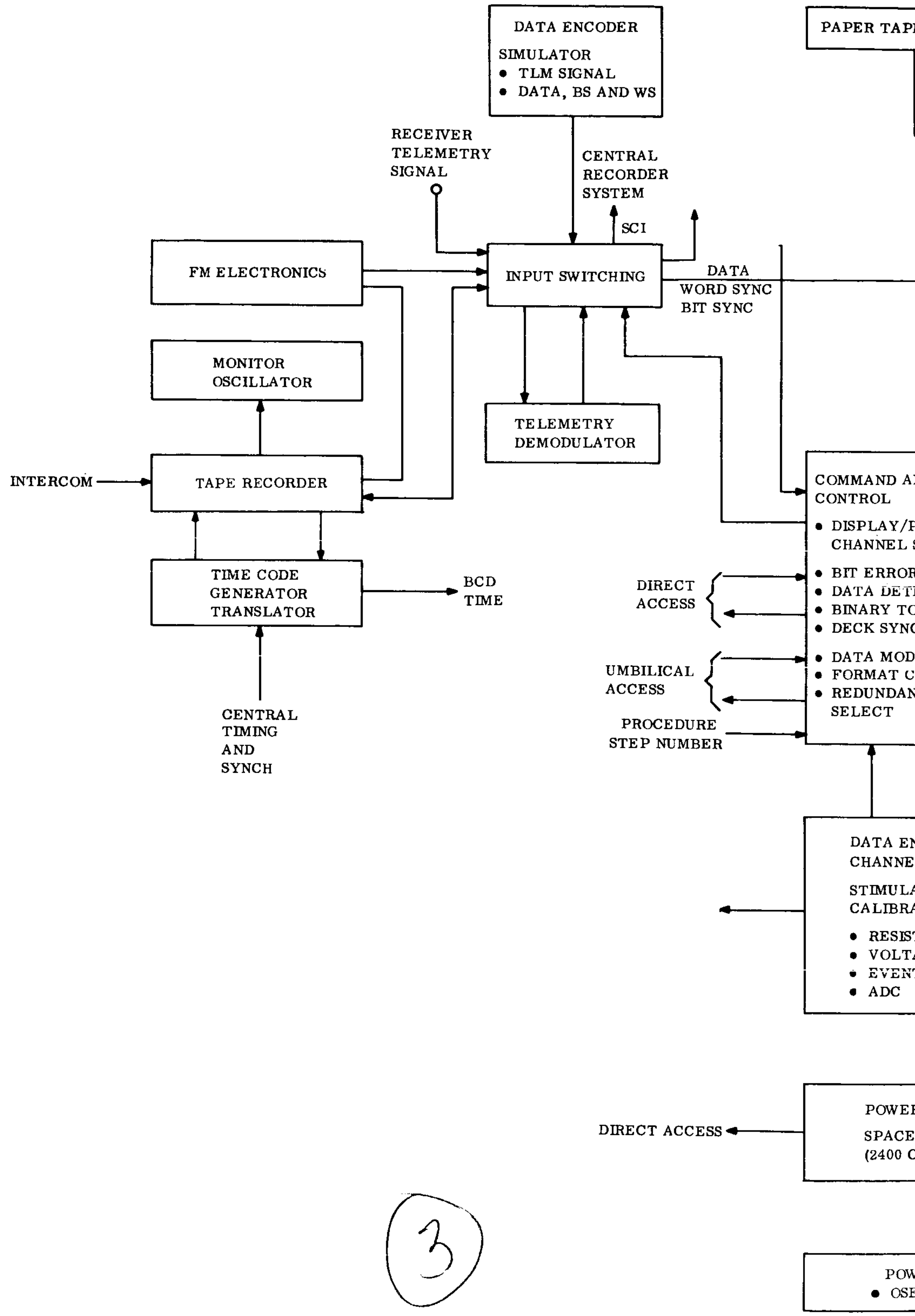
c. Data Encoder Simulator

d. Buffer

e. Preamble Sync Code Recognizer

f. Command and Control

g. DE Stimulator/Calibrator



DATA ENCODER
SIMULATOR
• TLM SIGNAL
• DATA, BS AND WS
PAPER TAP
RECEIVER TELEMETRY SIGNAL
CENTRAL RECORDER SYSTEM
SCI
FM ELECTRONICS
INPUT SWITCHING
DATA
WORD SYNC
BIT SYNC
MONITOR OSCILLATOR
TELEMETRY DEMODULATOR
INTERCOM
TAPE RECORDER
COMMAND A
CONTROL
• DISPLAY/
CHANNEL
TIME CODE GENERATOR TRANSLATOR
BCD TIME
DIRECT ACCESS
• BIT ERRO
• DATA DET
• BINARY TO
• DECK SYN
UMBILICAL ACCESS
• DATA MOD
• FORMAT C
• REDUNDAN
SELECT
PROCEDURE STEP NUMBER
CENTRAL TIMING AND SYNCH
DATA E
CHANNE
STIMUL
CALIBR
• RESIS
• VOLT
• EVEN
• ADC
DIRECT ACCESS
POWE
SPACE
(2400 C
POW
• OSE

3

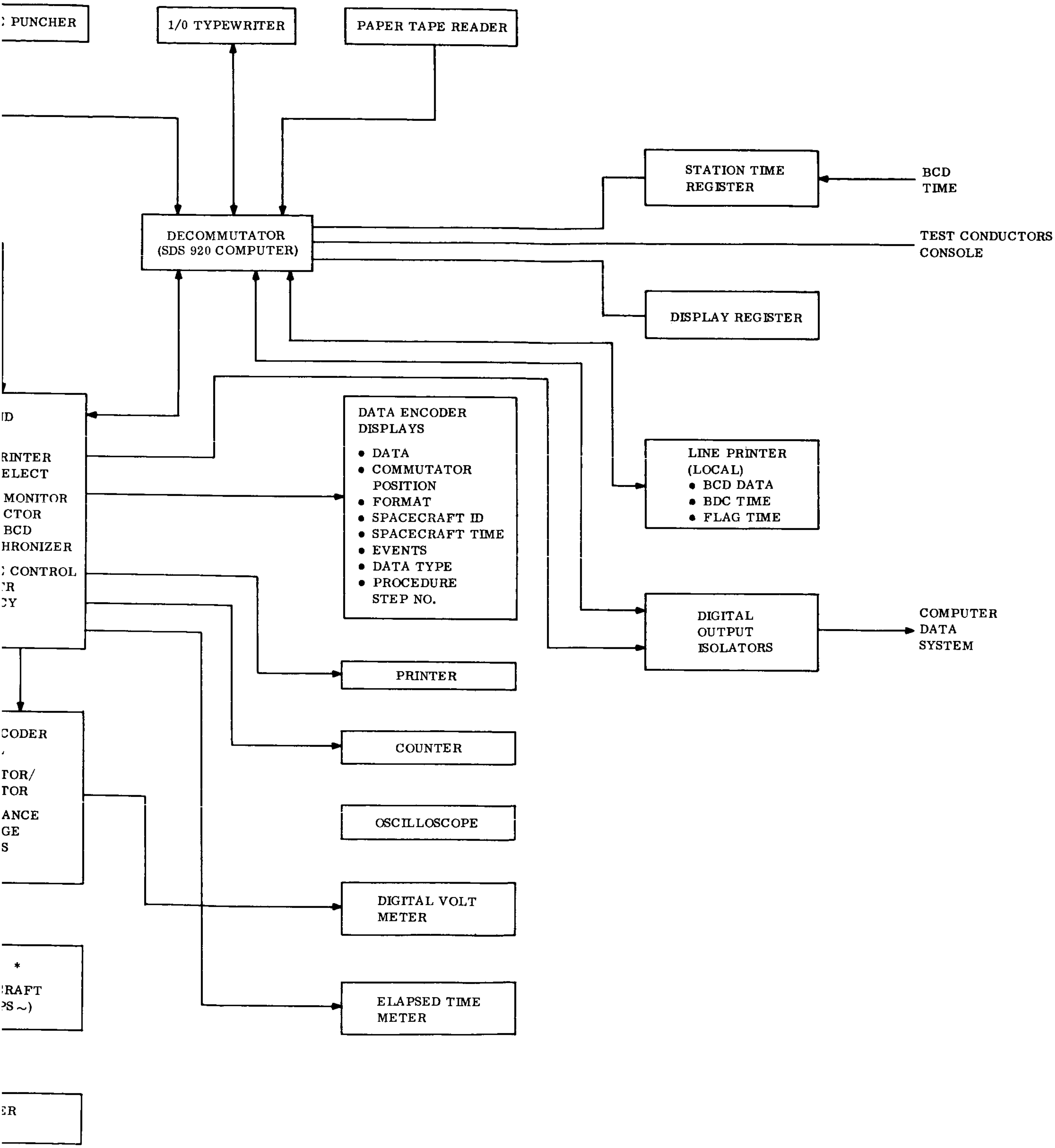


Figure 3-1. Data Encoder

h. Decommutator

3.1.1 INPUT SWITCHING

The input switching unit receives: (1) the composite telemetry signal bi-phase modulated with data and sync information from the radio subsystem OSE, tape recorder subsystem, or data encoder simulator. (2) modulated telemetry signals, data, word sync, bit sync, from the telemetry modulator, data encoder simulator, tape recorder, or command and control unit. The information received is routed to the appropriate user which is one of several of the following:

a. Tape recorder

b. Data decommutator

c. Telemetry demodulator

d. Computer data system

e. Central recording system. (SCI only)

The telemetry demodulator sync condition indication is supplied to the input switching subsystem for routing to the computer data system, central recorder, data decommutator and command and control subsystem.

3.1.2 TELEMETRY DEMODULATOR

a. General — The telemetry demodulator has the task of detecting and decoding a noise corrupted bi-phase modulated telemetry signal from ground telemetry receivers, magnetic tape recorders, or data encoder simulator. The telemetry signal is a composite signal containing both data and synchronization information on a single subcarrier. The demodulator output is a serial or parallel binary train of data pulses, bit sync, word sync and demodulator sync condition indication (SCI). The output data and sync information will be sent to the decommutator for processing and to the tape recorder system for storing. Bit sync, word sync and data are time synchronous.

The demodulator will be required to operate at the six different bit rates (R) listed below during the Voyager mission.

R_1 = 8,533 1/3 bps	R_4 = 533 2/3 bps
R_2 = 4,266 2/3 bps	R_5 = 106 2/3 bps
R_3 = 2,133 1/3 bps	R_6 = 3 1/3 bps

The demodulator is used during system and subsystem tests in the STC. The demodulator is identical to the demodulator in the Deep Space Instrumentation Facility.

b. Description — Upon receipt of the noise-corrupted telemetry signal the signal is sent through a low-pass filter ($f_c = 8f_s$) and amplifier, as shown in Figure 3-2. The low-pass filter masks out a portion of the unwanted noise while the amplifier serves as an isolation device and amplitude compensator. The amplitude compensation allows closer control of the signal level entering the demodulator. The filter amplifier presents the resulting signal to three detector filter branches; clock, data and sync.

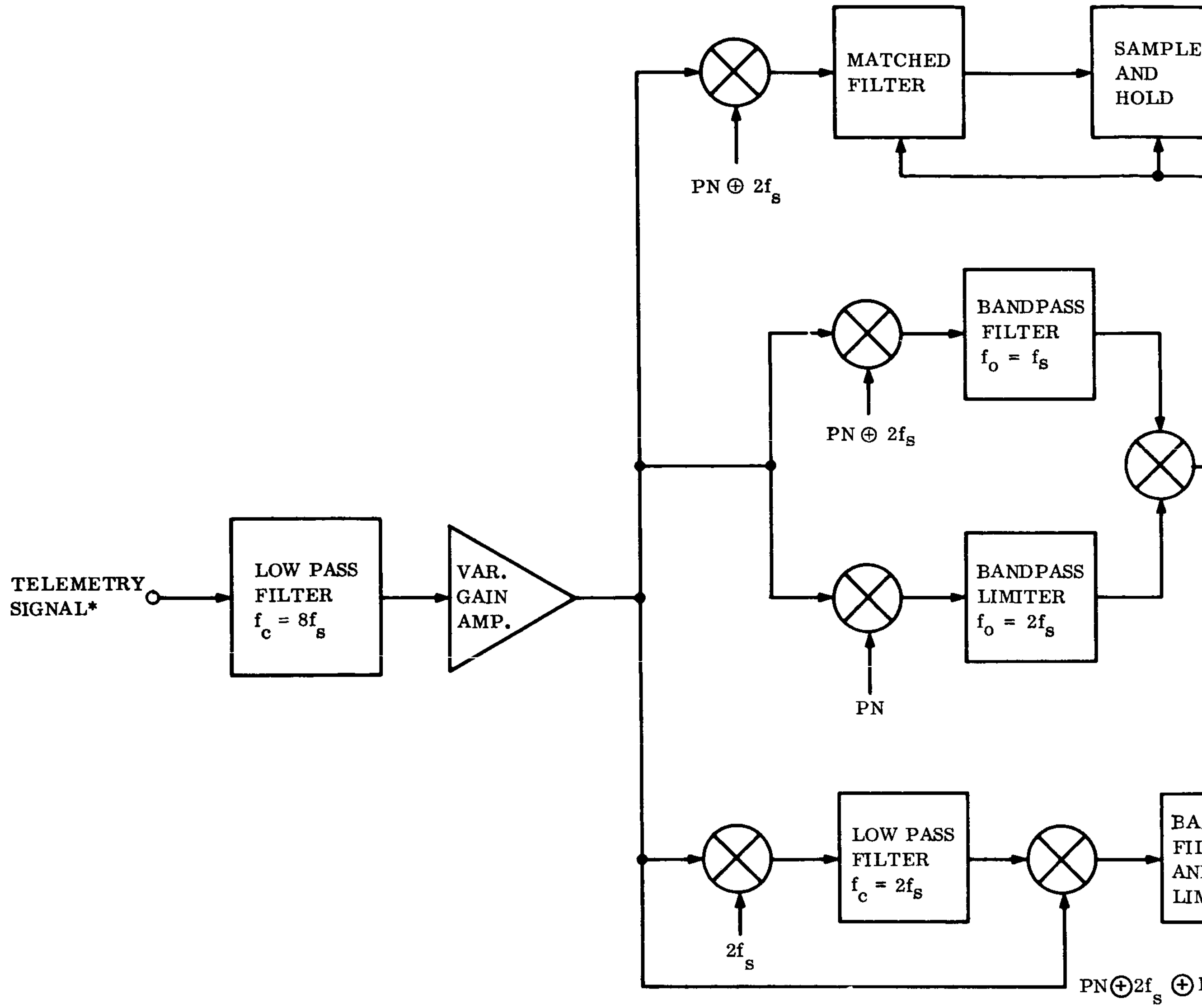
Demodulation of data is accomplished by first acquiring clock and then synchronizing the sync channel. Clock is acquired by locking a phase lock loop on the incoming clock after all modulation has been removed by mixing and squaring. The noise-free clock from the PLL is used for driving the binary logic circuits including the pseudo-noise (PN) code generator. To acquire word sync, the demodulator PN code generator must align or correlate with the received PN code. Thus, synchronization is determined by matched-filter detection of the quadrature channel as the internally generated code is correlated with the received code. When the two codes are aligned, synchronization is accomplished, thus, emitting code lock or sync condition indication (SCI), data, bit sync, word sync, and termination of the acquisition process.

Code slide-by, thus enabling acquisition, is accomplished by adding one 4 f_s clock pulse every 63 PN bits until acquisition is complete. When acquisition is complete, the addition of one clock pulse every 63 PN bits is discontinued by inhibiting the add-in process with SCI. Data is matched-filter detected in the data channel after removing PN code and 2 f_s clock from the bi-phase modulated signal.

The Serial-to-Parallel Converter and Preamble Recognizer receives the serial data and word sync, and converts the serial data to seven-bit parallel data. It also scans incoming data, seven words at a time, in search of the unique 49-bit preamble sync that precedes every new data type. The logic used permits recognition of the preamble sync with up to three errors. When the preamble sync is recognized, a Preamble-Recognize pulse is generated and sent to the decommutator for enabling the interrogation of the data type word. The preamble sync is a 28-bit word identifying the start of a new data type. The seven-bit data type word always follows the preamble sync.

3.1.3 DATA ENCODER SIMULATOR

The data encoder simulator functionally duplicates the S/C DE in order that the DE OSE can be checked and verified as to its operation without the use of a data encoder.



* TELEMETRY SIGNAL = $2f_s$ ⊕ PN ⊕ DATA

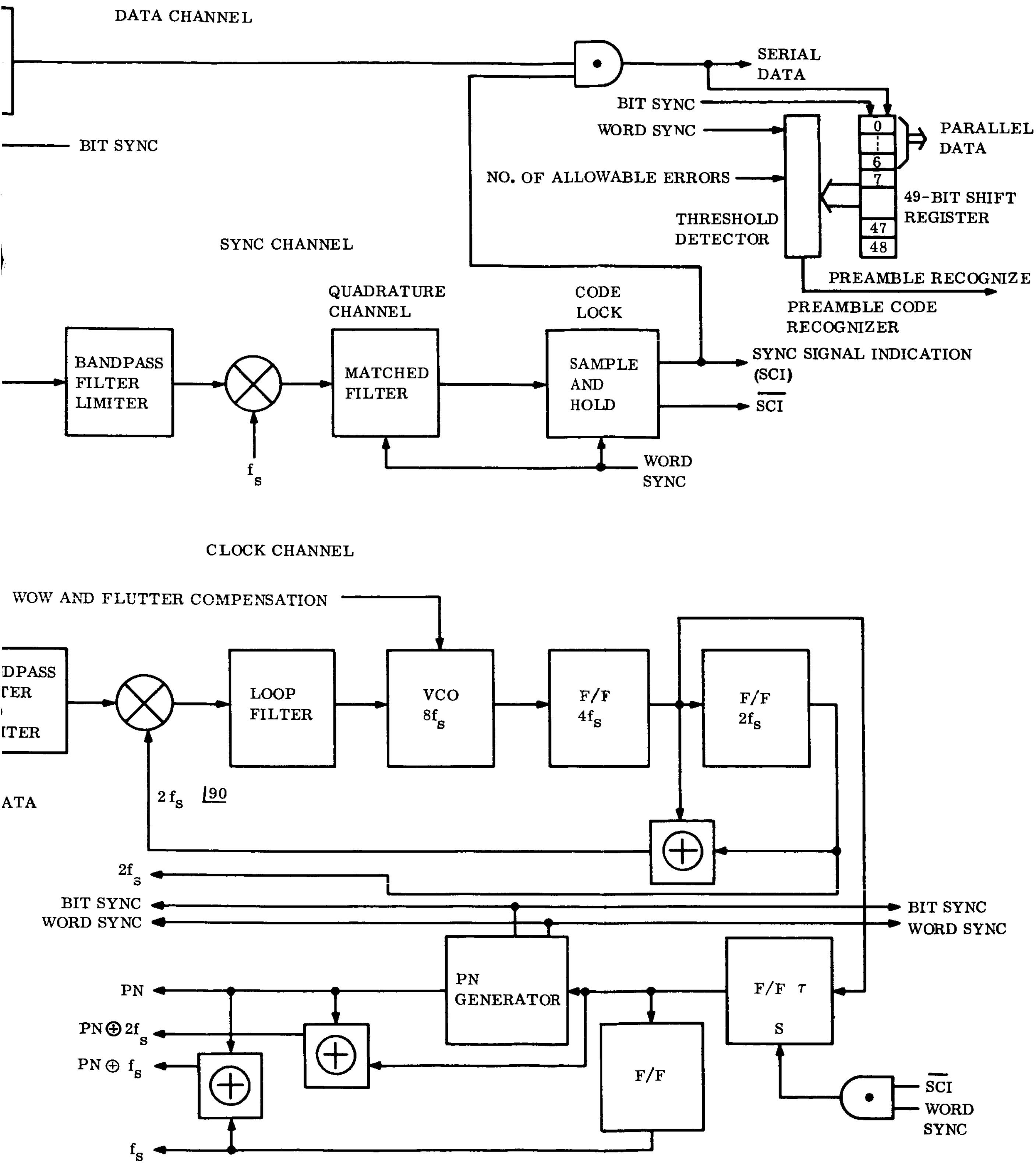


Figure 3-2. DSIF Telemetry Demodulator

The data encoder simulator generates data including preamble sync and data type as shown in Figure 3-3. Preamble sync is a unique 49-bit word used for all transmission modes, whereas the data type word is unique for each data type. The generated data is engineering and non-scan science data as shown in Figure 3-4. The simulated data contains frame sync, data mode, medium deck position and low-speed deck position and data. The data contains different words of known value for each channel. A degree of selection or programming of words is available as front panel control. Data collection formats are shown in Table 3-1 and Figures 3-5 through 3-10.

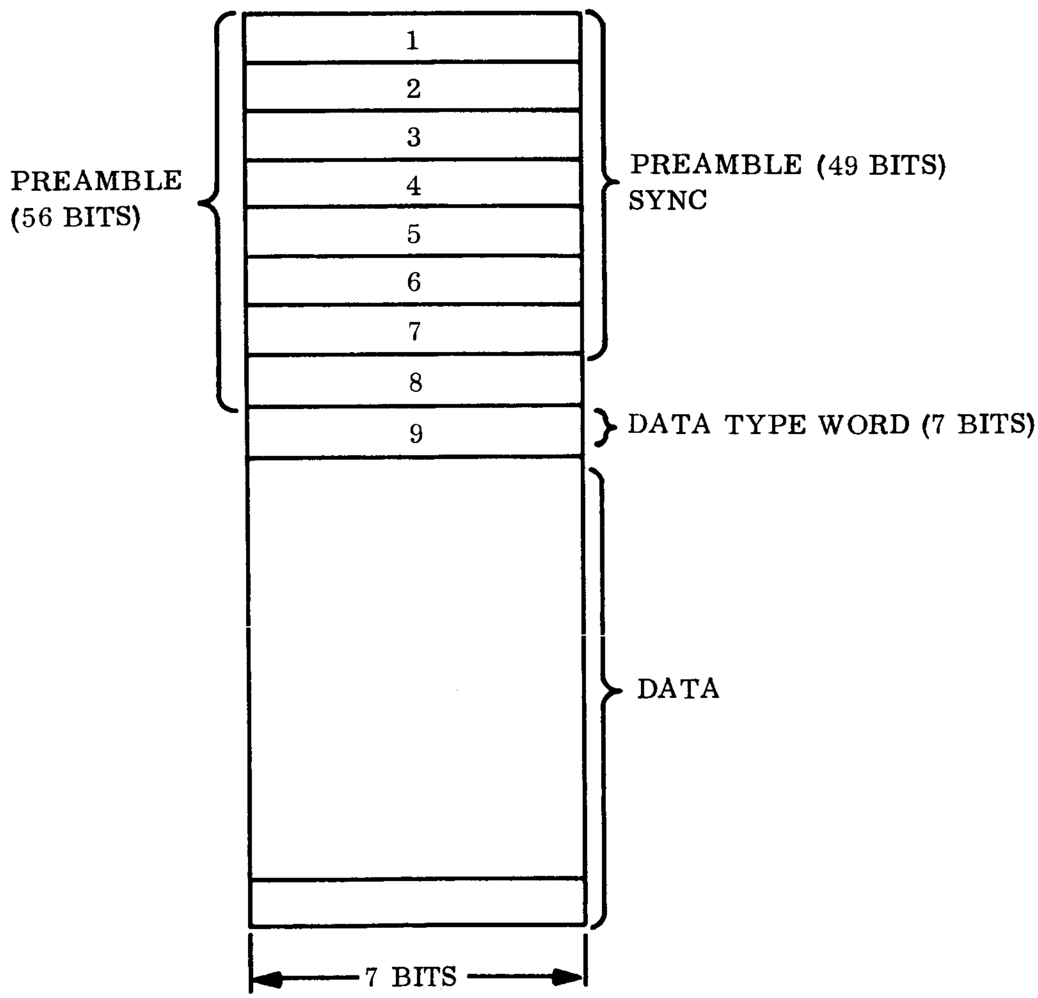


Figure 3-3. Data Type Format

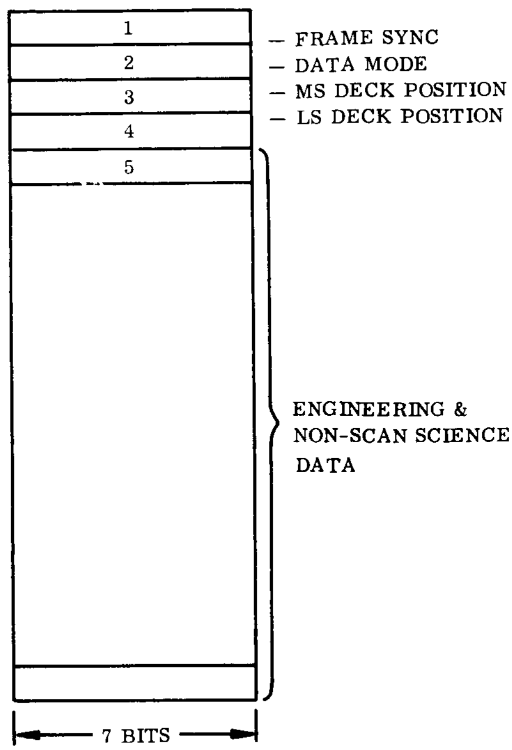
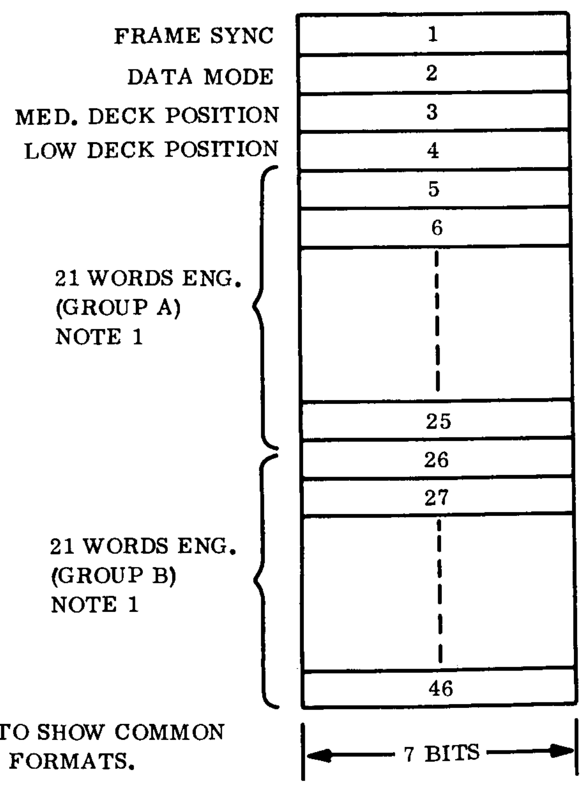


Figure 3-4. Non-Scan Data Mode Format



NOTE

GROUPS ARE SHOWN TO SHOW COMMON CHANNELS BETWEEN FORMATS.

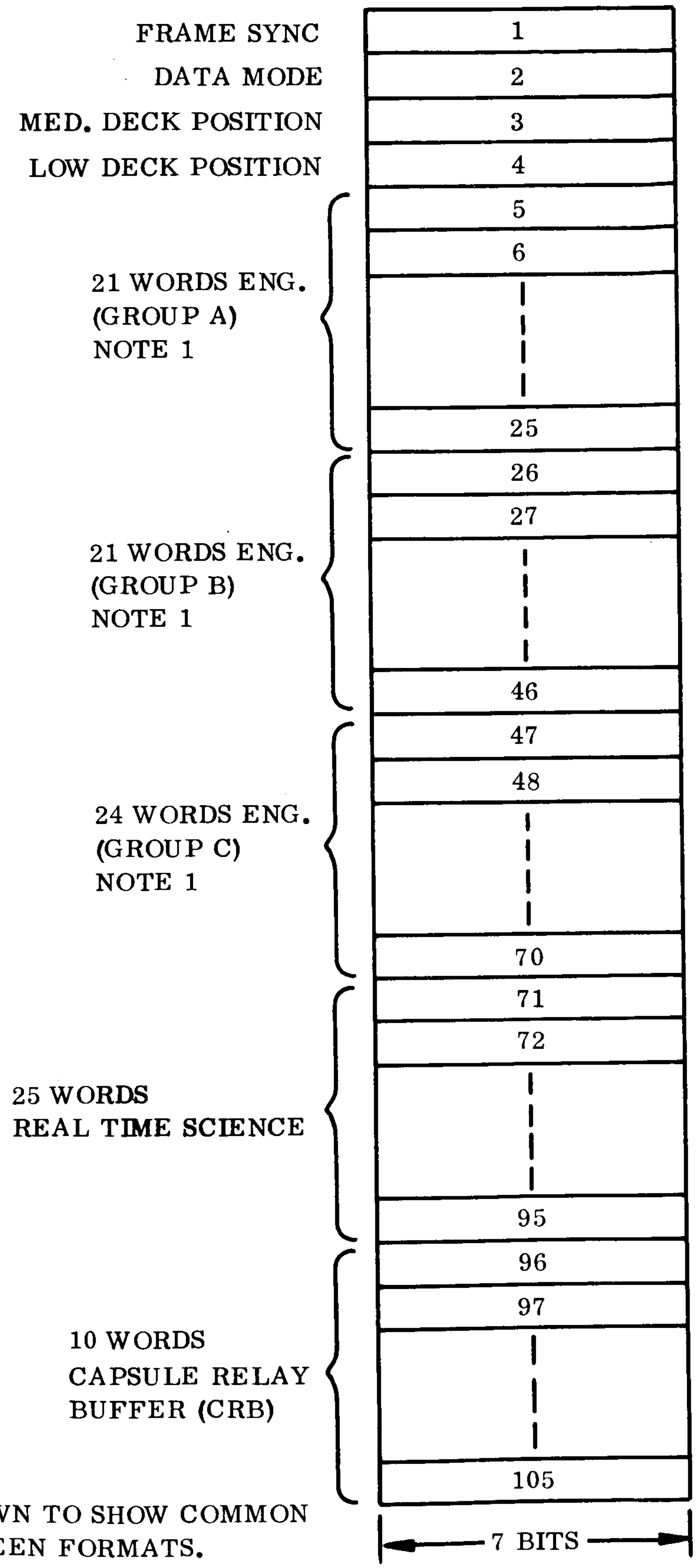
Figure 3-5. Maneuver Mode (Mode I)

Table 3-1. Data Mode Commutation Formats

Mode	High Deck Channels	Medium Deck Channels	Low Deck Channels	CRB Channels	Real Time Science Channels	Approach Guidance Channels
Maneuver (I)	46	70	---	---	---	---
Cruise (II)	70	110	100	10	25	---
Normal Orbit (III)						
(a) Buffered Non Scan	90	110	100	---	90	---
(b) Planet Scan Data		1×10^6 Bits MTR Planet Scan Data				
MCM Dump (IV)			---	2 MCM Blocks		
(a) Engineering and Capsule Data	1 MCM Block Maneuver Format Format Mode I					---
(b) Flare Date					3MCM Blocks of Flare Data	
Non-Scan Orbital (V)	90	110	100	---	90	---
Approach Guidance (VI)	70	110	100	14	25	38

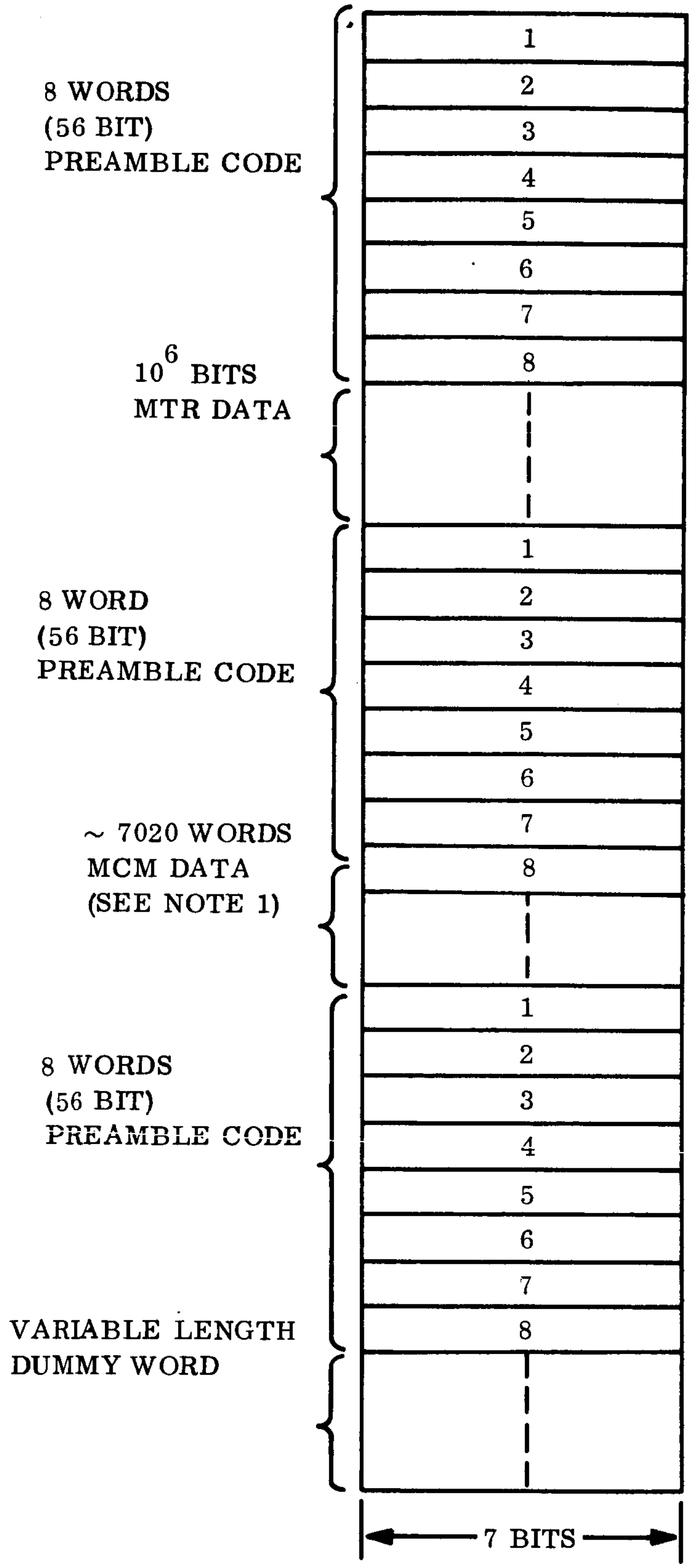
NOTE

Medium Deck Channels are ten in length
Low Deck Channels are twenty in length.



NOTE 1 - GROUPS ARE SHOWN TO SHOW COMMON CHANNELS BETWEEN FORMATS.

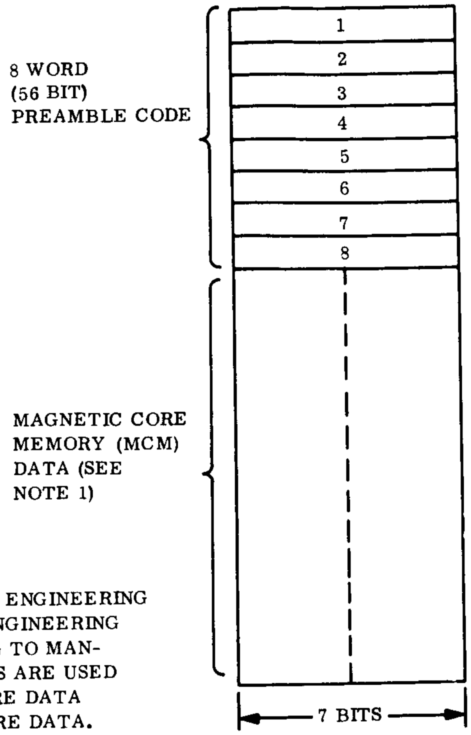
Figure 3-6. Normal Cruise (Mode II)



NOTE 1

THE MCM DATA TRANSMITTED BETWEEN SCAN BLOCKS IS COLLECTED DURING DATA TRANSMISSION. THE COLLECTION FORMAT IS THE SAME AS THE NON SCAN ORBITAL MODE (MODE V)

Figure 3-7. Normal Orbit Mode (Mode III)



NOTE

MEMORY DUMP MODE DATA CONSISTS OF ENGINEERING AND CAPSULE DATA OR FLARE DATA. ENGINEERING DATA IS STORED IN ONE MCM ACCORDING TO MANEUVER MODE (MODE I) WHILE TWO MCM'S ARE USED TO STORE CAPSULE DATA. DURING FLARE DATA DUMPS ALL THREE MCM'S CONTAIN FLARE DATA.

Figure 3-8. Memory Dump Mode (Mode IV)

3.1.4 COMMAND AND CONTROL

The command and control unit has hard-line control and monitoring responsibilities of the spacecraft data encoder as well as control of the DE OSE. The command and control unit exercises spacecraft DE subsystem controls. The command and control subsystem provides for the following:

a. Rate commands

b. Data type commands

c. Transmission mode commands

d. Block redundancy select commands

e. Display and/or Printer Channel Select

f. Commutator synchronization

g. Simple data detection (noise free and hard line)

h. Data conversion - binary to BCD

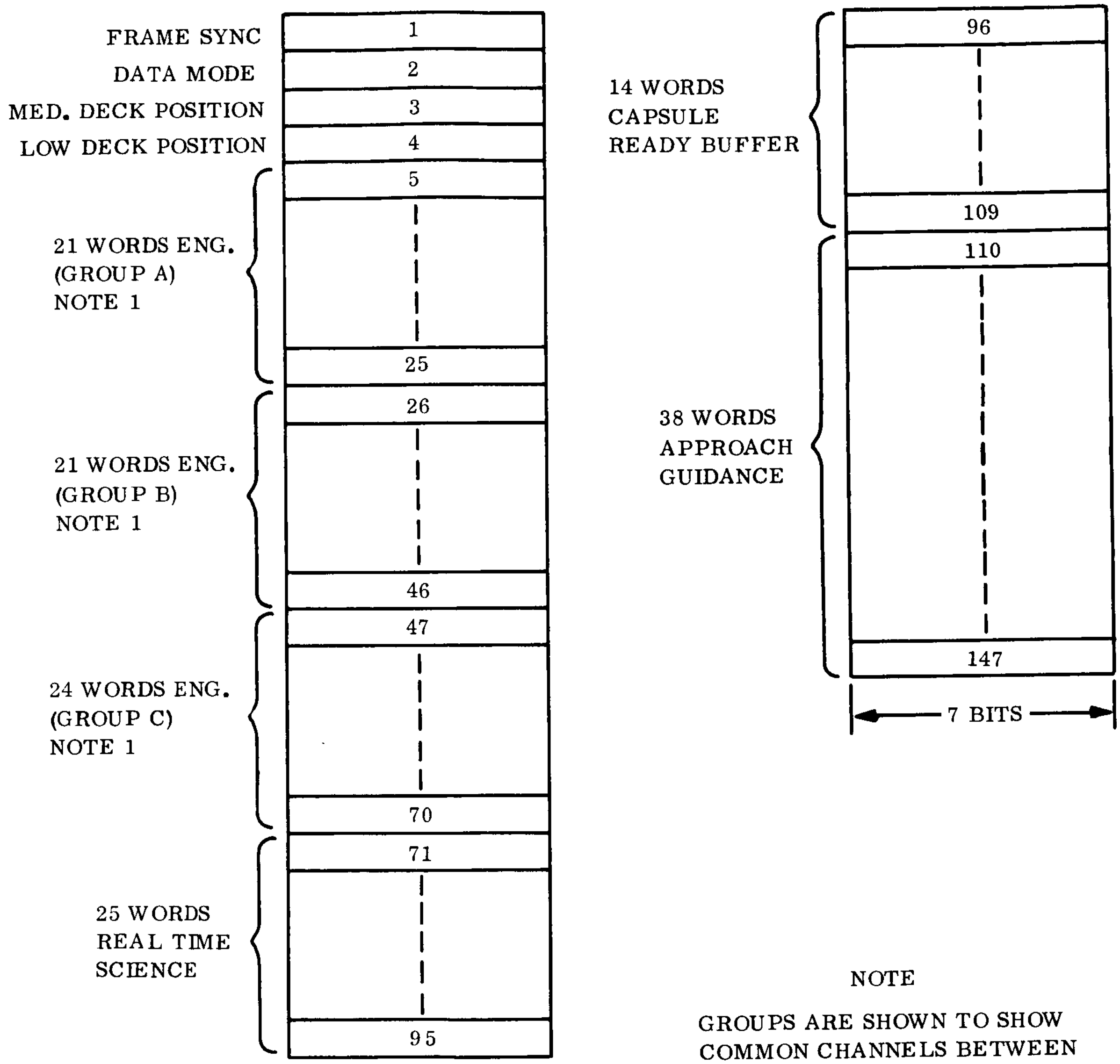
1	FRAME SYNC
2	DATA MODE/
3	MED. DECK POSITION
4	LOW DECK POSITION
5 – 25	21 WORDS ENG. (GROUP A) NOTE 1
26 – 46	21 WORDS ENG. (GROUP B) NOTE 1
47 – 70	24 WORDS ENG. (GROUP C) NOTE 1
71 – 90	20 WORDS ENG. (GROUP D) NOTE 1
91 – 180	90 WORDS REAL TIME SCIENCE

← 7 BITS →

NOTE

GROUPS ARE SHOWN TO SHOW COMMON CHANNELS BETWEEN FORMATS.

Figure 3-9. Non-Scan Orbital Mode (Mode V)



NOTE

GROUPS ARE SHOWN TO SHOW COMMON CHANNELS BETWEEN FORMATS.

Figure 3-10. Approach Guidance (Mode VI)

i. Bit error monitoring

j. Basic clock generation

While receiving and distributing:

a. Data

b. Data mode

c. Commutator deck position

d. Spacecraft ID

e. Spacecraft Time

f. Spacecraft Events

g. Data Type

h. Procedure step number

3.1.5 DATA ENCODER STIMULATOR/CALIBRATOR

The outputs on the stimulator/calibrator can be connected to the data encoder case harness for signal simulation for all data channels in the data encoder. The DE stimulator is mechanized so as to accommodate automatic calibration. The data encoder inputs are divided into two categories, analog and digital.

a. The simulated analog inputs are as follows:

   Voltage simulators provide seven discrete voltages to the data encoder in the following three voltage ranges: 0 to 100 mv, 0 to ±1.6 volts, and 0 to 3.2 volts.

b. The simulated digital signals consist of the following:

   1. The event simulator provides either a serial pulse train or two 3-bit parallel words to the DE.

   2. Simulated capsule relay buffer, magnetic core-memory and magnetic tape recorder data signals are provided.

3.1.6 DECOMMUTATOR

a. Function — The main decommutator is an SDS 920 computer shown in Figure 3-11. The decommutator receives demodulated telemetry data,

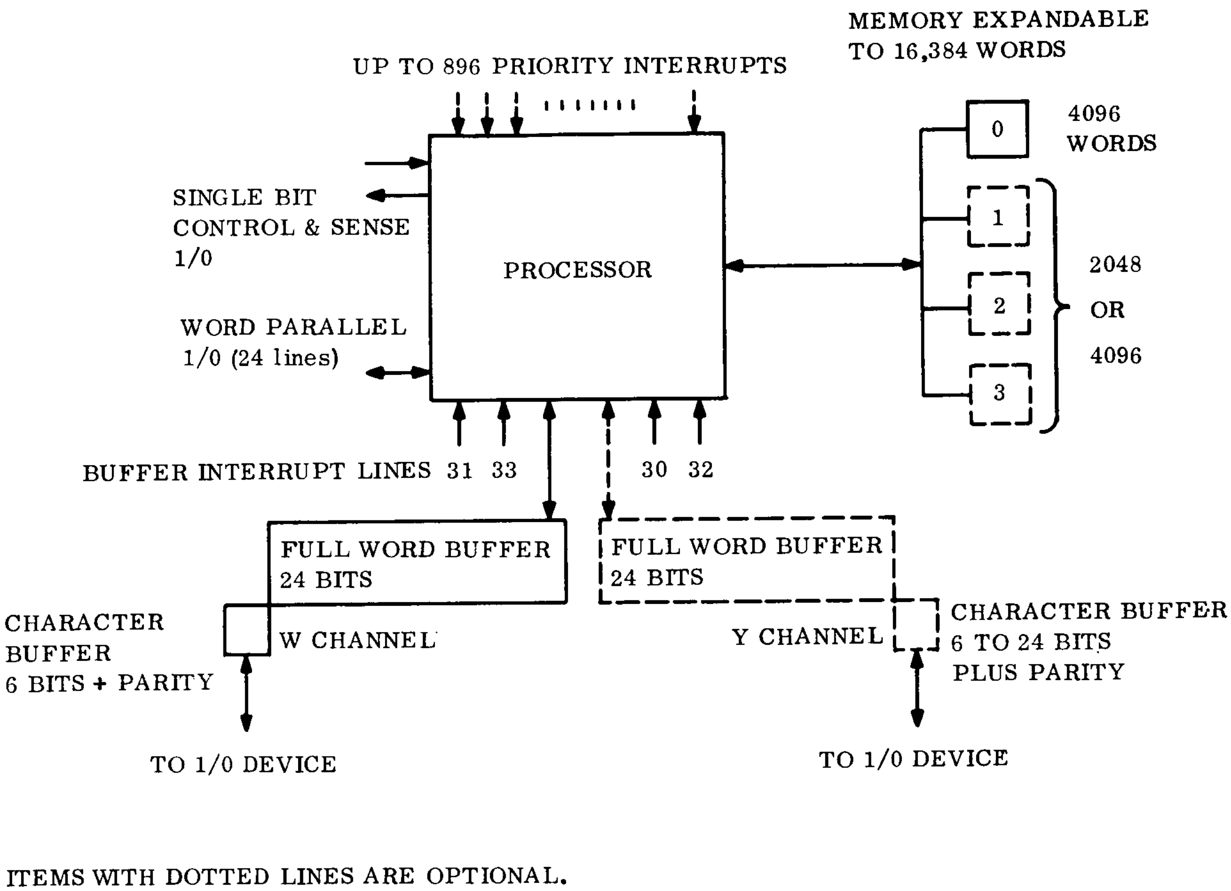


Figure 3-11. Decommutator Block Diagram

word sync, SCI, and preamble sync and performs the following program functions:

1. Determine data type and execute the necessary processing instructions or data type programs at all data rates.

2. Generate and identify the data condition information as to the validity of received frame sync, mode, MS deck position, LS deck position, and word checks for each channel of data. The word checks performed are limit checks or delta checks.

3. Synchronize and process the submultiplexed engineering data.

4. Synchronize and process science data (scan and non-scanned).

5. Synchronize and process capsule data.

6. Format selected telemetry data for visual display.

7. Format split word telemetry data for display.

8. Channel address each of the engineering, capsule, and non-scanned science data channels, including special engineering split channels.

9. Provide for preamble override.

10. Provide flywheel action for subcommutation program for frame sync, Mode, MS Deck position, and LS Deck position words.

11. Format telemetry data to binary or BCD parallel output.

12. Tag each data channel with flag word or DCI (Data Condition Indicator).

b. Decommutation Program — The decommutation program required to decommutate the telemetry data is described in three parts; input, processing and output. The program times and memory utilization are referenced to a Scientific Data System 920 general purpose computer with a 8192-word capacity of 24 bits each. The computer cycle time is eight microseconds and has typical execution times of 16 microseconds for adding and 32 microseconds for multiplying. The input and processing portion of the decommutation program is identical to that of the DSIF.

   1. Input — The input to the computer enters through the 24-pin external connector. The input information is received in parallel form as a 24-bit computer word. The 24-bit received word consists of seven data bits in the least significant position and the telemetry demodulator sync condition indicator (SCI) bit in the most significant bit position,

with the remainder of the 24-bit word unused, as shown in Figure 3-12. When the demodulator is synchronized, the SCI bit is a ONE; when it is out of lock, the SCI is a ZERO. This eight-bit data word is presented to the 24 pin connector at the time of word sync.

2. Internal Processing — The flow through the machine is characterized by the input, output, storage and processing phases. The program is loaded into the computer via paper tape along with data via the pin connector to complete the storage tables. The tables are organized into seven columns for program instructions, one for each data type as shown in Table 3-2. Two columns are used for storing received data so that word comparisons may be performed (detecting a word change from the previous word in that same location). A third column is used for limit checking of the data. An alarm signal is generated when the received word is out of limits. The length of each table corresponds to each data type format. See Table 3-1 and Figures 3-5 to 3-10.

   Along with data, which consist of a seven-bit telemetry word plus SCI, the decommutator receives word sync and preamble sync. Word sync initiates a priority interrupt enabling the computer to readin the new data word. The SDS 920 computer has a system of priority interrupts which allows the computer to be time shared according to fixed priority schedule. For this system, highest priority is given to word sync which commands the interrupt to whatever task the machine was performing at that time. The machine branches to the read instruction to accept incoming data. After the last write occurs, the machine returns to the next higher order interrupt or returns to the task being performed when interrupt was received.

   Preamble sync is used for interrogating the data type word, which follows preamble sync in the preamble code word. Preamble sync is also used to inhibit engineering frame sync search during Data Types C and F, scan and flare data, respectively, or more specifically during Data Modes III and IV. During all other Data Modes the preamble code word check is deleted.

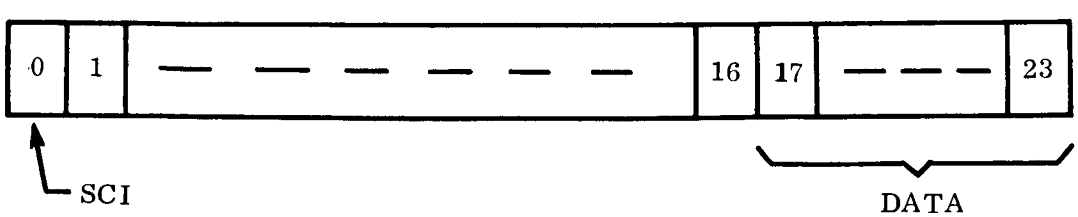


Figure 3-12. Decommutator Input Data Format

Table 3-2. Program Data Types

Transmission Mode	Data Mode	Data Rate	Data Type	Data Type Description
Maneuver Mode	I	3-1/3	A	Engineering Data
Cruise Mode	II	106-2/3	B	1. Engineering data 2. Capsule data 3. Non-scan science
Orbit Mode	III	8533-1/3	C	Planet scan data
			D	1. Engineering data 2. Non-scan science
Memory Dump Mode	IV	2133-1/3	A	One MCM Block of Mode I Engineering data
			E	Two MCM Blocks of CRB Data
			F	Three MCM Blocks of Flare Data
Non-Scan Orbital Mode	V	8533-1/3	D	1. Engineering data 2. Non-scan science data
Approach Guidance Mode	VI	106-2/3	G	1. Engineering 2. CRB 3. Non-scan science data 4. Approach guidance data

The seven data types required to be processed are categorized into two basic types: (1) engineering, capsule and non-scan science (ECNS) data, and (2) scan and flare (SF) data.

For ECNS data, a search is made for frame sync, a unique seven-bit word of all ONES in the high-speed deck. Upon receipt of frame sync, the next step is to read and check the Data Mode word as to its Data Type so as to incorporate the appropriate instruction table. Upon identifying the Data Type, the transmitted format is identified as to its minor frame length, subcommutated channels, addresses and use instructions.

After identification of the data type, if subcommutation is employed, the medium-speed deck position (MSDP) and the low-speed deck position (LSDP) words will follow mode word. These two words identify the subcommutator and commutator positions, and are used for synchronizing the subcommutator counters in the computer.

Figure 3-13 through 3-20 are flow charts describing the general operation of the decommutation program.

Once acquisition is accomplished, an inertia (or flywheel) feature is incorporated that allows for an error in the frame sync, mode, MSDP and LSDP words for one or two consecutive times depending on how programmed. Once acquired, and an error exists in one of the words mentioned above, a data condition indicator (DCI) is generated; one bit for each word or a total of four bits. The DCI is used as a flag to question the value of the data rather than drop or delete the data questioned.

Frame sync is continually searched for during all engineering data, and is inhibited during capsule and non-scan science data.

The SF data is processed similarly to the ECNS data, although there are no subcommutated data words. The scan data is formatted for CDS and tape storage after it has been stripped from all other incoming data.

3. Output — The output instructions for the channels of the seven data types are recognized so that the processed data may be routed in proper form to the appropriate user. The output data is presented in 24-bit parallel form, either in binary or BCD, as determined by the users need. Three users require the output data: the DE OSE, the Test Conductors Console, and the Computer Data System. The data is presented to the users in one of three formats. The three output formats are shown in Figure 3-21.

The DE OSE uses the format of Figure 3-21A for its own subsystem checkout. The first five bits are used for a flag word to yield a quick status check of the DE OSE and the data validity and are as follows:

(a) Δ – out of limit indication.

(b) Data Mode – indicates if data mode word has changed since last reading.

(c) Low-Speed Deck Position – indicates when the low-speed deck position word has not increased by one since the last reading.

(d) Medium-Speed Deck Position – indicates when the medium-speed deck position word has not increased by one since the last reading.

(e) Frame Sync – indicates loss of frame sync.

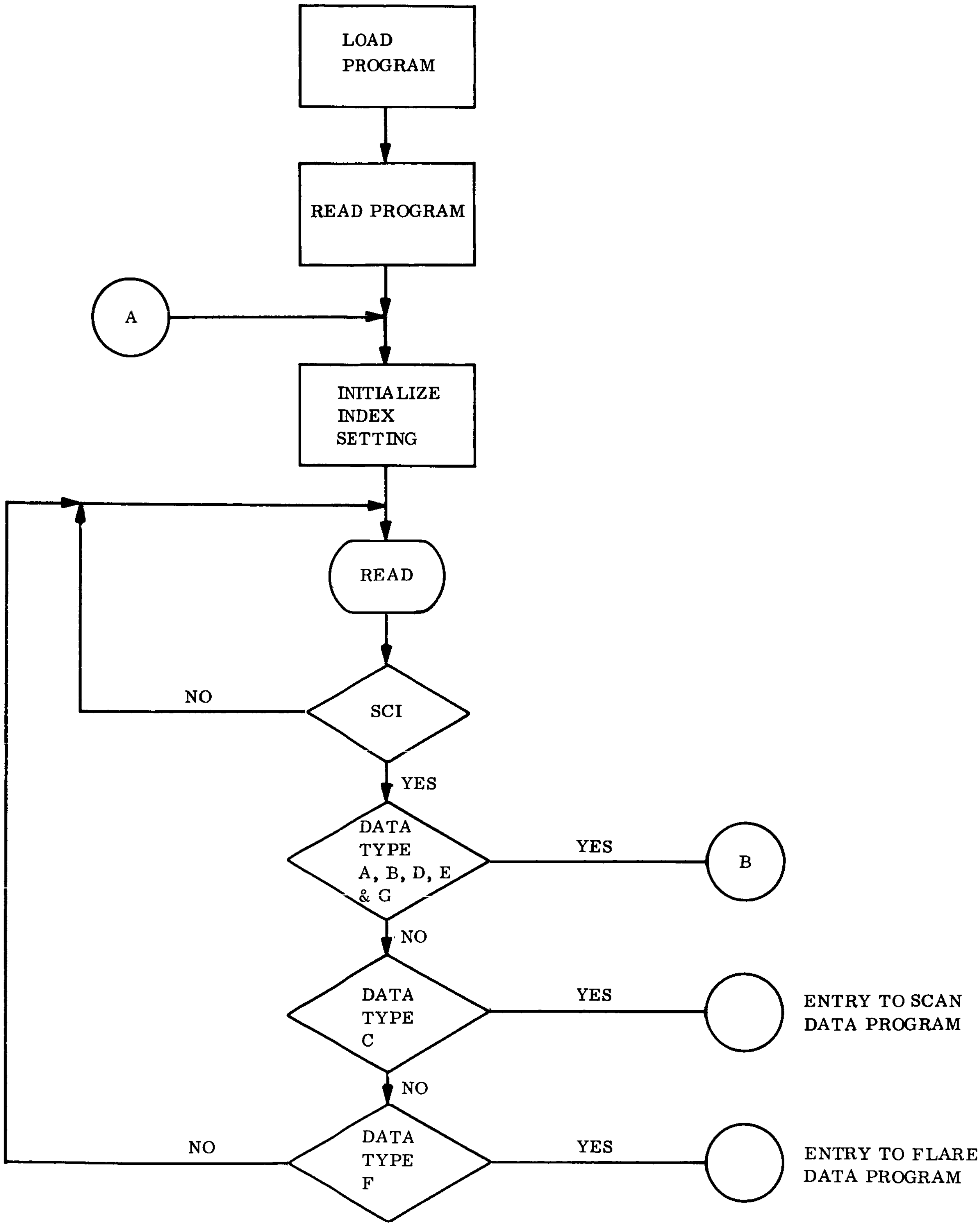


Figure 3-13. Flow Chart 1

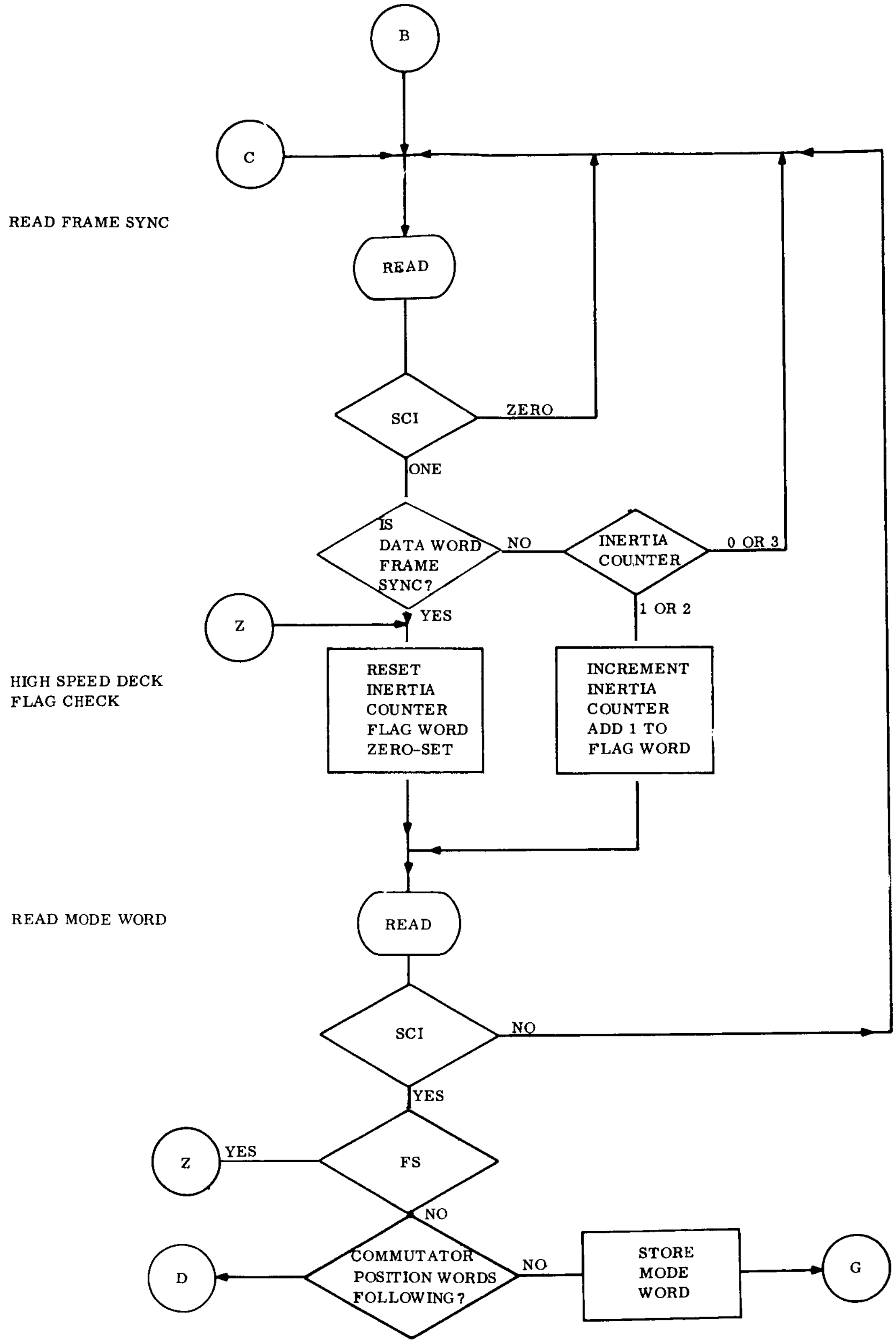


Figure 3-14. Flow Chart 2

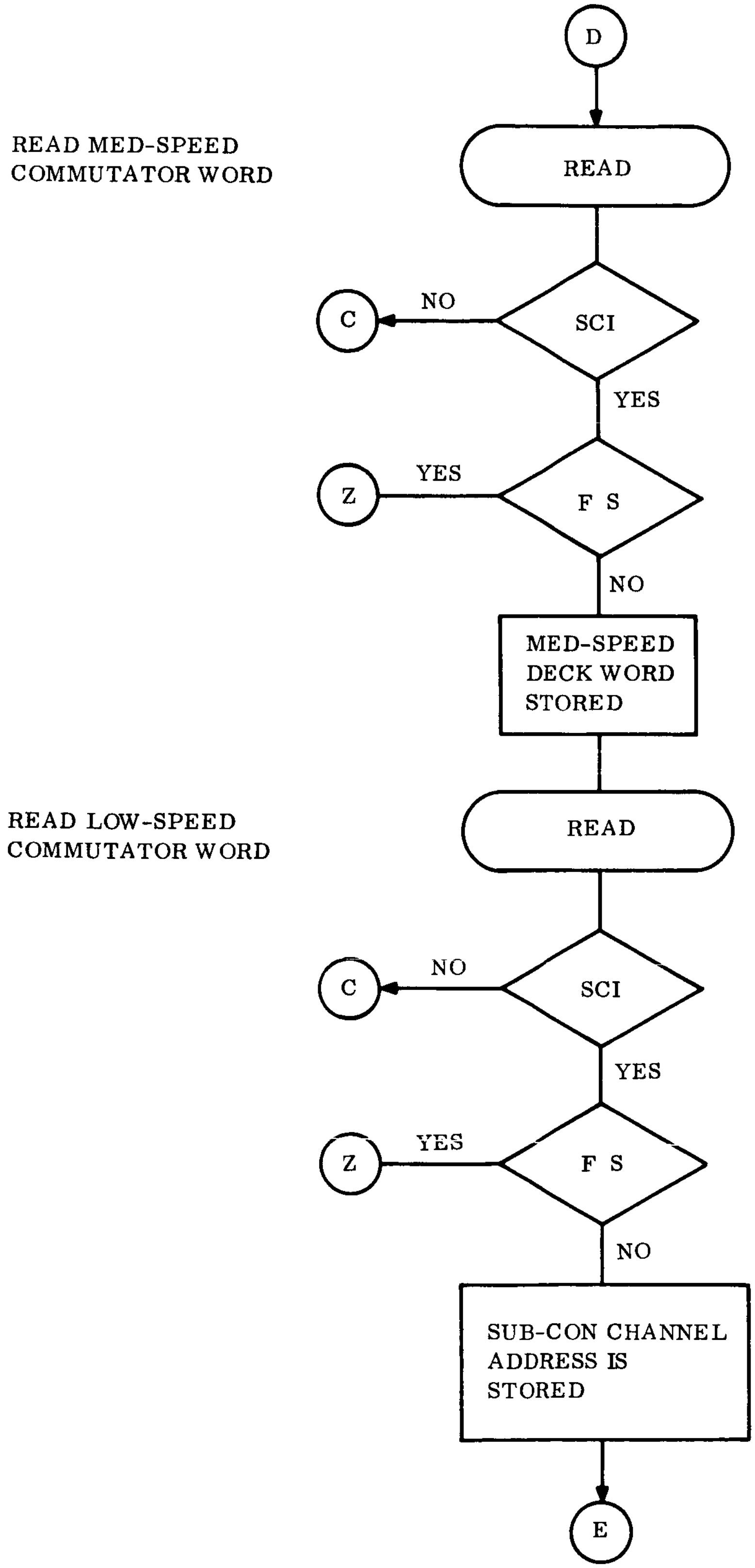


Figure 3-15. Flow Chart 3

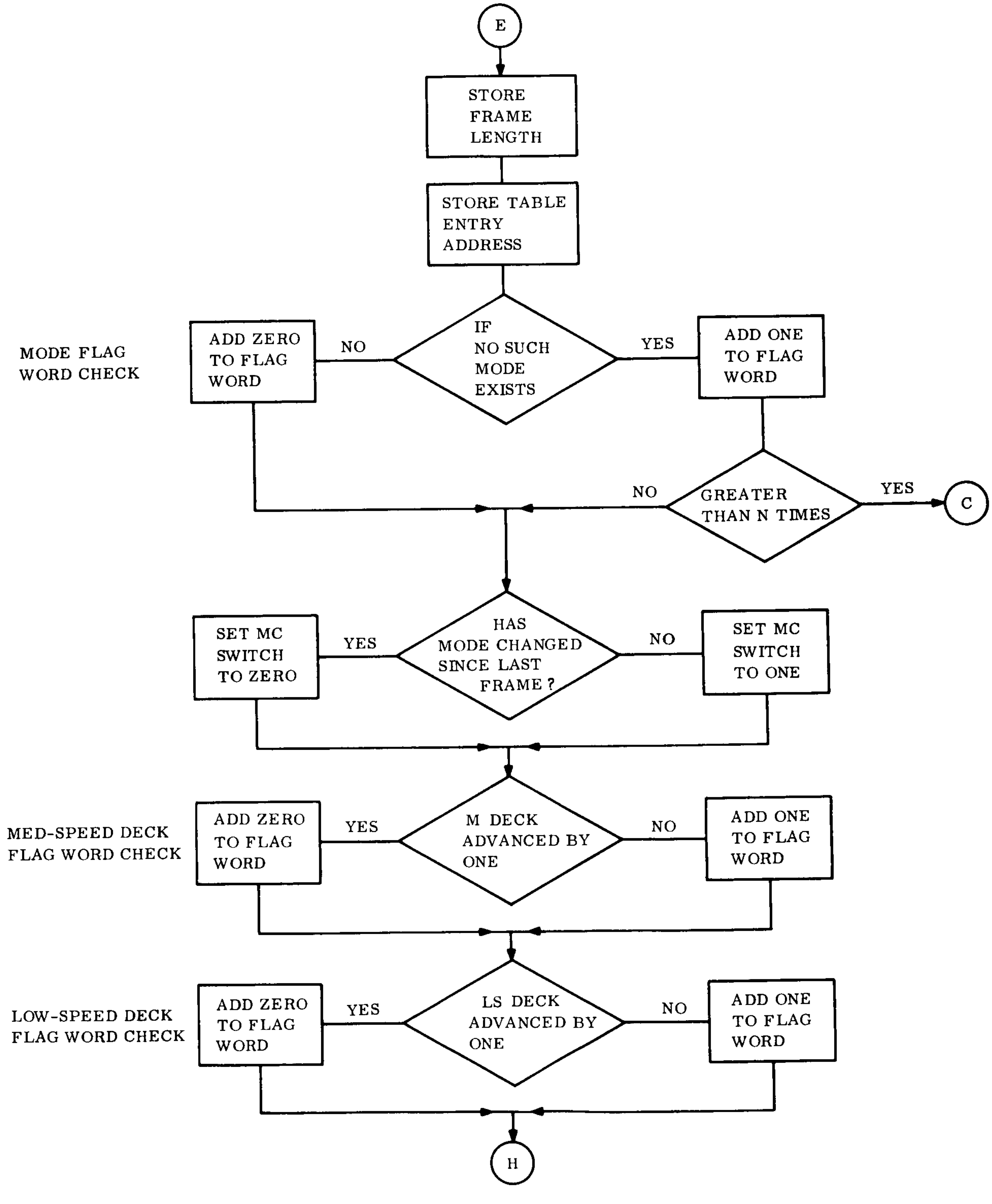


Figure 3-16. Flow Chart 4

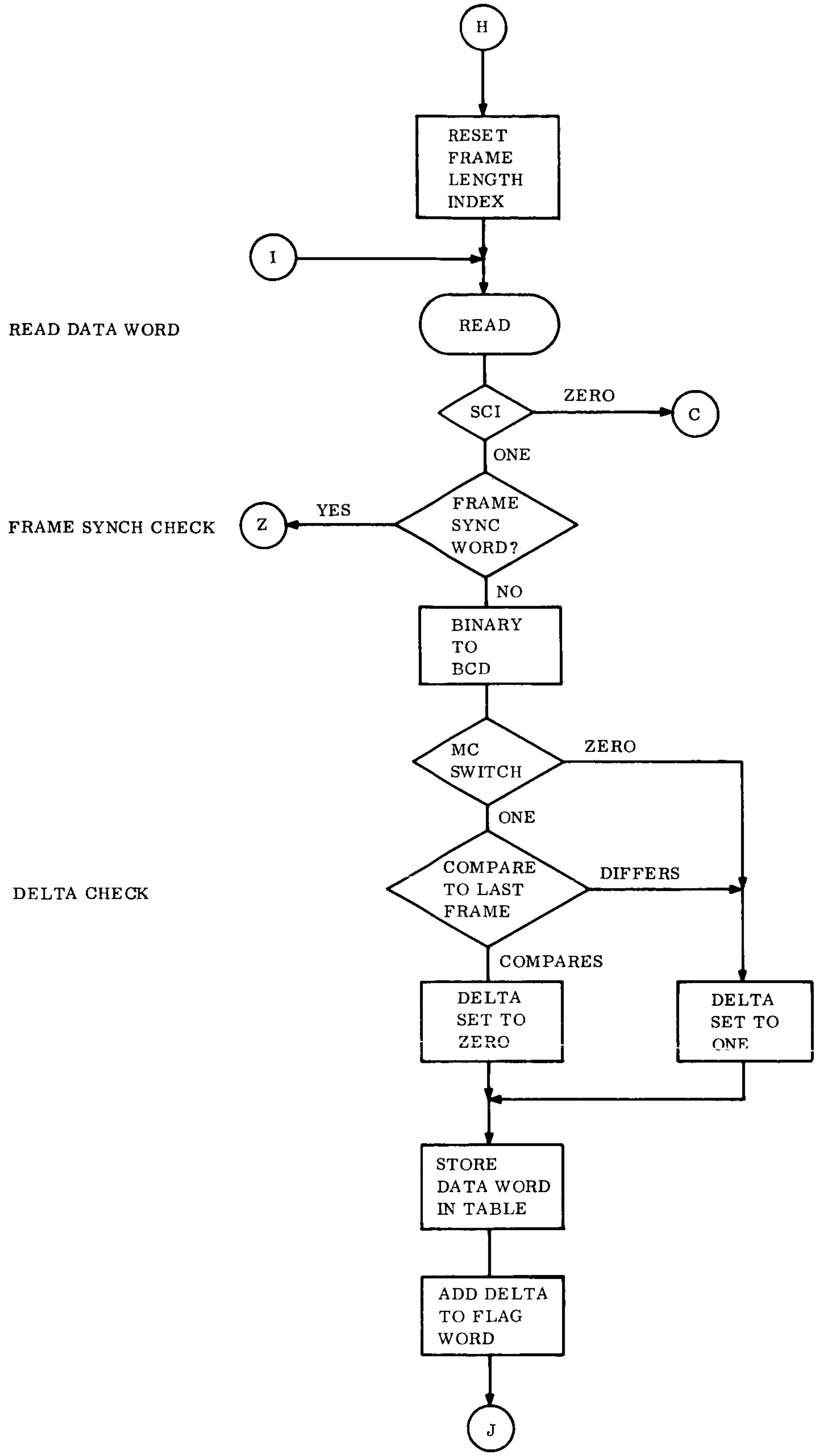


Figure 3-17. Flow Chart 5

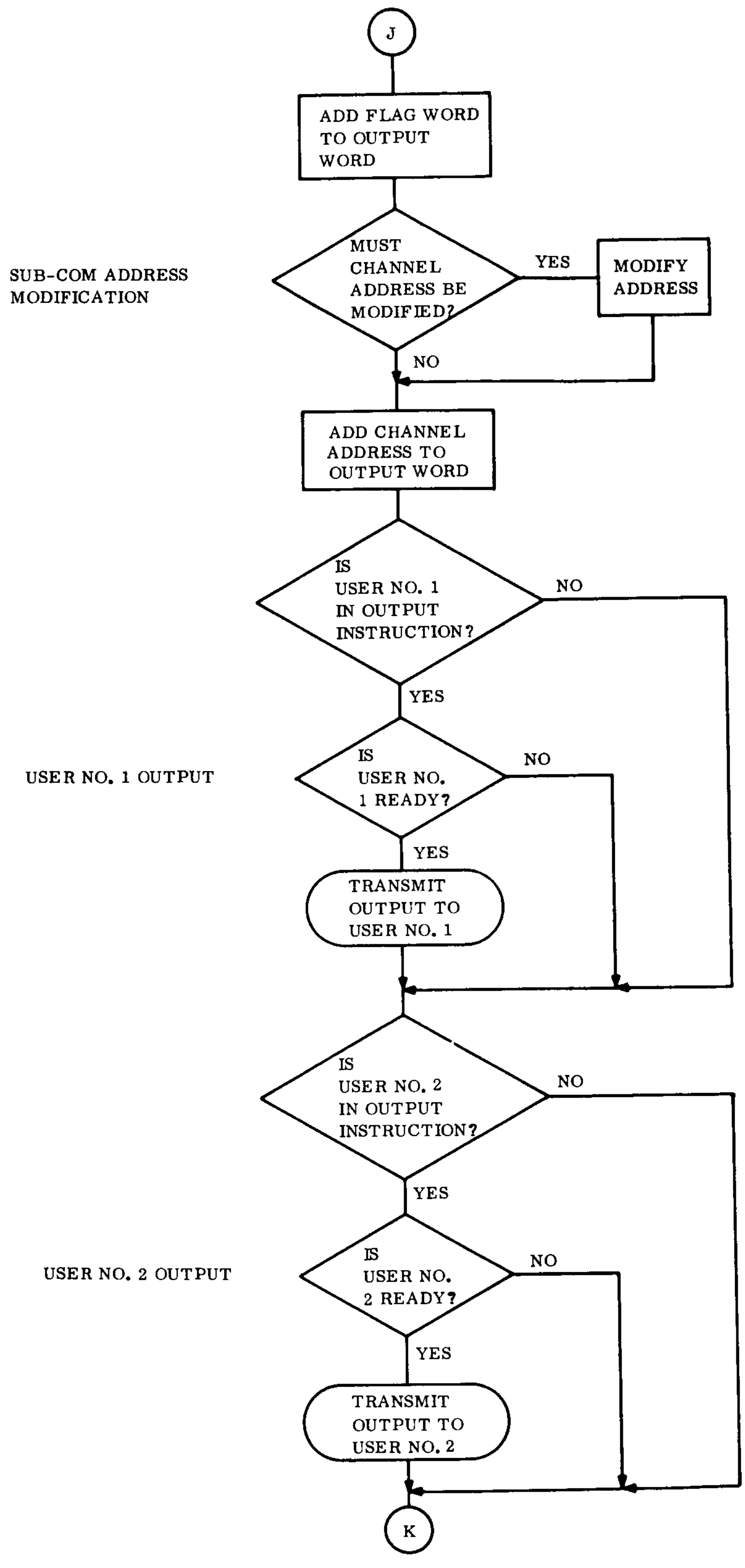


Figure 3-18. Flow Chart 6

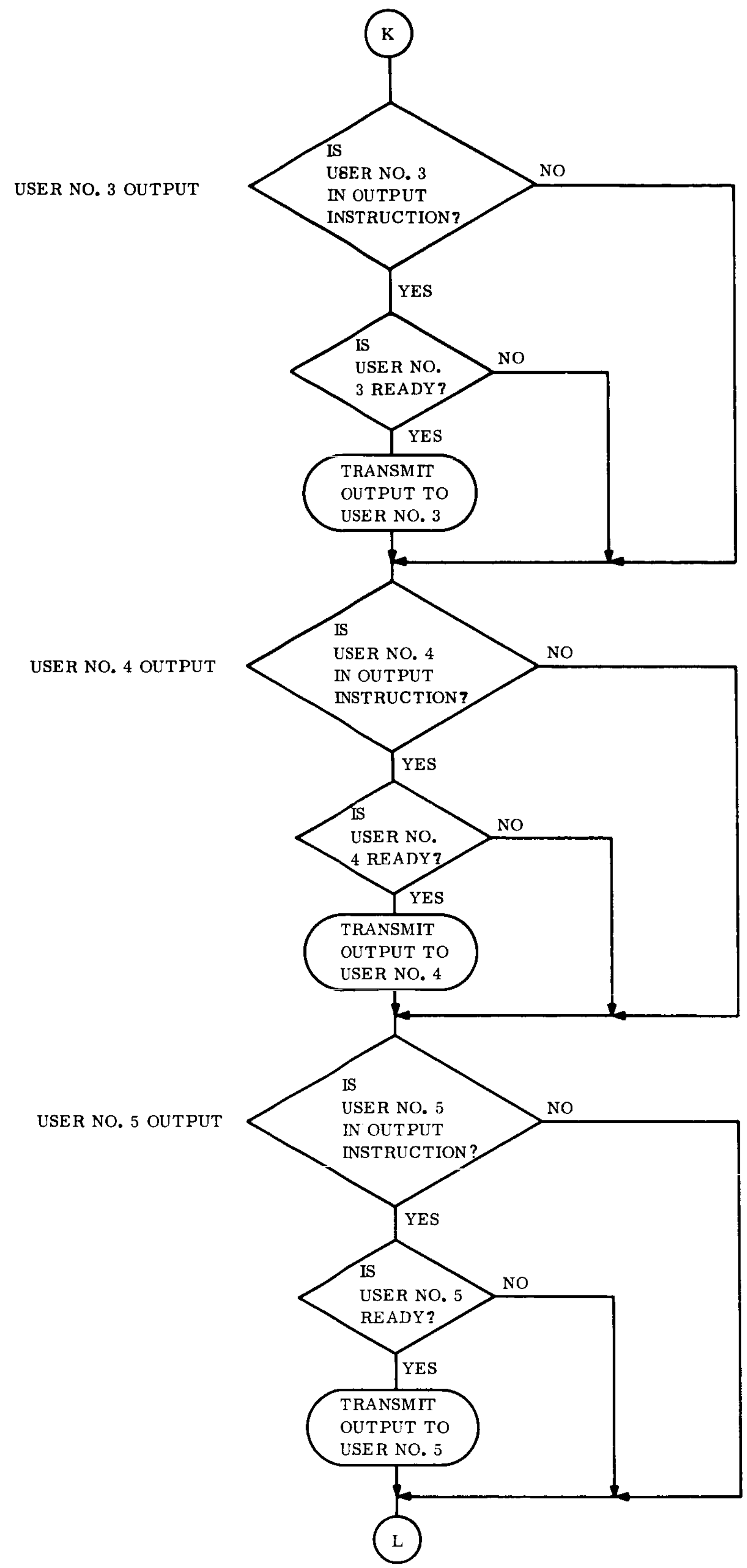


Figure 3-19. Flow Chart 7

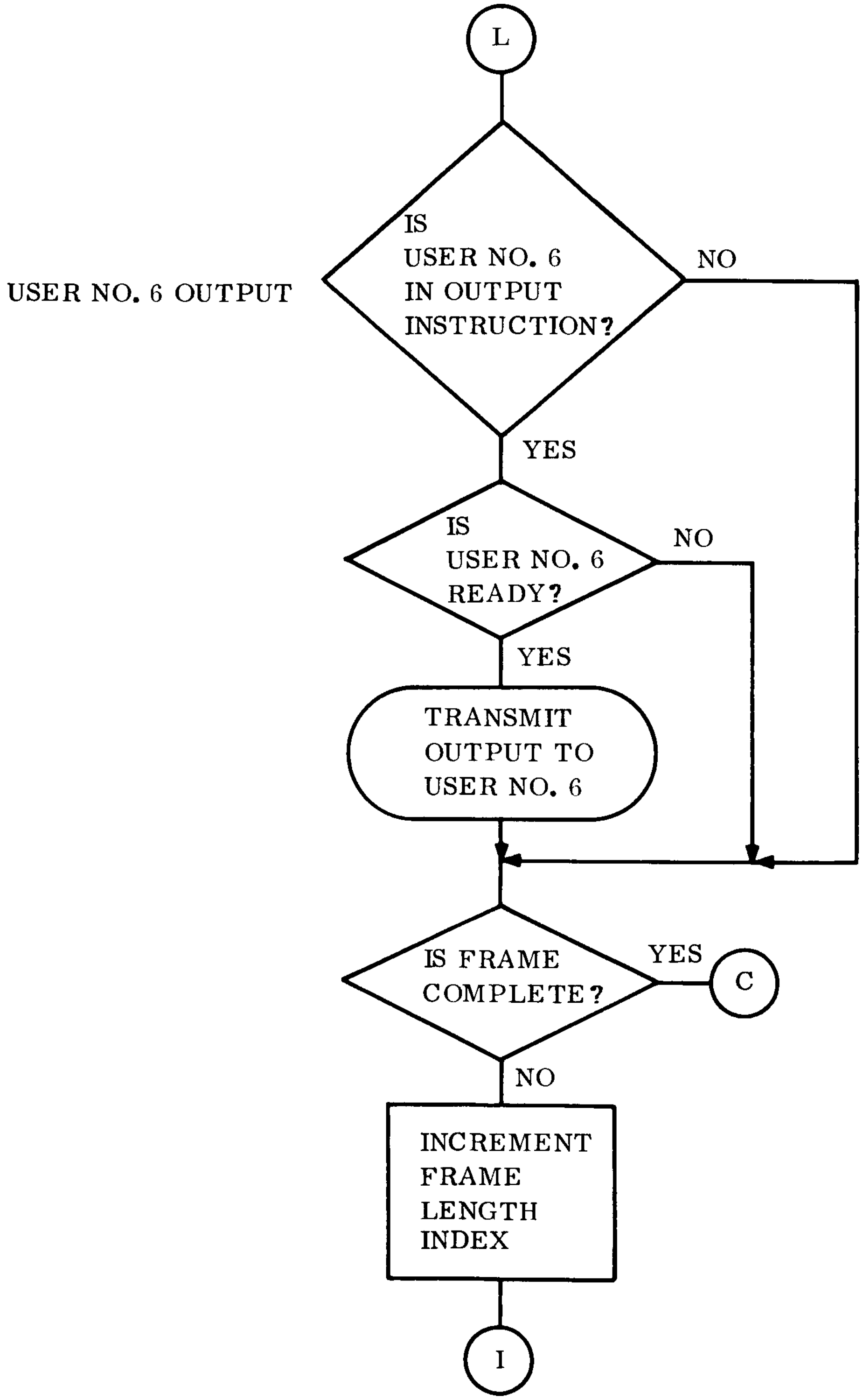
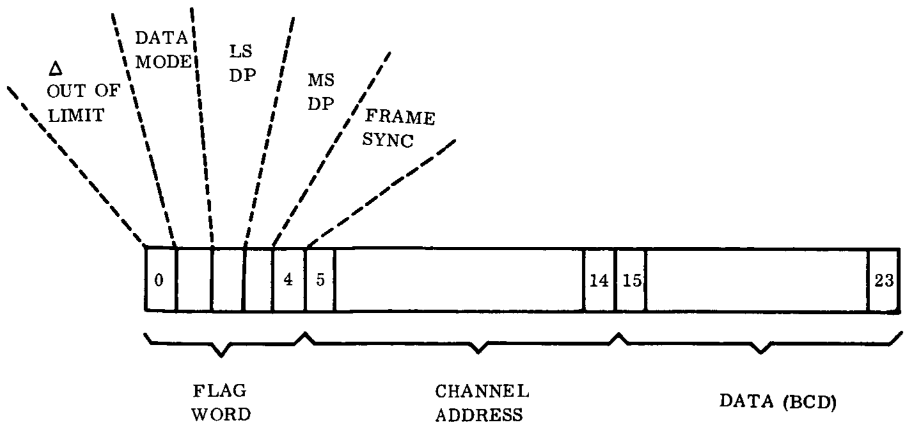
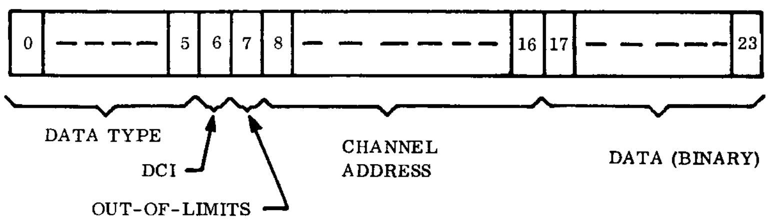


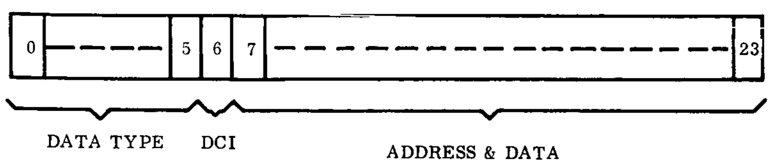
Figure 3-20. Flow Chart 8



A. DE OSE OUTPUT DATA FORMAT



B. ENGINEERING DATA



C. NON ENGINEERING DATA

NOTE

1. CDS ~ COMPUTER DATA SYSTEM
2. TDC ~ TELEMETRY DATA CONVERTER (B ONLY)
3. TCC ~ TEST CONDUCTOR'S CONSOLE

Figure 3-21. Data Encoder OSE and CDS, and Telecommunications Data Formats

The next ten bits are the channel address, while the last nine bits are the data (BCD).

The CDS requires an 18-bit word input while the decommutator output is a 24-bit word. Therefore, for engineering data, the input to the CDS is:

Δ Out of limit	1 bit
DCI - Data Condition Indicator (DCI is derived from the last four bits of the previously defined flag word OR'd together).	1 bit
Channel Address	9 bits
Data (Binary)	7 bits

For science, the data is presented to the CDS in parallel, one science word at a time. One bit is DCI, while the remaining 17 bits are channel address and data bits.

To identify the data type, the remaining six bits of the SDS 920 24-bit computer word is used as data type bits. The CDS can input these bits separately from its 18-bit data input. Figure 3-21B and C show the effective output format to the CDS and TCC. The TCC receives the same formats as the CDS.

3.1.7 TAPE RECORDER

The tape recorder is used to record and play back data during system test. The tape recorder will record seven channels with a track assignment as follows:

a. Ground instrumentation (FM subcarrier mix)

b. Data (Digital FSK)

c. Composite telemetry signal (FM)

d. Bit sync (Digital FSK)

e. Time and wow/flutter compensation (Direct)

f. Word sync (Digital FSK)

g. Voice lable and Intercom (Direct)

A tape recorder capable of a linear phase response in the bandwidth 0.45R to 72 R (R Δ data rate) is included in the subsystem.

3.1.8 TIME CODE TRANSLATOR

A time code translator capable of synchronizing to a serial 36-bit NASA time code is included in the STC console. It generates a BCD parallel data format for use by the data printout equipment and displays this time on the front panel in decimal form (days, hours, minutes, and seconds).

3.1.9 BIT ERROR RATE MONITOR

A bit error rate monitor unit is included in the STC console which can automatically check and monitor the bit error rate of the noise-corrupted data being detected and decoded by the demodulator using the hard-line data encoder as its reference.

3.1.10 ELAPSED TIME METER

An elapsed time meter is provided for recording total operating time for the S/C data encoder.

3.1.11 S/C POWER SUPPLY

A 2400-cps square wave power supply is provided. The power supply provides primary power to the spacecraft data encoder during subsystem testing.

3.1.12 OSE POWER SUPPLY

The OSE generates its own special power requirements. The OSE is provided with 120-volt, 60-cps, single-phase power for converting to its own individual needs.

3.1.13 LINE PRINTER

A high-speed line printer capable of printing 800 lines per minute and 132 characters per line is provided. The printer can be programmed for printing data, channel address, flag information time, procedure step number, data type and data mode.

3.1.14 PAPER TAPE PUNCH

A paper tape punch is provided for punching computer programs, readout instructions, etc. The puncher is capable of punching 60 characters per second.

3.1.15 PAPER TAPE READER

A paper tape reader is provided for reading into the computer decommutation programs, users instructions, format information, etc. The paper tape reader is capable of reading 300 characters per second.

3.1.16 I/O TYPEWRITER

An input/output typewriter capable of 15 characters per second input and output rate is provided.

3.1.17 SPECIAL EQUIPMENT

Special Equipment such as J-boxes, isolation amplifiers, digital output isolators, and cables are supplied as necessary.

3.1.18 DATA ENCODER DISPLAY

The DE OSE provides information displays such as panel lighted windows or lamps. These displays are on the basis of aiding system and subsystem checkout. Examples of these are:

a. Channel Data (selected and BCD)

b. Commutator Deck Position

c. Data Type

d. Data Mode

e. Data condition indicator

f. Spacecraft ID

g. Spacecraft time

h. Events

i. Procedure step number

j. Redundant configuration

3.1.19 TEST EQUIPMENT

a. Counter – This counter receives data encoder bit sync and displays bit rate of operation.

b. Monitor Oscilloscope – In the tape recorder console a monitor oscilloscope is included to verify performance of the tape machine during all recording operation.

c. Digital Volt Meter – This is used to monitor the simulated analog signals from the OSE.

d. Oscilloscope – This oscilloscope is used to monitor OSE and data encoder service during checkout.

e. Recorder – This recorder provides a printed record of the data encoder operation and a channel address and a digital value of the information, in the recording channel.

3.2 SELF TEST

The DE OSE is capable of performing self test during system and subsystem test. Self test may be run without interrupting system test procedures with the exception that the DE OSE cannot decommutate simulated and real data simultaneously. The DE simulator simulates the various types of data and at any one of the six bit rates, thus allowing a complete subsystem check out.

3.3 SYSTEMS TEST

The DE OSE, shown in Figure 3-22 is required to support systems and subsystems test at the STC. The OSE provides all the command and control requirements, data simulation or stimulation equipment as well as monitoring the spacecraft performance. S/C DE performance is determined by decommutating telemetry data via hard line or RF link. While using hard-line monitors a deck synchronizer contained in the command and control unit, will be employed to supplement the main decommutator. The deck synchronizer is slaved to the S/C, however, it does not allow complete check out of the spacecraft though it does add to the flexibility of the DE OSE. Shown in Figure 3-22 is a block diagram of the system test configuration.

3.4 SUBSYSTEM TEST

The DE may be tested as an electrically isolated subsystem while still installed in the spacecraft at the STC. S/C power will be supplied by the OSE during this particular test while performing similar tests as in the system test configuration. DE calibration is performed during subsystem testing, though calibration is not limited to subsystem testing. The DE OSE will automatically stimulate each DE channel for testing and calibrating, displaying, printing or storing the results.

4.0 INTERFACE DEFINITION

The DE OSE interfaces are defined in the tables listed below:

Table 4-1	DE OSE and Radio Subsystem OSE
Table 4-2	DE and DE OSE Umbilical Interfaces
Table 4-3	DE and DE OSE Direct Access Interfaces
Table 4-4	DE OSE and CDS Interface

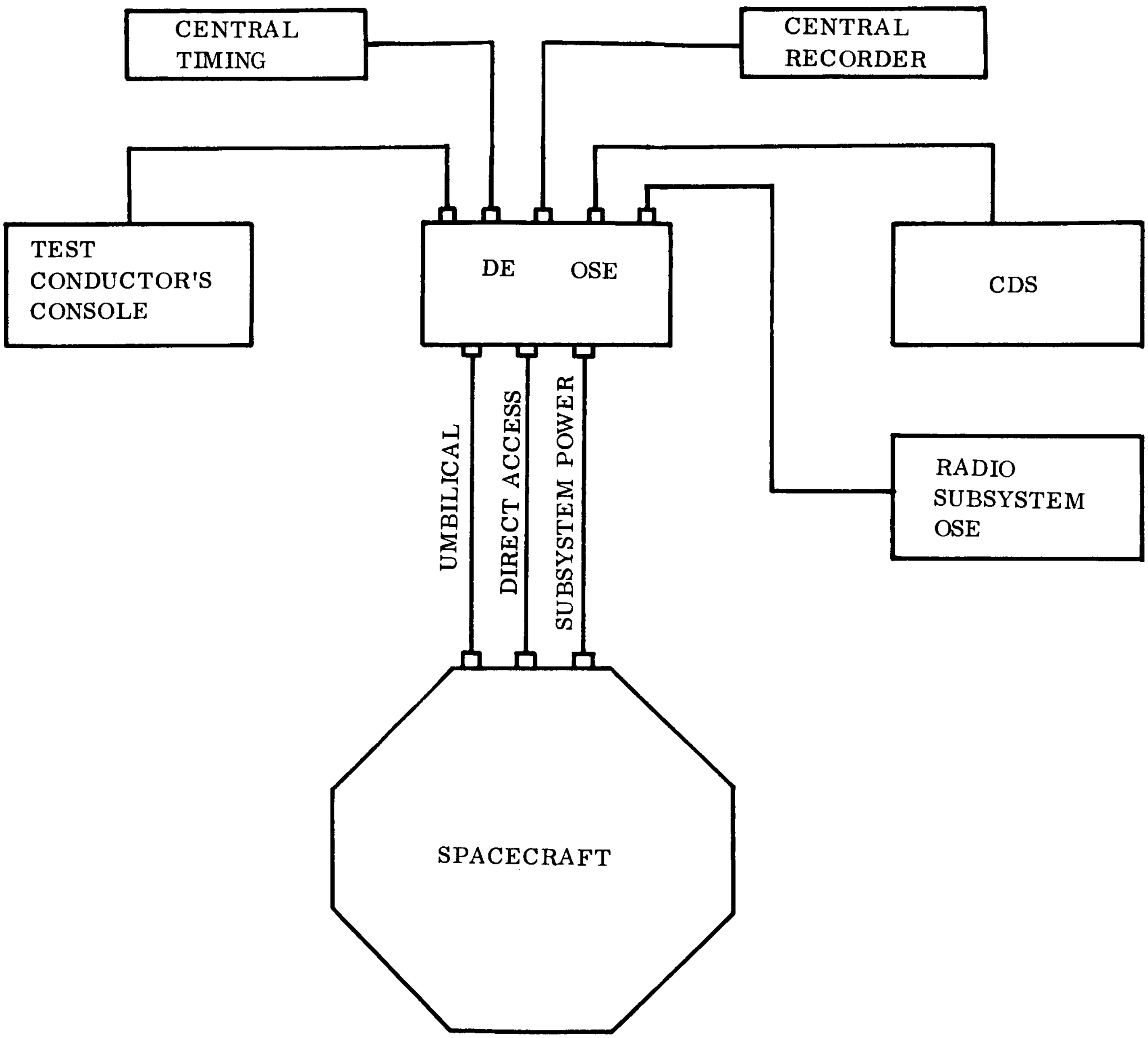


Figure 3-22. Data Encoder OSE, System Test Complex, System and Subsystem Test Configuration

Table 4-5	DE OSE and Test Conductor's Console Interface
Table 4-6	DE OSE and Central Recorder Interface
Table 4-7	DE OSE and Central Timing Interface
Table 4-8	DE OSE and CVE Interface

Table 4-1. DE OSE and Radio Subsystem OSE

Function	Characteristic	Impedance (OHMS)	Comments
S/C Telemetry Signal	0.3 to 3.0 v rms	10K	Single Subcarrier Bi-phase modulated with data and sync

Table 4-2. DE and DE OSE Umbilical Interface

Function	Characteristic
1. Modulated Subcarrier A	Modulated square wave
2. Modulated Subcarrier B	Modulated square wave
3. Modulated Subcarrier C	Modulated square wave
4. Word Sync	Positive going 3.5-volt pulse
5. Bit Sync	Positive going 3.5-volt pulse
6. Frame Sync	Positive going 3.5-volt pulse
7. Mode II Command	Positive going 3.5-volt pulse 3 ms or greater duration
8. 2 f_s	Square wave
9. Data Encoder Return	

Table 4-3. DE and DE OSE Direct Access Interfaces

Function	Signal Characteristic
1. Monitor A/D Converter No. 1	Voltage level
2. Monitor A/D Converter No. 2	Voltage level
3. Monitor A/D Converter No. 3	Voltage level
4. Monitor P/N Generator No. 1	Voltage level
5. Monitor P/N Generator No. 2	Voltage level
6. Monitor P/N Generator No. 3	Voltage level
7. Monitor Programmer No. 1	Voltage level
8. Monitor Programmer No. 2	Voltage level
9. Monitor Programmer No. 3	Voltage level
10. -20-volt TR Monitor	Voltage level
11. +3.5-volt TR Monitor	Voltage level
12. Data Encoder Return	

Table 4-4. DE OSE and CDS Interface

Function	Characteristic
1. Decommutated telemetry data	See Figure 3-21
2. Command request	Switch closures
3. Telemetry data	Analog inputs from telemetry data converter

Table 4-5. DE OSE and Test Conductor's Console Interface

Function	Characteristic
1. DE subsystem status and mode indicators	Switch closures
2. Decommutated telemetry data	See Figure 3-21

Table 4-6. DE OSE and Central Recorder Interface

Function	Characteristic
Sync Condition Indication (SCI)	Voltage level indicating demodulator Synchronization

Table 4-7. DE OSE and Central Timing Interface

Function	Characteristic
Central timing signal	Binary time code

Table 4-8. DE OSE and Command Verification Equipment Interface

Function	Characteristic
Lock Indication	Switch

5.0 PERFORMANCE PARAMETERS

5.1 TRANSMISSION MODES

The telemetry date received from the spacecraft is decommutated for the six Data Modes listed below:

Maneuver Mode - Mode I (3 1/3 bps)	Selected engineering data
Cruise Mode - Mode II (106 2/3 bps)	Engineering data multiplexed with capsule data and non-scanned science data.
Orbit Mode - Mode III (8533 1/3 bps)	Planet scan data blocks, with engineering and non-scanned science data inserted between scan data playback intervals.
Memory Dump Mode - Mode IV (2133 1/3 bps)	Blocks of engineering and capsule data or blocks of flare data.
Non-Scan Orbital Mode - Mode V (8533 1/3 bps)	Engineering data multiplexed with non-scanned science data.
Approach Guidance Mode - Mode VI (106 2/3 bps)	G&C data multiplexed with the normal cruise format.

The operational modes are keyed to a nominal bit rate as listed, but may be documented at any of the possible bit rates.

5.2 DATA TYPES

The seven data types and their associated modes are given in Table 3-2. At each data type change a 56-bit preamble code word is transmitted prior to sending data. The preamble consists of a unique 49-bit preamble sync followed by a seven-bit data type word. The decommutator is programmed to accomodate all data types.

5.3 BIT RATES

The DE OSE operates over six different bit rates as shown in Table 5-1. The bit rates will have an accuracy of 0.01% or better. The bit rate is one ninth of the subcarrier frequency, 2 f_s (2 f_s = 9R).

5.4 DYNAMIC RANGE

The demodulator operates over an input dynamic range of 0.3 volt to 3.0 volt rms signal plus noise in an 8 f_s bandwidth.

Table 5-1. DE OSE Bit Rates

Rate Number (N)	Bit Rate (bps)	R_1/R_N	2 f_s (C/S)
1	8533 1/3	1	76.8K
2	4266 2/3	2	38.4K
3	2133 1/3	4	19.2K
4	533 1/3	16	4.8K
5	106 2/3	80	960
6	3 1/3	2560	30

5.5 BIT ERROR RATE

The bit error rate at system threshold for each data rate is 5 x 10^{-3}.

5.6 DATA CHANNEL THRESHOLD SNR

The threshold signal-to-noise ratio in a one-cps noise bandwidth that must be present at the demodulator data detector are shown in the Table 5-2 for the six data rates.

Table 5-2. Data Channel Threshold SNR

Bit Rate (bps)	E/No (db)	S/No = (E/No)R (db)
8533 1/3	5.9	45.2
4266 2/3	5.9	42.2
2133 1/3	5.9	39.2
533 1/3	5.9	33.2
106 2/3	5.9	26.2
3 1/3	8.0	13.2

5.7 CLOCK CHANNEL THRESHOLD SNR

The clock tracking phase-locked loop threshold signal-to-noise ratio in the threshold noise bandwidth is given in Table 5-3. Acquisition and tracking bandwidth values are shown also and will exhibit an accuracy of ± 10%.

Table 5-3. Clock Tracking PLL Threshold SNR

Bit Rate	Acquisition BW $2B_{LO}$ (cps)	Tracking BW $2B_{LO}$ (cps)	Tracking SNR (db)
8533 1/3	32	8	>24.0
4266 2/3	$16\sqrt{2}$	$4\sqrt{2}$	>24.0
2133 1/3	16	4	>24.0
533 1/3	8	2	>24.0
106 2/3	4	1	24.0
3 1/3	1	1/4	17.0

5.8 AVERAGE ACQUISITION TIME

The average acquisition time, or mean time to acquire phase lock, for the threshold SNR's, thus allowing data detection, is shown in Table 5-4 for each bit rate, and no decision errors.

Table 5-4. Average Acquisition Time

Bit Rate (bps)	Mean Acquisition Time (seconds)
8333 1/3	0.258
4266 2/3	0.517
4133 1/3	1.034
533 1/3	4.13
106 2/3	20.64
3 1/3	132.3

5.9 PROBABILITY OF LOSS OF SYNC

The probability of loss of bit and word sync dhring a one-second period for threshold SNR at the data detector is approximately:

$$P_L^b = P_L^w = \begin{cases} <10^{-7} & \text{for 3-1/3 bps} \\ <10^{-6} & \text{for 106 2/3 bps or greater.} \end{cases}$$

5.10 FREQUENCY STABILITY

The frequency stability of the incoming 2 f_s subcarrier will be 0.01% or better.

5.11 SIMULATED DATA

a. HIGH-LEVEL ANALOG

Voltage - 0 to 3.2 volts dc (seven steps) Impedance - less than 10 k

b. LOW-LEVEL ANALOG

Voltage - 0 to 100 millivolts dc (seven steps) Impedance - less than 10 k

c. BIPOLAR ANALOG

Voltage - ±1.6 volts dc (seven steps) Impedance - less than 10 k

d. DIGITAL PULSES (EVENTS)

Voltage - "0" = 0 volts dc; "1" = 3.5 volts dc

Time Duration - 3 milliseconds

e. SERIAL DIGITAL DATA

Type - NRZ; synchronous with D/E bit sync

f. PARALLEL DIGITAL DATA

Voltage - "0" = 0 volts dc; 1 = 3.5 volts dc

Type - Pulse data dumped in parallel on receipt of D/E dump pulse.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 GENERAL

All incoming and outgoing signals are through the DE OSE junction box. The junction box will provide connector plugs for test cables, umbilical cables, ac power, and interfaces with the Central Recorder, Computer Data System, Science, Command Subsystem OSE, Radio Subsystem OSE, and Central Timing and Synchronization System. All in-going and outgoing signals are routed to the proper location in the junction box.

6.2 SIZE

The DE OSE is provided in eight 19-inch panel bay racks plus one line printer.

6.3 WEIGHT

The DE OSE does not exceed 8000 pounds.

6.4 POWER

The DE OSE does not require more than 6 kilowatts of 105-125-volt, 60-cps, single-phase power.

7.0 SAFETY CONSIDERATIONS

7.1 EQUIPMENT

The DE OSE is provided with isolation devices to minimize spacecraft-to-OSE operational difficulties as well as OSE-to-OSE operations. The equipment is designed with good engineering integrity and human engineering. The OSE is designed to minimize, if not negate, the chances of jeopardizing the spacecraft equipment due to engineering or faulty circuitry.

7.2 FACILITIES

The DE OSE is designed so as not to be hazardous to the facilities

7.3 PERSONNEL

The design of the DE OSE includes good human factors engineering from an operational and safety viewpoint.

CII - VB264FD105

SYSTEM TEST COMPLEX

CONTROLLER AND SEQUENCER OPERATIONAL SUPPORT EQUIPMENT

Index

1.0 SCOPE

This document contains a functional description and interface definitions for the Controller and Sequencer Operational Support Equipment (OSE) for the 1971 Voyager System Test Complex (STC) and for subsystem test.

2.0 APPLICABLE DOCUMENTS

The following documents form a part of this description:

VB260SR101	STC Test Objectives and Design Criteria
VB260SR102	STC Design Characteristics and Restraints
VB234FD107	Controller and Sequencer, Functional Description

3.0 DESCRIPTION

3.1 GENERAL

Figure 3-1 is a block diagram of the Controller and Sequencer (C&S), which is to be tested and evaluated by the Operational Support Equipment described in this document. Both equipments are digital devices.

The C & S OSE has four modes of operation:

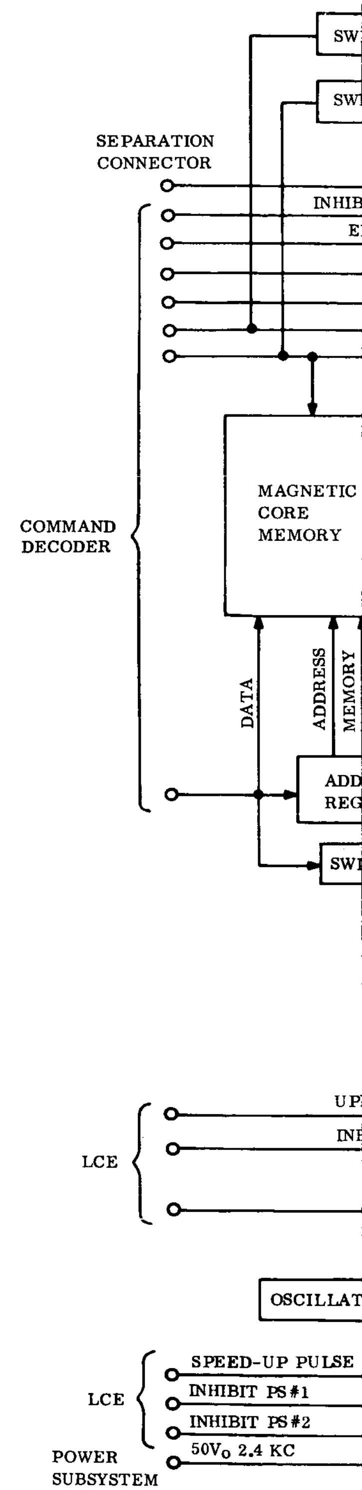
a. System Test - With the C&S installed and connected in the Spacecraft, the OSE determines whether or not the C&S is fully operational. This is accomplished by loading commands through the Command Decoder, and monitoring the response time of the Event output via Telemetry.

b. Subsystem Test - The OSE tests and evaluates the C&S by simulating or monitoring all interfaces. When the C&S is not connected to receive Spacecraft power the OSE simulates the power input.

c. Fault Isolation - The OSE utilizes the C&S interfaces plus test points to isolate malfunctions to the next lower level of assembly. (This mode is not used in conjunction with the STC.)

d. Self-Check - The OSE determines its own operability by monitoring test stimuli and exercising monitor inputs.

Figure 3-2 is a block diagram of the Controller and Sequencer OSE. It shows the functions necessary to perform the C&S tests listed above. In addition, it is necessary for the OSE to demonstrate its own operability. To that end, the OSE generates stimuli to exercise all input circuits and to test all output circuits.


SW
SW
SEPARATION
CONNECTOR
INHIB
COMMAND
DECODER
MAGNETIC
CORE
MEMORY
DATA
ADDRESS
MEMORY
ADD
REG
SW
UP
INH
LCE
OSCILLAT
LCE
SPEED-UP PULSE
INHIBIT PS #1
INHIBIT PS #2
50V₀ 2.4 KC
POWER
SUBSYSTEM

3

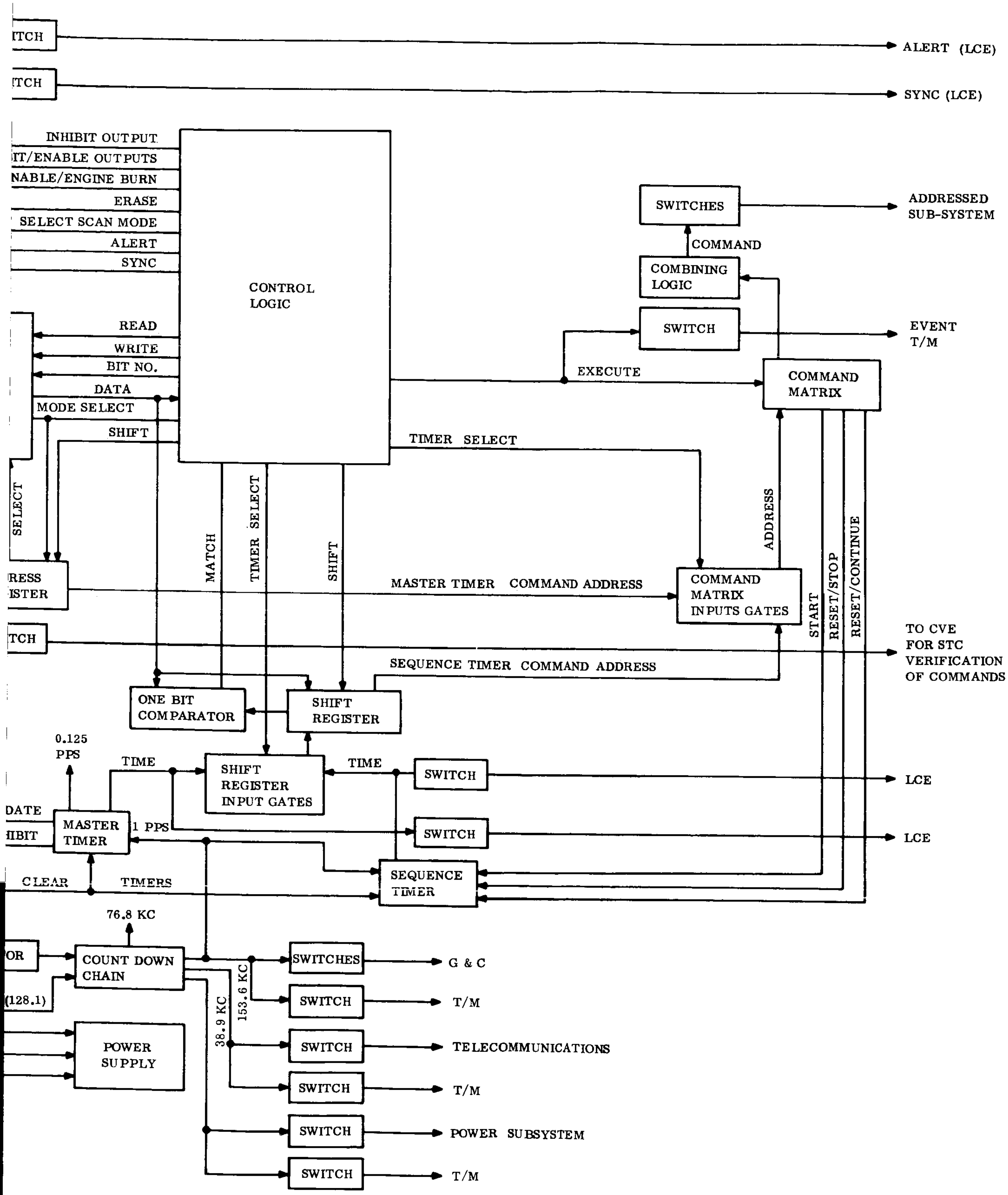


Figure 3-1. Controller and Sequencer, Block Diagram

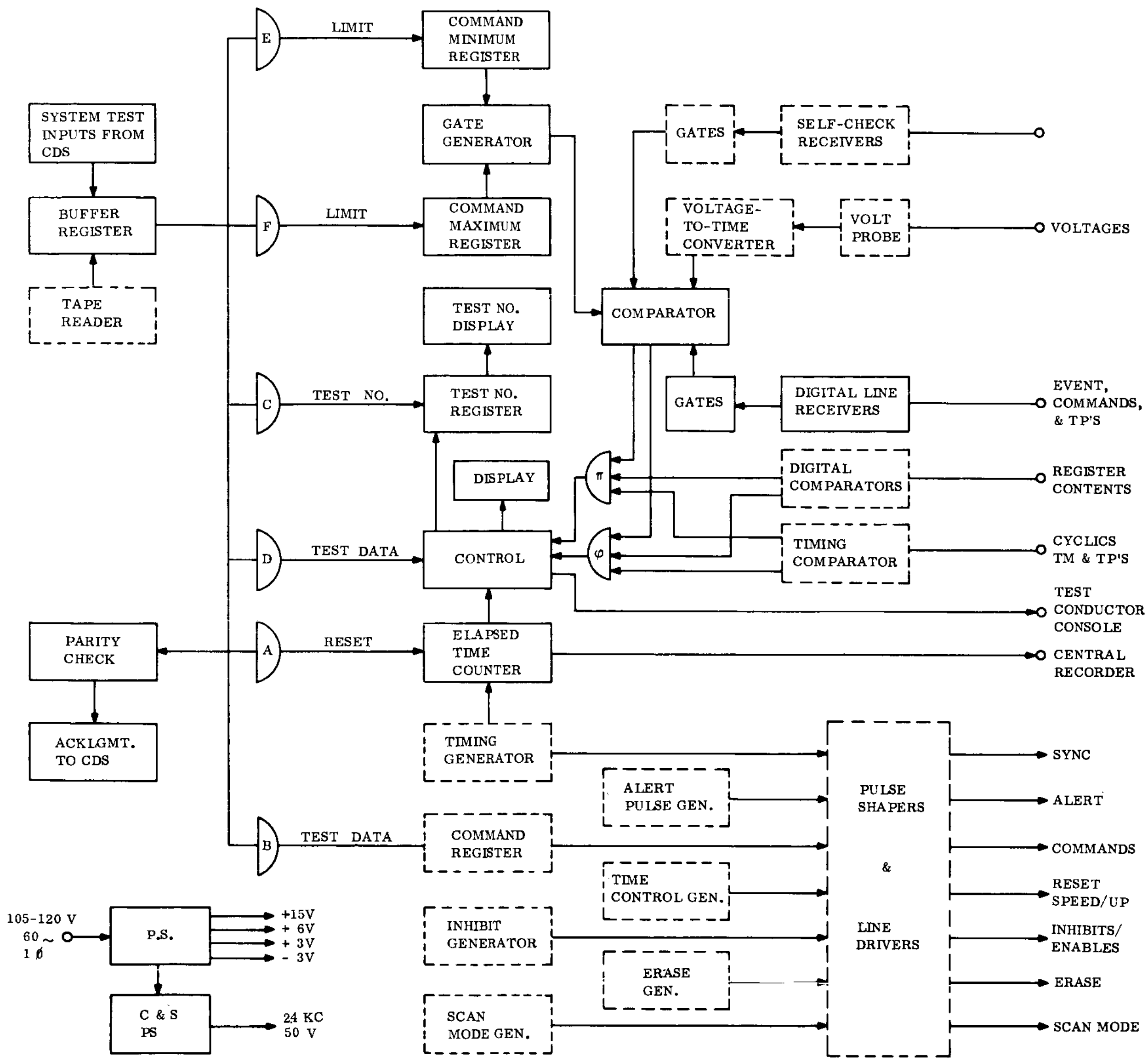


Figure 3-2. Controller and Sequencer OSE, Block Diagram

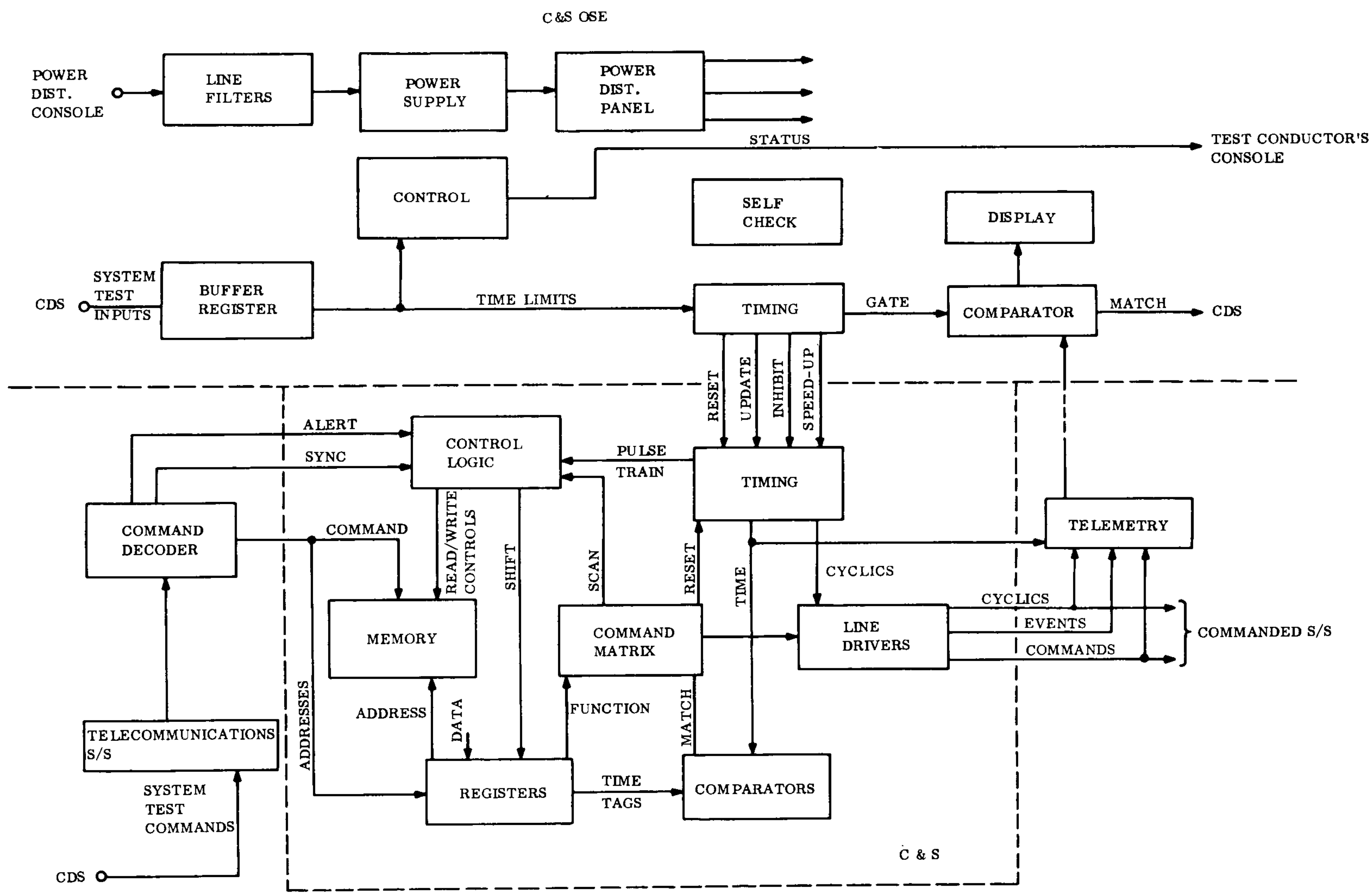


Figure 3-3. Controller and Sequencer at System Test Complex, Test Block Diagram

The primary mode of operation of the OSE is automatic. During System Test, test control inputs are received from the Computer Data System while the other test modes are controlled by test parameters stored on punched tape and read by a tape reader which is a part of the OSE.

Alternate test input modes, TAPE, MANUAL, and SYSTEM, are selected before the start of a test by means of INPUT SELECT switch. Provision is made for calibration of the power supply voltage and oscillator frequency and for introducing marginal test conditions. The Input Buffer converts input test data and propagates them into the Test Set Registers at appropriate speeds. (Alternate input buffers are provided for the various modes.)

In addition to providing the C&S with a burst of commands during subsystem tests, the OSE generates Sync Pulses, an Alert Pulse, a Timer Speed Up Pulse, a Timer Reset Pulse and a Timer Inhibit Pulse. (During system tests, however, these inputs are supplied by the Command Decoder as shown in Figure 3-3, the Test Block Diagram.) The inputs which the OSE provides to the C&S are described in Table 4-2. The conditions of all of these signals, which are needed to exercise the C&S, can be programmed for Nominal, Maximum Worst Case and Minimum Worst Case.

Gating within the OSE is generally of two kinds; repetitive and test dependent. The repetitive gating and clock pulses are provided by the Timing Generator. The other gates are operated by signals from the Control Logic, depending on the particular test being performed. (The timing and control paths are not shown in Figure 3-2 because it is a simplified block diagram.)

The Controller and Sequencer has two kinds of outputs: cyclics and stored commands. The cyclics are square wave signals used by Spacecraft subsystems for reference frequencies. The stored commands are pulses issued with one second resolution to control the timing and sequence of Spacecraft functions. The OSE checks the capability of the C&S to generate these signals, which are listed in Table 4-1.

In the event that the C&S detail design should incorporate resistors to produce voltage drops proportional to the degree of unanimity of voting logic, the C&S OSE will have the capability of signalling partial disagreement within majority logic voting circuits.

3.2 SELF TEST

Self-check is accomplished by feeding all OSE outputs back into the test circuits and stimulating the C&S test circuits in a prearranged sequence. Table 3-1 lists the sequence in which the OSE functions are tested. A failure is indicated by a test number referenced to the function which has malfunctioned.

Table 3-1. Self-Check Sequence

A.	Initiate Self-Check 1. Make Hermaphroditic Connections 2. Mount "Self-Check" Tape on Reader 3. Depress "Start" Switch
B.	Calibration Sequence 1. Power Supply Voltages 2. Oscillator Frequency
C.	Self-Check Sequence 1. Input Buffer 2. Control Logic 3. Sync Pulse 4. Alert Pulse 5. Timer Reset 6. Timer Inhibit 7. Timer Speed-Up 8. Command Register 9. Elapsed Time Counter 10. Test Number Register 11. Command Time Registers 12. Descrete Gate Generator 13. Timing Comparator 14. Parity Comparator 15. Digital Comparator 16. Line Receivers and Gates 17. Voltage Probe 18. Descrete Comparator
D.	Terminate Testing 1. Halt on failure and display number of test failed 2. Halt on successful completion of self-check and display "Test Complete"

The elements required for self-check are:

a. Punched tape test program

b. Test Stimuli:

   1. Cyclics

   2. Timed Commands

   3. Events

   4. Register Contents

c. Test Receivers

   1. Sync & Alert

   2. Command Bursts and Command Decoder Signals

   3. Timer Controls

d. Test Status Indication

e. Test Outcome Indication

The punched tape program includes all of the self-test parameters, including register reset pulses, test conditions and limits, and the test number. The tests are sequenced so that every function is tested before being used for the test of subsequent functions.

The test stimuli simulate the inputs which the OSE would normally receive from the C&S during operational tests. These stimuli are generated by the OSE, and are used with nominal and worst case characteristics.

The test receivers are designed to accept only stimuli within specified limits. Thus, the OSE must not only generate the proper stimuli for the C&S at the correct time, but pulse shapes must be acceptable also.

Table 3-2. Subsystem Test Sequence

A. Start Manually

1. Mount "Confidence Test" Tape on Reader
2. Depress OSE "Start" Switch

B. Preparation for Testing

1. Reset All Registers
2. Turn on C&S Power
3. Test Cyclic Outputs
4. Start Elapsed Time Counter, Reset C&S Timer
5. Check C&S Timer "Speed Up" Pulse

C. Load C&S Memory

1. Read Burst of Commands in from Tape
2. Convert Individual Commands from Characters to Words
3. Send Alert Pulse
4. Start Sending Sync Pulses
5. Transmit Commands to C&S

D. Individual Response Test Sequence

1. Read Test Number into Register
2. Read Each Test into Control Logic
3. Read Test Limits into Registers
4. Generate Response Gate
5. Compare Response with Response Gate
6. Proceed to Next Test or Halt and Display Number of Failed Test
7. Monitor Telemetry Outputs between Command Tests

E. Terminate Testing

1. If Unit Completes all Tests Successfully, Display "Test Complete"
2. Turn Off C&S Power and Inputs

3.3 SUBSYSTEM TEST

The test of the C&S as an isolated subsystem, uses the OSE to provide all stimuli and loads, similar to the performance of Self-Check. The OSE monitors all C&S interfaces and telemetry outputs, including:

a. Cyclics

b. Timed Commands

c. Power Supply Voltages

d. Events

The OSE stimulates and monitors the C&S until either the test sequence (Table 3-2) is complete or the C&S fails by:

a. Not accepting a correct command

b. Accepting an incorrect command

c. Generating cyclics out of tolerance limits

d. Failing to issue a command to the correct address at the correct time

e. Issuing a command to an incorrect address

f. Not complying with wave shape requirements

If the C&S fails the programmed test, the test sequence is stopped and the number of the failed test is displayed. This indicates the failure mode.

If the C&S passes all tests in the sequence, TEST COMPLETE is displayed.

3.4 FAULT ISOLATION

The C&S OSE can be used to isolate a fault to a replaceable assembly of a C&S which has failed. In the Fault Isolation Mode, the OSE automatically checks the C&S functions in the sequence listed in Table 3-3, stopping as soon as a test has been failed. The number of the failed test indicates the malfunctioning assembly. The test block diagrams of Figure 3-4 show how this sequence utilizes the interfaces and a minimum number of test points to accomplish this testing.

The OSE provides manual capability to permit greater test resolution. This includes a bank of toggle switches to generate a command word, an oscilloscope, a signal generator, and a frequency counter.

Table 3-3. Fault Isolation

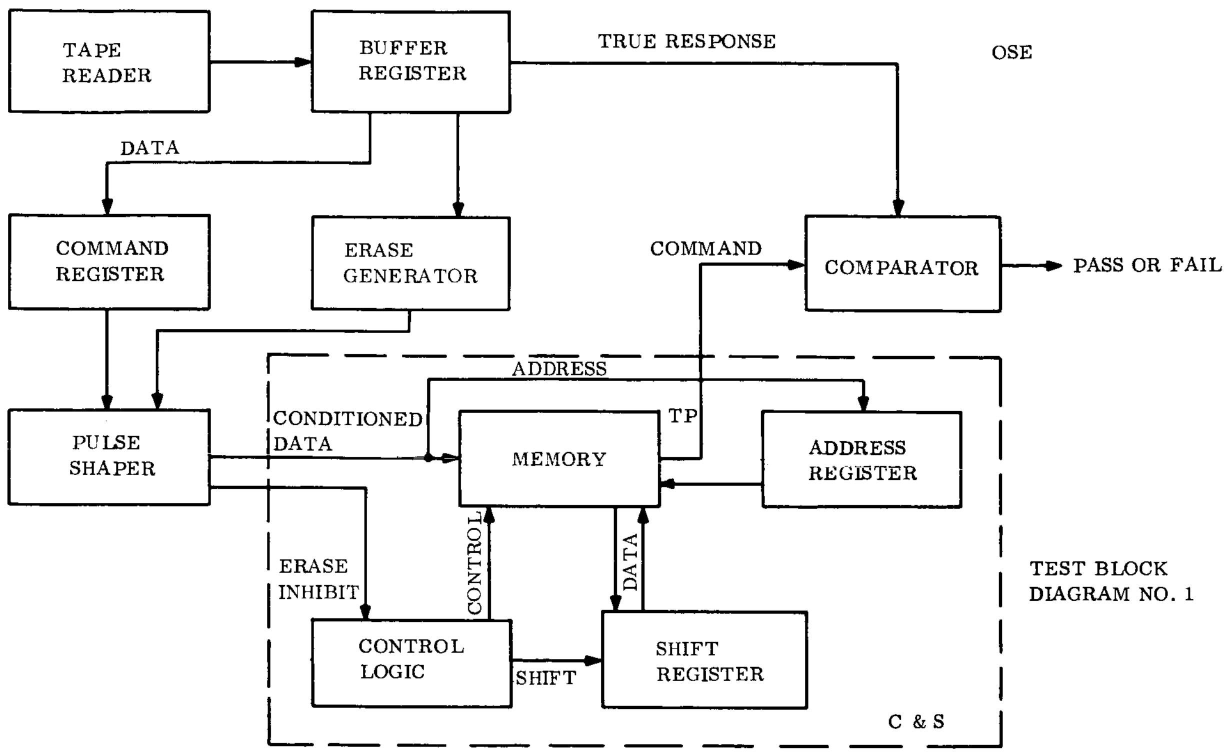
A. Initiate Fault Isolation

1. Mount "Fault Isolation" Tape on Reader
2. Depress OSE "Start" Switch
3. Perform "Confidence Test" Sequence
4. If C&S Fails to Pass any Test, Record the Test Number and Jump to "Fault Isolation" Sequence

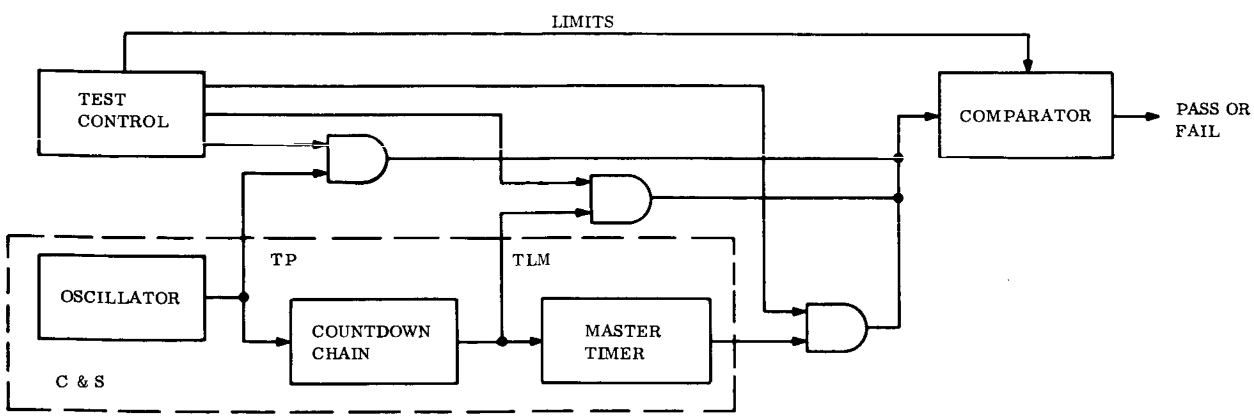
B. Fault Isolation Sequence	Refer to Test Block Diagram (Figure No.)
1. Erase Check	3-4A
2. Control Logic - Load Mode	3-4A
3. Shift Register - Shifting Words In	3-4A
4. Address Register - Shifting Addresses In	3-4A
5. Memory	3-4A
6. Oscillator	3-4B
7. Countdown Chain	3-4B
8. Master Timer	3-4B
9. Command Matrix - Timing Controls	3-4C
10. Sequence Timer	3-4C
11. Cyclics	3-4C
12. Timing Comparators	3-4C
13. Command Matrix - Scan No. 1	3-4D
14. Control Logic - Scan No. 1	3-4D
15. Shift Register - Recirculate	3-4E
16. Address Register - Counting	3-4E
17. Timing Selection Matrix	3-4F
18. Command Destination Gates	3-4F
19. Command Matrix - Scan No. 2	3-4D
20. Control Logic - Scan No. 2	3-4D
21. Combining Logic and Drivers	3-4D

C. Terminate Testing

1. Halt on Failure and Display Number of Failed Test
2. Halt on successful completion of all fault isolation tests and display "Test Complete."

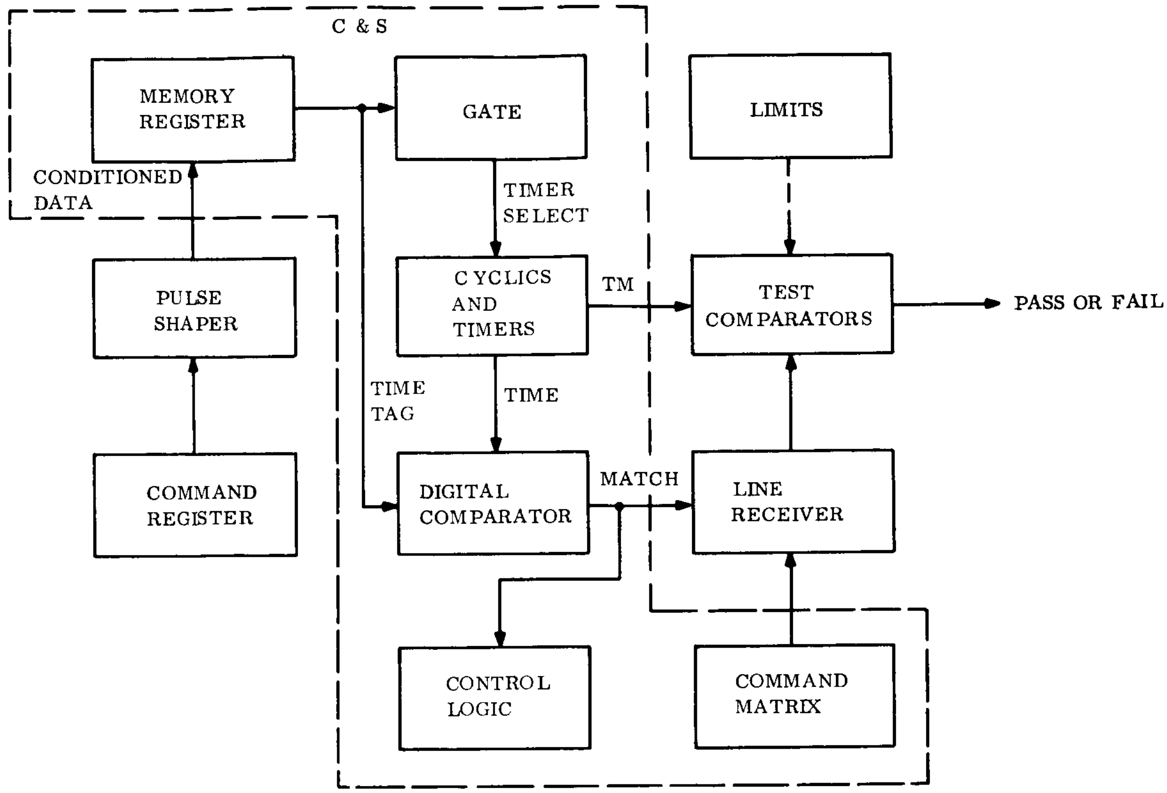


A. Test Block Diagram No. 1

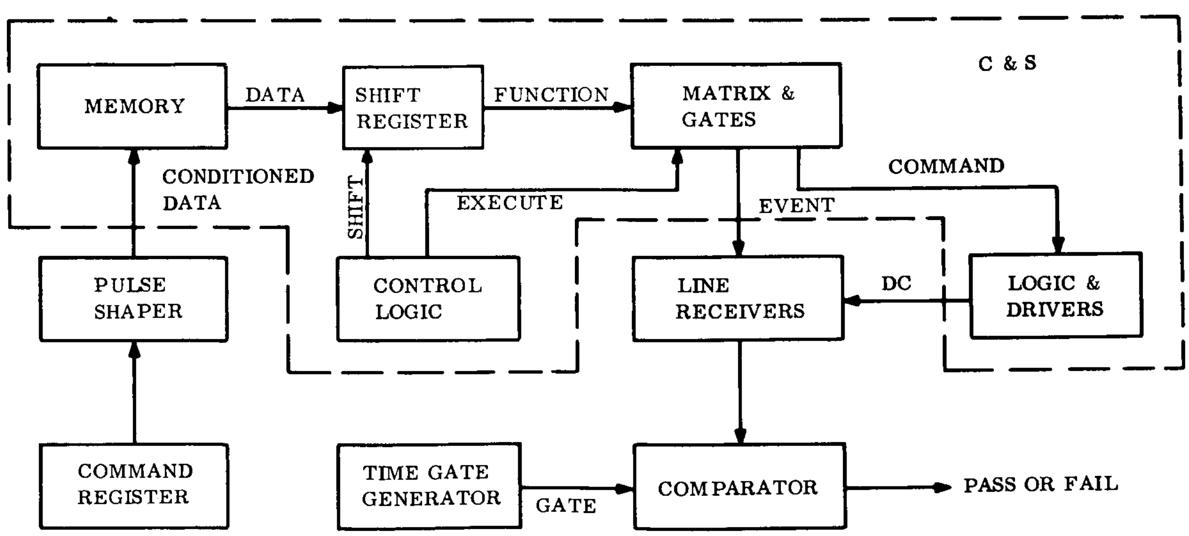


B. Test Block Diagram No. 2

Figure 3-4. Test Block Diagrams

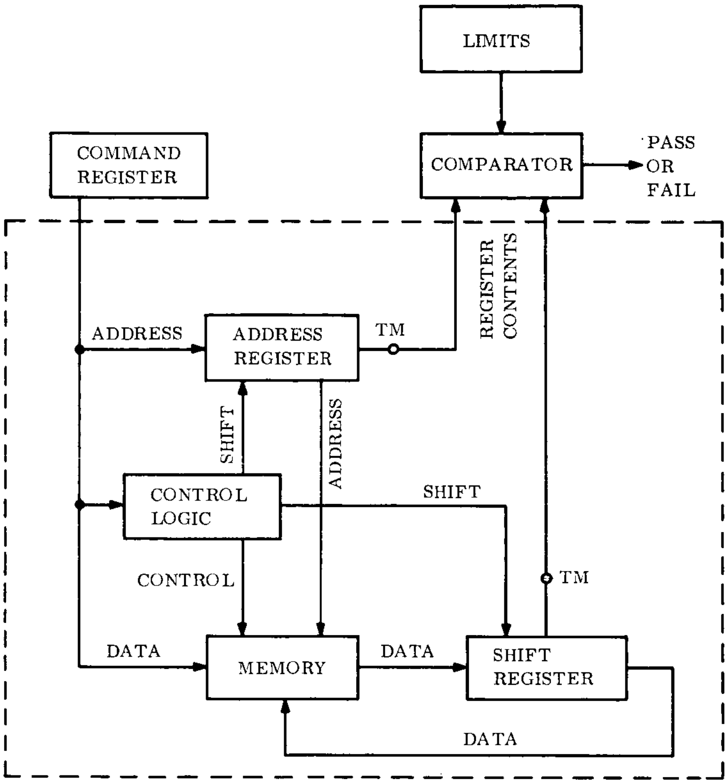


C. Test Block Diagram No. 3

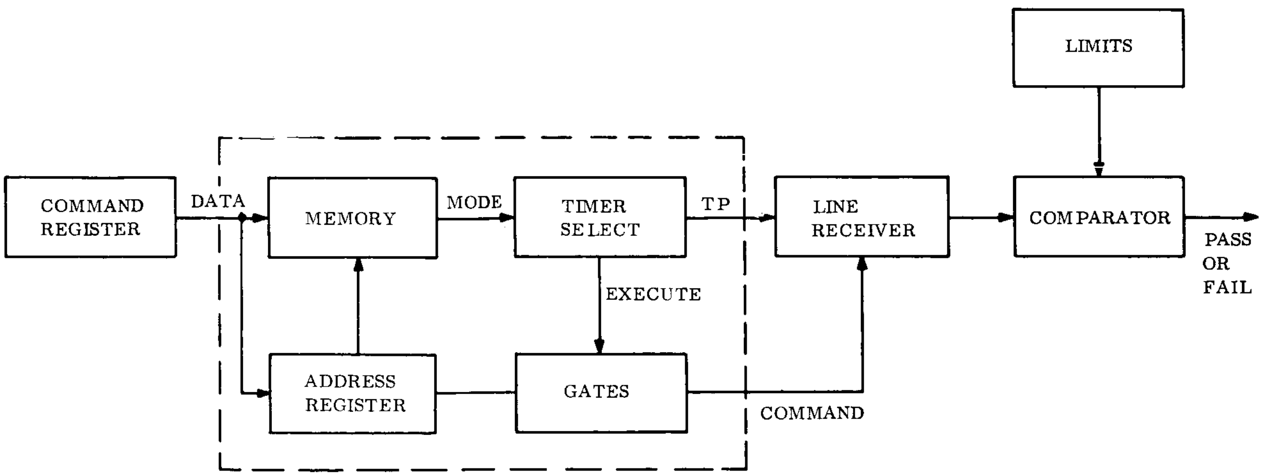


D. Test Block Diagram No. 4

Figure 3-4. Test Block Diagrams



E. Test Block Diagram No. 5



F. Test Block Diagram No. 6

Figure 3-4. Test Block Diagrams

3.5 SYSTEM TEST

3.5.1 During System Tests, the C&S operates as an integral part of the Spacecraft, and the C&S OSE operates as a part of the System Test Complex. C&S stimuli and loads are provided by the Spacecraft subsystems, such as the Command Decoder and Telemetry, so that the equivalent functions of the OSE are not utilized.

Figure 3-3 illustrates how the OSE is used to monitor the C&S while the Command Verification Equipment (CVE) exercises the command loop. The C&S OSE controls the C&S Master Timer through the umbilical lines, monitors inputs, Cyclics and Events via telemetry and indicates failure to the Computer Data System. Table 4-3 lists the status indicators used for C&S diagnostics.

4.0 INTERFACE DEFINITIONS

All significant interfaces are electrical.

4.1 C&S OSE inputs are tabulated in Table 4-1.

4.2 C&S OSE Outputs are tabulated in Table 4-2.

4.3 C&S OSE Displays are tabulated in Table 4-3.

5.0 PERFORMANCE PARAMETERS

5.1 ENVIRONMENTAL CONDITIONS

5.1.1 VIBRATION AND SHOCK

The C&S OSE is required to withstand the shock and vibration encountered in transportation.

5.1.2 TEMPERATURE AND HUMIDITY

The C&S OSE is required to be capable of proper operation at any temperature between 65° F and 90° F and at any humidity.

5.2 TIMING ACCURACY

The basic timing tolerance of the OSE is ± one part in 10^5. Derived frequencies are subject to this tolerance.

Table 4-1. C&S OSE Inputs

Interface	Function	Parameter	Description
Test Director's Console	Power	Turn On	6v pulse, 2ms. min.
	Power	Turn Off	6v pulse, 2ms. min.
	Mode	System Test	6v pulse, 2ms. min.
		Subsystem Test	6v pulse, 2ms. min.
		Self Test	6v pulse, 2ms. min.
Computer Data System	Command Data	Pulse Code	6v sq. wave, 1 cps
	Test Limits	Pulse Code	6v sq. wave, 1 cps.
	Events*	Timing	6v pulse, 2 ms. min.
	Master Timer*	Pulse Code	6v 26-bit train, one/sec.
	Sequence Timer*	Pulse Code	6v 18-bit train, one/sec.
Controller & Sequencer	Cyclic	1pps	6v sq. wave, 10 ma. max.
	Cyclic	38.4 kc	6v sq. wave, 10 ma. max.
	Cyclic	153.6 kc	6v sq. wave, 10 ma. max.
	DC Power	Voltage	± 3 v, ± 5%
	DC Power	Voltage	+ 6 v, ± 5%
	DC Power	Voltage	+ 15 v, ± 5%
	Commands, Input	Pulse Code	6v sq. wave 1pps
	Discrete Commands**	(225 Max.)	Switch closure (see Figures 5-1 and 5-2)
	Alert	Pulse	6v 2ms. min
	Sync	Pulse	6v, 1 pps

*During systems tests, these C&S functions are monitored by Telemetry
**Monitored during subsystem tests

Table 4-2. C&S OSE Outputs

Interface	Function	Parameter	Description
Test Director's Console	Status	Operable & On	Switch Closure
	Status	Inoperable	Switch Closure
Controller & Sequencer	Power*		2.4 kc sq. wave; xfmr isolated
	Command**	Alert	6v pulse, 2ms min.
	Command**	Sync	6v sq. wave, 10 ma max
	Command**	Data	6v sq. wave, 10 ma max †
	Decoder Controls**	Erase	6v pulse, 2ms min.
	Decoder Controls**	Enable Engine Burn	Switch opening
	Decoder Controls**	Select Scan Mode	6v pulse, 2ms min.
	Decoder Controls**	Inhibit Outputs	6v level
	Timer Controls***	Clear Timers	6v pulse, 2ms min.
	Timer Controls***	Inhibit Master Timer	6v level
	Timer Controls***	Update Master Timer	6v level
	Timer Controls***	Speed-Up	6v level
	Majority Logic***	Inhibit P.S. No. 1	6v level
	Majority Logic***	Inhibit P.S. No. 2	6v level
Command Verification Unit	C&S Commands, Input	Pulse Code	6v sq. wave 1 pps.

Table 4-3. C&S OSE Displays

Functions	Indications
Separation Connector Functions:	
Output Inhibit	OK/Alarm
LCE Functions	
Timer Clear	OK/Alarm
Timer Load	OK/High/Low
Timer Inhibit	OK/Alarm
Command Decoder Functions	
Inhibit Outputs	OK/Alarm
Select Scan Mode	OK/Alarm
Erase	OK/Alarm
Enable Engine Burn	OK/Alarm
Alert	OK/Alarm
Sync	OK/Alarm
Parity	OK/Alarm
Command	Binary Display (37 bits)
Event	OK/Early/Missing/Late
Power	On
Mode	System/Subsystem/Self Check
Test Failure	Test Number (4 Digits)
Test Status	In Test/Test Complete

5.3 SWITCH CLOSURES

The characteristics of the switch closures listed in Tables IV and V are shown in Figures 5-1 and 5-2. Figure 5-1 shows the step switch used both in the Controller and Sequencer and in the OSE. Figure 5-2 shows the pulse switch used in both equipments.

5.4 POWER

The OSE uses 105-120-volt, 60-cycle, single-phase (three-wire) filtered power which is regulated to within 5%. Eight-volt transients are tolerated. During subsystem and Fault Isolation tests, the OSE provides power to the C&S in accordance with the Spacecraft Specification. This power is nominally at 50 volts, 2400 cycles per second, and is square wave.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 GENERAL CHARACTERISTICS

6.1.1 POWER REQUIREMENTS

20 amp., 120 volts AC, 60 cps.

6.1.2 WEIGHT

300 pounds

6.1.3 SIZE

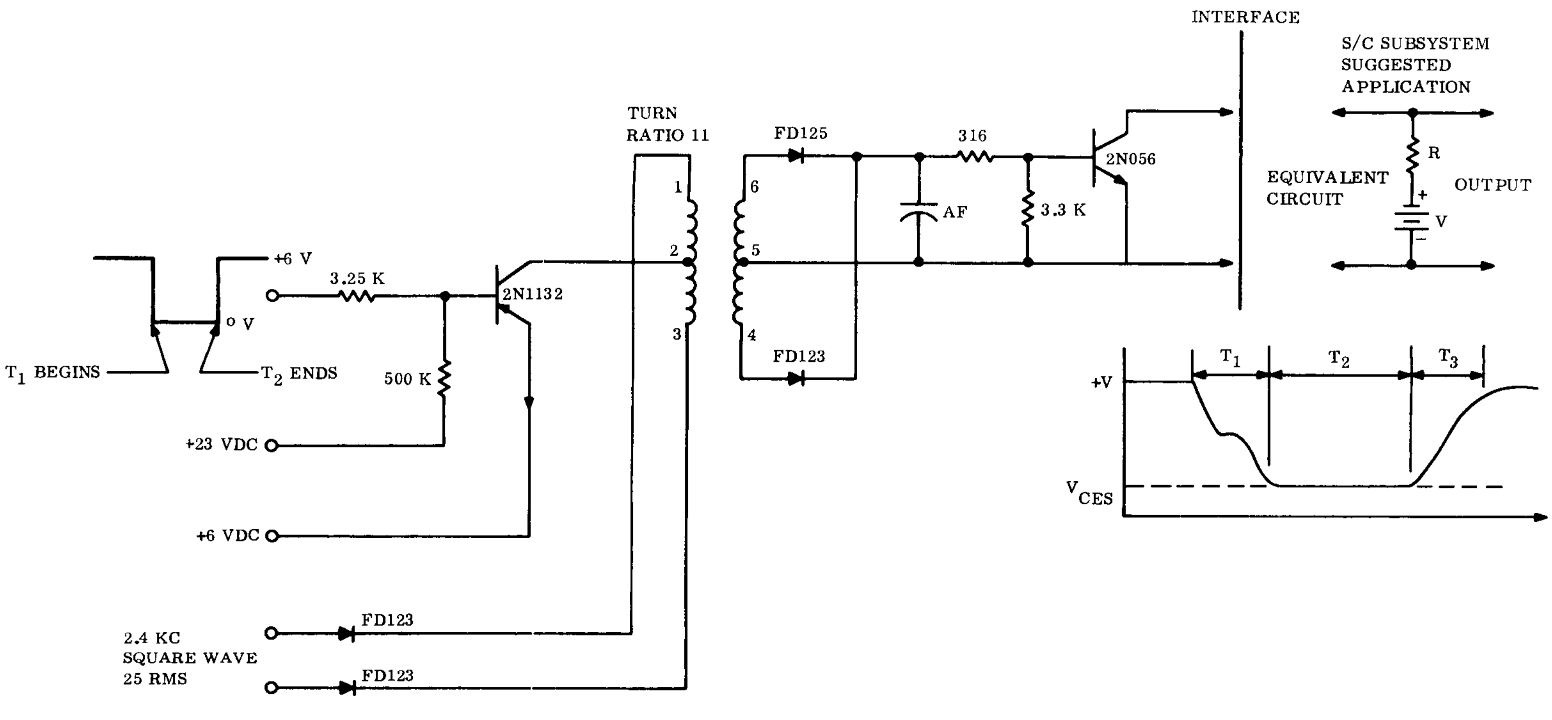
One-bay console, 2' W, 6' H, 42" D (C&S Test Console)
One-bay equipment, 2' W, 6' H, 2' D (Auxilliary Test Equipment)

6.2 RELIABILITY AND MAINTAINABILITY

The OSE is required to be worst case designed, utilizing parts, materials, processes, circuits and assemblies which are coordinated with other Operational Support Equipment as well as the Spacecraft and other end items. Use of proven circuits and logic elements is emphasized. Construction is to be modular, with like elements being interchangeable and provision made to facilitate inspection, test, fault isolation and repair.

6.3 RF INTERFERENCE

The OSE is to be kept free of conducted and radiated RF Interference by filtering and shielding. It should not be susceptible to such noise as leaks into it.



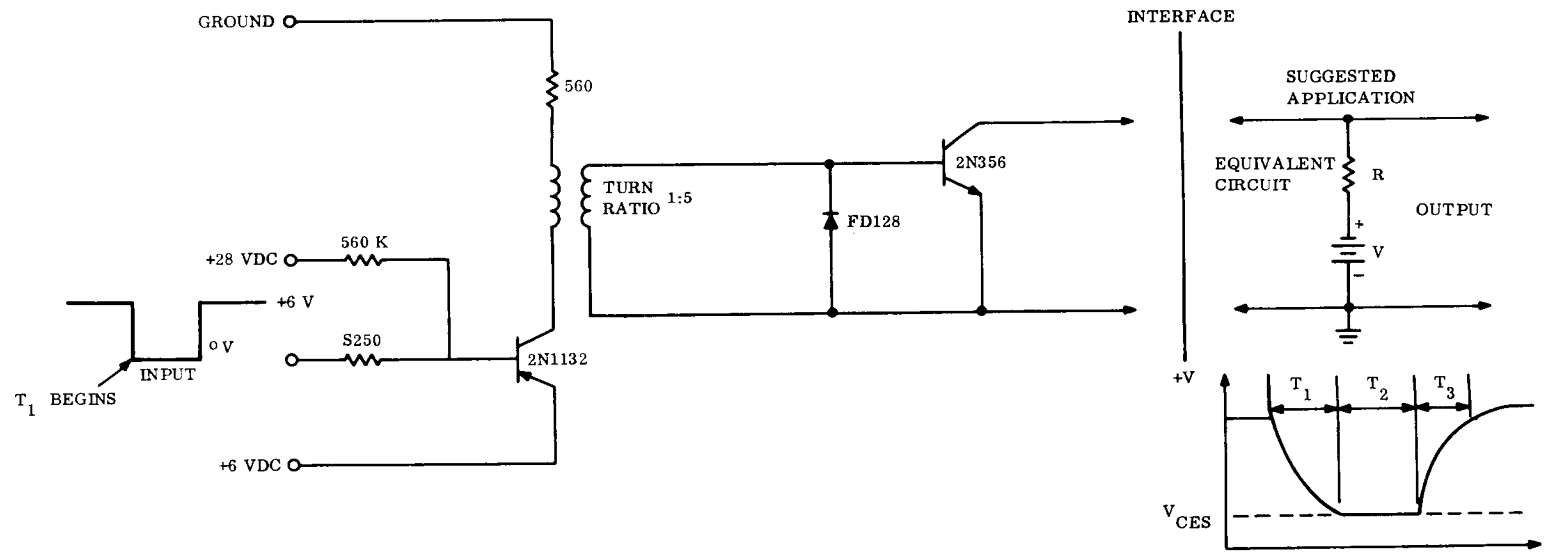
ISOLATED STEP SWITCH SPECIFICATIONS

PARAMETER	MAXIMUM	NOMINAL	MINIMUM
LEADING EDGE SWITCH TIME (T_1)	1.5 MILLISEC.	.2 MILLISEC.	-
SWITCH CLOSURE DURATION (T_2)	-	-	-
TRAILING EDGE SWITCH TIME (T_3)	5.0 MILLISEC.	1.5 MILLISEC.	-
OPEN SWITCH VOLTAGE	35. VOLTS	-	3. VOLTS
CLOSED SWITCH CURRENT (I_c)	100. MILLIAMP.	-	-
OPEN SWITCH RESISTANCE	1000. MEGOHM	10. MEGOHM	1. MEGOHM
*V_{ces} (TRANSISTOR SATURATION VOLTAGE)	2.1 V (I_c = 100 ma)	1.1 V (I_c = 100 ma)	.35 V (I_c = 100 ma)
V_{os} (TRANSISTOR OFFSET VOLTAGE)	.1 VOLTS	.1 VOLTS	.05 VOLTS
R_{sat} (TRANSISTOR SATURATION RESISTANCE)	20 ohms	10 ohms	3 ohms

$\left[{}^*V_{ces} = V_{os} + I_c R_{sat}\right]$ TIME T_2 IS CONTROLLED BY THE DECODER

$T_2 = 100 \pm 10$ MILLISECOND FOR DC WORD OUTPUTS

Figure 5-1. Isolated Step (IS) Switch Characteristics



ISOLATED PULSE SWITCH SPECIFICATIONS

PARAMETER	MAXIMUM	NOMINAL	MINIMUM
LEADING EDGE SWITCH TIME (T_1)	5 MICROSEC.	0.5 MICRO SEC.	-
SWITCH CLOSURE DURATION (T_2)	10. MILLISEC.	6 MILLI SEC.	2 MILLISEC.
TRAILING EDGE SWITCH TIME (T_3)	1. MILLISEC.	0.7 MILLI SEC.	-
OPEN SWITCH VOLTAGE	35. VOLTS	-	3. VOLTS
CLOSED SWITCH CURRENT (I_c)	10. MILLIAMP.	-	-
OPEN SWITCH RESISTANCE	1000. MEGOHM	10. MEGOHM	1. MEGOHM
*V_{ces} (TRANSISTOR SATURATION VOLTAGE)	.3 V (I_c = 10 ma)	.2 V (I_c = 10 ma)	.08 V (I_c = 10 ma)
V_{os} (TRANSISTOR OFFSET VOLTAGE)	.1 VOLTS	.1 VOLTS	.05 VOLTS
V_{sat} (TRANSISTOR SATURATION RESISTANCE)	20 ohms	10 ohms	3 ohms

$\left[{}^{*}V_{ces} = V_{os} + I_c R_{sat} \right]$

Figure 5-2. Isolated Pulse (IP) Switch Characteristics

6.4 TAPE READER AND SPOOLER

A commercial panel-mounted tape reader and spooler is the primary input device for subsystem test. It will read 1 inch, eight-level, punched paper (or mylar) tape. Tape preparation equipment is required "off-line." This includes a keyboard card punch, a card-to-tape converter and a tape punch.

6.5 POWER SUPPLIES

a. OSE Power - A regulated power supply provides ± 3 volts DC, + 6 volts DC and + 15 volts DC for OSE circuitry.

b. Controller and Sequencer Power - A 2.4-kc square wave power supply simulates the nominal and worst-case power supplied to the C&S by the spacecraft power system.

6.6 LOGIC PANEL ASSEMBLY

All OSE logic is to be contained in a five-channel logic panel assembly. This assembly is hinge mounted to provide access both to test points and to the panel-mounted connectors and cabling. The five channels contain:

Input buffers and control logic

Memory registers and drivers

Timing generator

Input circuits

Output circuits

6.7 OSCILLOSCOPE

A commercial, panel mounted oscilloscope is required for evaluation of pulse wave shapes.

6.8 SQUARE WAVE GENERATOR

A commercial, panel mounted square wave generator is required for signal tracing.

6.9 FREQUENCY COUNTER

A commercial panel mounted frequency counter is required for calibrating and checking Test Set frequencies and for counting events per unit time.

6.10 DISPLAY PANEL

The OSE display panel mounts necessary test displays listed in Table VI plus the four character number of a failed test, C&S running time, OSE running time, test mode, Test Director Console status, Central Recorder status and a failure alarm light.

6.11 MANUAL CONTROLS PANEL

The Control Panel provides the capability of selecting test modes and monitor points. It also provides a bank of switches for manually entering test words into the C&S, for selecting nominal or worst case test conditions, and for selecting marginal conditions (beyond worst case limits). These manual controls vary frequency, rise and fall times, width and amplitude of the pulses sent to the C&S by the Pulse Shapers.

6.12 INTERFACE CABLES AND CONNECTORS

The console desk extension mounts the connectors which contain the interconnections to the C&S for both the Subsystem Test and for Fault Isolation. The OSE includes the interconnecting cables. Interface cables for System Test in the STC enter the OSE from under the false floor, and attach to connectors provided for that purpose.

7.0 SAFETY CONSIDERATIONS

The OSE is designed to present no hazard to operating personnel or facilities as low voltages are used throughout.

All direct access and umbilical functions, as well as C&S outputs, are isolated from within the C&S OSE. This isolation is provided by Line Receivers and Line Drivers.

Provision is made to monitor OSE operation and give an alarm when operating limits are exceeded. Test jacks are provided to minimize the need for probing test points of the OSE and the C&S.

All connectors are keyed to prevent improper connection of interfaces and sub-assemblies.

Where, because of test equipment tolerances, automatic testing leaves the C&S operability uncertain, the C&S is rejected. The true condition of the C&S can then be determined manually.

Each of the major items of OSE is provided with overload protection.

CII - VB266FD101

SYSTEM TEST COMPLEX

POWER SUBSYSTEM OPERATIONAL SUPPORT EQUIPMENT

Index

1.0 SCOPE

This document describes the functional requirements for the Power Subsystem OSE in the System Test Complex. This equipment will simulate solar power and monitor power subsystem performance during spacecraft system tests, and perform detailed tests on the power subsystem independently of the rest of the spacecraft.

2.0 APPLICABLE DOCUMENTS

VB236FD101	Functional Description of Power Subsystem
VB260SR101	STC Test Objectives and Design Criteria
VB260SR102	STC Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The Power subsystem OSE provides all the test equipment required for support of the Voyager power subsystem in the System Test Complex. The STC can support:

a. Initial application of power

b. Power subsystem evaluation and fault isolation

c. Spacecraft system tests

The power subsystem OSE is also capable of supporting power subsystem test.

To support these test activities, the power OSE has the capability to simulate the solar array electrical output, monitor and evaluate direct access test points continuously, record all test parameters and events, perform a complete self test and provide access to signal lines for special test equipment. The power subsystem OSE can, when testing the power subsystem separately from a system test, simulate the command/power subsystems interface and introduce fault indications to test redundant modes of power subsystem operation.

3.2 FUNCTIONAL BLOCK DIAGRAMS

The following block diagrams illustrate the characteristics and functions of the Power subsystem OSE:

Figure 3-1 - Power Subsystem OSE - This figure shows in functional block form all of the elements included in the Power Subsystem OSE.

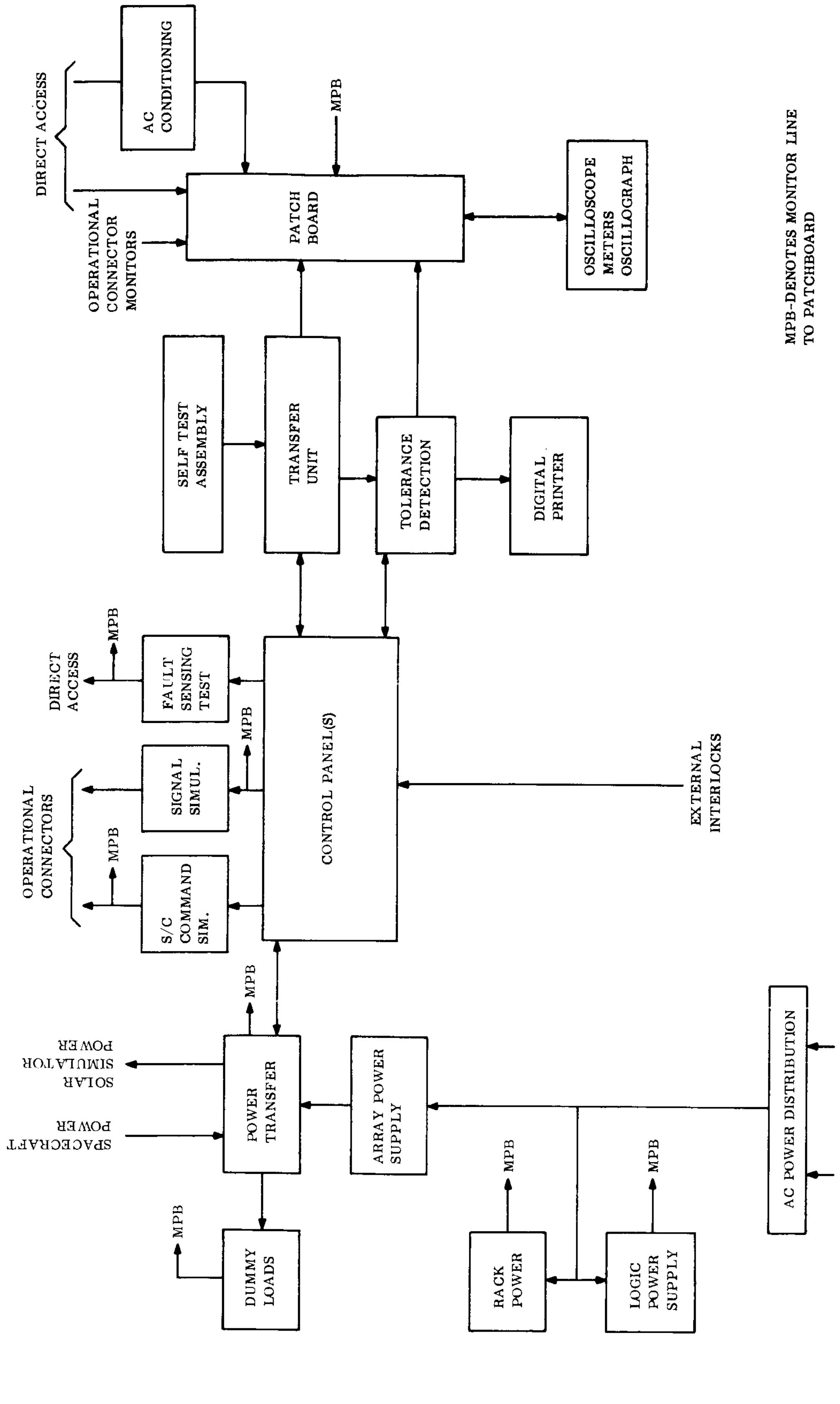


Figure 3-1. Power Subsystem OSE, Functional Block Diagram

Figure 3-2 - Power Subsystem, System Test Block Diagram - This is a test block diagram showing the functional elements of the power subsystem OSE used to support a system test on the spacecraft.

Figure 3-3 - Power Subsystem, Subsystem Test Block Diagram - This figure is a test block diagram showing the functional elements of the power subsystem OSE used to support a subsystem test on the Voyager power subsystem.

3.3 SELF TEST CAPABILITIES

The power subsystem OSE is capable of performing test on its own power and monitor circuits to ensure that it is fully operational and capable of carrying out its support mission. The self test features may be used before beginning a power system test, periodically during a test sequence or at any time the operator decides a self check sequence is required. The self test sequence can be used on a non-interference basis during a spacecraft test.

The self test is implemented by having the transfer assembly switch premeasured test signals into the tolerance detection assembly and observing whether they are correctly identified as being in tolerance or out of tolerance.

The switching of the transfer assembly is self tested by observation of the self tests of the tolerance detection and test signal generation functions.

Power output is self tested by switching high power level outputs into the variable dummy loads and making measurements of voltage and delivered current.

3.4 FUNCTIONAL DESCRIPTION - SYSTEM TEST

Refer to Figure 3-2 - Power Subsystem, System Test Block Diagram.

3.4.1 TEST SET-UP

During system test, the power OSE is connected to the spacecraft direct access connections and the spacecraft umbilical connector. The power subsystem OSE interconnections are established to support system testing through use of a PATCH BOARD which carries all signals to be monitored.

3.4.2 POWER INPUT TO SPACECRAFT

External d-c power is introduced into the spacecraft through the umbilical connector in such a manner as to simulate the solar array output to the power subsystem. The solar array power supply provides this d-c power. The power to the spacecraft

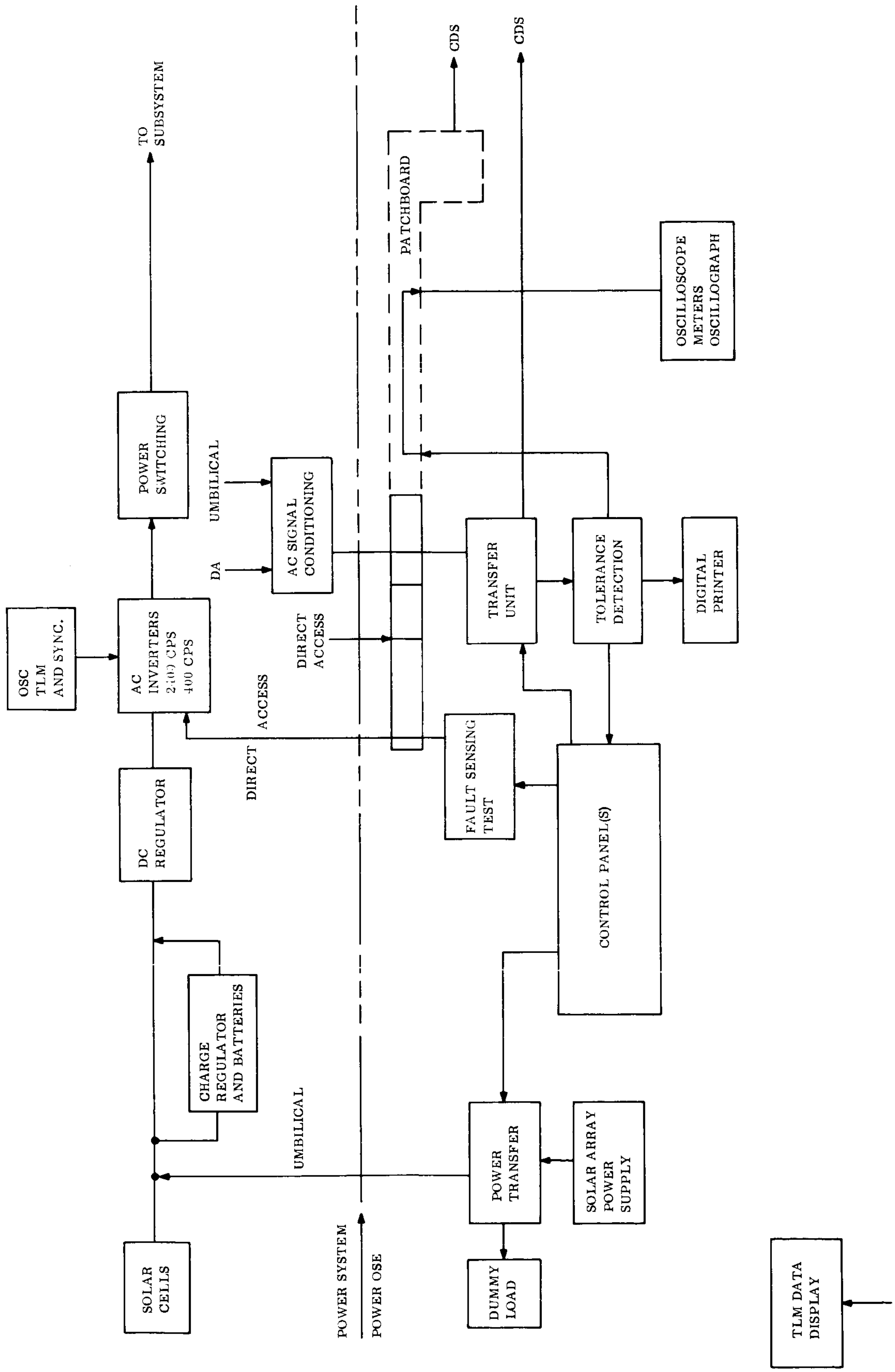


Figure 3-2. Power Subsystem -System Test, Block Diagram

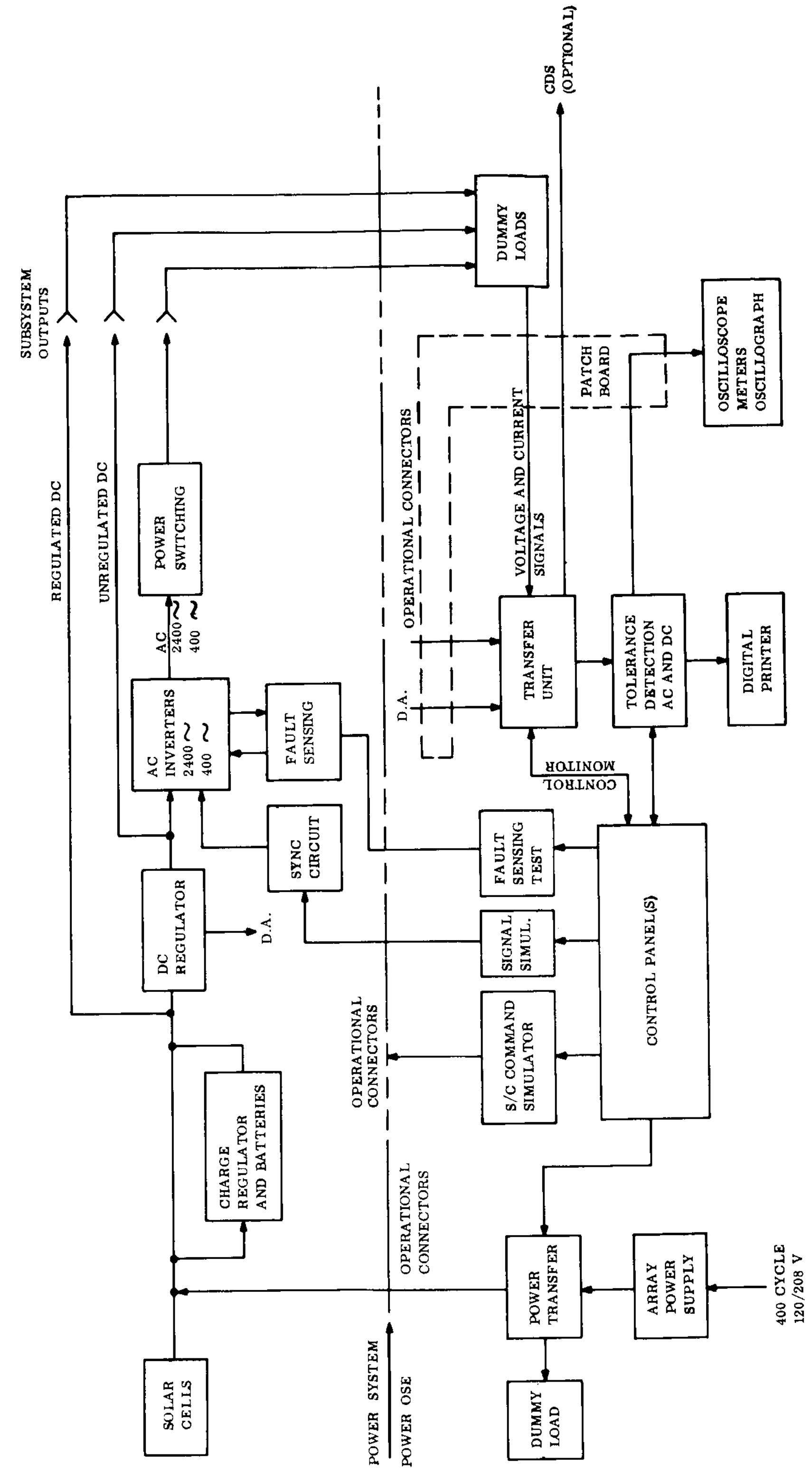


Figure 3-3. Power Subsystem - Subsystem Test, Block Diagram

umbilical is routed through a Power Transfer assembly which switches the output power from a variable Dummy Load to the vehicle bus on command from the Control Panel.

The Power Transfer assembly receives interlock signals from the Control Panel and the Transfer Unit to prevent accidental application of power to the spacecraft when there is a possibility of damage.

3.4.3 MONITORING THE TEST SIGNALS

A majority of the test signals monitored during a system test come from the power subsystem direct-access connectors, other signals come from from the umbilical, and the remainder come from the power subsystem OSE itself. All signals to be monitored pass through the Transfer Assembly which controls the input lines to the Tolerance Detection Assembly. All signals are normally routed to the Tolerance Detection unit.

The Tolerance Detection Assembly is capable of determining if a monitor point is in or out of its prescribed tolerance and on which side of the tolerance band it is out. When a function is out of tolerance an indication appears on the control panel informing the operator of the condition. The operator may select a latching mode with the indication remaining after the out of tolerance condition disappears or he may elect to have the indication only as long as the out of tolerance condition exists. (The Self Test Unit provides marginal out of tolerance signals and signals within tolerance for a complete check of the Tolerance Detection Assembly.)

An A-C Conditioning Unit is provided with the power OSE. The conditioning unit is placed near the umbilical and provides impedance conversion and isolation for umbilical signals coming from the power subsystem. The circuits used in the A-C Conditioning Unit are identical to the circuits in the umbilical "J" Box at the launch complex.

3.4.4 RECORDING

Permanent records are desirable for analysis and review. The signals available from direct-access test points and the umbilical can be recorded. (During subsystem tests, approximately 200 signals can be recorded.) The recording device will be a digital printer tied to the tolerance detection assembly. The printer provides a rapid, easily read display which is also a permanent record.

Recording of continuous signals is provided by a twenty-four channel oscillograph. The oscillograph inputs are selected at the patchboard. In addition to the internal recording capability, analog signals are also sent to the Computer Data System for analysis and to the Central Recorder during a system test. These signals are also selected at the patchboard.

3.4.5 SPACECRAFT INPUTS FROM OSE

Hardwire Commands sent to the Spacecraft power system from the OSE are initiated at the Control Panel. The Power Subsystem OSE controls mode switching through the umbilical connector with the controls mounted on this panel.

3.4.6 POWER SUBSYSTEM OSE CONTROL AND DISPLAY

The Control Panel provides a central control point for operating the OSE and performing a test sequence. Most of the controls for the power OSE are located on the control panel. These controls include power subsystem switching, control of the Transfer Assembly modes, Power Transfer, and any other displays required for the convenience of the operator.

A telemetry data display, while not a part of the power OSE, is available and conveniently located so that during a system test sequence the telemetry signals which come from the power subsystem are printed out where the operator of the Power OSE can review them.

There is also an oscilloscope and meters which may be used as monitors when required, although they are primarily troubleshooting and calibration aids.

3.5 FUNCTIONAL DESCRIPTION - S/S TEST MODES

Refer to Figure 3-3 - Power Subsystem, Subsystem Test Block Diagram.

3.5.1 SPACECRAFT INTERFACE

When testing the power subsystem alone, the subsystem OSE provides all of the required interconnecting cables and harnesses. Cables connect the OSE to the subsystem and a harness interconnects the three subsystem racks which contain the power subsystem so that the subsystem may be tested in its entirety. The OSE also has the capability to test each of the three power subsystem bays as a separate entity.

3.5.2 SPACECRAFT SIMULATION

The test sequences for system test and subsystem test can be identical. The major difference between the two tests is that during the subsystem test the OSE must simulate the rest of the spacecraft to the power subsystem. Spacecraft simulation is accomplished by providing dummy loads for all subsystem power outputs, a command simulation assembly for simulating the vehicle command system and routing the signals normally going to the telemetry system to the transfer assembly. All of the additional signals associated with the subsystem test come from the operational connectors. The direct-access connectors are still used and have the same characteristics. The umbilical functions still exist, but the interface is now at the operational connectors.

3.5.3 SPACECRAFT FAULT SIMULATION

Fault signals which come from the Fault Sensing Test Assemby (controlled from the Control Panel) are the only signals entering the S/C Power Subsystem through direct-access connectors. These signals provide a false fault indication to vehicle fault sensing circuits to test alternate and redundant modes of operation in the power subsystem. The purpose of this function is to determine if the spacecraft power subsystem will sense its own faults and properly switch itself over to its redundant capability. The fault signals are normally used during the subsystem test, but may also be injected during the system test.

3.5.4 ADDITIONAL EQUIPMENT REQUIRED

The capability to perform a subsystem test requires additional equipment over the system test. In addition to the command signal simulator and the dummy loads already mentioned, the OSE would have an expanded patchboard capability for the additional signals, and larger capacity in the transfer assembly, tolerance detection assembly and self test unit.

4.0 POWER SUBSYSTEM OSE INTERFACE

The important interfaces with the Power Subsystem OSE are electrical interfaces with the spacecraft and other OSE.

4.1 POWER SUBSYSTEM ELECTRICAL INTERFACE

a. Direct-Access Connectors - The power OSE will connect to the power subsystem direct access connectors; Table 4-1 lists the direct access signals.

b. Umbilical Connector - The power OSE connects to the umbilical connector through the STC cables and junction box. A list of umbilical functions appears in Table 4-2.

c. Operational Connectors - During subsystem tests the power OSE interfaces with the operational connectors associated with the power subsystem. This interface includes simulated spacecraft commands and simulated solar power generated by the OSE. It also includes telemetry sensor signals generated by the power subsystem.

d. Solar Panel Connectors - During system tests when the solar panels are not connected, the power OSE will supply power via the panel/structure connectors and will provide TLM sensor simulation.

4.2 COMPUTER DATA SYSTEM - ELECTRICAL INTERFACE

The Power Subsystem OSE will relay sixty analog signals to the Computer Data System for system evaluation during system test sequences. These will be high impedance d-c signals.

4.3 CENTRAL RECORDER - ELECTRICAL INTERFACE

The power subsystem OSE will transmit sixty high impedance d-c signals to the central recorder during the system test. These signals will be the same as the ones sent to the CDS.

Table 4-1. Systems Test Direct Access Functions

	Function		Characteristics
1.	Main Regulator Voltage		0 to 3.2 v Sensor Range
2.	Main Regulator Current		0 to 3.2 v Sensor Range
3.	2.4 kc Bus Current		0 to 3.2 v Sensor Range
4.	Capsule Current		0 to 3.2 v Sensor Range
5.	Transmitter Current		0 to 3.2 v Sensor Range
6.	Battery No. 1 Current		0 to 3.2 v Sensor Range
7.	Battery No. 2 Current		0 to 3.2 v Sensor Range
8.	Battery No. 3 Current		0 to 3.2 v Sensor Range
9.	Solar Array Current		0 to 3.2 v Sensor Range
10. thru 36.	Battery Cell Voltages Batteries No. 1, 2, 3	3 to 5 VDC	Sensed in three cell groups on each of the three batteries. Single cell voltages range 1.0 to 1.6 VDC.
	Battery Charge Regulator No. 1		
	(Charge Rate Setting)		
37.	Off		
38.	Set A (1.0 A Rate)		
39.	Set B (1.5 A Rate)		
40.	Set C (3.0 A Rate)		
	Battery Charge Regulator No. 2		
	(Charge Rate Setting)		Four lines required per regulator.
41.	Off		
42.	Set A		
43.	Set B		
44.	Set C		

Table 4-1. Systems Test Direct Access Functions (Continued)

	Function		Characteristics
	Battery Charge Regulator No. 3		
	(Charge Rate Setting)		
45.	Off		
46.	Set A		
47.	Set B		
48.	Set C		
49.	Main Regulator No. 1	On/Off	Voltage Signal for On and for Off.
50.	Main Regulator No. 2	On/Off	
51.	2.4 kc Inverter No. 1	On/Off	
52.	2.4 kc Inverter No. 2	On/Off	
53.	400 cps, 3 θ Inverter No. 1	On/Off	
54.	400 cps, 3 θ Inverter No. 2	On/Off	
55.	Back Up Sync (38.4 kc) Oscillator	On/Off	
	2.4 kc Power Distribution		
56.	Fuel Tank Heaters	On/Off	
57.	Capsule	On/Off	
58.	Science	On/Off	
59.	Relay	On/Off	
60.	Antenna Electronics	On/Off	
61.	Scan Electronics	On/Off	
62.	Autopilot Electronics	On/Off	
63.	Approach Guidance	On/Off	

Table 4-2. Umbilical Tabulation Systems Test Function

Function	Characteristics
1. External Power	44 - 55 vdc
2. External Power Return	
3. Enable Array/Battery	Pulse to Relay (On)
4. Disable Array/Battery	Pulse to Relay (Off)
5. Enable/Disable Return	
6. Enable/Disable State	On/Off
7. Array/Battery Bus Voltage	30 - 55 vdc
8. Raw Battery Bus Voltage	30 - 45 vdc
9. Battery No. 1 Voltage (Course)	30 - 45 vdc
10. Battery No. 2 Voltage (Course)	30 - 45 vdc
11. Battery No. 3 Voltage (Course	30 - 45 vdc
12. Battery No. 1 Temperature	5 - 10 vdc
13. Battery No. 2 Temperature	5 - 10 vdc
14. Battery No. 3 Temperature	5 - 10 vdc
15. Temperature Sensor Return	
16. 400 cps, 3 Φ Bus PH A	22 - 28 vac
17. 400 cps, 3 Φ Bus PH B	22 - 28 vac
18. 400 cps, 3 Φ Bus PH C	22 - 28 vac
19. 2.4 kc Bus	50 vac
20. 2.4 kc Bus Return	

4.4 STC FACILITY - ELECTRICAL INTERFACE

The Power Subsystem OSE will receive power and interlock signals from the STC Power Distribution and other OSE. The required power is:

a. 120/208-VAC, 400-cycle, 3-phase, 4-wire.

b. 120-VAC, 60-cycle, 1-phase, 3-wire.

c. Interlock signals from other subsystem OSE consoles.

4.5 TEST CONDUCTOR'S CONSOLE - ELECTRICAL INTERFACE

The Power Subsystem OSE will send test status signals to the Test Conductor's Console.

5.0 PERFORMANCE PARAMETERS

Estimated performance parameters are as follows:

a. Solar Array Power Supply - Range 0 - 55 VDC

b. Tolerance Detection - Accuracy ±1%

c. Dummy Loads - within ±5% of real S/C loads

d. Transfer Assembly - Capacity 200 Vehicle Signals

e. Digital Printer - 11 lines per second

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Voyager Power Subsystem OSE will be similar in configuration to the Mariner C Power subsystem OSE. The design and packaging approach should be similar. The basic differences are those imposed by the different power levels and additional features of the Voyager flight power subsystem.

6.1 GENERAL

Size - Three Standard Racks

Weight - Approximately 900 lbs.

Power Required - 120-VAC, 1-Phase, 60-Cycle, 3-Wire, 20-Amp
120/208-VAC, 3-Phase 400-Cycle, 4-Wire, 15-Amp

6.2 CABINET CONFIGURATION

Figure 6-1 shows the anticipated placement of assemblies in the Power Subsystem OSE Cabinet.

6.3 CONTROL PANEL

The Control Panel will contain the displays and controls required to perform a test. The anticipated display and control requirements are listed below.

a. Controls

1. Ground Power On/Off

2. Solar Array and Battery - Enable (On)

3. Solar Array and Battery - Disable (Off)

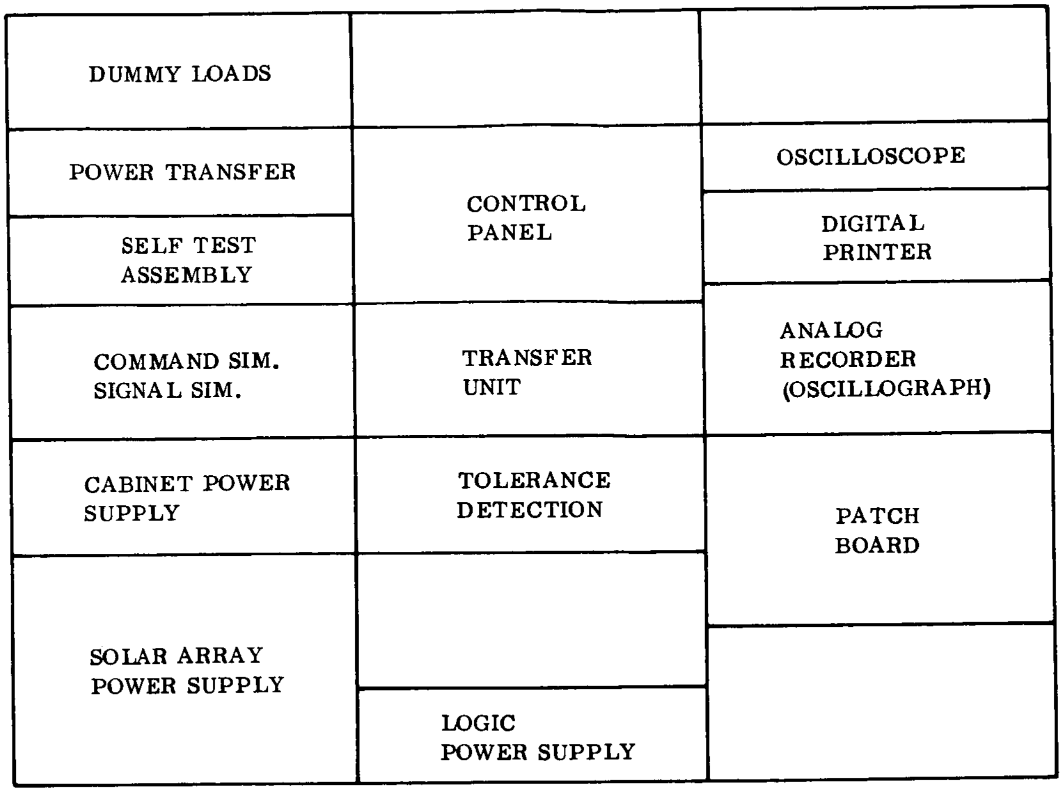


Figure 6-1. Power S/S OSE, System Test Mode

4. OSE Operate or Transfer to S/C

5. OSE Self Test

6. Emergency Disconnect

7. Out of Tolerance Failure Latch

b. <u>Monitors, Visual/Audio</u>

1. Visual indication of the control status above

2. External Power Voltage/Current (Array Simulator)

3. Battery status charge/discharge

4. Battery Voltage/Current

5. Self Test: Start, In Process, Complete

6. External Power: On, Off; In Tolerance/Out of Tolerance

7. "Out of Tolerance" for the following monitors:

   (a) Battery Voltage
   (b) Main Regulator
   (c) 400 CPS Inverter, Phases A, B and C
   (d) Battery Current
   (e) 2.4 KC Inverter

8. Battery Temperature

6.4 TRANSFER ASSEMBLY

The transfer assembly will be a rack mounted electronics assembly consisting of switching circuits on removable cards mounted on a holding rack.

6.5 TOLERANCE ASSEMBLY

The tolerance assembly will be a rack mounted electronics assembly consisting of switching and comparator circuits on removable cards mounted in a holding rack.

6.6 SELF TEST PANEL

The Self Test Panel will consist of front panel with controls and signal level adjustments and circuit elements mounted on removable cards. The cards will be mounted in a holding rack.

6.7 SOLAR ARRAY POWER SUPPLY

The Solar Array Power Supply will simulate the approximate output voltage versus current curve of the solar array. The power supply will be a self-contained unit with voltage and current meters and all necessary controls.

6.8 DUMMY LOAD ASSEMBLY

The Dummy Load Assembly consists of resistors mounted on a chassis with appropriate cooling. Load currents will be monitored with meters mounted on the front panel. Separate loads will be provided for solar array simulator power supply, 400-cycle spacecraft power, 2.4-kc spacecraft power and spacecraft d-c power. Any other loads required for testing will be mounted on this assembly, which will have adequate cooling and ventilation.

6.9 POWER TRANSFER

The Power Transfer Assembly will be a separate assembly containing power switches, contactors and circuit breakers. This panel will have no controls except for reset of circuit breakers.

6.10 S/C COMMAND SIMULATOR AND SIGNAL SIMULATOR

The S/C Command Simulator and Signal Simulator will be an electronics assembly with front panel controls. Circuits used should simulate the command output circuits in the vehicle.

6.11 A-C CONDITIONING ASSEMBLY

The A-C Conditioning Assembly is a small self-contained unit with its own power supply. It contains AC-DC Conversion circuits and isolation amplifiers. It will be normally mounted in a junction box near the vehicle, but may be used by itself.

6.12 S/C INTERFACE

The S/C Interface includes all the cable assemblies required to support the use of the Power Subsystem OSE. It will include direct access cables and any other necessary test cables. The cable assemblies will use a standard family of connectors, but care will be taken through selection of specific connectors and rotation of inserts to avoid any possibility of errors in connections.

6.13 POWER SUBSYSTEM OSE TERMINATION

The Power Subsystem OSE termination is a connector and distribution assembly, mounted in the Power OSE cabinet, which provides for cables coming from the spacecraft and distributes the signals throughout the power OSE cabinet.

6.14 PATCHING BOARD

Patching and reconnection of the OSE assemblies to conform to particular test modes or to monitor specific signals is accomplished through a removable patch board. This board is an AMP or MAC patchboard.

6.15 STANDARD UNITS

Listed are the commercially available assemblies and the type of equipment envisioned:

a. Oscillograph — Visicoder or equivalent

b. Digital Recorder — Hewlett Packard or equivalent

c. Oscilloscope — Tektronics 535 or equivalent

d. Meter — NLS Digital Voltmeter

6.16 CABINET POWER SUPPLIES

The cabinet power supply consists of the power supplies required for operating the logic circuits, control displays and amplifiers on the various electrical assemblies contained in the Power Subsystem OSE. One power supply provides 28 vdc rack power for indicators and switches. A second power supply provides various d-c voltages required for the assemblies using logic cards.

7.0 SAFETY CONSIDERATIONS

7.1 SPACECRAFT PROTECTION

Series resistors and/or isolation transformers are provided in all monitoring lines to assure protection of S/C circuits. Direct power inputs to the vehicle have isolated grounds to avoid the possibility of uncontrolled ground loops. Interlocks are provided on the vehicle power input circuits to prevent application of power at times when it may result in S/C or OSE damage. Cable keying should be used, where appropriate.

7.2 OSE AND FACILITIES PROTECTION

Equipment other than the spacecraft is protected by conventional circuit breakers and fuses in the appropriate power lines.

7.3 PERSONNEL SAFETY

In addition to normal procedural precautions, there will be an external and visible cabinet ground connection to a central building ground. The safety ground line will be no smaller than No. 2 AWG. The oscillograph will have an interlock for switching off dangerous voltages during equipment maintenance.

CII - VB264FD102

SYSTEM TEST COMPLEX

ARTICULATION SUBSYSTEM OSE

Index

1.0 SCOPE

This document is a functional description of the Articulation Subsystem OSE which forms a part of the 1971 Voyager System Test Complex. The document describes the functions of this equipment during system test and during subsystem tests.

2.0 APPLICABLE DOCUMENTS

GE

VB260SR101	1971 Voyager Spacecraft System Test Complex Test Objectives and Design Criteria
VB260SR102	1971 Voyager Spacecraft System Test Complex Design Characteristics and Restraints
VB220FD110	1971 Voyager Telemetry Channel Assignment
VB220FD112	1971 Voyager Flight Sequence

3.0 FUNCTIONAL DESCRIPTION

3.1 SYSTEM TESTS

During system tests, the articulation subsystem functions in the same manner as in flight, to the maximum practicable extent. The OSE described herein is utilized primarily to monitor and evaluate the performance of the subsystem. Figure 3-1 is a functional block diagram of this equipment, showing its configuration and use in performing system tests.

The functions required during system tests are as follows:

a. Time, Event and Status Displays and Controls - The Articulation Subsystem OSE requires that test status, certain discrete events, and time be displayed for the cognizant engineer. This panel (or panels) also provides for entry into the STC communication system. Events and switch closure coordination is obtained between the Test Conductor and the Attitude Control and Approach Guidance engineer, through this function.

b. Strip Recorder - The display and analysis of the electronic signals from the planet sensor package, articulation electronics package, and other parts of the articulation subsystem, is implemented with a multichannel recorder. Control of scaling, calibration, speed and selection of channel response is to be implemented at this equipment.

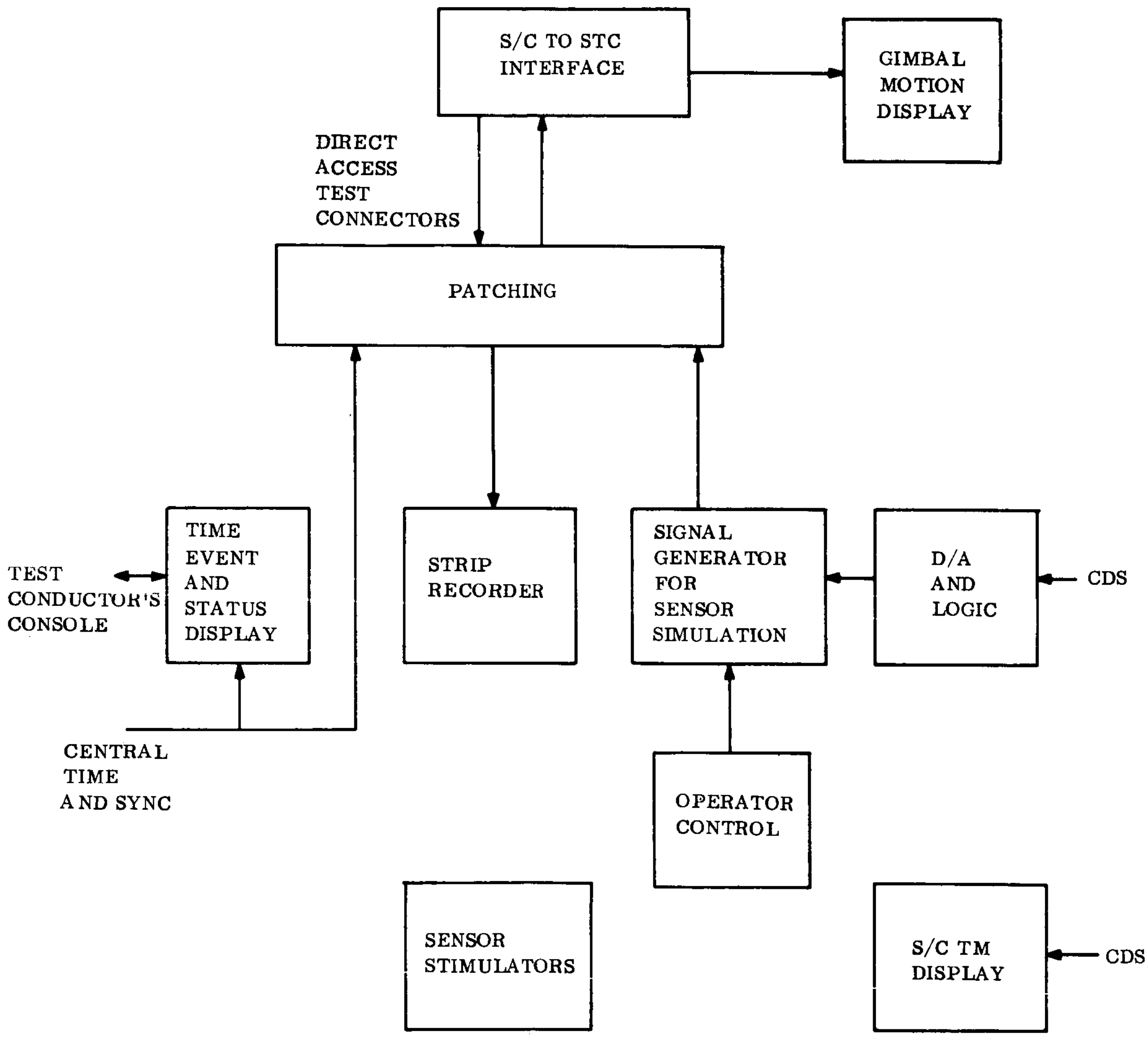


Figure 3-1. Articulation Test Equipment, System Test Configuration, Functional Block Diagram

c. Sensor Signal Generation - The articulation subsystem IR sensor will be simulated during system tests. The signal generator, in response to commands from the CDS or in response to controls of the cognizant engineer, will select and vary appropriate simulated detector signals from a program of available signals, and insert these signals into the electronics of the articulation subsystem. These signals have to have amplitude, phase and other information characteristic of the sensor, so that the subsystem responses are developed.

d. Telemetry Display - A telemetry display driven by the Computer Data System is a functional part of the articulation subsystem monitoring, although not physically integral with the OSE.

e. Power Distribution - Power distribution, test time and running time meters, and communications, although not shown in Figure 3-1, are required.

f. Gimbal Motion Display - A monitor will be provided to give confirmation of operation and phasing characteristics of antenna and instrument package gimbal motion.

g. Sensor Stimulation - The system test will be accomplished using simulated signals, generated by the OSE, in lieu of sensor outputs, to accurately simulate the flight profile. The sensors have to be tested periodically, however, in order to provide a basis for determining articulation subsystem readiness as part of the system test. Stimulators for these sensors are used for this purpose.

3.2 SUBSYSTEM TEST

During articulation Subsystem tests, the Articulation Subsystem OSE will be configured as shown in Figure 3-2. To evaluate subsystem performance, stimuli ordinarily available from interfacing Spacecraft subsystems are required. The Articulation subsystem OSE is required to generate and apply these stimuli during subsystem tests. It also simulates and applies the normal loads to the Articulation subsystem. In this configuration an exhaustive performance test is conducted to ensure high confidence and integrity of flight equipment for use.

Sensor stimulation is the same as for system test. The OSE stimulator auxiliary equipment motion limits will be checked out at more points to give better performance data than is required in system test.

The spacecraft subsystems required to interface with the articulation subsystem such as power, command, telecommunications and pyro will be simulated to the extent required to fully exercise the articulation subsystem.

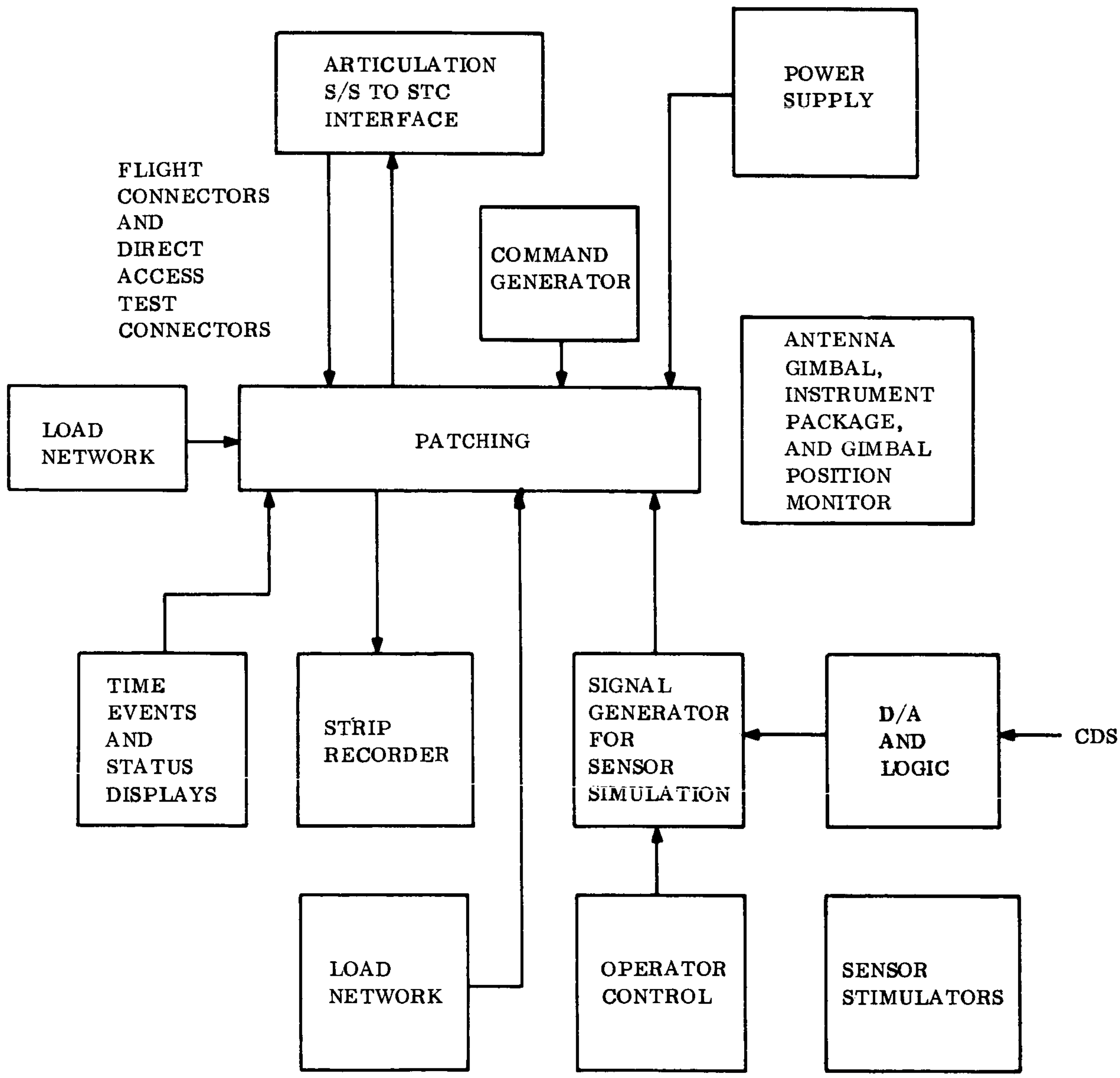


Figure 3-2. Articulation Test Equipment, Subsystem Test Configuration, Functional Block Diagram

The functions required during subsystem tests are as follows:

a. Time, Event and Status Displays and Controls - The Articulation Subsystem OSE requires that test status, certain discrete events, and time be displayed for the cognizant engineer. This panel (or panels) also provides for entry into the STC communication system. Events and switch closure are obtained for the articulation cognizant engineer, through this function.

b. Strip Recorder - The display and analysis of the electronic signals from the planet sensor package, articulation electronics package, and other parts of the articulation subsystem, is implemented with a multichannel recorder. Control of scaling, calibration, speed and selection of channel response is to be implemented at this equipment.

c. Sensor Signal Generation - The articulation subsystem IR sensors will be simulated during subsystem tests as required. The signal generator, in response to commands from the CDS or in response to controls of the cognizant engineer, will react and vary appropriate simulated detector signals from a program of available signals, and insert these signals into the electronics of the Articulation Subsystem. These signals have to have amplitude, phase and other information characteristic of the sensors, so that the proper subsystem responses are developed.

d. Power Distribution - Power distribution, test time and running time meters, and communications, although are not shown in Figure 3-1, are required.

e. Gimbal Motion Display - A monitor will be provided to give confirmation of operation and phasing characteristics of antenna and instrument package gimbal motion.

f. Commands - Since the command subsystem is not available for articulation tests, a command generator equivalent to the S/C command S/S is provided to generate and process Guidance and Control commands.

g. Power - The S/C power pack equivalent will be provided in the A/C, A/G subsystem area.

h. Load - A load network is required to substitute for the electrical load normally seen by the Articulation Subsystem.

i. Sensor Stimulation - Stimulators are required to determine the accuracy and sensitivity of the planet and other sensors.

3.3 SELF TEST

The self test technique to be used for the articulation OSE will be closed loop. The stimulator equipment will be tested through calibrated positions and intensity measurements. The electronic OSE will have end-to-end signal tracing with fault isolation indicators which function in parallel, and which provide isolation from S/C circuits. Data Process/Display equipment will have self test circuits incorporated in the calibration techniques. Articulation actuator monitors will be self tested through the use of dummy loads and positive position indicators. Self test procedures will be incorporated in software and enforced as part of standard test procedure.

4.0 INTERFACE DEFINITION

The interfaces of the Articulation Subsystem OSE are given in Table 4-1.

Table 4-1. Articulation Subsystem Interfaces

Interfacing Item	Input/Output	Characteristics
A. Optical Interfaces		
Planet Sensors	Input	Simulated planet energy
B. Mechanical Interfaces		
Antennae Gimbals	Output	Vernier protractor measurements of positions 0°-180° 0°-200°
Instrument Package Gimbals	Output	same as above
C. Electrical Interfaces		
Planet Sensor	Output	Video waveform
Power Distribution	Input	120V, 60 cps 208V, 400 cps, 3 ϕ
CDS	Input	Sensor stimulation mode and execute - switch closures
	Output	Measured data, test status analogue voltages
Test Conductor	Output	Status (switch closures) and communication
	Input	Communication

Table 4-1. Articulation Subsystem Interfaces (Continued)

Interfacing Item	Input/Output	Characteristics
Articulation SS Electronics	Output	Direct Access test points, monitored data TM sensor outputs, via flight connector
	Input	Simulated S/C power loading and C&S commands, via flight connector
Central Time and Synch	Input	Time

5.0 PERFORMANCE PARAMETERS

a. Command Functions - The command functions for articulation are given in document VB220FD112 Voyager Flight Sequence.

b. Telemetry Functions - The telemetry functions for articulation are given in document VB220FD110 Voyager Telemetry Channel Assignment.

c. Direct Access Monitored Functions - The required direct access functions to be monitored are as listed below:

1.	Gimbal Motor Temperature (1-5)	5
2.	Gimbal Motor Pressure (1-5)	5
3.	Gimbal Motor Voltage + (1-5)	5
4.	Gimbal Motor Voltage - (1-5)	5
5.	Gearbox Pressure (1-5)	5
6.	Gearbox Temp. (1-5)	5
7.	Stepping flipflops	10
8.	Horizon Sensor (Coax)	1

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The articulation OSE will consist of the items in Table 6-1 housed in a portion of a standard console-rack unit with appropriate power adaptors, cooling and communications

Table 6-1. Guidance and Control Subsystems Common Components Usage Matrix

OSE End Item	Articulation S/S	Autopilot S/S	Attitude Control S/S
1 Power Supply	X	X	X
2 Sun Sensor Stimulator			X
3 Star Sensor Stimulator			X
4 Planet Sensor Stimulator	X		X
5 Rate Table		X	X
6 Oscilloscopes	X	X	X
7 Digital Counters	X	X	X
8 Spacecraft Simulator	X	X	X
9 Special Purpose Computer	X	X	X
10 Actuator Monitors		X	X
11 Multi Meter	X	X	X
12 Hardware Monitors		X	X
13 Actuator Simulators		X	X
14 Command Signal Generator	X	X	X
15 Telemetry Point Terminal		X	X
16 Guidance and Control Console	X	X	X
17 Self Test Equipment	X	X	X
18 Alignment Equipment		X	X
19 Calibration Equipment (Unique)	X	X	X
20 Simulated Load Devices		X	X
21 Data Processors/Recorders	X	X	X
22 Gimbal Monitors	X		
23 Test Point	X		
24 Load Network	X		

network receptacals. In addition, the uniqueness of the articulation sensor requires a custom-geometry fixture for stimulator and gimbal motion devices. Auxiliary equipment will include necessary signal generators to isolate faults to modular units as determined by S/C construction technique.

The Articulation, Attitude Control - Approach Guidance, and Autopilot Subsystems of the Spacecraft form the Spacecraft Guidance and Control Subsystem. They use many flight components in common. Similarly, the Articulation OSE, Attitude Control - Approach Guidance OSE and the Autopilot OSE form separate functional support entities, but will use OSE components in common. The total of these components (or end items) forms the total Guidance and Control OSE for the STC.

Table 6-1 is a matrix which delineates this common usage of components in the OSE for these three Guidance and Control Subsystems.

End Item Characteristics

a. Power Supply - Provides equivalent to spacecraft power supply as described in Volume A, Book 3.

b. Planet Sensor Stimulator (Articulation) - Provide energy source to stimulate sensor defined in Volume A, Book 3. IR 14 μ to 35 μ range.

c. Oscilloscopes - Rack mounted.

d. Digital Counters - Rack mounted.

e. Spacecraft Simulator - Appropriate inputs and outputs to permit acceptance tests on G&C S/S.

f. Special Purpose Signal Generator - A signal generator which provides well-behaved functions in lieu of G&C signals for gain, and event determination.

g. Gimbal Monitors (Articulation) - Phasing device to observe proper motion and polarity of valves.

h. Multimeters - Portable display devices for hardwire monitors.

i. Test Point - Patch panel structure suitable to terminate hardwire monitors in articulation equipment and route them 15 for display.

j. Load Network - Loading device to permit testing and troubleshooting actuator circuit in absence of actuators.

k. Command Signal Generator - Hardwire command generator to permit testing of G&C S/S, in absence of CC&S S/S. Items are identified in Articulation Command Functions.

7

l. Guidance and Control Console - Primary equipment for Articulation support of system acceptance tests. Contains switching controls, test points, monitors, self test circuits, junction terminals, system time and safety devices to support Attitude Control, Autopilot and Articulating functions in Mission profile test.

m. Self Test Equipment - As necessary to prevent loading S/C circuitry and capable of fault isolation.

n. Calibration Equipment (Unique) - To meet requirements of the many stimulators and actuator motion detectors used in articulation checkout.

o. Data Processors/Recorders - Display terminals for the approximately 60 hardwire monitors.

7.0 SAFETY CONSIDERATIONS

The articulation Equipment and its inherent Operational Support Equipment consists of mechanical, electrical, pneumatics, and optical components and parts. In addition, standard industrial safety procedures must be used.

7.1 EQUIPMENT

The Articulation subsystem equipment requires special precautions to ensure safety of handling and operation. They are in the area of (1) optical materials, (2) dust protection and coating cleanliness, (3) optical alignment and cleanliness requirements, (4) sensor handling and care, (5) actuator alignment and cleanliness, and (6) electronic equipment voltage and stress limits.

7.2 FACILITIES

No special safety provisions are required to safeguard facilities.

7.3 PERSONNEL

Protection from gimbal motion and motion monitors must be defined.

CII - VB264FD103

SYSTEM TEST COMPLEX

AUTOPILOT

Index

1.0 SCOPE

This document is a functional description of the Autopilot Subsystem OSE which forms a part of the 1971 Voyager System Test Complex. The functions of this equipment during system test and during subsystem tests are described.

2.0 APPLICABLE DOCUMENTATION

VR260SR101	1971 Voyager Spacecraft System Test Complex Test Objectives & Design Criteria
VR260SR102	1971 Voyager Spacecraft System Test Complex Design Characteristics and Restraints
VR210PM018	1971 Voyager Telemetry Channel Assignment
VR210PM013	1971 Voyager Phase IA Command List

3.0 FUNCTIONAL DESCRIPTION

3.1 SYSTEM TEST

During system test, the Autopilot subsystem functions in the same manner as in flight, to the maximum practicable extent. The OSE described herein is utilized primarily to monitor and evaluate the performance of the S/C subsystem.

Figure 3-1 is a functional block diagram of this equipment, showing its configuration and use in performing system tests.

3.1.1 FUNCTIONS

The functions required during system tests are:

a. Time, Event and Status Displays and Contracts - The autopilot OSE requires that test status, certain discrete events, and time be displayed for the engineer. Events and switch closure coordination is obtained between the Test Conductor and the autopilot engineer, through this function.

b. Strip Recorder - The display and analysis of the electronic signals from the several areas of the Gyro package, and other parts of the A/P subsystem, is implemented with a multichannel recorder. Control of scaling, calibration, speed and selection of channel response is to be implemented at this equipment.

c. Gyroscope Output Display - The waveform of gyroscope outputs is to be displayed on an oscilloscope. The signals are routed through the direct-access test connectors. Since this display may also require more precise frequency

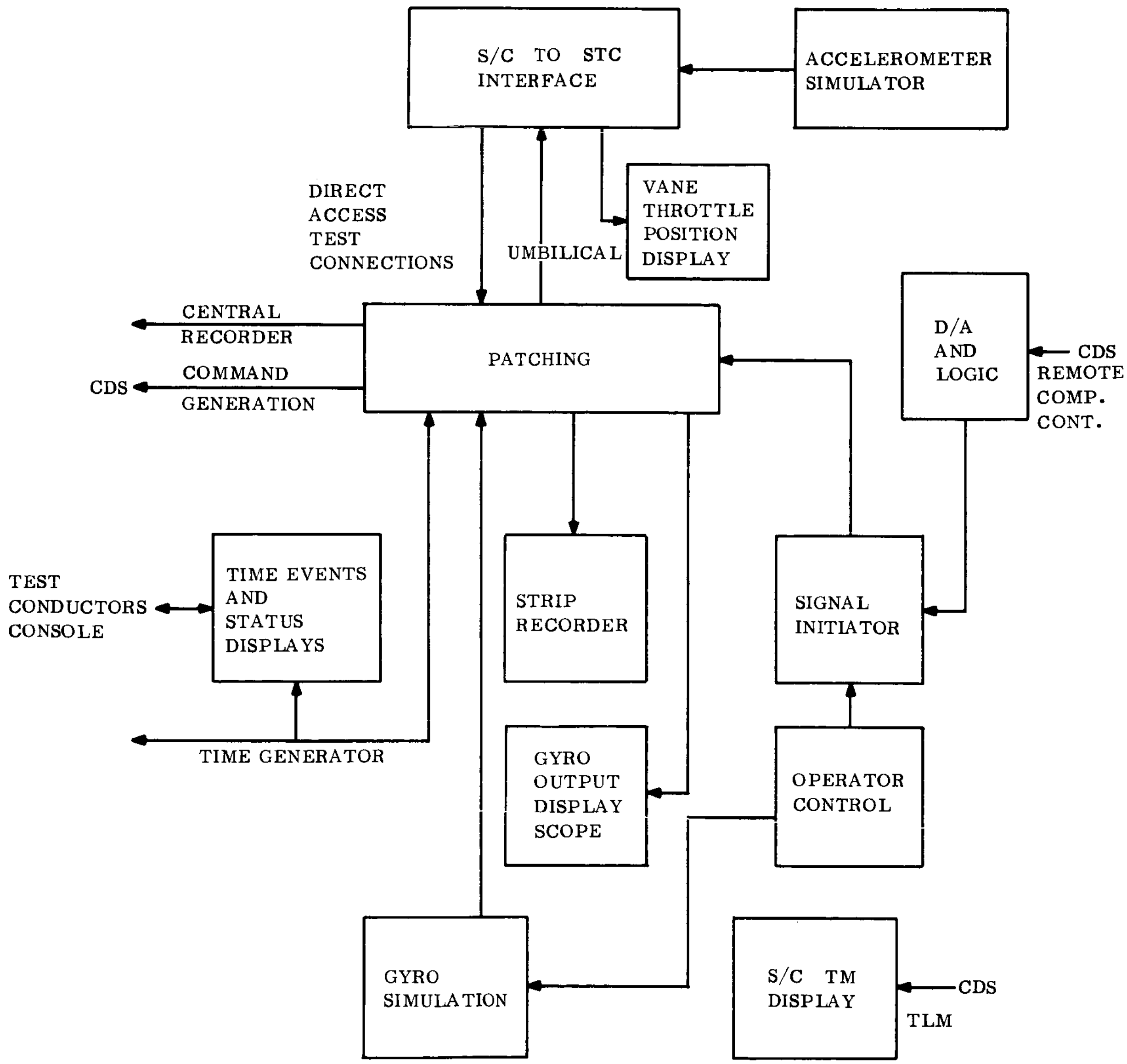


Figure 3-1. Autopilot Test Equipment, System Test Configuration, Functional Block Diagram

measurement and display instrumentation than an oscilloscope can provide, a frequency meter (counter) may be required.

d. Sensor Signal Generation - The A/P subsystem acceleration and gyro sensors will be simulated during system tests. The signal generator, in response to commands from the CDS or in response to controls of the cognizant engineer, will select and vary appropriate simulated detector signals from a program of available signals, and insert these signals into the electronics of the A/P subsystem. These signals have to have amplitude, phase and other information characteristic of the sensors, so that the proper subsystem responses are developed.

e. Telemetry Display - A telemetry display, driven by the Computer Data System, is a functional part of the A/P subsystem monitoring, although not physically integral with the OSE.

f. Power Distribution - Test time and running time meters, and communications, although not shown in Figure 3-1, are required.

g. Actuator Monitors - A Display Monitor for engine (4) throttle position and engine (4) vane position is provided to enable the subsystem engineer to confirm operation and phasing of A/P actuators.

3.2 SUBSYSTEM TEST

During the STC subsystem test cycle the A/P equipment will be configured as shown in Figure 3-2. In this configuration an exhaustive performance test is conducted to ensure high confidence and preparation of flight equipment for use.

Sensor stimulators are the same as for system test except for the Gyro rate table which is used separately to test the quality of gyro package performance prior to their being mounted on spacecraft. (Gyros from A/C, S/S.)

The OSE vane and throttle position equipment motion limits will be checked out at more points to give better performance data than is required in the system test.

The spacecraft subsystems required for interface with the A/P such as power, command, telecommunications and pyro will be simulated to the extent required to fully exercise the A/P subsystem.

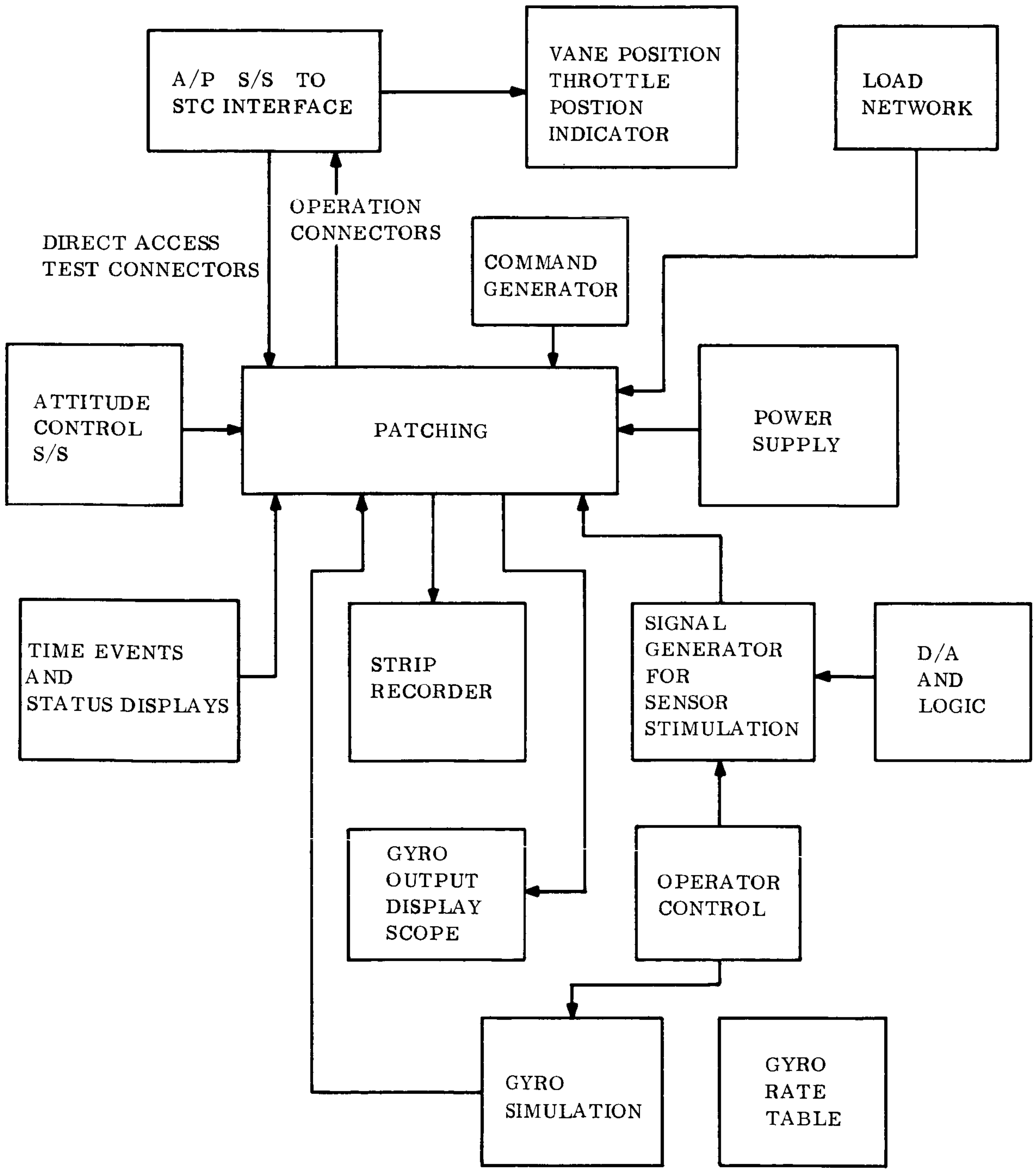


Figure 3-2. Autopilot Test Equipment, Subsystem Test Configuration, Functional Block Diagram

3.2.1 FUNCTIONS

The functions required during subsystem tests are:

a. Time, Event and Status Displays and Controls - The autopilot subsystem OSE requires that test status, certain discrete events, and time be displayed for the cognizant engineer. This panel (or panels) also provides for entry into the STC communication system. Events and switch closure is obtained through this function.

b. Strip Recorder - The display and analysis of the electronic signals from the several areas of the Gyro package, Attitude Control Electronics package, and other parts of the A/P subsystem, is implemented with a multichannel recorder. Control of scaling, calibration, speed and selection of channel response is to be implemented at this equipment.

c. Gyroscope Output Display - The waveform of gyroscope outputs, is to be displayed on an oscilloscope. The signals are routed through the direct access test connectors.

d. Telemetry Display - A telemetry display, driven by the S/C sensor via hardline, is an integral part of the A/P subsystem monitoring.

e. Accelerometer Simulator and Activator - The equivalent output of the accelerometer and gyros will be simulated signals inserted in their loops for certain phases of the subsystem test. At the times designated by his status and event displays, the cognizant engineer will select from a fixed program of gyro and accelerometer outputs the proper ones to insert into the subsystem.
its timing and

f. Power Distribution - Test time and running time meters, and communications required as part of the subsystem displays.

g. Actuator Monitors - A display monitor for engine (4) throttle position and engine (4) vane position is provided to enable the subsystem engineer to confirm operation and phasing of A/P actuators.

h. Commands - Since the command subsystem is not available for A/P S/S tests a command generator equivalent to the S/C command S/S is provided to generate and process guidance and control commands.

i. Power - The S/C power pack equivalent will be provided in the A/P subsystem area.

j. Gyro Rate Table - The gyro package must be tested separately prior to replacement of the bay containing the gyro package within the bay. A rate table is required for these tests of drift, coast-down, bias, scale, etc.

k. A load network is required to substitute for the electrical load normally seen by the A/P.

3.3 SELF TEST OF OSE

The self test technique to be used for the A/P, OSE will be closed loop non-interference with S/C test. The stimulator equipment will be tested through calibrated positions and intensity measurements. The electronic support OSE will have end-to-end signal tracing with fault isolation indicators which function in parallel and isolation with S/C circuitry. Data Process/Display equipment will have self test circuits incorporated in calibration techniques. A/P actuator monitors will be self test through dummy loads and positive position indicators. Self test procedures will be incorporated in software and enforced as part of standard test procedure.

4.0 INTERFACE DEFINITION

The interfaces of the Autopilot Subsystem OSE are given in Tables 4-1 and 4-2.

5.0 PERFORMANCE PARAMETERS

5.1 COMMAND FUNCTIONS

The command functions for A/P is given in document VR210PM013 Voyager Phase 1A Command List.

5.2 TELEMETRY FUNCTIONS

The telemetry functions for A/P is given in document VB220FD110 Volume A Voyager Telemetry Channel Assignment.

5.3 DIRECT ACCESS MONITORED FUNCTIONS

The required hardwire functions for A/P are as listed in Table 5-1.

5.4 UMBILICAL HARDWIRE FUNCTIONS

The umbilical functions for A/P are as listed in document VB280SR102 for LCE (Pitch Gyro, Yaw Gyro, Roll Gyro, Heater power, Power return signal ground, shield return). Specific use of each function will depend upon test configuration and problems encountered.

Table 4-1. Mechanical Interfaces

Interfacing Item	Input/Output	Characteristics
Accelerometer	Input	0-5 g
(subsystem testing only)		0-10000 Fps

Table 4-2. Electrical Interfaces

Interfacing Item	Input/Output	Characteristics
Ground Power Distn.	Input	120 v, 60~ 208 v, 400~, 3φ
Computer Data System	Input	Commands for sensor and gyro simulation - switch closures
	Output	Monitored analogue test data
Control Recorder	Output	Monitored analogue test data
Test Conductor's Console	Input	Switch closures
	Output	Test status
Autopilot Subsystem	Output	Direct access functions umbilical functions, simulated s/c power
	Input	Umbilical and flight connector, C & S commands and loading Autopilot power Simulated torque and sensor output signals
Control Time and Synch	Input	Time

Table 5-1. Hardwire Monitors - Autopilot

		WIRES
1	Accelerometer Output Amplifier	1
2	Accelerometer Torque Input	1
3	Accelerometer Logic Circuits	2
4	MCS Engine Control Error Signals	8
5	MCS Engine Control Feedback Signals	8
6	Velocimeter Engine Cutoff	1
	TOTAL	21

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The autopilot OSE will consist of the items listed in Table 6-1 housed in a standard three-bay console-rack unit with appropriate power adaptors, cooling and communications network recepticals. In addition, the uniqueness of the A/P sensors requires custom-geometry fixtures for stimulators and actuator-motion devices. Auxiliary equipment will include necessary signal generators to isolate faults to modular units as determined by S/C construction techniques.

The Articulation, Attitude Control — Approach Guidance, and Autopilot Subsystems of the Spacecraft form the Spacecraft Guidance and Control Subsystem. They use many flight components in common. Similarly, the Articulation OSE, Attitude Control — Approach Guidance OSE and the Autopilot OSE form separate functional support entities, but will use OSE components in common. The total of these components (or end items) forms the total Guidance and Control OSE for the STC.

Table 6-1 is a matrix which delineates this common usage of components in the OSE for these three Guidance and Control Subsystems.

End Item (EI) Characteristics are as follows:

a. Power Supply - Provides equivalent to spacecraft power supply as described in Volume A, Book 3.

   Provide energy source to stimulate sensors defined in Volume A, Book 3.

b. Rate Table - Standard or special rate table of the Fecker type to excite gyros used in A/P S/S Block Diagram. Required for position measurements with a 0 to ±6-degree capability.

Table 6-1. Guidance and Control Subsystems Common Use Matrix

	OSE End Item	Articulation S/S	Autopilot S/S	Attitude Control S/S
1	Power Supply	X	X	X
2	Sun Sensor Stimulator			X
3	Star Sensor Stimulator			X
4	Planet Sensor Stimulator	X		X
5	Rate Table		X	X
6	Oscilloscopes	X	X	X
7	Digital Counters	X	X	X
8	Spacecraft Simulator	X	X	X
9	Special Purpose Computer	X	X	X
10	Actuator Monitors		X	X
11	Multi Meters	X	X	X
12	Hardware Monitors		X	X
13	Actuator Simulators		X	X
14	Command Signal Generator	X	X	X
15	Telemetry Point Terminal		X	X
16	Guidance and Control Console	X	X	X
17	Self Test Equipment	X	X	X
18	Alignment Equipment		X	X
19	Calibration Equipment (Unique)	X	X	X
20	Simulated Load Devices		X	X
21	Data Processors/Recorders	X	X	X
22	Gimbal Monitors	X		
23	Test Point	X		
24	Load Network	X		

c. Oscilloscopes

d. Digital Counters

e. Spacecraft Simulator - Appropriate inputs and outputs to permit acceptance tests on G&C S/S without using other S/V prime S/S components.

f. Special Purpose Computer - A signal generator which provides well-behaved functions in lieu of G&C signals for gain, switching lines and event determination.

g. Actuator Monitors (A/P) - Phasing, device to ensure proper firing and polarity of valves.

h. Multimeters - Portable display devices for hardwire monitors.

i. Hardwire Monitors - Patch panel structure suitable to terminate hardwire monitors in A/P equipment and route them to EI 18 for display.

j. Actuator Simulators - Loading device to permit testing and troubleshooting actuator circuit in absence of actuators.

k. Command Signal Generator - Hardwire command pattern to permit testing of G&C S/S, in absence of C&S S/S. Items are identified in A/P, Command Functions.

l. Telemetry Point Terminal - Monitor board for telemetry functions in absence of TLM S/S in G&C S/S tests. Items are identified in Voyager Telemetry Requirements.

m. Guidance and Control Console - Primary EI for A/P support of system acceptance tests. Contains switching controls, test points, monitors, self test circuits, junction terminals, system time and safety devices to support Attitude Control, Autopilot and Articulating functions in Mission profile test.

n. Self Test Equipment - Necessary to prevent loading S/C circuitry and capable of fault isolation.

o. Calibration Equipment - To meet requirements of the many stimulators and actuator motion detectors used in A/P checkout.

p. Simulated Load Devices - S/C equivalent interfaces with the A/P needed is subsystem evaluation.

q. Data Processors/Recorders - Display terminals for the approximately 118 hardwire monitors.

7.0 SAFETY CONSIDERATIONS

The A/P Equipment and its inherent Operational Support Equipment consists of mechanical, electrical components and parts. Standard industrial safety procedures must be used in addition.

7.1 EQUIPMENT

Gyro and Accelerometer Package requires special handling and operation procedures.

7.2 FACILITIES

No special safety provisions are required to safeguard facilities.

7.3 PERSONNEL

Protection from stimulator energy and actuator motion monitors must be defined.

CII - VB264FD104

SYSTEM TEST COMPLEX

ATTITUDE CONTROL AND APPROACH GUIDANCE

Index

1.0 SCOPE

This document is a functional description of the Attitude Control and Approach Guidance Control Subsystems OSE which forms a part of the 1971 Voyager System Test Complex. The document describes the functions of this equipment during system test and during subsystem tests.

2.0 APPLICABLE DOCUMENTS

GE

VR260SR101	1971 Voyager Spacecraft System Test Complex Test Objectives and Design Criteria
VR260SR102	1971 Voyager Spacecraft System Test Complex Design Characteristics and Restraints
VR210PM018	1971 Voyager Telemetry Channel Assignment
VR210PM014	1971 Voyager Phase IA Command List

3.0 FUNCTIONAL DESCRIPTION

3.1 SYSTEM TEST

During system tests, the Attitude Control and Approach Guidance subsystems function in the same manner as in flight, to the maximum practicable extent. The OSE described herein is utilized primarily to monitor and evaluate the performance of the subsystem. To evaluate subsystem performance, stimuli not available from interfacing Spacecraft subsystems are required. The Attitude Control and Approach Guidance Subsystem OSE is required to generate and apply these stimuli during System Test.

Figure 3-1 is a functional block diagram of this equipment, showing its configuration and use in performing system tests.

The functions required during system tests are:

a. Time, Event and Status Displays and Controls - The Attitude Control and Approach Guidance Subsystems OSE require that test status, certain discrete events, and time be displayed for the cognizant engineer. This panel (or panels) also provides for entry into the STC communication system. Events and switch closure coordination is obtained between the Test Conductor and the Attitude Control and Approach Guidance cognizant engineer, through this function.

b. Strip Recorder - The display of and analysis of the electronic signals from the several areas of the Gyro package, Attitude Control Electronics package, and other parts of the A/C and A/G subsystem, is implemented with a multichannel recorder.

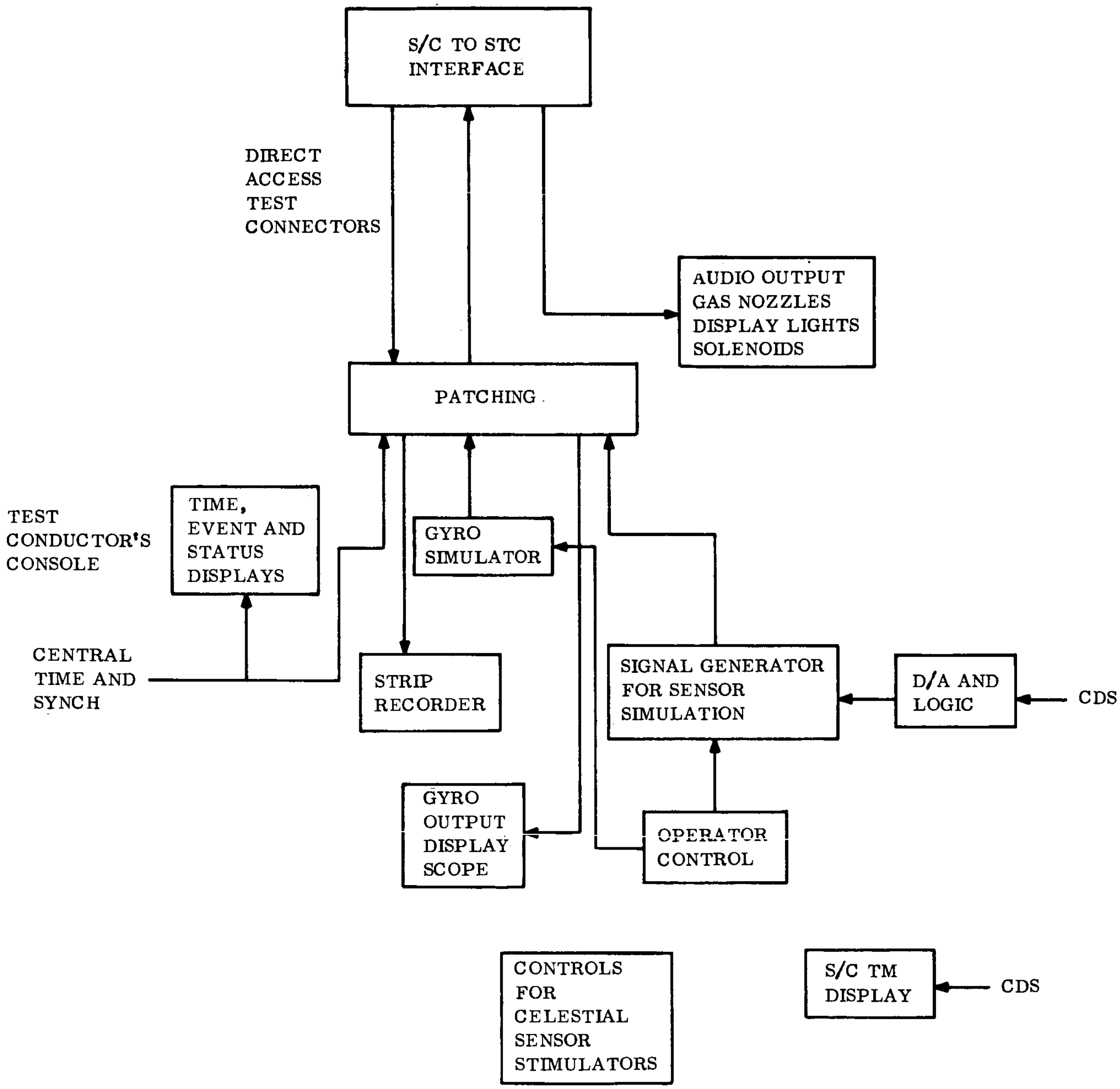


Figure 3-1. Attitude Control and Approach Guidance Test Equipment, System Test Configuration Functional Block Diagram

Control of scaling, calibration, speed and selection of channel response is to be implemented at this equipment.

c. Gyroscope Output Display - The waveform of gyroscope outputs is to be displayed on an oscilloscope. The signals are routed through the direct access test connectors. Since this display may require more precise frequency measurement and display instrumentation than an oscilloscope can provide, a frequency meter (counter) will be provided.

d. Sensor Signal Generation - The A/C and A/G subsystem optical and IR sensors will be simulated during system tests. The signal generator, in response to commands from the CDS or in response to controls of the cognizant engineer, will select and vary appropriate simulated detector signals from a program of available signals, and insert these signals into the electronics of the A/C and A/G subsystem. These signals have to have amplitude, phase and other information characteristics of the sensors, so that the proper subsystem responses are developed.

e. Telemetry Display - A telemetry display, driven by the Computer Data System, is a functional part of the A/C and A/G subsystem monitoring, although not physically integral with the OSE.

f. Gyro Simulator - The equivalent output of the gyros will be simulated signals inserted in their loops for certain phases of the system test. At the time designated by his status and event displays, and as instructed by the Test Conductor, the cognizant engineer will select from a fixed program of gyro and accelerometer outputs, the proper ones to insert into the subsystem.

g. Optical Stimulation - Optical stimulation is characteristic of subsystem testing, rather than system tests. However, the cognizant engineer will have to make confidence tests of the optical sensors either just before, after, or during system tests. The controls for the electro-optical stimulation equipment, i.e., angle off, intensity, contrast, etc., have to be available to him, and the stimulating fixtures also have to be available.

h. Power Distribution - Power distribution, test time and running time meters, and communications, although not shown in Figure 3-1, are required.

i. Actuator Monitors - A monitor of audio and light displays will be provided to give confirmation of operation and phasing characteristics of A/C actuators. The gas supply is to be partially charged in order to excersize actuators.

3.2 SUBSYSTEM TEST

During the STC subsystem test cycle the A/C and A/G equipment will be configured as shown in Figure 3-2. In this configuration an exhaustive performance test is conducted to ensure high confidence of flight equipment for use.

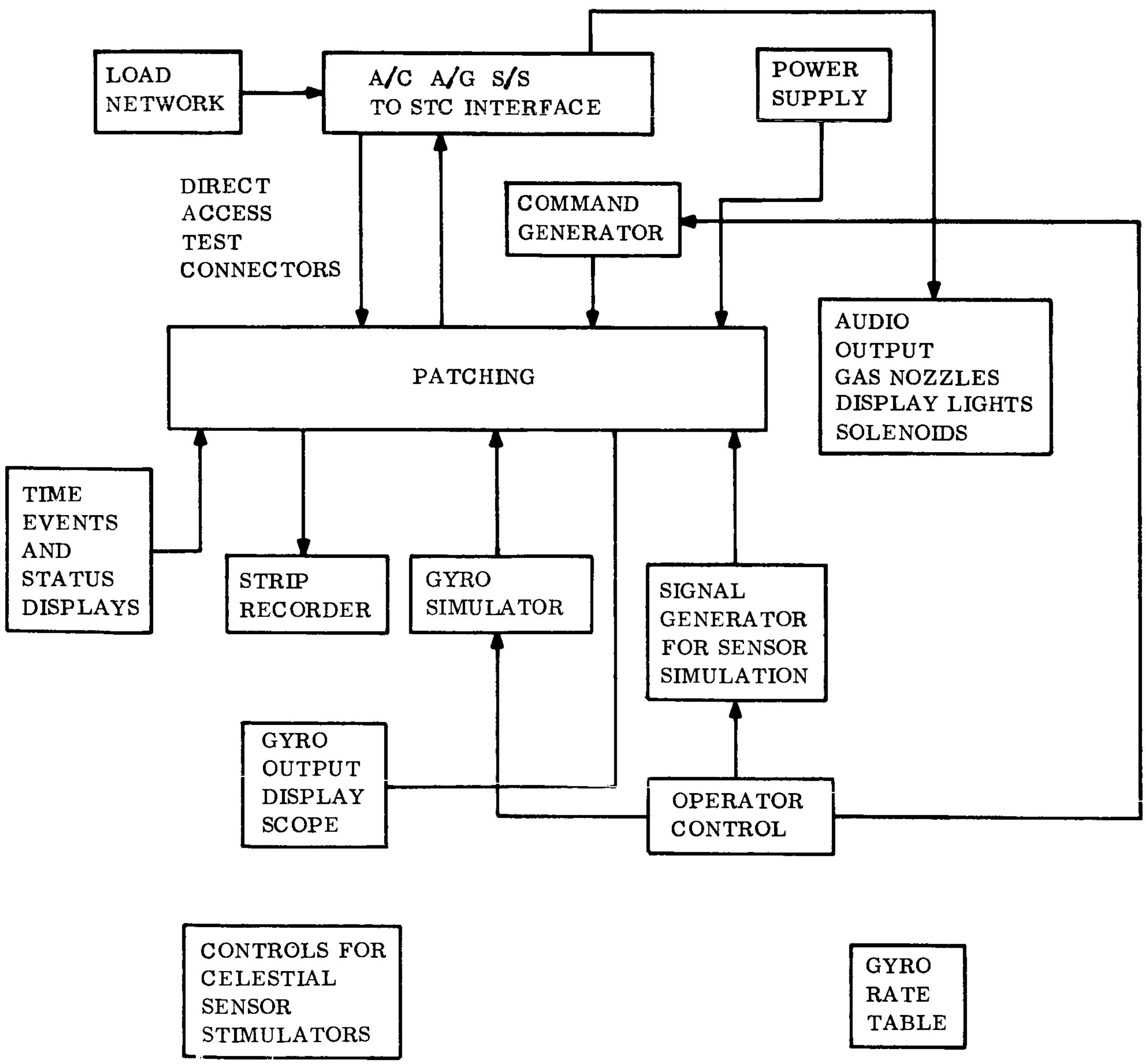


Figure 3-2. Attitude Control and Approach Guidance Test Equipment, Subsystem Test Configuration, Functional Block Diagram

Sensor stimulators are the same as for system test except for Gyro rate table which is used to separately test the quality of gyro package performance prior to their being mounted on the spacecraft.

The OSE stimulator auxiliary equipment motion limits will be checked out at more points to give better performance data than is required in system test.

The spacecraft subsystems required to interface with the A/C and A/G subsystems for power, command, telecommunications and pyro will be simulated to the extent required to fully exercise the A/C and A/G subsytems.

The functions required during subsystem tests are:

a. Time, Event and Status Displays & Controls - The Attitude Control & Approach Guidance Subsystems OSE require that test status, certain discrete events, and time be displayed for the cognizant engineer. This panel (or panels) also provides for entry into the STC communication system.

b. Strip Recorder - The display of and analysis of the electronic signals from the several areas of the Gyro Package, Attitude Control Electronics package, and other parts of the A/C and A/G subsystem, is implemented with a multichannel recorder. Control of scaling, calibration, speed and selection of channel response is to be implemented at this equipment.

c. Gyroscope Output Display - The waveform of gyroscope outputs is to be displayed on an oscilloscope. The signals are routed through the direct access test connectors. This display may also require more precise frequency measurement and display instrumentation than an oscilloscope.

d. Sensor Signal Generation - TheA/C and A/G subsystem optical and IR sensors will be simulated during subsystem tests as required. The signal generator, in response to commands from the CDS or in response to controls of the cognizant engineer, will react and vary appropriate simulated detector signals from a program of available signals, and insert these signals into the electronics of the A/C and A/G subsystem. These signals have to have amplitude, phase and other information characteristics of the sensors, so that the proper subsystem responses are developed.

e. Gyro Simulator - The equivalent output of the gyros will be simulated signals inserted in their loops, for certain phases of the subsystem test. At the time designated by his status and event displays, the cognizant engineer will select from a fixed program of gyro and accelerometer outputs, the proper ones to insert into the subsystem. This function will be under the engineer's control.

f. Optical Stimulation - Optical stimulation is characteristic of subsystem testing, rather than system tests. The controls for the electro-optical stimulation equipment, e.g., angle off, intensity, contrast, etc., have to be available, and the stimulating fixtures also have to be available.

g. Power Distribution - Power distribution, test time and running time meters, and communications are required as part of the subsystem displays.

h. Actuator Monitors - A monitor of audio and light displays gives confirmation of operation and phasing of A/C actuators.

i. Commands - Since the command subsystem is not available for A/C and A/G subsystem tests, a command generator equivalent to the S/C command Subsystem is provided to generate and process Guidance and Control commands.

j. Power - The S/C power pack equivalent will be provided in the A/C and A/G subsystem area.

k. Gyro Rate Table - The Gyro package must be tested separately prior to replacement of the bay containing gyros, or the gyro package within the bay. A rate table is required for these tests of drift, coast down, bias, scale, etc.

3.3 SELF TEST OF OSE

The self test technique to be used for the A/C and A/G OSE will be closed loop. The stimulator equipment will be tested through calibrated positions and intensity measurements. The electronic support OSE will have end-to-end signal tracing with fault isolation indicators which functions in parallel and isolation with S/C circuitry. Data Process/Display equipment will have self test circuits incorporated in calibration techniques. A/C and A/G actuator monitors will be self test through dummy loads and positive position indicators. Self test procedures will be incorporated in software and enforced as part of standard test procedure.

4.0 INTERFACE DEFINITION

The interfaces of the Attitude Control and Approach Guidance Subsystem OSE are given in the following tables:

Table 4-1. Optical Interfaces

Interfacing Item	Input/Output	Characteristics
Sun Sensors	Input	Zenon light, 32 min arc, ±1 sun equivalent
Star Trackers	Input	Fibre optic pencil, ±3 stellar magnitudes
Approach Guidance Sensor	Input	Visual light (Mars, sun, Canopus)

Table 4.2. Mechanical Interface

Interfacing Item	Input/Output	Characteristic
Vehicle Nozzle Gas Stream	Output	Detect gas flow inception

Tabel 4.3 Electrical Interfaces

Interfacing Item	Input/Output	Characteristics
Ground Power Distn.	Input	120 V, 60 ~ 208 V, 400 ~, 3Ø
Control Time & Synch	Input	Time signals
Test Conductor's Console	Input	Switch closures, communication
	Output	Test status, switch closure and communication
Computer Data System	Input	Execute commands
	Output	Test data, status
Control Recorder	Output	Test Data
Attitude Control and Approach Guidance Subsystem	Output	Simulated spacecraft power,
		Simulated sensor signals,
		Simulated commands,
		Simulated loading,
	Input	Direct access test point and umbilical monitored data TM sensor outputs from flight connectors.

5.0 PERFORMANCE PARAMETERS

5.1 COMMAND FUNCTIONS

The command functions for A/C and A/G is given in document VR210PM014 Voyager Phase IA Command List.

5.2 TELEMETRY FUNCTIONS

The telemetry functions for A/C and A/G is given in document VB220FD110 Volume A Voyager Telemetry Channel Assignment.

5.3 DIRECT-ACCESS MONITORED FUNCTIONS

The required functions for A/C and A/G are as listed in Table 5-1.

Table 5-1. A/C and A/G Direct-Access Monitored Functions

1.	Acquisition Sun Sensor simulation inputs three per two axes	6
2.	Cruise Sun Sensor simulation inputs three per two axes	6
3.	Canopus simulation input three per one axis	3
4.	Threshold Detector Outputs six per three axes	18
5.	Majority gates output four per three axes	12
6.	Solenoid Drive Current four per three axes	12
7.	Roll Integrator Output	1
8.	Sun gate Amplifier Output	2
9.	Gyro Output Amplifier one per three axes	3
10.	Accelerometer Output Amplifier	1
11.	Accelerometer Torque Input	1
12.	Gyro Logic Circuits	7
13.	Accelerometer Logic Circuits	2

5.4 UMBILICAL HARDWIRE FUNCTIONS

The umbilical functions for A/C and A/G are as listed in document VB280SR102, for LCE (Pitch Gyro, Yaw Gyro, Roll Gyro, Heater power, Power return signal ground, shield return). Specific use of each function will depend upon test configuration.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The attitude control and guidance sensor OSE will consist of the following items housed in two standard console-rack units with appropriate power adaptors, cooling and communications network receptacles. In addition, the uniqueness of the A/C and A/G sensors requires custom-geometry fixtures for stimulators and actuator-motion devices. Auxiliary equipment will include necessary signal generators to isolate faults to modular units as determined by S/C construction technique.

The Articulation, Attitude Control - Approach Guidance, and Auto Pilot Subsystems of the Spacecraft form the Spacecraft's Guidance and Control Subsystem. They use many flight components in common. Similarly, the Articulation OSE, Attitude Control - Approach Guidance OSE and the Autopilot OSE form separate functional support entities, but will use OSE components in common. The total of these components (or end items) forms the total Guidance and Control OSE for the STC.

Table 6-1 is a matrix which delineates this common usage of components in the OSE for these three Guidance and Control Subsystems.

Table 6-1

OSE End Item	Articulation S/S	Autopilot S/S	Attitude Control S/S
1. Power Supply	X	X	X
2. Sun Sensor Stimulator			X
3. Star Sensor Stimulator			X
4. Planet Sensor Stimulator	X		X
5. Rate Table		X	X
6. Oscilloscopes	X	X	X
7. Digital Counters	X	X	X
8. Spacecraft Simulator	X	X	X
9. Special Purpose Computer	X	X	X
10. Actuator Monitors		X	X
11. Multi Meters	X	X	X
12. Hardware Monitors		X	X
13. Actuator Simulators		X	X
14. Command Signal Generator	X	X	X

Table 6-1. (Continued)

OSE End Item	Articulation S/S	Autopilot S/S	Attitude Control S/S
15. Telemetry Point Terminal		X	X
16. Guidance and Control Console	X	X	X
17. Self Test Equipment	X	X	X
18. Alignment Equipment		X	X
19. Calibration Equipment (Unique)	X	X	X
20. Simulated Load Devices		X	X
21. Data Processors/Recorders	X	X	X
22. Gimbal Monitors	X		
23. Test Point	X		
24. Load Network	X		

End Item Characteristics follow:

a. Power Supply - Provides equivalent to spacecraft power supply as described in Volume A, Book 3.

b. Sun Sensor Stimulator - Carbon arc on Xenon lamp suitable in spectral quantities to excite and test pitch and yaw, coarse, fine and sun gate sun sensors as described in Voyager 1971, Attitude Control S/S functional block diagram. The carbon arc affords the closest approximation to the sun's spectrum; the stimulator is designed for 32 minutes collimation with a radiance of 900 watts per square centimeter per steradian. The intensity is ± 1 Sun. An alternative approach would be a Xenon lamp (5 KW), 32 arc minutes collimation, giving ± 1 Sun.

c. Star Sensor Stimulator - A tunsten source with fiber optics at the focal point of a parabolic mirror is used to excite the Canopus Star Sensor. The stimulator is designed for a five arc second collimation. The tunsten intensity is varied to cover a range of -3 to +3 magnitude star.

d. Planet Sensor Stimulator (A/G) - Provide energy source to stimulate sensors defined in Volume A, Book 3.

e. Rate Table - Standard or special rate table of the Fekkar type to excite gyros used in A/C Subsystem Block Diagram. Required for rate and position measurements with a 0 to ±6 degree/sec capability.

f. Oscilloscopes

g. Digital Counters

h. Spacecraft Simulator - Appropriate inputs and outputs to permit tests on G&C Subsystem.

i. Special Purpose Signal Generator - A signal generator which provides well-behaved functions in lieu of G&C signals for gain, switching lines and event determination.

j. Actuator Monitors (A/C) - Phasing device to ensure proper firing and polarity of valves.

k. Multimeters - Portable display devices for hardwire monitors.

l. Test Point Monitors - Patch panel structure suitable to terminate hardwire monitors in A/C and A/G equipment and route them to EI 19 for display.

m. Load Network

n. Command Signal Generator - Hardwire command pattern to permit testing of G&C Subsystem in absence of CC&S Subsystem Items are identified in A/C and A/G Command Functions.

o. Guidance and Control Console - Primary EI for A/C and A/G support of system acceptance tests. Contains switching controls, test points, monitors, self test circuits, junction terminals, system time and safety devices to support Attitude Control, Autopilot and Articulating functions in Mission profile test.

p. Self Test Equipment

q. Calibration Equipment - To meet requirements of the many stimulators and actuator motion detectors used in A/C and A/G checkout.

r. Simulated Loads - S/C equivalent interfaces with the A/C and A/G needed for subsystem tests.

s. Data Displays/Recorders - Display terminals for the approximately 118 test points.

7.0 SAFETY CONSIDERATIONS

The A/C and A/G Equipment and its inherent Operational Support Equipment consists of mechanical, electrical, pneumatics, and optical components and parts. Standard industrial safety procedures must be used in addition.

7.1 EQUIPMENT

The A/C and A/G subsystem equipment requires special precautions to insure safety of handling and operation. They are in the area of (1) Sun sensor cell structure and coatings, (2) optical materials and alignment, (3) star tracker light intensity, dust protection and coating cleanliness, (4) approach guidance sensor optical alignment and cleanliness requirements, (5) sensor handling and care, (6) actuator alignment and cleanliness, and (7) electronic equipment voltage and stress limits.

7.2 FACILITIES

No special safety provisions are required to safeguard facilities.

7.3 PERSONNEL

Protection from stimulator energy and actuator gases should be provided.

CII - VB265FD101

SYSTEM TEST COMPLEX

PYROTECHNIC SUBSYSTEM OPERATIONAL SUPPORT EQUIPMENT

Index

1.0 SCOPE

This document is a functional description of the Pyrotechnic Subsystem OSE for use in the System Test Complex. This equipment will be used for performing tests on the Pyrotechnic Subsystem and for supporting spacecraft system tests which exercise the Pyrotechnic Subsystem.

2.0 APPLICABLE DOCUMENTATION

VB235FD104	1971 Voyager Spacecraft Functional Description of Pyrotechnic Subsystem
VB260SR101	1971 Voyager Spacecraft System Test Complex - Test Objectives and Design Criteria
VB260SR102	1971 Voyager Spacecraft System Test Complex - Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

3.1 SYSTEM TESTS

Refer to Figure 3-1 - Pyrotechnic Subsystem OSE and Figure 3-2, Pyrotechnic Test Block Diagram.

During the system test, all arming and firing commands will come to the pyro controllers from within the spacecraft. The COMMAND GENERATION PANEL will have the capability of initiating firing signals through the Computer Data System. Reset commands and mode setting commands will be applied through the umbilical connector from the PYROTECHNIC CONTROL PANEL. All umbilical functions related to the Pyrotechnic Subsystem will be displayed on this panel.

Telemetry event signals from the Pyrotechnics Subsystem will be processed through the spacecraft telemetry system, decommutated by the Date Encoder OSE and displayed on a CDS controlled digital printer mounted near the Pyrotechnic Subsystem OSE.

Squib simulators will be used to simulate the resistance characteristics of the squibs during tests. When all simulators are connected, the system will be electrically the same as the spacecraft flight configuration. Each simulator will have capability for connecting monitoring lines to a direct writing recorder which can record the voltage pulse across the simulators when the firing pulse is received. The normal mode of operation in system test will be to monitor direct access points on the controller and record the voltage across series resistors in the pyro firing line. The recorded waveform will be examined to determine if the pulse is of sufficient amplitude and duration. The signal will also actuate a holding circuit tied to a display lamp showing the operator

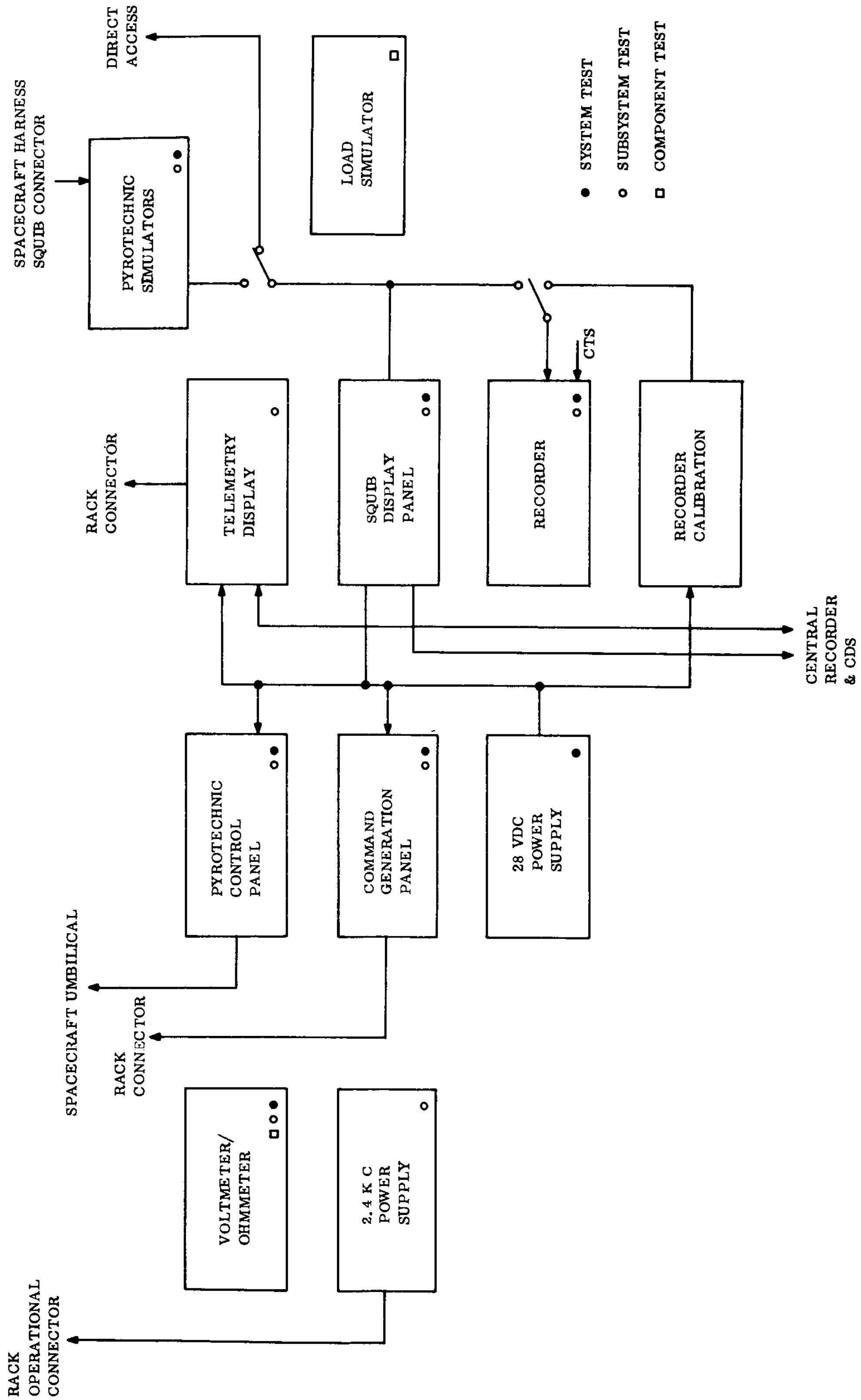


Figure 3-1. Pyrotechnic Subsystem OSE, Block Diagram

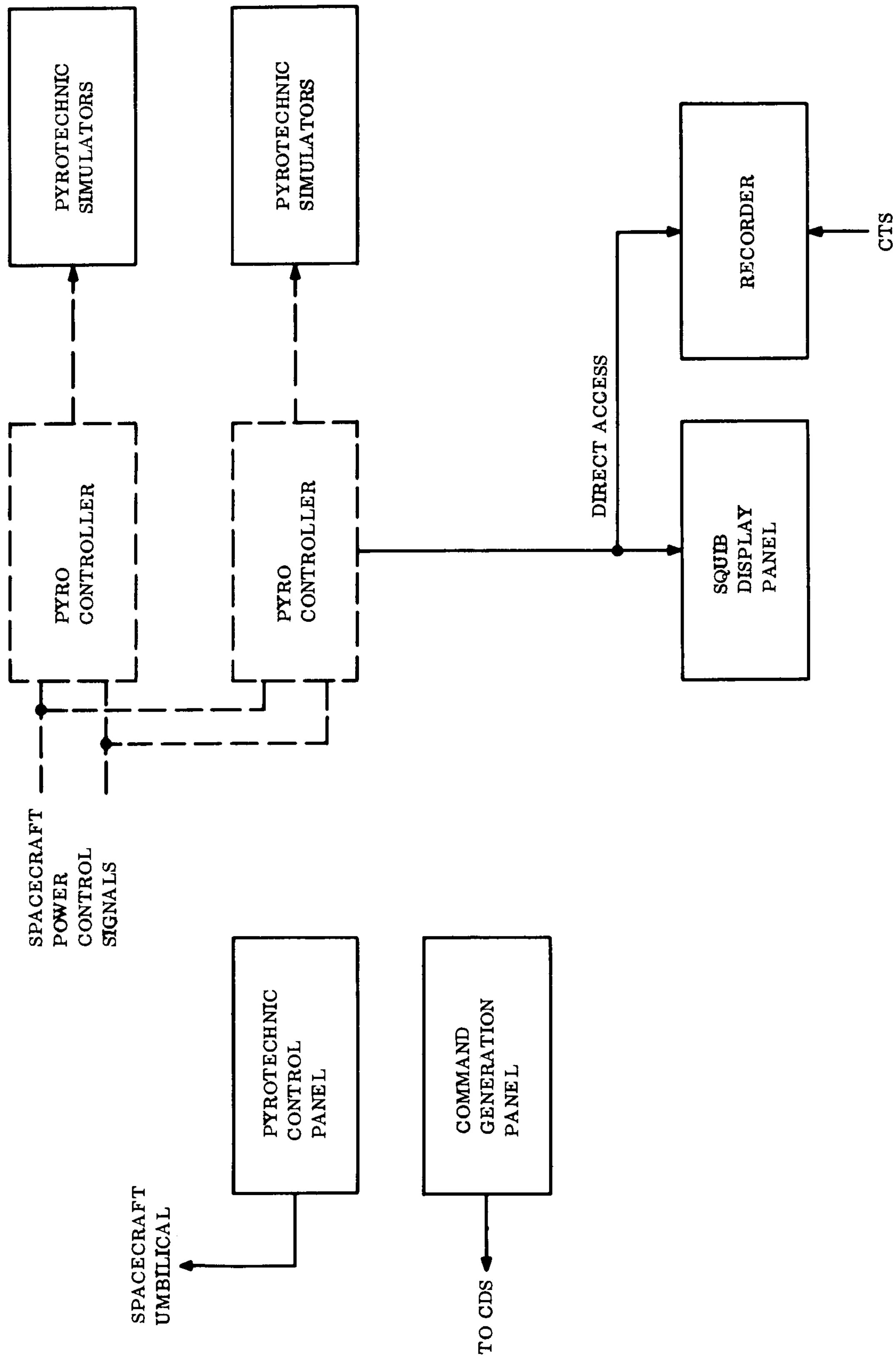


Figure 3-2. Pyrotechnic Subsystem, System Test Configuration, Test Block Diagrams

that the event occurred. An event signal will be sent to the Central Recorder and the Computer Data System from the SQUIB DISPLAY PANEL.

Before using the Pyrotechnic OSE for a system test it should be completely checked using its own self test circuits, as follows:

a. Simulation - A simulation circuit will be provided which will simulate the pyro firing circuits for checking out the pyrotechnic OSE. The simulator circuit will be internal and switched in as desired.

b. Calibration - Calibration circuits will be included to assist the the operator in calibration of the direct writing recorder.

c. Other Features - There will be continuous voltage and current monitors on power supplies and a rack mounted volt/ohnmeter to measure capacitor voltages and squib simulator resistances.

3.2 SUBSYSTEM TESTING

Refer to Figure 3-1, Pyrotechnics Subsystem OSE; and Figure 3-3, Pyrotechnic Subsystem Test Block Diagram.

The Pyrotechnic subsystem can be tested when the rack containing the controllers is mounted in the spacecraft or when the bay containing the controller is separate from the spacecraft. The Pyrotechnic Subsystem test equipment can be used in either test configuration. The Pyrotechnic OSE can also be used in conjunction with other OSE to perform a complete test on the entire bay.

During the subsystem test all inputs to the bay containing the pyro controllers will come from the Pyrotechnic Subsystem OSE. The spacecraft power will be simulated by a 2.4 kc power supply; the firing commands will be generated by the COMMAND GENERATION PANEL; and the mode setting commands will be generated by the PYROTECHNIC CONTROL PANEL. Telemetry event signals from the pyrotechnic controller will be displayed on the TELEMETRY DISPLAY. The Central Recorder may be connected.

SQUIB SIMULATORS will be employed in the same manner as in the system test if the bay is installed in the spacecraft. The option exists to monitor squib current at the simulators or to monitor at the direct access connectors. If the bay is not mounted in the spacecraft, the LOAD SIMULATOR PANEL will be used to provide dummy loads. In either case, the RECORDER measures and records the squib current while the signal event is displayed on the SQUIP DISPLAY PANEL. The Computer Data System need not be employed in the subsystem test.

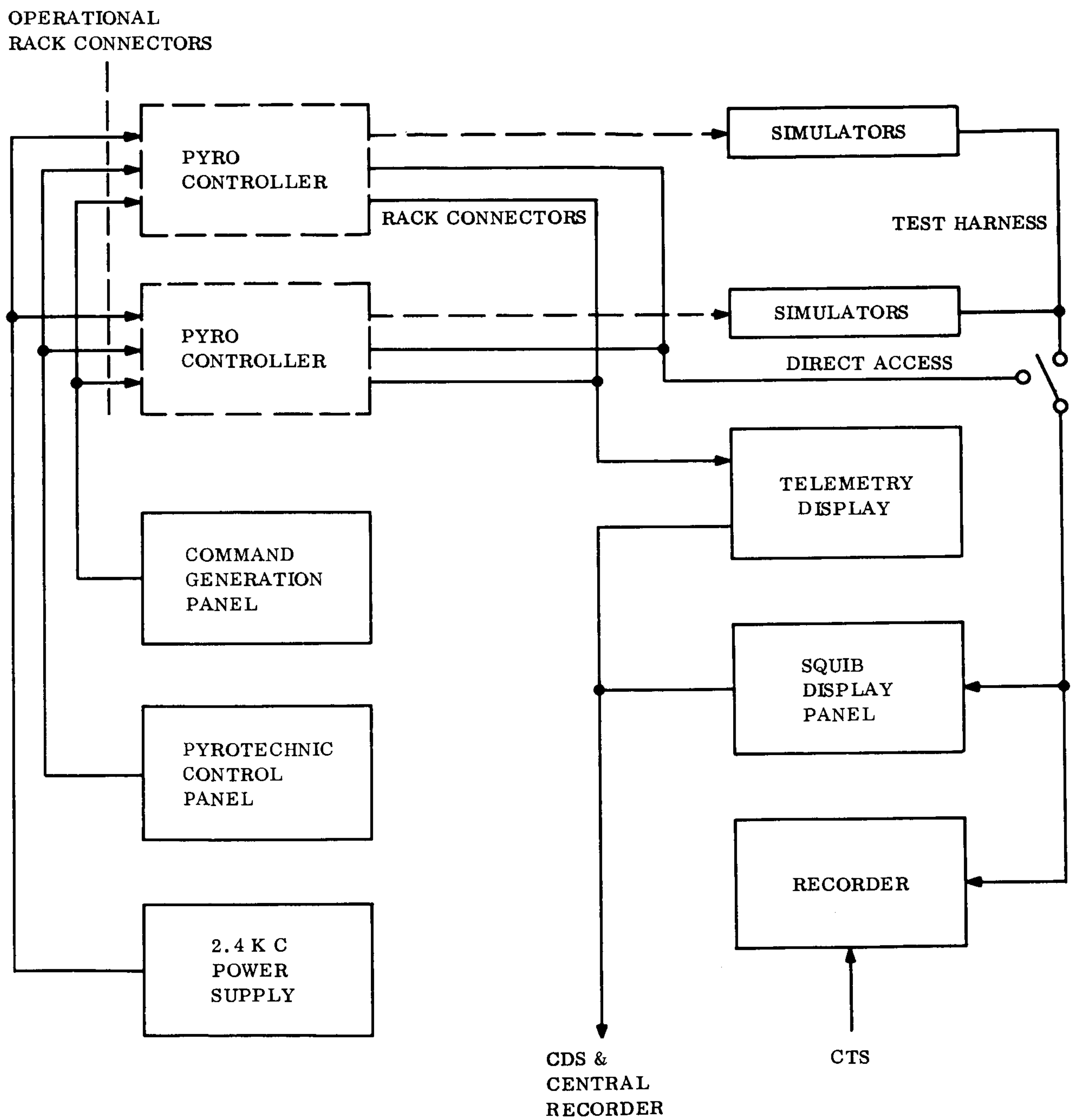


Figure 3-3. Pyrotechnic Subsystem, Subsystem Test Configuration Test Block Diagram

Self test features used during, and associated with, the subsystem test are the same as those used with the system test.

3.3 COMPONENT TESTS

Refer to Figure 3-1, Pyrotechnic Subsystem OSE and Figure 3-4, Pyrotechnic Subsystem Component Test Block Diagram.

The pyrotechnic controllers may be tested separately using the subsystem OSE. Command and control signals are provided by the COMMAND GENERATION PANEL and the PYROTECHNIC CONTROL PANEL. The required power is supplied by the 2.4 KC POWER SUPPLY. The SIMULATOR PANEL will provide the proper loads to the controllers under test. The TELEMETRY DISPLAY PANEL will provide telemetry event monitoring and the RECORDER will make a recording of the firing pulses.

Any internal testing or troubleshooting of the controllers can be accomplished by using facility test equipment in support of the subsystem OSE.

4.0 INTERFACE DEFINITION

4.1 GENERAL

The Pyrotechnic Subsystem OSE for the STC has interfaces with the equipment listed below:

a. Pyrotechnic Subsystem Direct Access and Umbilical Points.

b. Central Recorder

c. Computer Data System

4.2 PYROTECHNIC SUBSYSTEM INTERFACE

Pyrotechnic Subsystem interfaces are as given below.

Interface	Input	Output	Characteristics
Pyro Controller	Simulated Commands		30 commands, 15-volt pulses
	Reset Signals		
	Power		2.4 kc ac, 2 watts
		Telemetry Signals	
		Firing Pulses	100 firing pulses to bridge wires

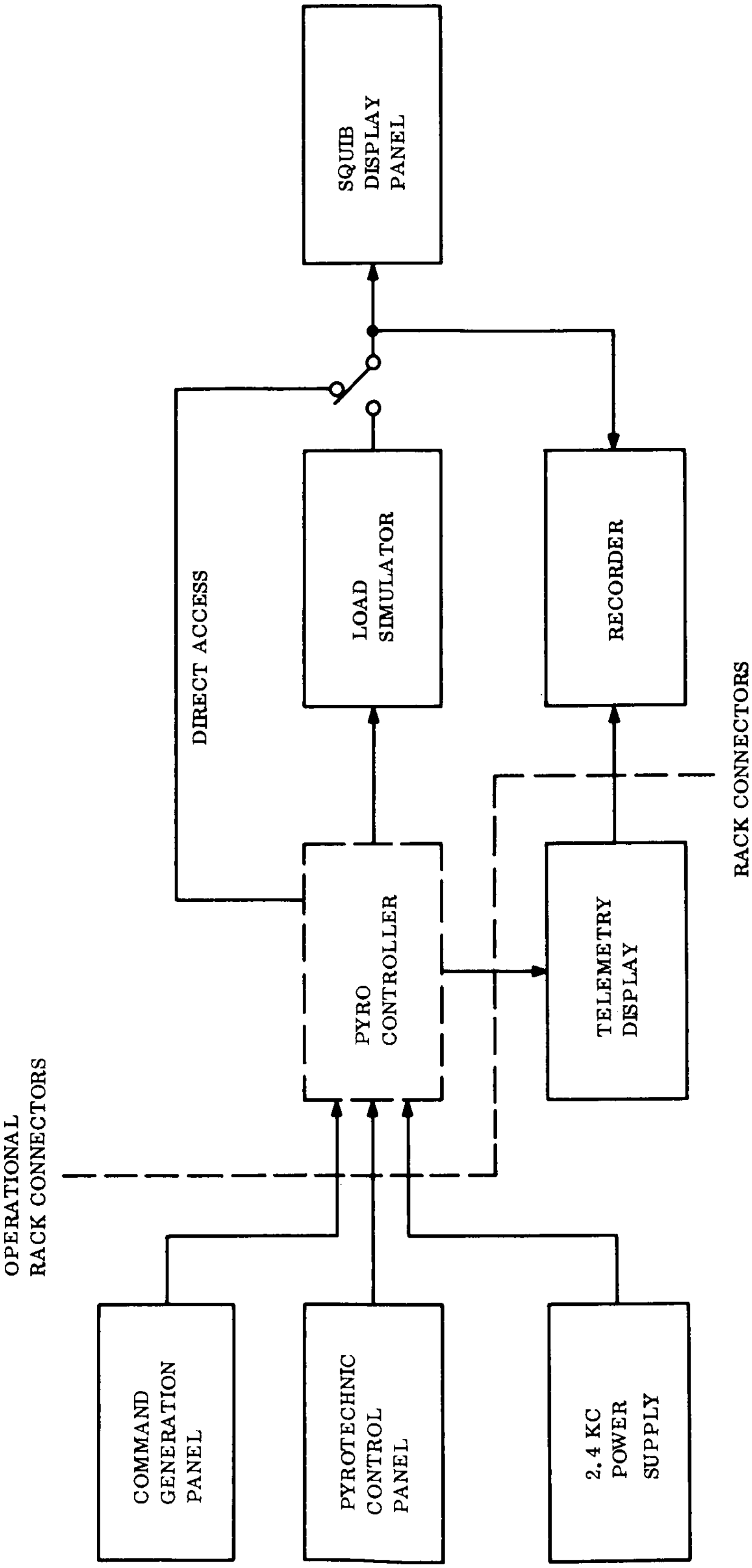


Figure 3-4. Pyrotechnic Subsystem, Controller Test Configuration, Test Block Diagram

4.3 SYSTEM TEST COMPLEX INTERFACE

System Test Complex interfaces are as given below.

Interface	Input	Characteristics
Central Recorder and Computer Data System	Telemetry Events	8 events
	Squib Firing Timing	100 events
	Command Timing	30 events
	Reset Events	2 events
Central Time and Synch	Timing	Code

5.0 PERFORMANCE PARAMETERS

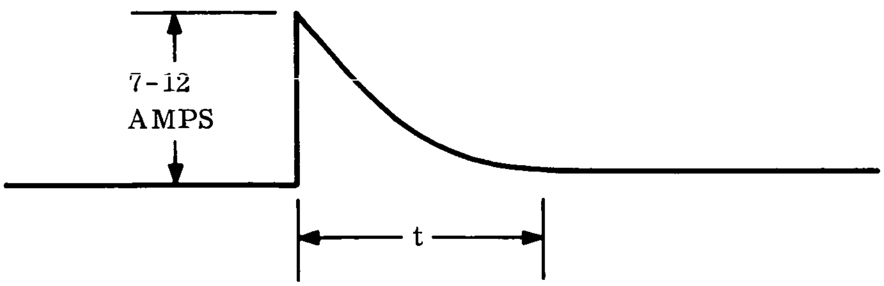
5.1 SPACECRAFT SYSTEM TEST AND SUBSYSTEM TEST

Interface	Output	Characteristics
Test Conduction Console	Test Status	Switch closures

a. SQUIP SIMULATORS must have the same electrical resistance characteristics as the actual squibs.

Range 0.9 ohm to 1.1 ohms.

b. The RECORDER shall provide an accurate record of the current pulse delivered to the squib simulator. The record shall be sufficiently accurate to show whether or not a real squib would have fired.



c. A printer will provide current information from the spacecraft telemetry system to the operator of the pyrotechnic subsystem OSE. This printer should not be physically integrated into this OSE, but should be located in proximity to the OSE.

d. Reset signals shall have sufficient amplitude and duration to provide an unambiguous reset command to the spacecraft pyrotechnic controllers.

e. The simulated vehicle power shall be variable over the full range of allowable vehicle power characteristics.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 UTILIZATION OF MARINER "C" EQUIPMENT

The basic approach to testing is the same as that of Mariner "C" except for those changes made necessary by the increased quantity of explosive devices and differences in controller configuration. Except for squib simulators, the Mariner "C" OSE design approach should be used for Voyager.

6.2 EQUIPMENT SKETCH

See Figure 6-1.

6.3 GENERAL

Power Requirements	20 amp, 120 vac, 60 cycle
Weight	approx. 700 lbs
Size	Two standard equipment bays plus cabling and squib simulator packages

6.4 PYROTECHNIC CONTROL PANEL

A standard display panel with self-contained supporting circuits. This panel will display pyrotechnic safe and arm conditions and control the reset circuits. An identical panel will be provided in the LCE.

6.5 SQUIB DISPLAY PANEL

A standard display panel showing which squib have received firing pulses. The signal lights on this panel will remain on indicating a fired condition until a reset signal is given.

6.6 RECORDER

A direct write galvanometer recorder with its associated amplifiers. A 24-channel instrument such as the Visicoder.

6.7 REMOTE TELEMETRY PRINTER

A digital printer is provided as part of the Data Encoder OSE displays, but is mounted near the Pyrotechnic OSE.

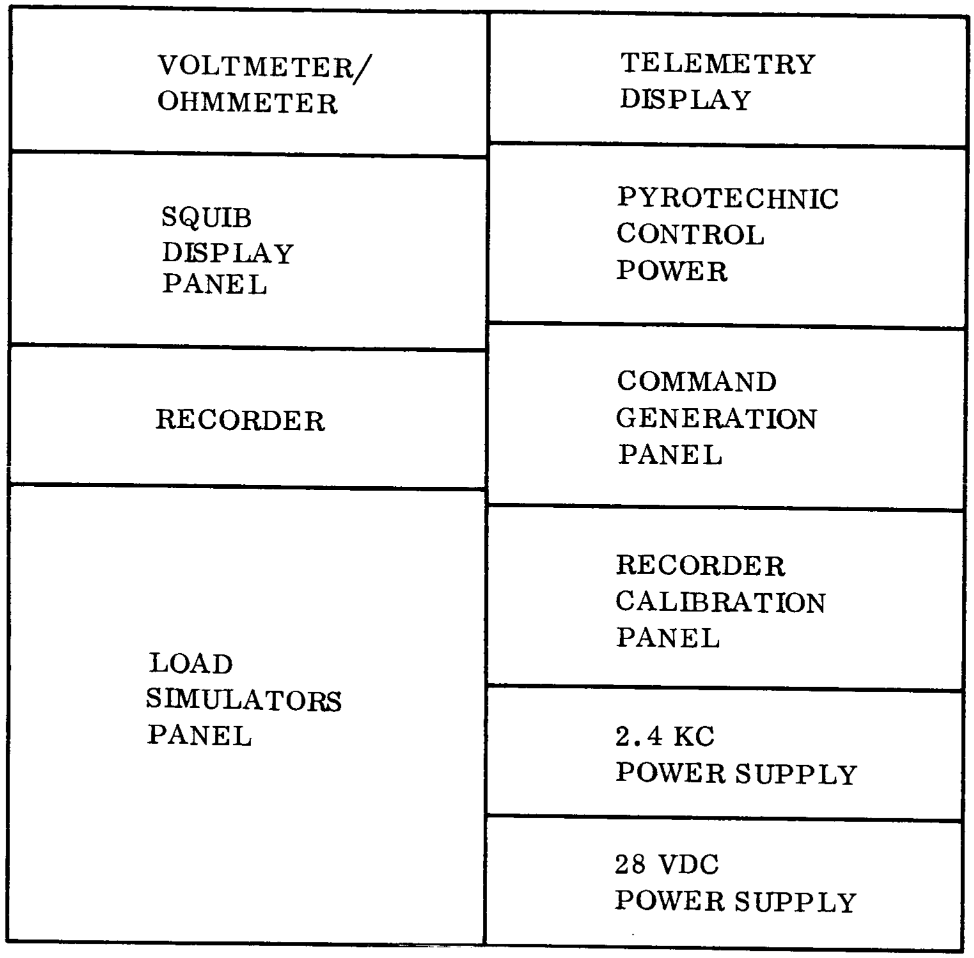


Figure 6-1. Pyrotechnic Subsystem OSE, Equipment Layout Diagram

6.8 SQUIB SIMULATORS

Small devices with a fixed resistance which simulate the resistance of the squib during repetitive tests. The simulators also contain sensing leads for connection to the Squib Display Panel and the Recorder.

6.9 COMMAND GENERATION PANEL

This is a standard control panel which has the capability of generating commands which simulate commands normally provided by the spacecraft command system.

6.10 TELEMETRY DISPLAY

A Telemetry Display panel will provide a display of all of the telemetry events and levels normally sent to the vehicle telemetry system.

6.11 2.4 KC POWER SUPPLY

A 2.4 kc power supply will accurately simulate the power supplied to the controllers from the spacecraft power system. This power supply shall be variable over the range of voltage and frequency specified as acceptable for the vehicle power supply.

6.12 LOAD SIMULATOR

This panel will contain fixed resistors which will be used to simulate squib and harness resistances during component tests.

6.13 28 VDC POWER SUPPLY

A cabinet power supply will be provided to give internal power to the Pyrotechnic Subsystem OSE.

6.14 CALIBRATION PANEL

An assembly will be provided which will include a calibration circuit and a selector switch. This assembly will be used to calibrate the recorder before and during tests.

7.0 SAFETY CONSIDERATIONS

7.1 VEHICLE SAFETY

A preliminary test of the OSE prior to use with the vehicle should be made to verify the signal levels, power supply voltages, and load resistances.

7.2 PERSONNEL

There are no unusual hazards associated with this equipment. During use the external cabinet must be grounded to a central bus to avoid a shock hazard between cabinets.

CII - VB262FD101

SYSTEM TEST COMPLEX

PROVISIONS FOR THE SCIENCE AND CAPSULE OSE

Index

1.0 SCOPE

This document describes the provisions recommended for the Science (and DAS) and Capsule OSE in the Voyager STC. It contains an interface definition and a listing of the physical characteristics of the equipment which will be used in the STC.

2.0 APPLICABLE DOCUMENTS

VB260SR101	1971 Voyager STC Test Objectives and Design Criteria
VB260SR102	1971 Voyager STC Design Characteristics and Constraints

3.0 DESCRIPTION

The OSE must conform to the constraints defined in the Design Characteristics and Constraints document, VB260SR102.

The capsule OSE must provide a simulator which can be used during system testing when the regular capsule is not available. It should simulate to the S/C and use the same interfaces as the flyable unit.

4.0 INTERFACE DEFINITION

a. The Capsule OSE will interface with the following units when in the STC.

1. Central Recorder - For recording analog signals

2. Computer Data System

   (a) Request S/C commands for capsule tests

   (b) Analog and Digital Data - For computer analyses

   (c) Computer Control - For remotely controlling the console during repetitive system test, if remote control is required

3. Data Encoder OSE - Digital TM Data

4. Umbilical "J" Box

5. Direct Access connectors

6. S/C Subsystem power connector

7. Test Conductors Console - Status Indicators

8. Power Distribution Console - 60-cycle power

9. S/C capsule interface - For capsule simulator

b. The Science OSE will have the following interface when in the STC:

1. Central Recorder

2. Computer Data System

   (a) Request S/C commands for Science Tests

   (b) Analog and Digital Data - For computer analyses

   (c) Computer Control - For remotely controlling the console during repetitive system tests, if remote control is required.

3. Data Encoder - Digital TM data

4. Umbilical "J" Box

5. Direct Access

6. S/C Subsystem power connector

7. S/S Status - For test conductors use

8. Power Distribution Console - 60-cycle power

5.0 PERFORMANCE PARAMETERS

To be determined.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 CAPSULE OSE

a. Standard electronic bays, adjacently located

b. 2.4 KVA, 120 VAC, 60-cycle, single-phase power

c. Weight 1600 lbs.

6.2 SCIENCE OSE

a. Standard electronic bays, adjacently located

b. 8.0 KVA, 120 VAC, 60-cycle, single-phase power

c. Weight 4000 lbs.

7.0 SAFETY PROVISIONS

To be determined.

CII - VB280SR101

LAUNCH COMPLEX EQUIPMENT

TEST OBJECTIVES AND DESIGN CRITERIA

Index

1.0 SCOPE

This document defines the test objectives and design criteria which should be applied to the OSE required to support the launch preparations and terminal countdown. This equipment is the Launch Complex Equipment.

This document also defines the test objectives and design criteria which should be applied to subsystem and system support equipment to be utilized in conjunction with prelaunch preparations and testing in the explosives safe area. This includes support of propellant loading, gas loading and installation of explosives.

2.0 APPLICABLE DOCUMENTS

The Launch Complex Equipment should conform to the following documents:

JPL Specifications

V-MA-004-002-14-03	Voyager 1971 Mission Guidelines
V-MA-004-001-14-03	Voyager 1971 Mission Specifications

3.0 TEST OBJECTIVES

3.1 LAUNCH COMPLEX SUPPORT OBJECTIVES

The Launch Complex Equipment must support all pad operations related to the Voyager spacecraft including normal test and launch operations and unscheduled operations made necessary by malfunctions in the spacecraft.

3.1.1 SPACECRAFT POWER

a. Provide power for the spacecraft by simulated solar and battery power.

b. Monitor spacecraft batteries.

c. Switch the spacecraft from externally supplied power to its internal power subsystem.

3.1.2 TELECOMMUNICATIONS

a. Provide means to initialize command and data systems and preset the spacecraft prior to receiving a command load.

b. Provide the capability of loading digital commands for both storage and immediate execution in the spacecraft. The STC shall be used for command generation and program loading via RF to the pad area. The LCE has to provide the capability of insertion into the spacecraft via RF from a link at the pad into a coaxial umbilical lead. The received and loaded commands and programs are made available to the STC for verification by the LCE.

3.1.3 CONFIDENCE TESTS

a. Operate the spacecraft as a system and monitor its operation through the umbilical connector.

b. Permit use of the system test complex which will control spacecraft tests through an RF link inserting commands generated at the STC and deriving the spacecraft response from the telemetry reduction equipment in the STC. The confidence test will be sufficiently detailed to indicate which subsystem if any has a malfunction.

3.1.4 FAULT ISOLATION TESTS

The intent is to utilize the LCE supported by the STC to troubleshoot in as much detail as possible before removing a spacecraft from the pad for maintenance. The information received by the LCE comes only from the spacecraft umbilical connector. There is no provision for direct access connectors or for decommutation of the telemetry composite signal in the LCE. Any meaningful malfunction analysis performed while the spacecraft is on the pad must depend heavily on the support of the data reduction equipment in the STC.

3.1.5 RF LINK

a. Provide for an RF link between the spacecraft on the pad and the System Test Complex.

b. Provide for an RF link between the spacecraft on the pad and the local DSIF station. The intent is to provide capability for exchange of information between the launch ready spacecraft and the SFOF, verifying spacecraft compatibility with both DSIF and SFOF.

3.2 EXPLOSIVES SAFE AREA SUPPORT OBJECTIVES

The LCE must support all Explosives Safe Area operations related to the Voyager spacecraft. This includes support of both scheduled operations and support of operations made necessary by equipment malfunctions.

3.2.1 GENERAL CAPABILITY

a. All test objectives listed in paragraph 3.1 for the Launch Complex are also objectives for the ESA.

b. The objectives must be met with the spacecraft covered by the shroud or uncovered, inside the buildings or outside.

3.2.2 EXPLOSIVES LOADING

a. Check the pyrotechnic subsystem for safety and set it to the correct mode before loading explosives.

b. Preset any arming devices.

c. Monitor the condition of the pyrotechnics and the pyrotechnic subsystem after explosives are loaded.

3.2.3 PNEUMATICS TESTING

a. Provide for leak testing and proof testing of all subsystems subject to high pressures.

b. Pressurize all pneumatics and propellant systems.

3.2.4 COLD GAS LOADING

a. Complete the loading of all cold gases in the proper quantities at the specified temperatures and pressures and verify the loading operation when it is completed.

b. Prepare the subsystems for final loading with any purging, cleaning or other operations required.

3.2.5 PROPELLANT LOADING

a. Load all liquid propellants, accurately measuring the weight of the propellant loaded and verify that the loading operation has been completed in a satisfactory manner.

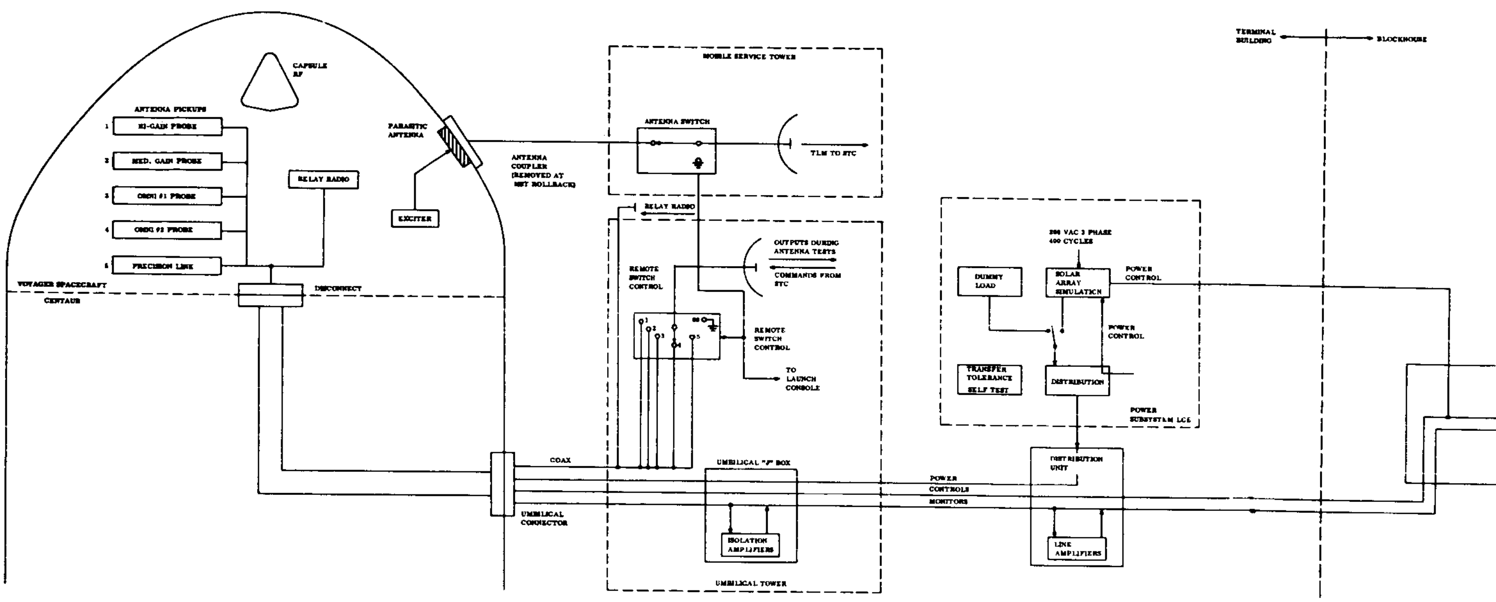
b. Prepare the subsystem for propellant loading with any purging or cleaning operations required.

4.0 DESIGN CRITERIA

4.1 LAUNCH COMPLEX

4.1.1 FUNCTIONAL CONFIGURATION

The LCE used at the launch complex must be designed so that its design and functions implement functional block diagram, Figure 4-1.


CAPSULE RF
ANTENNA PICKUPS
1 HI-GAIN PROBE
2 MED. GAIN PROBE
3 OMNI #1 PROBE
4 OMNI #2 PROBE
5 PRECISION LINK
RELAY RADIO
PARASITIC ANTENNA
EXCITER
ANTENNA COUPLER (REMOVED AT MST ROLLBACK)
VOYAGER SPACECRAFT
CENTAUR
DISCONNECT
MOBILE SERVICE TOWER
ANTENNA SWITCH
TLM TO STC
RELAY RADIO
OUTPUTS DURING ANTENNA TESTS
COMMANDS FROM STC
REMOTE SWITCH CONTROL
REMOTE SWITCH CONTROL
TO LAUNCH CONSOLE
COAX
UMBILICAL "J" BOX
ISOLATION AMPLIFIERS
UMBILICAL CONNECTOR
UMBILICAL TOWER
POWER
CONTROLS
MONITORS
DISTRIBUTION UNIT
LINE AMPLIFIERS
400 CYCLES
DUMMY LOAD
SOLAR ARRAY SIMULATION
POWER CONTROL
POWER CONTROL
TRANSFER TOLERANCE SELF TEST
DISTRIBUTION
POWER SUBSYSTEM LCE
TERMINAL BUILDING
BLOCKHOUSE

5

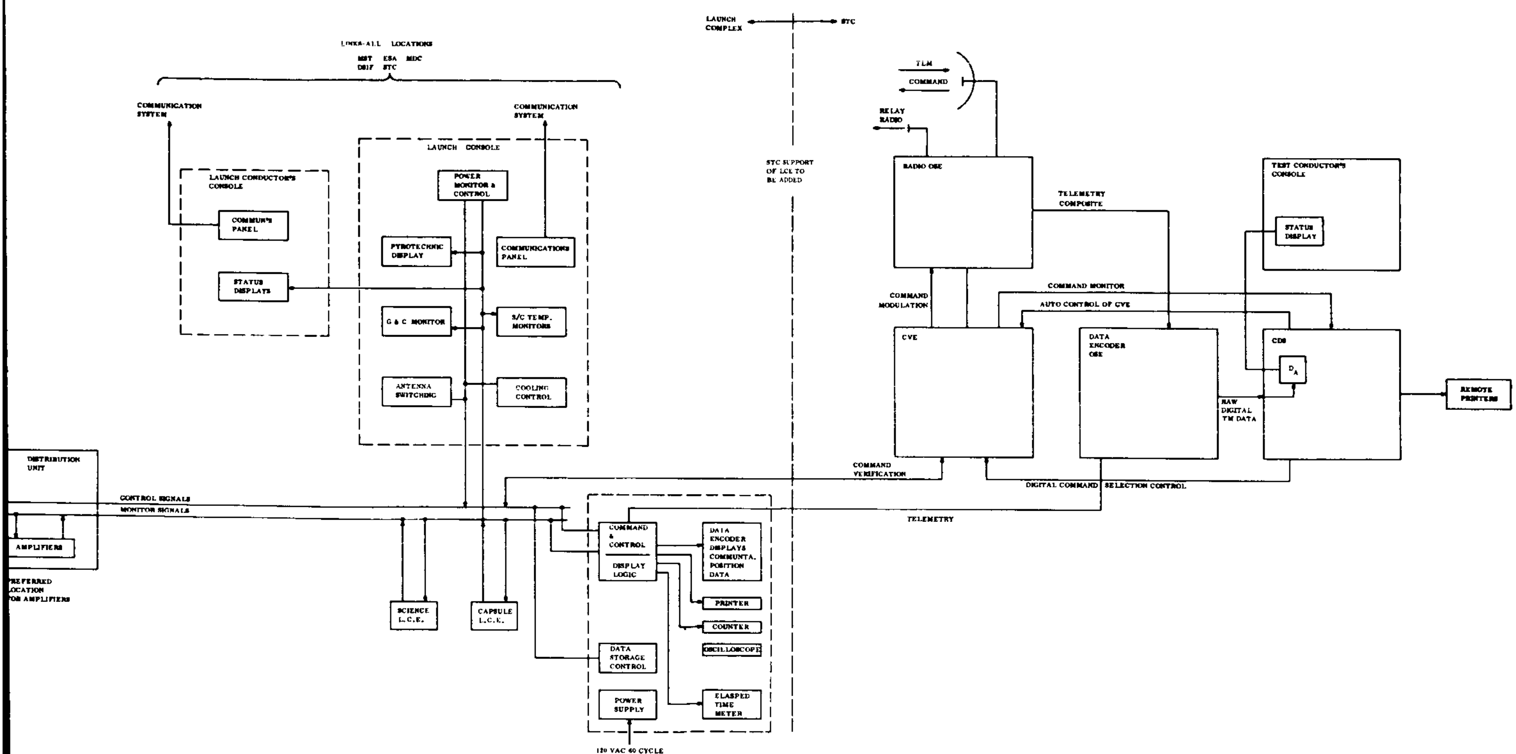


Figure 4-1. Launch Complex Equipment, Block Diagram

4.1.2 BLOCKHOUSE EQUIPMENT

The equipment provided for indoor use at the launch complex must be configured to be physically compatible with the space available at the launch pad. The indoor environment will be controlled and comfortable for human beings.

4.1.3 OUTDOOR EQUIPMENT

The equipment provided for outdoor use must withstand a coastal environment with salt atmosphere and elevated temperatures.

4.1.4 ELECTRIC POWER

The launch complex equipment must provide power to the spacecraft in a form compatible with the spacecraft power system. The equipment will operate off the available power in the blockhouse.

4.1.5 SELF TESTING

The LCE must be completely checked out from the umbilical interface before the umbilical is connected to a spacecraft. A separate article of equipment will be provided to substitute for the spacecraft during LCE tests. After a spacecraft is connected to the LCE, self checking features will be used to verify that the LCE is operating properly and providing accurate monitoring of spacecraft condition.

4.1.6 COMMUNICATIONS

The LCE must provide for adequate communications between the Voyager launch crew and launch conductor inside the launch complex and further must provide adequate communications between the launch conductor in the blockhouse and the mission director at SFOF.

4.1.7 RELIABILITY

There must be a combination of reliability and maintainability and that the possibility of missing a launch window due to LCE failures will be extremely small.

4.1.8 STC SUPPORT

The RF and hardwire links between the STC and the spacecraft shall be such that the STC can be used to perform analysis of subsystem performance and analysis of critical parameters.

4.1.9 CABLE RUNS

The equipment provided must be suitable for use with long cable runs. Line drivers and impedance matching devices should be in the blockhouse wherever possible. Cable runs and patching will be provided by ETR.

4.1.10 SEPARATION OF FUNCTIONS

Power control and spacecraft power switching equipment will be separately racked. Wherever feasible, subsystem LCE will be separated on a panel basis such that no two subsystems will be combined on one panel. If the quantity of LCE required for a particular subsystem fills one or more racks that subsystem should be racked separately.

4.1.11 S/C SIMULATOR

A spacecraft simulator must be a physically separate piece of LCE. It must be transportable to the Service Tower location where it can be connected directly to the S/C umbilical connector.

4.1.12 TELECOMMUNICATIONS INTERFACE

The interfaces between the spacecraft telecommunications subsystems and the LCE must be functionally identical with the interface at the STC and the interface with the DSIF. The same type of command generation and telemetry reduction equipment must be used but the physical configuration of the RF Link will differ at the pad and the STC.

4.1.13 RF COMMUNICATIONS

The spacecraft on the pad will be totally enclosed in an RF opaque shroud. There will be a parasitic antenna on the shroud and several RF carrying coaxial cables coming out of the umbilical connector. The LCE must provide passive means for linking the parasitic shroud antenna to the STC and the local DSIF station. The LCE must also provide means for linking RF umbilical signals to the STC and the DSIF.

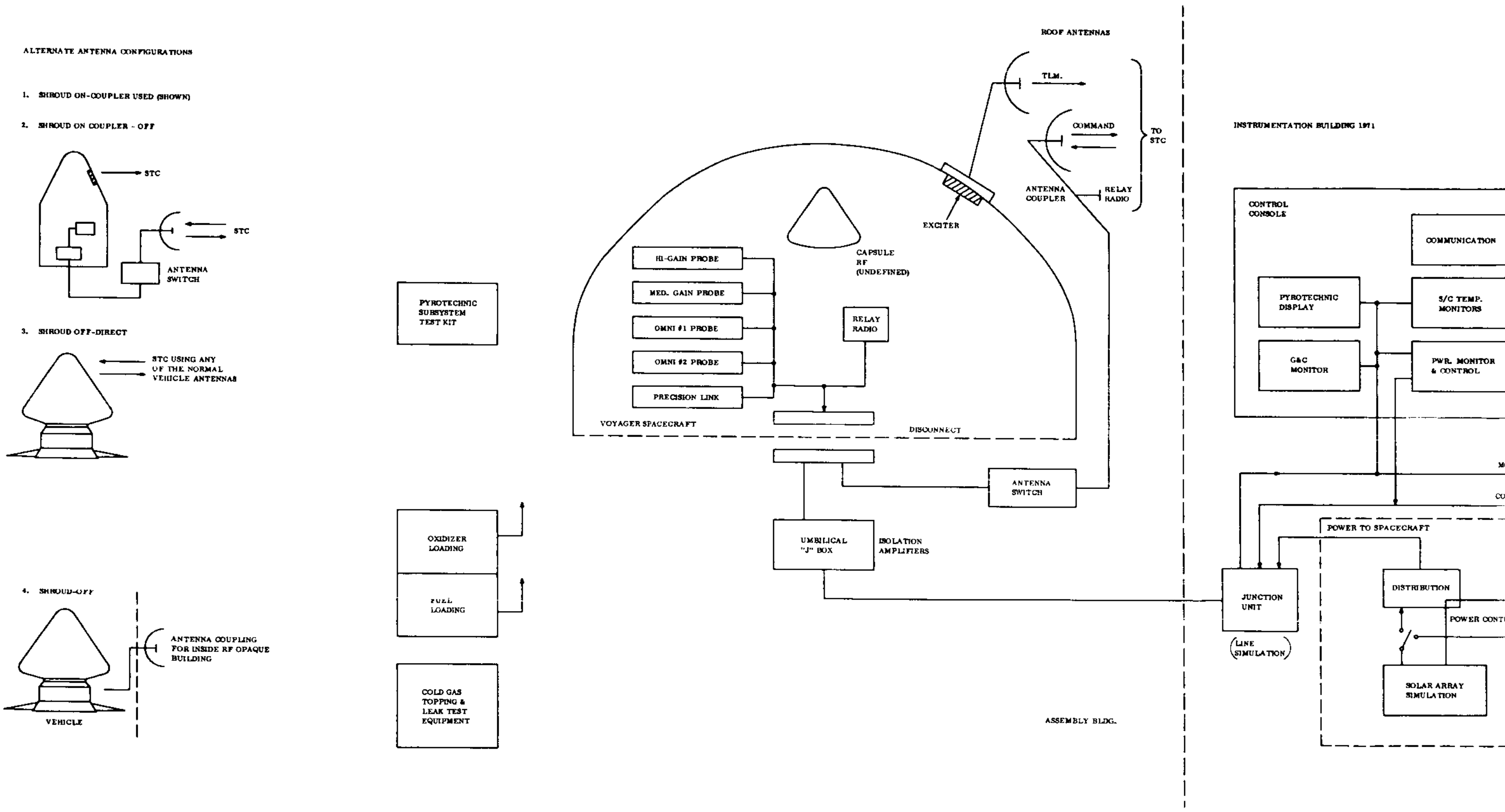
4.2 EXPLOSIVES SAFE AREA

4.2.1 FUNCTIONAL CONFIGURATION

The ESA equipment must be designed so that the design and functions of the ESA equipment implement functional block diagram, Figure 4-2.

4.2.2 CAPABILITY

All of the functions of the LC configuration must be available at the Explosives Safe Area.



ALTERNATE ANTENNA CONFIGURATIONS
1. SHROUD ON-COUPLER USED (SHOWN)
2. SHROUD ON COUPLER - OFF
STC
STC
ANTENNA SWITCH
3. SHROUD OFF-DIRECT
STC USING ANY OF THE NORMAL VEHICLE ANTENNAS
4. SHROUD-OFF
ANTENNA COUPLING FOR INSIDE RF OPAQUE BUILDING
VEHICLE
PYROTECHNIC SUBSYSTEM TEST KIT
OXIDIZER LOADING
FUEL LOADING
COLD GAS TOPPING & LEAK TEST EQUIPMENT
ROOF ANTENNAS
TLM.
COMMAND
TO STC
ANTENNA COUPLER
RELAY RADIO
EXCITER
CAPSULE RF (UNDEFINED)
HI-GAIN PROBE
MED. GAIN PROBE
OMNI #1 PROBE
OMNI #2 PROBE
PRECISION LINK
RELAY RADIO
VOYAGER SPACECRAFT
DISCONNECT
ANTENNA SWITCH
UMBILICAL "J" BOX
ISOLATION AMPLIFIERS
ASSEMBLY BLDG.
INSTRUMENTATION BUILDING 1971
CONTROL CONSOLE
COMMUNICATION
PYROTECHNIC DISPLAY
S/C TEMP. MONITORS
G&C MONITOR
PWR. MONITOR & CONTROL
POWER TO SPACECRAFT
JUNCTION UNIT
(LINE SIMULATION)
DISTRIBUTION
SOLAR ARRAY SIMULATION

9

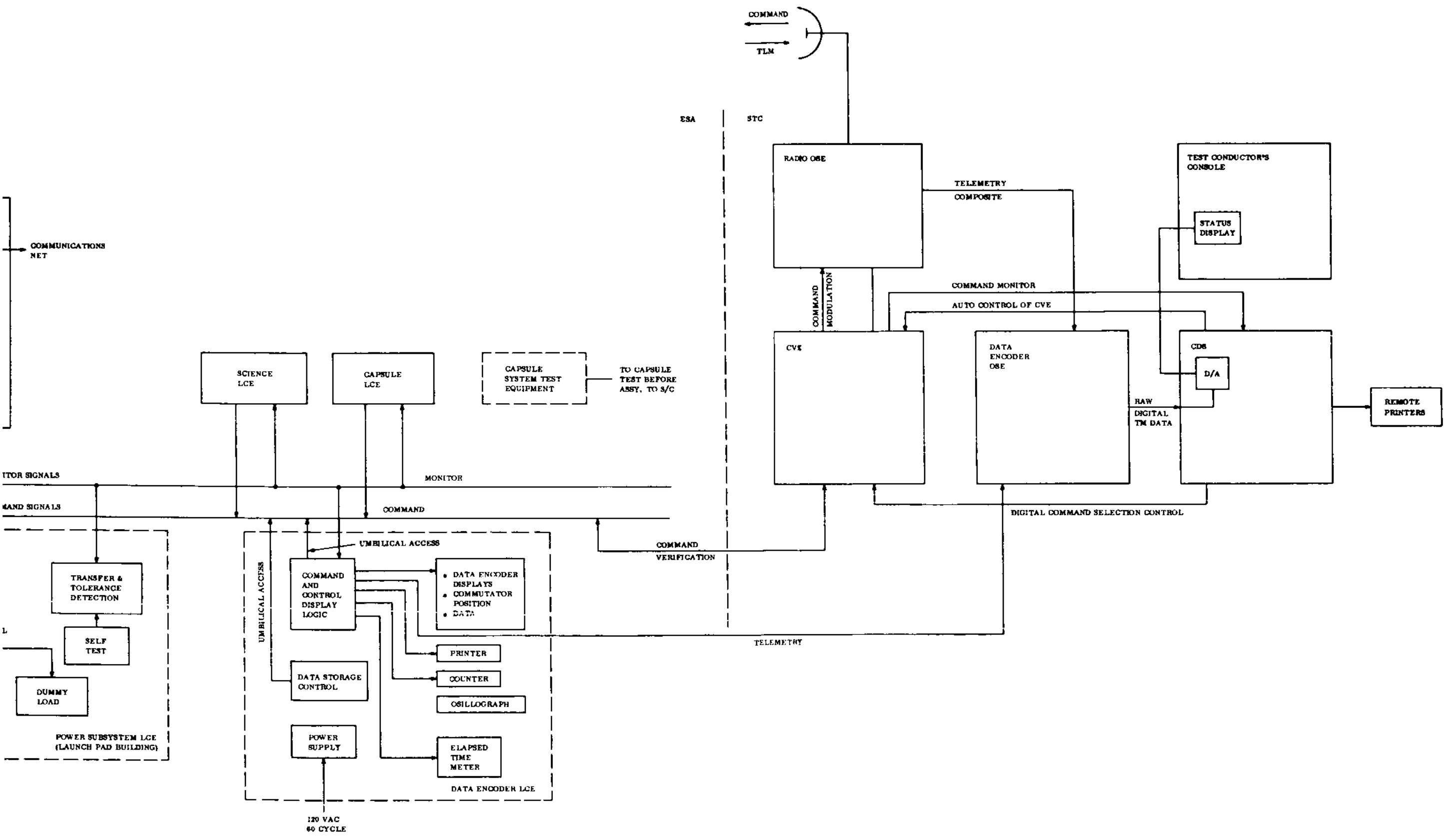


Figure 4-2. Explosive Safe Area, Block Diagram

4.2.3 INSTRUMENT LAB EQUIPMENT

The equipment provided at the ESA for indoor use must be physically compatible with the available space. The equipment will be physically identical to the pad LCE except for changes made necessary by the space limitations in the instrument lab.

4.2.4 SPACECRAFT SHROUD

The ESA equipment must provide for operation and test of the vehicle with the shroud on, covering the vehicle, or the shroud off and the vehicle exposed. This must be true inside the ESA buildings and outside.

4.2.5 PROPELLANT LOADING EQUIPMENT

Propellants for orbit injection and midcourse correction will be loaded in the ESA. The capability of performing the necessary operations in a safe manner is extremely important. In addition to loading the propellants with a minimum safety hazard the equipment provided must also perform any necessary preparations or testing on the propellant subsystem.

4.2.6 GAS LOADING EQUIPMENT

High pressure gases for attitude control propellants and for pressurization of the rocket propellants will be loaded in the ESA. Safety will be an important consideration. In addition to loading the gases with a minimum safety hazard the equipment provided must also perform any necessary preparations or testing on the subsystems using high pressure gases. This includes leak checks.

4.2.7 PYROTECHNIC SUBSYSTEM EQUIPMENT

Explosive devices will be loaded at the ESA. Personnel safety and spacecraft protection will be extremely important in all operations associated with loading pyrotechnics and in spacecraft handling after pyrotechnics are loaded. The equipment provided must have capability to condition the pyro firing circuits before loading, verify the safety of the loading operation and check subsystem condition after loading.

4.2.8 AHSE COMPATIBILITY

The ESA equipment must be compatible in function and fit with the AHSE equipment used with and associated with the operations at the ESA.

CII - VB280SR102

LAUNCH COMPLEX EQUIPMENT

DESIGN CHARACTERISTICS AND RESTRAINTS

Index

1.0 SCOPE

It is the intent of this document to define the requirements which will be applied to the Operational Support Equipment to be used at the Launch Complex and at the Explosives Safe Area (ESA).

2.0 APPLICABLE DOCUMENTS

2.1 The Launch Complex Equipment shall conform to the following specifications.

JPL Specifications

V-MA-004-002-14-03 Voyager 1971 Mission Guidelines
V-MA-004-001-14-03 Voyager 1971 Mission Specification (Preliminary)

14902	Wiring, Ground and Missile Electronic Equipment
20027	Securing of Components
20057	General Specification, Ground Support Equipment, General Design Requirements for Electrical Interconnections
30236	Environmental Test Specification, RF Interference Control for S/C and Ground Control
30600	Standard Specification, Paint for Ground Equipment
30602	Standard Specification, Engraving on Ground Equipment Control Panel Faces
30603	Standard Specification, Knob Standard for Ground Equipment (Single Turn Knobs)
30604	Standard Specification, Meters for Ground Equipment Control Panels
30605	Standard Specification, Color Indicator Light and Switch Light
30609	Standard Specification, Racks for Ground Equipment
30610	Standard Specification, GSDS Equipment, Enclosures (Racks)

GE

VB 280SR101 1971 Voyager Spacecraft LCE Test Objectives and Design Criteria

3.0 DESIGN CHARACTERISTICS

3.1 GENERAL DESIGN CHARACTERISTICS

The design objective of the Launch Complex Equipment is to produce a set of homogenous equipment for use at the launch pad and a set of equipment to support operations at the ESA. The equipment must be compatible with the spacecraft, compatible with the operating environment and have an efficient interface with the operators. The subassemblies and assemblies making up the LCE must function together without mutual interference.

The LCE at the Launch Complex and at the ESA must each be able to perform a confidence test on the spacecraft. A confidence test consists of excercising the spacecraft system in order to demonstrate that it is a functioning unit capable of receiving and transmitting information and that there are no major malfunctions in the system. The LCE must be compatible with the STC to the extent that the STC will support all confidence tests.

The LCE circuits shall be electrically identical to the circuits used in the STC. Whereever possible the LCE assemblies shall be identical to the assemblies used in the STC.

3.2 LAUNCH AREA EQUIPEMENT CHARACTERISTICS

The Launch Area equipment must be capable of performing or supporting the performance of specific tasks at the launch pad. These tasks are as follows:

a. Pad Verification
b. Assembly to Booster
c. Confidence Testing
d. Initializing
e. Data Loading
f. Power Control
g. Countdown Monitoring against abort criteria

Before a spacecraft is installed on the booster the Spacecraft Simulator will be used to check the operation of the LCE. The verification test must show that the LCE is functioning properly and there is no possibility of damage to the spacecraft.

All LCE and umbilical functions shall be controlled from equipment located in the blockhouse. Appropriate displays will be provided in the blockhouse equipment to support the control functions.

A console will be provided for central control of the launch system in the blockhouse. This console will house any subsystems control panels used for subsystems not requiring an entire LCE console. The console will include, but not be limited to, a

Pyrotechnic Subsystem Panel, a Controller and Sequencer Panel, a Power Control Panel, a Temperature Limits Panel, an Attitude Control Display Panel and communications equipment.

The LCE will be used as the principle monitor equipment during all phases of the terminal countdown. Panels will be so configured as to minimize the possibility of operator error at this critical time.

3.3 ESA EQUIPMENT CHARACTERISTICS

The ESA equipment must be capable of performing particular tasks some of which are unique to the ESA. These tasks are as follows:

a. Leak Testing
b. Fuel and Oxidizer Loading
c. High Pressure Gas Loading
d. Pyrotechnic Loading
e. Shroud Installation
f. Confidence Testing

The equipment provided for loading fuels and oxidizers will function independently of the other LCE. This equipment will be capable of loading an inactive vehicle and activating any sensing capability needed to monitor S/C tanks during loading. A separate set of loading equipment will be provided for each type of propellant to be loaded. The equipment will also provide for any required preparation of the propellant system including pressurization and proof testing of the pressurant tanks. Accuracy of loading will be sufficient that the spacecraft does not need to be weighed after loading.

The OSE will make use of a facility gas supply to load all cold gas tanks used to supply reaction fuel to the attitude control nozzles. The equipment will be capable of functioning independently of other LCE. It will measure the temperature and pressure of the loaded gas for an accurate determination of weight. This equipment will also perform any required leak checks or proof testing required in the ESA.

The Pyrotechnic Portable Test unit will be capable of preparing the vehicle for safe loading of explosive devices. The pyrotechnic devices will be delivered to the ESA and checked. The portable test unit will preset the pyrotechnic controllers to the proper mode, ascertain that it is safe to connect the squibs to the pyrotechnic controllers and safely check circuit continuity after squibs are connected. The unit will operate independently of all other OSE, will be self powered, portable and contain any necessary operating cables.

The LCE provided at the ESA must have the capability of performing confidence tests at various stages of ESA activities. The equipment provided will allow confidence tests to be performed with or without the shroud, provide for vehicle communications with the STC through RF link and hardwire and support any unscheduled tests which must be performed.

3.4 UMBILICAL INTERFACE

The major electrical interface with the spacecraft is the umbilical connector. The spacecraft umbilical is an in flight disconnect between the Voyager and the Centuar. Centuar cables connect this disconnect to the fly away connector in the shroud. Cables connecting the Centuar umbilical to the Voyager LCE are required. A listing of the functions appearing in the umbilical connector is in Table 3-1.

Table 3-1. Umbilical Functions

RADIO SUBSYSTEM	
1. (Coax) High gain antenna probe.	3. (Coax) Primary low gain antenna probe.
2. (Coax) Medium gain antenna probe.	4. (Coax) Secondary low gain antenna probe.
RELAY RADIO SUBSYSTEM	
1. (Coax) Antenna Probe.	
COMMAND SUBSYSTEM	
None	
POWER SUBSYSTEM	
1. Array/Battery Bus Voltage	11. Battery No. 1 Temperature
2. Array Enable Switch SW-1 Monitor	12. Battery No. 2 Temperature
3. Turn-on Enable Switch, SW-1.	13. Battery No. 3 Temperature
4. Turn-off Enable Switch, SW-1.	14. Battery Sensor Temperature Return
5. Enable Switch, SW-1, return.	15. 400 cps, phase 1 voltage.
6. External power, 44-55 VDC, 15 amps.	16. 400 cps. phase 2 voltage.
7. External power return.	17. 400 cps, phase 3 voltage.
8. Battery No. 1 Voltage.	18. 2.4 KC voltage.
9. Battery No. 2 Voltage.	19. 2.4 KC voltage.
10. Battery No. 3 Voltage.	20. Raw Battery Bus Voltage.

Table 3-1. Umbilical Functions (Continued)

CONTROLLER AND SEQUENCER	
1. Alert Signal.	6. Inhibit Master Timer.
2. Sync Signal.	7. Update Master Timer.
3. Command Information.	8. Clean Timers.
4. Engine burn enable state.	9. Inhibit Power Supply No. 1.
5. Speed-Up Timing (128:1)	10. Inhibit Power Supply No. 2.
DATA ENCODER	
1. Modulated Subcarrier A.	6. Frame Sync.
2. Modulated Subcarrier B.	7. Mode II Command.
3. Modulated Subcarrier C.	8. Data Encoder Return.
4. Word Sync.	9. Recorders to launch mode.
5. Bit Sync.	10. 2 f_s
GUIDANCE AND CONTROL	
1. Gyro Heater Power.	5. Common Return.
2. Gyro Rebalance Amplifier 1.	6. Shield Return.
3. Gyro Rebalance Amplifier 2.	7. Approach Guidance Update 1
4. Gyro Rebalance Amplifier 3.	8. Approach Guidance Update 2
PYROTECHNIC SUBSYSTEM	
1. Continuity.	3. Arm Signal 1.
2. Continuity Return.	4. Arm Signal 2.
DATA STORAGE SUBSYSTEM	
None	
SCIENCE	
Undetermined. Twelve umbilical lines are assumed for this subsystem.	

Table 3-1. Umbilical Functions (Continued)

CAPSULE	
1. Ground Power Main Bus.	16. Sequence Enable
2. Ground Power Main Bus Monitor.	17. Sequence Rate Select.
3. Internal/External Power Transfer.	18. Ground Telemetry Clocks.
4. Internal/External Power Monitor.	19. External/Internal Telemetry.
5. Barrier Pressure Monitor.	20. Telemetry On/Off.
6. Checkout Sequence Signal.	21. Tape recorder On/Off.
7. Spacecraft/Capsule External Clock Select.	22. Tape recorder Record/Playback Select.
8. Telemetry Mode 1 Select.	23. Telemetry Bit Rate Output.
9. Telemetry Mode 2 Select.	24. Telemetry Frame Rate Output.
10. Telemetry Mode 3 Select.	25. Telemetry Encoder Output.
11. Telemetry Mode 4 Select.	26. Sensor Power Supply Monitor.
12. Telemetry Mode 5 Select.	27. Pyrotechnic Continuity.
13. UHF Transmitter On/Off.	28. Telemetry Mode Monitor.
14. UHF Power Amplifier On/Off.	29. Tape Recorder Reset.
15. VHF Transmitter On/Off.	

4.0 DESIGN RESTRAINTS

4.1 PERFORMANCE

4.1.1 RELIABILITY

The reliability of the LCE must be such that the mean time before failure must be greater than 200 times the maximum daily launch window.

4.1.2 RESTORATION TIME

In the event of failure the total time taken to restore the LCE to an operational condition, including diagnosis of trouble and replacement of defective subassembly must be no greater than 1/4 of the minimum daily launch window for subassemblies with a MTF of less than 2500 hours.

4.1.3 SIGNAL ATTENUATION

The equipment provided in the Launch complex must be suitable for cable runs of 2000 feet between blockhouse and pad. ESA equipment must be suitable for cable runs of 500 feet between instrument lab and vehicle.

4.2 ELECTRICAL RESTRAINTS

4.2.1 POWER REQUIREMENTS

All elements of the LCE shall operate from 105 - 125 volt, 60 cps, single-phase three-wire power source and/or 120/208 volt, 400 cps, three-phase, four-wire power source. Each power input line to the LCE shall have an appropriate overload protective device mounted in the LCE.

4.2.2 CIRCUIT ISOLATION

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

4.2.3 CALIBRATION

Instruments used in the LCE will require calibration. The LCE design will permit rapid, convenient calibration of all instruments.

4.2.4 ELECTROMAGNETIC INTERFERENCE

Electromagnetic interference and susceptibility will be minimized. Power line filters will be incorporated in all equipment, diodes placed across relay coils and capacitors across switches. Solid state switching circuits should be used in noise critical circuits. Steps will be taken to suppress all EMI at its source.

4.2.5 GROUNDING

Careful attention will be devoted to ground returns in the LCE. To prevent ground loops and maintain control of grounds, there will be several ground systems in each cabinet.

Each power supply used in a cabinet will have its return maintained separately and returned to a central ground point unique for that power supply in the cabinet. If various returns must be connected together, this will be done by connecting the appropriate ground points together. Chassis grounds will be connected together by braided wire connected to a central bus bar ground in each cabinet. Ground loops in the chassis ground system will be avoided. An external ground connection will be provided to connect the chassis ground system to the building ground.

4.3 ENVIRONMENTAL CONDITIONS

4.3.1 VIBRATION AND SHOCK

Except for portable test units, the most severe shock and vibration conditions will be met during transportation when the equipment is not operating.

4.3.2 TEMPERATURE AND HUMIDITY

Indoor equipment will be subjected to temperature ranges of 65° to 90° F. and 50% relative humidity. Outdoor equipment in exposed areas must be designed to operate in a temperature range of 25° to 160° F. and 95% relative humidity. Except for equipment to be installed in the umbilical J-Box, these requirements are design goals and the equipment need not be tested to them.

4.3.3 STORAGE

The temperature limits for storage are 32° F. and 120° F.

4.3.4 MISCELLANEOUS CONDITIONS

Requirements for sand and dust, salt spray, rain and fungus are applicable only to outdoor equipment and should be stated separately for each piece of equipment to which they apply.

4.4 EQUIPMENT RACKS

4.4.1 STANDARD RACKS

A standard rack, JPL Specification 30609, will be used for Voyager. For convenience in handling, racks longer than two bays shall be avoided.

4.4.2 PAINT

A standard finish, JPL Specification 30600, will be used for equipment racks and all front panels.

4.4.3 CONSOLE AND PANEL ACCESSORIES

Accessories common to all racks will be standardized. (Switches, indicators and knobs fall in this category.)

4.4.4 COOLING

All racks will have provision for installation of fans if required. Fans shall be electrically noise free.

4.4.5 IDENTIFICATION

All parts and assemblies will be identified. Larger assemblies will carry an approved identification plate.

4.5 CABLES

4.5.1 LONG LINES

Cables between buildings are required range facilities. Transmission of power between blockhouse and pad will be avoided. Cable currents will be limited to 100 milliamps. To the maximum practicable extent, high power levels will be transfered at the pad while the power control signals will be generated in the Blockhouse LCE.

4.5.2 UMBILICAL CABLES

The umbilical cables will connect the vehicle umbilical with the boom connector plate and then to a junction box located near the connection plate. Isolation amplifiers will be provided at this point if needed.

4.5.3 SPARE WIRES

It is desirable to keep a reasonable number of wires in each cable as designated spares which may be required for any signals added late in the program.

4.5.4 GROUNDING OF CABLES

All LCE equipment located in the same building will be tied together at a common point with a ground wire No. 2 AWG or greater.

4.5.6 SHIELDING

Shield leads shall be carried individually through each connector and into the equipment racks. If shields are connected together, it must be done with jumpers in the equipment racks. The only exception will be the umbilical cables. All shielded leads must have insulated shielding. Under no circumstances may a shield carry current (coaxial cables are excluded).

4.6 CONNECTORS

Standard connectors will be specified for different categories of use.

4.6.1 POWER INPUT

A specific type of connector for each kind of power.

4.6.2 OSE TO OSE

A family of connectors will be designated for use in the LCE.

4.6.3 TEST EQUIPMENT INPUTS

Wherever possible, standard signal input connectors will be specified for particular pieces of test equipment such as printers, oscilloscopes.

4.7 SPECIAL EQUIPMENT

4.7.1 SIGNAL CONDITIONERS

Isolation amplifiers and line amplifiers will be used where required, particularly in the umbilical junction boxes and between buildings. Isolation amplifiers shall be the same as those used in the STC.

5.0 LAUNCH COMPLEX REQUIREMENTS

The use of the Voyager LCE at any particular pad will place certain requirements on that pad. The following is a description of the requirements currently estimated.

5.1 UMBILICAL TOWER

The umbilical tower will provide electrical cabling between the spacecraft and the LCE. The approximate wire requirements from the Launch Pad Building to the umbilical connector are two power wires 1/0 or greater and up to 150 shielded wires No. 20 AWG or greater.

A junction box will be mounted as close as feasible to the umbilical catenary termination point providing access to all umbilical wires in the vicinity of the spacecraft. Amplifiers will be located in this box. An RF switch box will also be mounted on the tower near the junction box.

Two antennas will be mounted on the umbilical tower in clear view of the hanger. One antenna will be a parabolic dish for operation in the 2000 MC region. The other antenna will operate at approximately 200 MC.

5.2 MOBILE SERVICE TOWER

The Mobile Service Tower will have a 2000-mc parabolic dish antenna mounted on it. This antenna will be used when the MST surrounds the spacecraft. The MST must also support an antenna coupler which will connect to the antenna.

5.3 LAUNCH PAD BUILDING

Space for four racks of equipment will be provided under the umbilical tower in the vicinity of the pad. The space must have a controlled environment suitable for electronic equipment and personnel. Storage space must also be provided for tools and miscellaneous test equipment. Power required from the facility will be four 120VAC, 60-cycle, 3-wire circuits, and one 120/208VAC, 400-cycle, 4-wire circuits.

5.4 BLOCKHOUSE

Space will be required in the blockhouse for eight upright racks of equipment and one four-bay sit down console in the control area. The space must have a controlled environment suitable for electronic equipment and personnel.

Power required from the facility will be eight 120VAC, 60-cycle, 3-wire circuits.

5.5 INTER-BUILDING CABLES

Cables installed between the blockhouse and pad are required for LCE use. There will be at least 100 shielded pair of No. 20 AWG wire. Between the blockhouse and hangar (STC) there will be at least four video pairs.

6.0 DOCUMENTATION

A minimum set of documents for the Launch Complex and the Explosives Safe Area shall include a system schematic for each location, individual schematics, functional descriptions of the equipment and assembly drawings.

CII - VB280FD101

LAUNCH COMPLEX EQUIPMENT

TELECOMMUNICATIONS

OSE

Index

1.0 SCOPE

This document describes the Data Handling and Storage (DH&S) Launch Complex Equipment (LCE) to be used at the Launch Complex and Explosive Safe Area in support of the Voyager Project

2.0 APPLICABLE DOCUMENTS

VB280SR101	1971 Voyager Spacecraft Launch Complex Equipment Test Objectives and Design Criteria.
VB280SR102	Launch Complex Equipment Design Characteristics and Restraints.
VB263FD105	Functional Description, Data Storage OSE.
VB263FD106	Functional Description, Data Encoder OSE.

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The Telecommunications Subsystem includes the Radio, Relay Radio, Command, Data Handling and Storage subsystems. The LCE required to support the telecommunications equipment will involve only the Data Handling and Storage parts of the TCM subsystem. The remainder of the TCM subsystem is supported by the System Test Comples through an r-f link between pad and hangar.

Only Data Handling and Storage LCE (DH&S LCE) is required to support tests at the Launch Complex and Explosives Safe Area. The DH&S LCE operates in conjunction with the STC Telecommunications equipment. The LCE consists of assemblies which are identical to assemblies used in the Data Handling and Storage equipment in the STC. Only those assemblies required for use in the launch complex and explosives safe area are used in the DH&S LCE.

3.2 BLOCK DIAGRAM

A block diagram of the DH&S LCE, including external interfaces, is given in Figure 3-1.

3.3 REQUIRED FUNCTIONS

The DH&S LCE is capable of performing the following functions:

a. Monitoring the modulated subcarrier output of the spacecraft Data Encoder.

b. Providing limited decommutation and display of telemetry data.

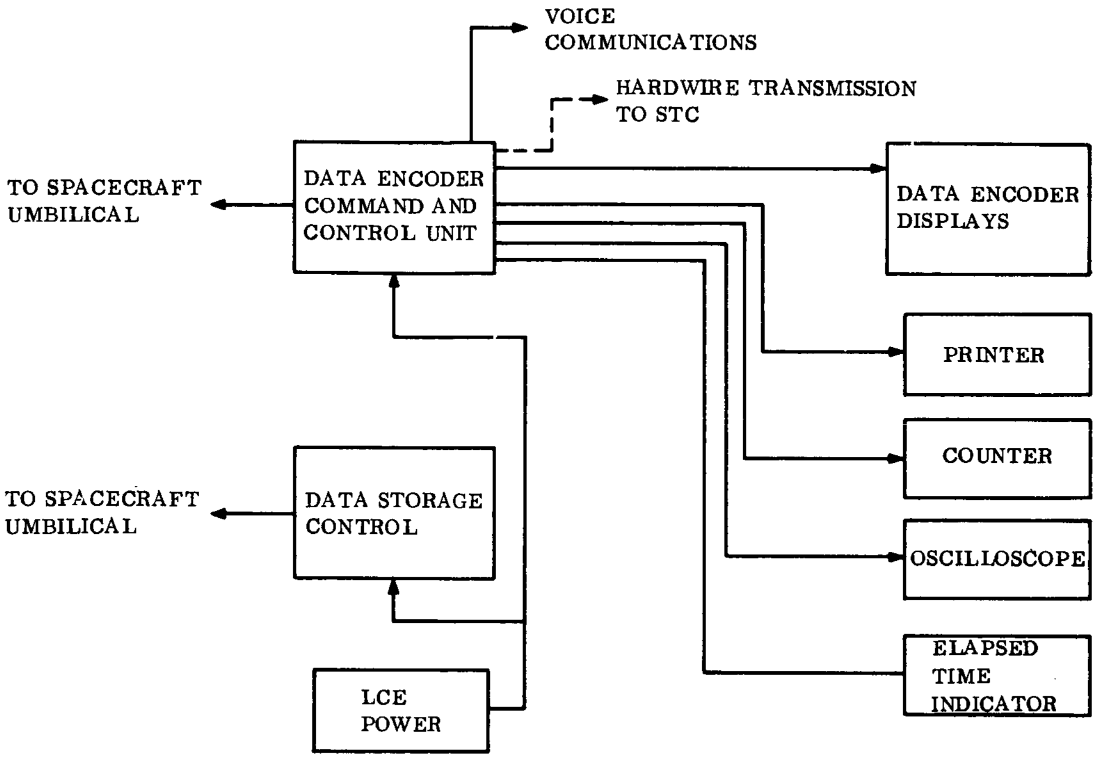


Figure 3-1. Data Handling and Storage LCE, Block Diagram

c. Seting the spacecraft Data Encoder to the Launch Mode.

d. Setting the spacecraft Data Storage Subsystem to the Launch Mode.

e. Providing the modulated squarewave output of the spacecraft Data Encoder to a hardwire transmission circuit.

3.4 COMPONENTS

The components of the DH&S LCE are:

a. Data Encoder Command and Control Unit

b. Data Storage Control Unit

c. Data Encoder Display

d. Printer

e. Oscilloscope

b. Display engineering data, capsule data, and non-scan science data received at any of the six bit rates.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINST

The LCE is housed in two rack enclosures and requires approximately 3000 watts. The weight is approximately 2000 pounds.

7.0 SAFETY

7.1 SPACECRAFT PROTECTION

The LCE is designed to interface safely with the spacecraft through the umbilical J-box.

7.2 EQUIPMENT

The LCE provides circuit protection and safety devices such as overload relays and voltage stabilization devices. Safety features such as proper weight distribution to prevent tipping are utilized.

7.3 PERSONNEL SAFETY

The cabinet is grounded with an external ground connection no smaller than No. 2 awg.

CII - VB280FD101

LAUNCH COMPLEX EQUIPMENT

TELECOMMUNICATIONS

OSE

Index

1.0 SCOPE

This document describes the Data Handling and Storage (DH&S) Launch Complex Equipment (LCE) to be used at the Launch Complex and Explosive Safe Area in support of the Voyager Project

2.0 APPLICABLE DOCUMENTS

VB280SR101	1971 Voyager Spacecraft Launch Complex Equipment Test Objectives and Design Criteria.
VB280SR102	Launch Complex Equipment Design Characteristics and Restraints.
VB263FD105	Functional Description, Data Storage OSE.
VB263FD106	Functional Description, Data Encoder OSE.

3.0 FUNCTIONAL DESCRIPTION

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The Telecommunications Subsystem includes the Radio, Relay Radio, Command, Data Handling and Storage subsystems. The LCE required to support the telecommunications equipment will involve only the Data Handling and Storage parts of the TCM subsystem. The remainder of the TCM subsystem is supported by the System Test Comples through an r-f link between pad and hangar.

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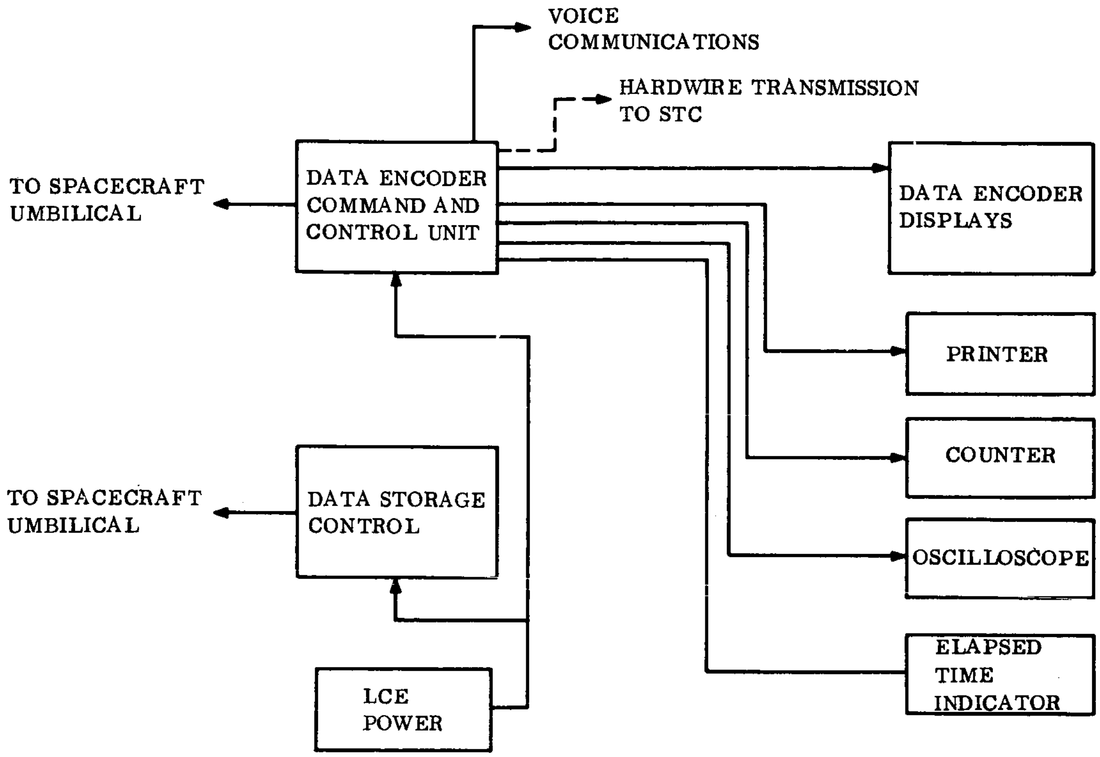

Figure 3-1. Data Handling and Storage LCE, Block Diagram

c. Seting the spacecraft Data Encoder to the Launch Mode.

d. Setting the spacecraft Data Storage Subsystem to the Launch Mode.

e. Providing the modulated squarewave output of the spacecraft Data Encoder to a hardwire transmission circuit.

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The components of the DH&S LCE are:

a. Data Encoder Command and Control Unit

b. Data Storage Control Unit

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d. Printer

e. Oscilloscope

b. Display engineering data, capsule data, and non-scan science data received at any of the six bit rates.

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The LCE is housed in two rack enclosures and requires approximately 3000 watts. The weight is approximately 2000 pounds.

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7.2 EQUIPMENT

The LCE provides circuit protection and safety devices such as overload relays and voltage stabilization devices. Safety features such as proper weight distribution to prevent tipping are utilized.

7.3 PERSONNEL SAFETY

The cabinet is grounded with an external ground connection no smaller than No. 2 awg.

CII - VB280FD102

LAUNCH COMPLEX EQUIPMENT CONTROLLER AND SEQUENCER OPERATIONAL SUPPORT EQUIPMENT

Index

1.0 SCOPE

This document contains a functional description and interface definitions for the Controller and Sequencer Operational Support Equipment, a part of the Launch Complex Equipment used during prelaunch operations for the 1971 Voyager Missions, and used at the Explosive Safe Area for post assembly confidence testing.

2.0 APPLICABLE DOCUMENTS

GE

VB280SR101	1971 Voyager Spacecraft LCE Test Objectives and Design Criteria
VB280SR102	1971 Voyager Spacecraft LCE Design Characteristics and Constraints
VB280FD101	1971 Voyager Spacecraft Telecommunications OSE, Launch Complex Equipment
VB264FD105	1971 Voyager Spacecraft Controller and Sequencer OSE, System Test Complex

3.0 DESCRIPTION

3.1 GENERAL

The Controller and Sequencer Operational Support Equipment for the Launch Complex Equipment is a part of the C & S subsystem OSE described in VB264FD105. It consists of the C & S Timer Control Panel, and is mounted in a rack which is part of the Launch Complex Equipment.

3.2 THE CONTROLLER AND SEQUENCER SUBSYSTEM

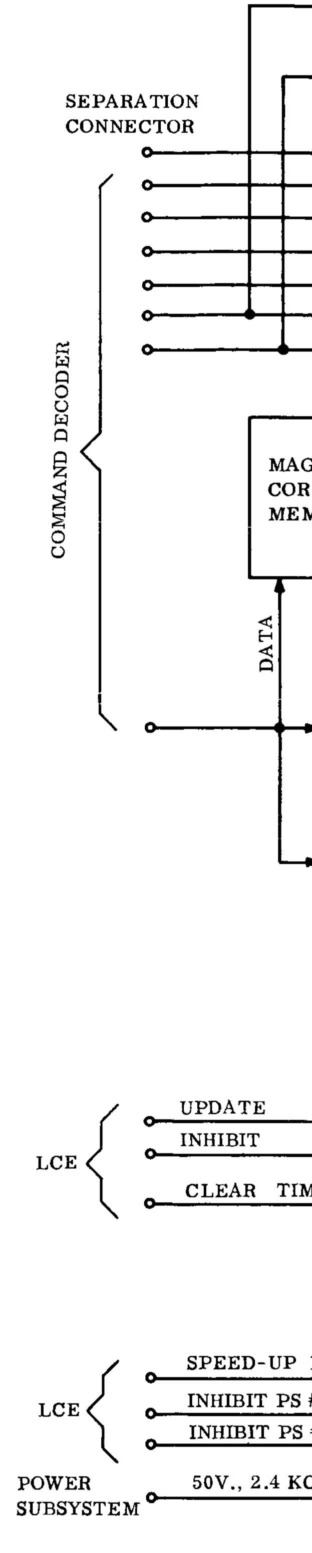
Figure 3-1 is a simplified block diagram of the S/C Controller and Sequencer. The C&S stores spacecraft and capsule commands including maneuver data, orbital sequences, and updating data for the high-gain antenna and scan platform. These commands are of two kinds: those which are utilized but once (associated with the Master Timer) during the mission, and those which occur in repetitive sequences (associated with the Sequence Timer). The S/C C&S also furnishes timing signals to other subsystems.

During countdown, the S/C C&S receives all commands from the Command Verification Equipment by way of the Telecommunications Subsystem. The S/C Telemetry Subsystem monitors the following C&S information:

a. C&S Events

b. Master Timer Time

c. Sequence Timer Time


SEPARATION
CONNECTOR
COMMAND DECODER
MAG
COR
ME
DATA
UPDATE
INHIBIT
LCE
CLEAR
SPEED-UP
INHIBIT PS
INHIBIT PS
LCE
POWER
SUBSYSTEM
50V., 2.4 K

3

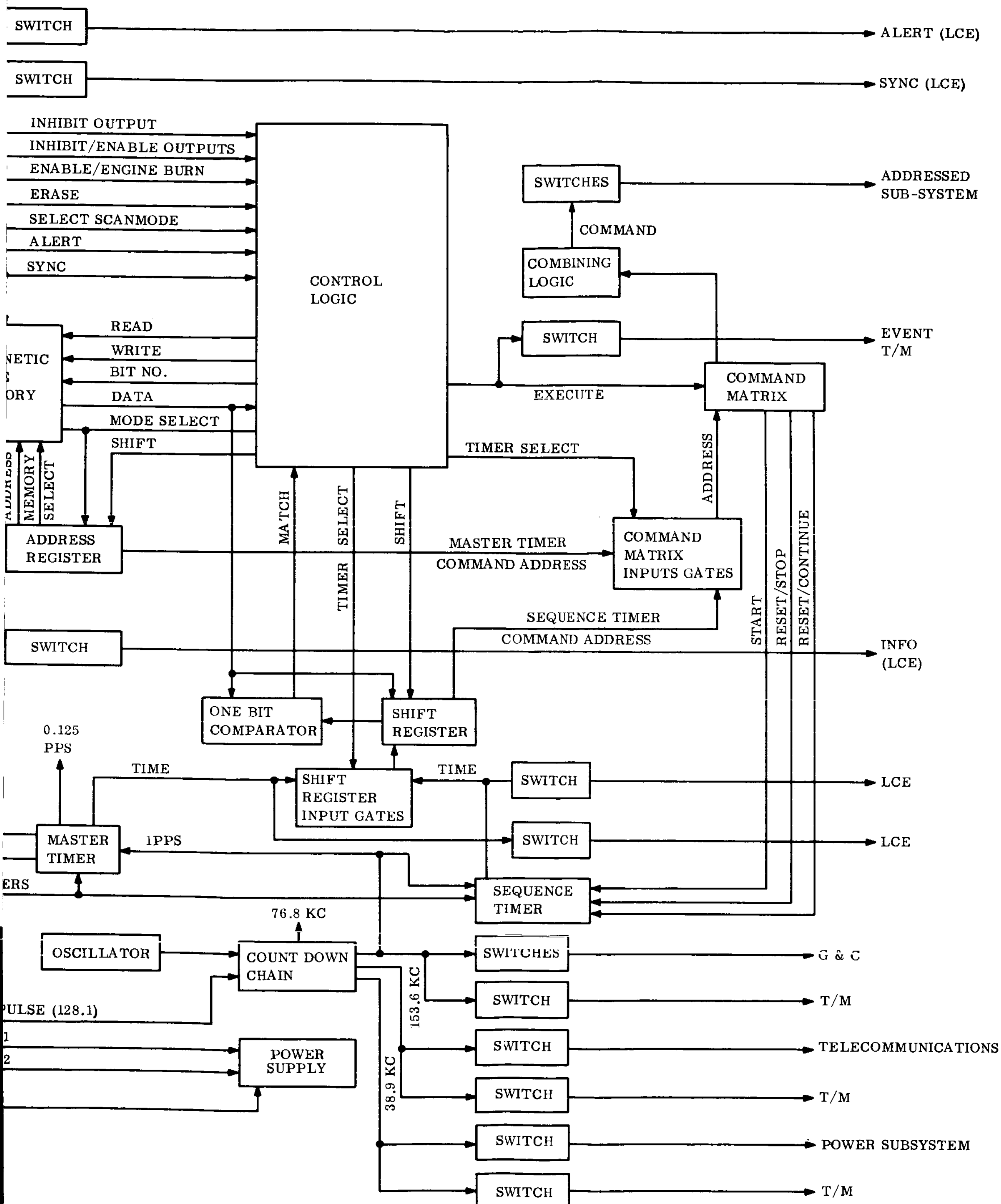


Figure 3-1. Controller and Sequences, Block Diagram

d. Shift Register Contents

e. Oscillator Oven Temperature

The C & S OSE in the Launch Complex Equipment, both at the Launch Area and at the Explosive Safe Area, does not require any hard line monitoring for confidence establishment. Confidence in the S/C C&S is established by:

a. Monitoring subsystem controlled by the S/C C&S.

b. Monitoring S/C C&S Telemetry data in the STC and relaying its status to the Launch Directors.

During countdown, the C & S Timer Control Panel (Figure 3-2), which is located in the blockhouse, controls the following functions:

a. Master Timer Speed-Up (128 : 1)

b. Master Timer Inhibit

c. Master Timer Update

d. Clear Timers

e. Inhibit Power Supply No. 1

f. Inhibit Power Supply No. 2

4.0 INTERFACE DEFINITIONS

All interfaces are electrical.

The C & S Timer Control Panel outputs tabulated in Table 4-1 go to the S/C C&S from the Launch Complex through the umbilical cable.

Table 4-1. Time Control Panel Outputs

Function	Parameter	Definition
Master Timer Speed-up	Voltage Level	6 v switch closure
Master Timer Inhibit	Voltage Level	6 v switch closure
Master Timer Update	Pulse	6 v, 2 ms min.
Clear Timers	Pulse	6 v, 2 ms min.
Inhibit Power Supply No. 1	Voltage Level	6 v switch closure
Inhibit Power Supply No. 2	Voltage Level	6 v switch closure

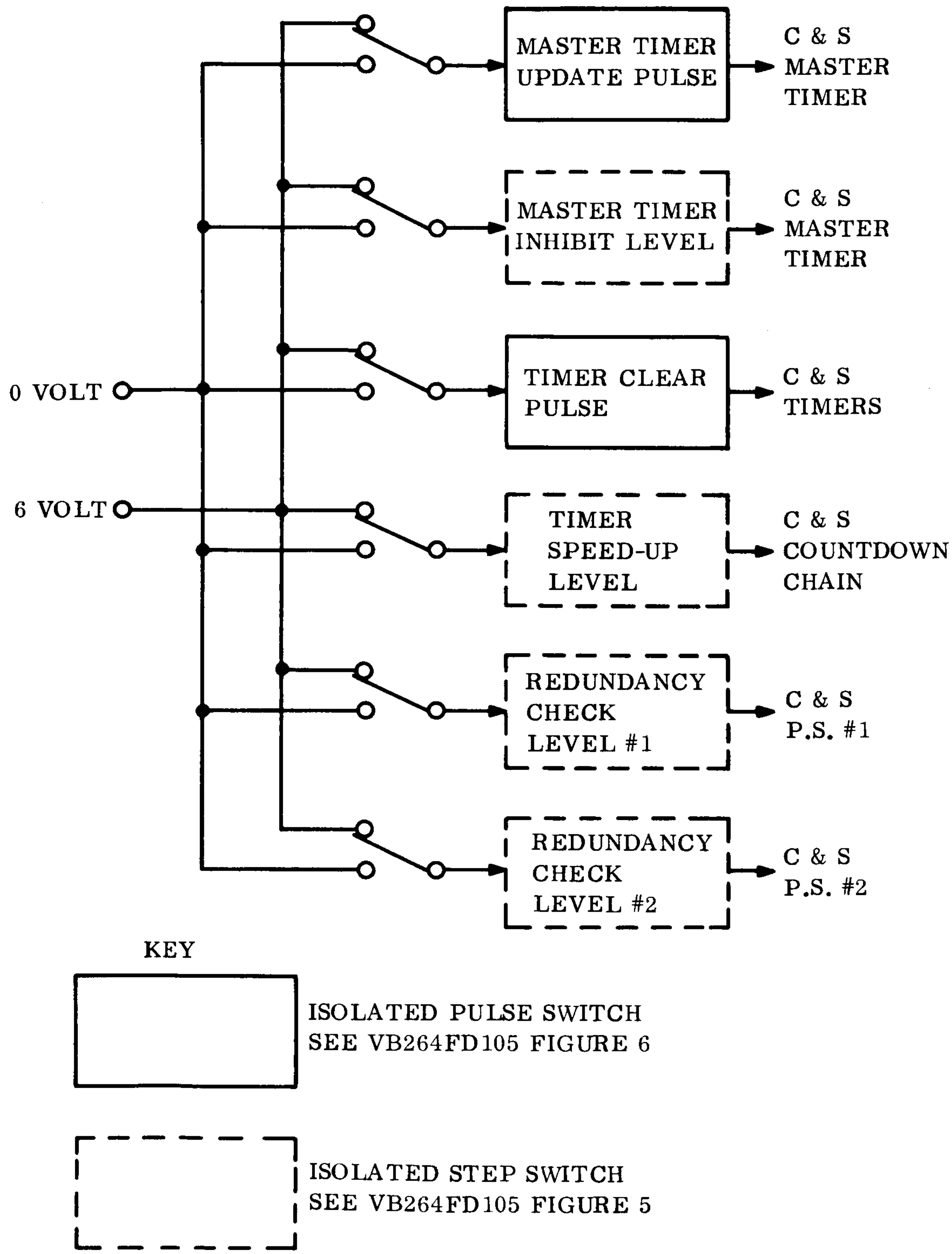


Figure 3-2. Controller and Sequencer OSE Control Panel

5.0 PERFORMANCE PARAMETERS

5.1 ENVIRONMENTAL CONDITIONS

5.1.1 VIBRATION AND SHOCK

The C & S OSE is to be capable of proper operation after the shock and vibration encountered in transportation to the operational site.

5.1.2 TEMPERATURE AND HUMIDITY

The C&S OSE is capable of proper operation at any temperature between 35° F and 125° F and at any humidity.

5.2 SWITCH CLOSURES

The characteristics of the switch closures listed in Table 4-1 are shown in Figures 5-1 and 5-2 of VB264FD105.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 POWER REQUIREMENTS

28 v dc, 1 ma.

6 v dc, 1 amp.

2.4 KC, 25 v square wave, 0.5 amp.

6.2 WEIGHT

Eight pounds

6.3 SIZE

7 inches x 19 inches, Standard rack mount panel.

7.0 SAFETY CONSIDERATIONS

The OSE is designed to present no hazard to operating personnel or facilities. Low voltages are used throughout.

CII - VB280FD103

LAUNCH COMPLEX EQUIPMENT POWER SUBSYSTEM

Index

1.0 SCOPE

This document describes the functional requirements for the Power Subsystem LCE. This equipment will simulate solar array power and monitor and control power subsystem operation during confidence tests and launch countdown.

2.0 APPLICABLE DOCUMENTS

VB236FD101	Functional Description of Power Subsystem
VB266FD101	Functional Description of Power Subsystem OSE in the STC
VB280SR101	LCE Test Objectives and Design Criteria
VB280SR102	LCE Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The Power Subsystem LCE provides all the necessary operational equipment required to support the spacecraft power subsystem during prelaunch testing and during final countdown.

To support these activities the Power Subsystem LCE has the capability to provide external d-c power for vehicle operation when the batteries are shut down, capability to control power subsystem mode switching through the umbilical and capability to monitor power subsystem condition through the umbilical.

The Power Subsystem OSE for the LCE also controls and monitors the spacecraft Power Subsystem in confidence tests performed at the Explosives Safe Area. ESA equipment is functionally identical to the equipment used at the Launch Area.

3.2 FUNCTIONAL BLOCK DIAGRAM

Refer to Figure 3-1, Launch Complex Equipment, Power Subsystem LCE Block Diagram. This diagram shows in functional form all of the elements included in the Power Subsystem LCE.

3.3 EQUIPMENT LOCATION

The Power Subsystem LCE is used at the Launch Complex and at the Explosive Safe Area. The contraints peculiar to these locations impose requirements on the LCE which have nothing to do with performance.

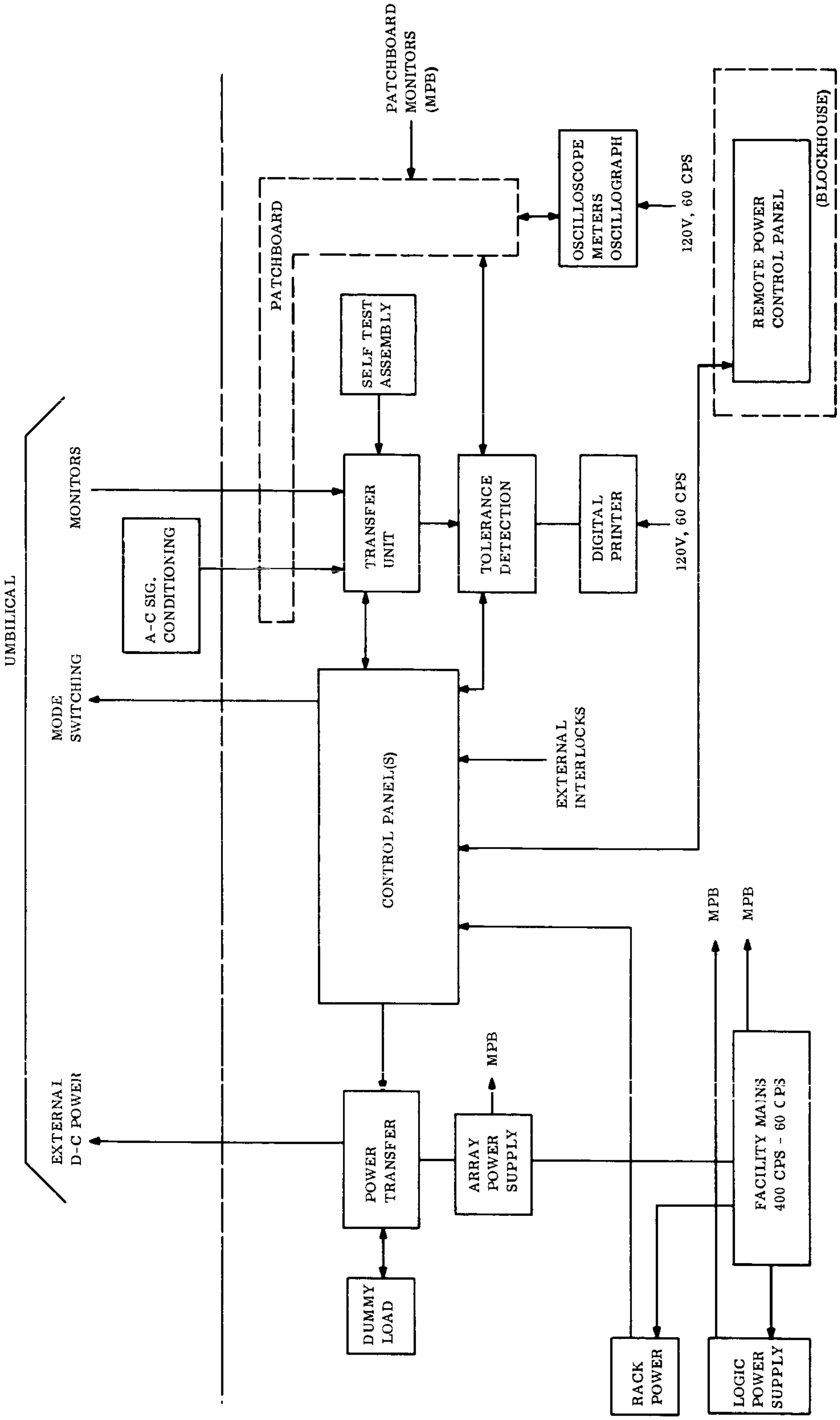


Figure 3-1. Launch Complex Equipment, Power Subsystem Block Diagram

At the launch complex all equipment must be controlled from the blockhouse which is at least 2000 feet from the pad. The external power supply for the vehicle must be much closer to the vehicle. The design solution is to divide the power LCE into three pieces. An AC Conditioning unit is placed on the umbilical tower as close as possible to the spacecraft for providing impedance matching and circuit isolation. The cabinet is placed in the launch pad building in the closest protected work area. Since this building is evacuated during launch a Remote Control Panel will be placed in the blockhouse for use during the terminal count.

3.4 PRELAUNCH ACTIVITIES

3.4.1 POWER INPUT TO SPACECRAFT

External d-c power is introduced into the spacecraft through the umbilical connector in such a manner as to simulate the solar array output to the power subsystem. The "solar array" power supply provides this d-c power. The power to the spacecraft umbilical is routed through a Power Transfer assembly which switches the output power from a variable Dummy Load to the vehicle bus on command from the Control Panel or the Remote Control Panel. Two redundant solar array power supplies are provided to minimize the possibility of launch delays due to power supply failure.

The Power Transfer assembly receives interlock signals from the Control Panel and the Transfer Unit to prevent accidental application of power to the spacecraft when there is a possibility of damage.

3.4.2 SPACECRAFT INPUTS FROM LCE

One of the most critical functions of the Power Subsystem LCE is the control of the spacecraft power system during the terminal countdown. The power subsystem LCE controls mode switching through the umbilical connector. Hardwire commands sent to the Spacecraft power subsystem from the LCE are initiated at the Control Panel or the Remote Control Panel.

3.4.3 MONITORING THE POWER SUBSYSTEM

All spacecraft signals monitored by the power subsystem LCE come from the power subsystem via the umbilical connector. The remaining monitored signals come from the power subsystem LCE itself. All signals to be monitored pass through the Transfer Assembly which controls the input lines to the Tolerance Detection Assembly. All signals are normally routed to the Tolerance Detection Unit.

The Tolerance Detection Assembly is capable of determining if a monitor point is in or out of its prescribed tolerance and on which side of the tolerance band it is out. When a function is out of tolerance an indication appears on the control panel and the remote control panel, informing the operator of the condition. The local operator may select a latching mode with the indication remaining after the out of tolerance condition disappears or he may elect to have the indication only as long as the out of tolerance condition exists.

(The Self Test Unit provides marginal out of tolerance signals and signals within tolerance for a complete check of the Tolerance Detection Assembly.)

An AC Conditioning Unit is provided with the power subsystem. The conditioning unit is placed in the umbilical junction box on the umbilical tower. It provides impedance conversion and isolation for umbilical signals coming from the spacecraft power subsystem. The circuits used in the AC Conditioning Unit are designed to transmit signals over 2000 feet of shielded wire. Identical isolation circuits are used in the STC.

3.4.4 RECORDING

Permanent records are desirable for analysis and review. The signals available from the umbilical can be recorded. The recording device will be a digital printer tied to the tolerance detection assembly. The printer provides a rapid, easily read display which is also a permanent record.

In addition to the digital printer, recording of continuous signals is provided by a twenty-four channel oscillograph. The oscillograph inputs are selected at the patchboard.

3.4.5 POWER SUBSYSTEM LCE CONTROL AND DISPLAY

The Control Panel provides a central control point for operating the LCE and performing a test sequence. Most of the controls for the power LCE are located on the control panel. These controls include power subsystem switching, control of the Transfer Assembly modes, Power Transfer, and any other displays required for the convenience of the operator. Most of the displays and controls are repeated on a Remote Control Panel located in the blockhouse. The Remote Control Panel contains all spacecraft mode switching controls, solar array power supply controls and malfunction indications. There is also an oscilloscope and meters which may be used as monitors when required, although they are primarily troubleshooting and calibration aids.

3.5 SELF TEST CAPABILITIES

The power subsystem LCE is capable of performing tests on its own power and monitor circuits to ensure that it is fully operational and capable of carrying out its support mission. The self test features may be used before connecting the umbilical to the spacecraft, periodically during prelaunch activities or at any time the operator decides a self check sequence is required. The self test sequence can be used on a non-interference basis during a spacecraft test or operational sequence. The self test is implemented by having the transfer assembly switch test signals into the tolerance detection assembly. The comparator circuits in these assemblies are then used to evaluate the test signals which are OSE generated. In tolerance and out of tolerance signals are thus checked. The tolerance detection assembly is self checked by using premeasured signals as the input and observing whether they are correctly identified as being in tolerance or out of tolerance.

The switching of the transfer assembly is self tested by observation of the self tests of the tolerance detection and test signal generation functions.

Power output is self tested by switching high power level outputs into the variable dummy loads and making measurements of voltage and delivered current.

4.0 POWER SUBSYSTEM LCE INTERFACES

The important interfaces with the Power Subsystem LCE are electrical interfaces with the spacecraft and other LCE.

4.1 POWER SUBSYSTEM ELECTRICAL INTERFACE

Umbilical Connector - The power LCE connects to the umbilical connector through the facility cables and umbilical junction box. A list of umbilical functions appears in Table 4-1.

4.2 LAUNCH PAD BUILDING - ELECTRICAL INTERFACE

The Power Subsystem LCE will receive power from the facility and interlock signals from other LCE. The required power is:

a. 120/208 VAC, 400 cycles, 3-phase, 4-wire

b. 120 VAC, 60 cycles, 1-phase, 3-wire

c. Interlock signals from other LCE.

5.0 PERFORMANCE PARAMETERS

Those performance parameters estimated are:

a. Solar Array Power Supply - Range 0 - 55 vdc

b. Tolerance Detection - Accuracy ± 1%

c. Dummy Load - within ± 5% of real load

d. Transfer Assembly - Capacity 50 vehicle Signals

e. Digital Printer - 11 lines per second

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Voyager Power Subsystem LCE will be similar in configuration to the Voyager power Subsystem OSE. The design and packaging approach is to delete those assemblies

Table 4-1. Umbilical Tabulation, Power Subsystem

Function	Characteristics
1. External Power	44 - 55 vdc
2. External Power Return	
3. Enable Array/Battery	Pulse to Relay (On)
4. Disable Array/Battery	Pulse to Relay (Off)
5. Enable/Disable Return	
6. Enable/Disable State	On/Off
7. Array/Battery Bus Voltage	30 - 55 vdc
8. Raw Battery Bus Voltage	30 - 45 vdc
9. Battery No. 1 Voltage (Course)	30 - 45 vdc
10. Battery No. 2 Voltage (Course)	30 - 45 vdc
11. Battery No. 3 Voltage (Course)	30 - 45 vdc
12. Battery No. 1 Temperature	5 - 10 vdc
13. Battery No. 2 Temperature	5 - 10 vdc
14. Battery No. 3 Temperature	5 - 10 vdc
15. Temperature Sensor Return	
16. 400 cps 3φ Bus PH A	22 - 28 vac
17. 400 cps 3φ Bus PH B	22 - 28 vac
18. 400 cps 3φ Bus PH C	22 - 28 vac
19. 2.4 kc Bus	50 vac
20. 2.4 kc Bus Return	

requiring direct access and retain the assemblies associated with the umbilical connector. Principle difference is the addition of another solar array power supply for redundancy during countdown.

6.1 GENERAL

Size - Three Standard Racks

Weight - Approximately 900 lbs.

Power Required - 120 vac, 1-Phase, 60-cycle, 3-Wire, 20-Amp
120/208 vac, 3-Phase, 400-Cycle, 4-Wire, 15-Amp

6.2 CABINET CONFIGURATION

Figure 6-1 shows the anticipated placement of assemblies in the Power Subsystem LCE Cabinet.

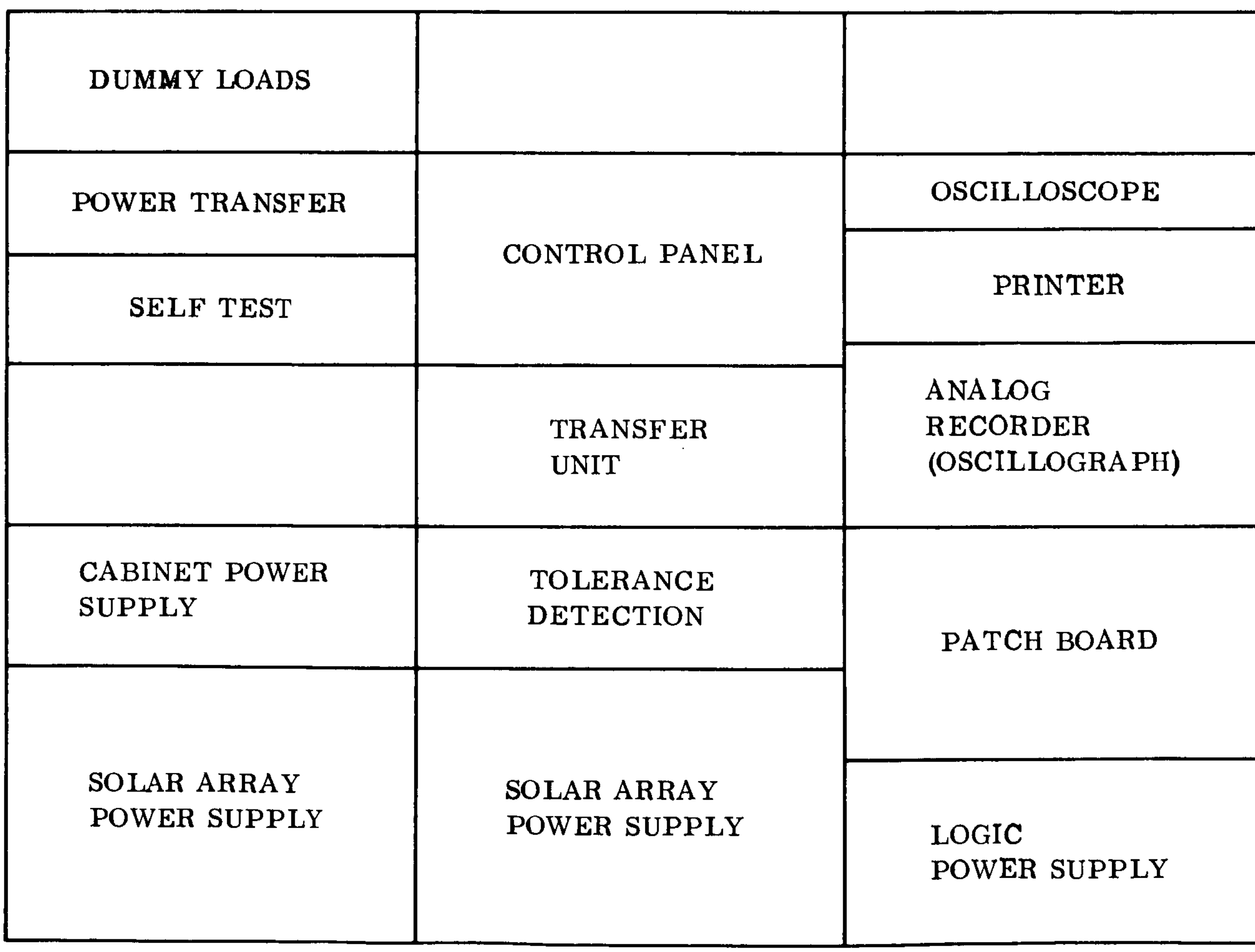


Figure 6-1. Power Subsystem LCE

6.3 CONTROL PANEL

The Control Panel will contain the displays and controls required to perform a test. The anticipated display and control requirements are listed below.

a. Controls

1. Ground Power On/Off
2. Solar Array and Battery - Enable (On)
3. Solar Array and Battery - Disable (Off)
4. OSE Operate or Transfer to S/C
5. OSE Self Test
6. Emergency Disconnect
7. Out of Tolerance Failure Latch

b. Monitors, Visual/Audio

1. Visual indication of the control status above
2. External Power Voltage/Current (Array Simulator)
3. Battery status charge/discharge
4. Battery Voltage/Current
5. Self Test: Start, In Process, Complete
6. External Power: on, off; In Tolerance/Out of Tolerance
7. "Out of Tolerance" for the following monitors:
   (a) Battery Voltage
   (b) Battery Current
   (c) Main Regulator

(d) 2.4-kc Inverter

(e) 400-cps Inverter, Phases A, B and C

8. Battery Temperature

6.4 REMOTE CONTROL PANEL

The Remote Control Panel will contain the displays and controls required during the terminal countdown. The anticipated display and control requirements are listed below.

a. Controls

1. Ground Power On/Off
2. Solar Array and Battery - Enable (On)
3. Solar Array and Battery - Disable (Off)
4. OSE Operate or Transfer to S/C
5. OSE Self Test
6. Emergency Disconnect
7. Remote Voltage Control (Array Simulator)

b. Monitors, Visual/Audio

1. Visual indication of the control status above
2. External Power Voltage/Current (Array Simulator)
3. Battery status charge/discharge
4. Battery Voltage/Current
5. Self Test: Start, In Process, Complete
6. External Power: On, Off; In Tolerance/Out of Tolerance
7. "Out of Tolerance" for the following monitors:

(a) Battery Voltage

(b) Battery Current

(c) Main Regulator

(d) 2.4-kc Inverter

(e) 400-cps Inverter, Phases A, B and C

8. Battery Temperature

6.5 TRANSFER ASSEMBLY

The transfer assembly will be a rack-mounted electronics assembly consisting of switching circuits on removable cards mounted on a holding rack.

6.6 TOLERANCE ASSEMBLY

The tolerance assembly will be a rack-mounted electronics assembly consisting of switching and comparator circuits on removable cards mounted in a holding rack.

6.7 SELF TEST PANEL

The Self Test Panel will consist of front panel with controls and signal level adjustments and circuit elements mounted on removable cards. The cards will be mounted in a holding rack.

6.8 SOLAR ARRAY POWER SUPPLY

The Solar Array Power Supply will simulate the approximate output voltage versus current curve of the solar array. The power supply will be a self-contained unit with voltage and current meters and all necessary controls.

6.9 DUMMY LOAD ASSEMBLY

The Dummy Load Assembly consists of resistors mounted on a chassis with appropriate cooling. Load currents will be monitored with meters mounted on the front panel. Loads will be provided for the solar array power supplies. Any other loads required for testing will be mounted on this assembly which will have adequate cooling and ventilation.

6.10 POWER TRANSFER

The Power Transfer Assembly is a separate assembly containing power switches, contactors and circuit breakers. This panel will have no controls except for reset of circuit breakers.

6.11 AC CONDITIONING ASSEMBLY

The AC Conditioning Assembly is a small self-contained unit with its own power supply. It contains AC-DC Conversion circuits and isolation amplifiers. It will be mounted in a junction box on the umbilical tower near the spacecraft umbilical.

6.12 POWER SUBSYSTEM LCE TERMINATION

The Power Subsystem LCE termination is a connector and distribution assembly, mounted in the Power LCE cabinet, which provides for cables coming from the other LCE and umbilical and distributes the signals throughout the power LCE cabinet.

6.13 PATCH BOARD

Patching and reconnection of the OSE assemblies to conform to particular test modes or to monitor specific signals is accomplished through a removable patch board. This board is an AMP or MAC patchboard.

6.14 STANDARD UNITS

Listed are the commercially available assemblies and the type of equipment envisioned.

Oscillograph - Visicorder or equivalent

Digital Recorder - Hewlett Packard or equivalent

Oscilloscope - Tektronics 535 or equivalent

Meter - NLS Digital Voltmeter

6.15 CABINET POWER SUPPLIES

The cabinet power supply consists of the power supplies required for operating the logic circuits, control displays and amplifiers on the various electrical assemblies contained in the Power Subsystem LCE. One power supply provides 28 vdc rack power for indicators and switches. A second power supply provides various d-c voltages required for the assemblies using logic cards.

7.0 SAFETY CONSIDERATIONS

7.1 SPACECRAFT PROTECTION

Series resistors and/or isolation transformers are provided in all monitoring lines to assure protection of S/C circuits. Direct power inputs to the vehicle have isolated grounds to avoid the possibility of uncontrolled group loops. Interlocks are provided on the vehicle power input circuits to prevent application of power at times when it may result in S/C or LCE damage.

7.2 LCE AND FACILITIES PROTECTION

Equipment other than the spacecraft is protected by conventional circuit breakers and fuses in the appropriate power lines.

7.3 PERSONAL SAFETY

In addition to normal procedural precautions, there will be an external and visible cabinet ground connection to a central building ground. The safety ground line will be no smaller than No. 2 AWG. The oscillograph will have an interlock for switching off dangerous voltages during equipment maintenance.

CII - VB280FD104

LAUNCH COMPLEX EQUIPMENT

GUIDANCE AND CONTROL OSE

Index

1.0 SCOPE

This document is a functional description of the Guidance and Control Subsystem LCE, a part of the 1971 Voyager Launch Complex Equipment. This equipment will provide control signals, monitor gyro temperatures and monitor gyro loop operation during prelaunch confidence tests and during launch countdown.

2.0 APPLICABLE DOCUMENTS

VB264FD104	-	Functional Description of Guidance and Control OSE in the STC
VB234FD102	-	Functional Description of Attitude Control Subsystem
VB280SR101	-	LCE Test Objectives and Design Criteria
VB280SR102	-	LCE Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

During launch operations, the Guidance and Control LCE (Figure 3-1) indicates that the guidance and control subsystem and, in particular, the gyro loops are functioning properly. This will be done by direct monitor through the umbilical, so that real time data will be available. These functions will be displayed in the Blockhouse on meters. The G&C LCE will also provide updating signals for the approach guidance system during the terminal countdown.

Rate signals from the gyros are monitored. The gyros measure vehicle motion on the pad. The vehicle motion is a result of earth rate and structural motion. The rate signal gives a gross indication of system operation.

Temperature signals from the gyros are also monitored. The performance of the gyros is affected by temperature making continuous monitoring of temperature necessary.

Rate and temperature signals from the vehicle umbilical are amplified in the umbilical J-Box by the G&C Amplifiers and transmitted to the G&C Display Panel 2000 feet away in the blockhouse.

Approach Guidance update signals are initiated with switches on the display panel and transmitted through the umbilical connector.

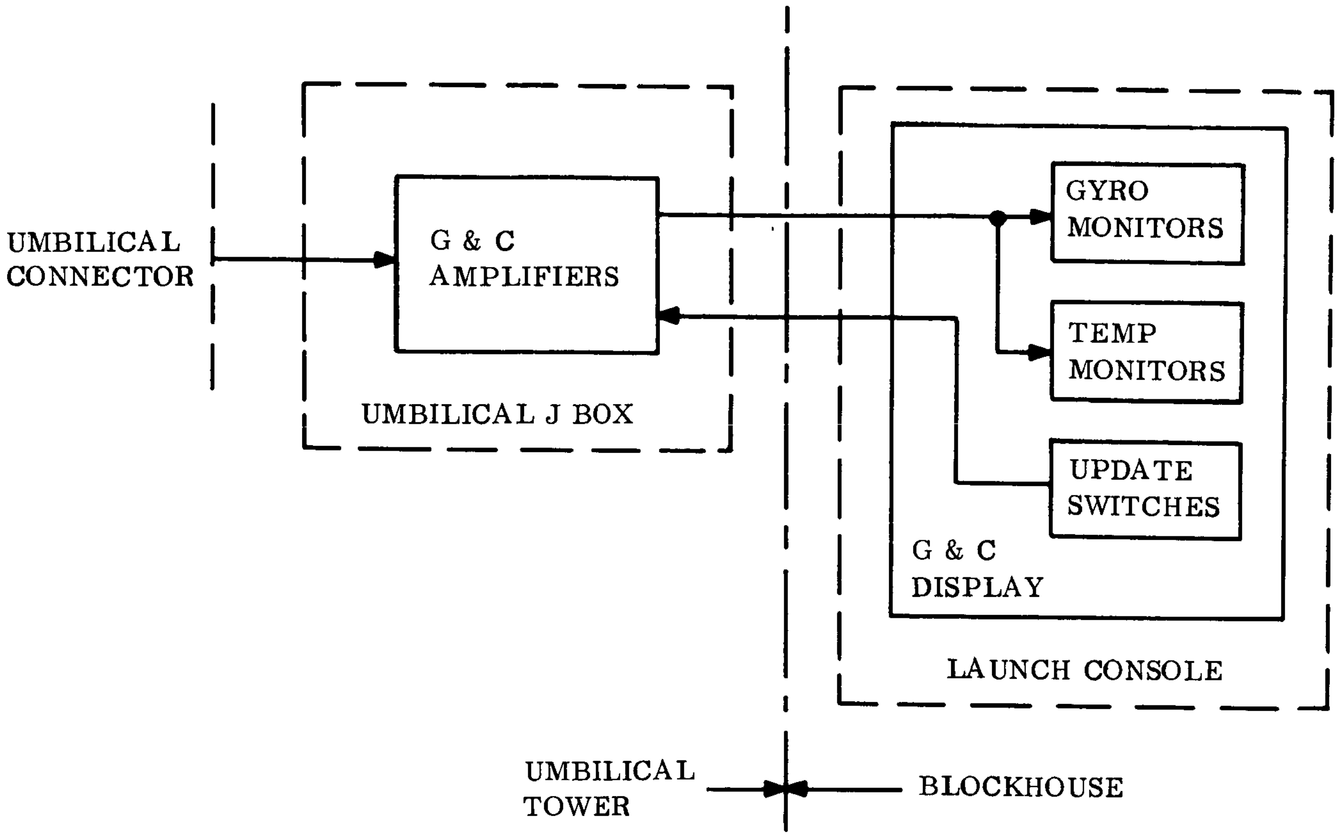


Figure 3-1. Guidance and Control LCE, Functional Block Diagram

4.0 INTERFACE DEFINITION

4.1 MECHANICAL INTERFACES

Interface	Characteristics
Launch Console	Panel Space 19 inches wide, 7 inches high, 18 inches deep
Umbilical J-Box	Space for package of isolation amplifiers and power supply

4.2 ELECTRICAL INTERFACES

Interface	Characteristics
Spacecraft Umbilical Pitch Rate Signal Yaw Rate Signal Roll Rate Signal	0 - 1 vac 400 Cycles

Interface	Characteristics
Pitch Gyro Temperature	0 - 3.2 vdc
Yaw Gyro Temperature	
Roll Gyro Temperature	
Shield Return	
Ground Return	
Approach Guidance Update 1	d-c signals
Approach Guidance Update 2	
Launch Console	115 vac, 60 cycle
Umbilical J-Box	115 vac, 60 cycle

5.0 PERFORMANCE PARAMETERS

	Measurement Accuracy
Gyro Rate Signals	± 1%
Gyro Temperatures (Meter Accuracy)	± 1%

6.0 PHYSICAL CHARACTERISTICS

6.1 G & C DISPLAY PANEL

Size 7 inches high, 19 inches wide

The Panel is a conventional rack panel with six display meters, switches, and any necessary calibration adjustments.

6.2 G & C AMPLIFIERS

A small self-contained package containing isolation amplifiers and the associated power supply.

7.0 SAFETY CONSIDERATIONS

7.1 EQUIPMENT

The spacecraft is protected from damage by isolation resistors or transformers in the G & C Amplifier box. Protection of the LCE is provided by circuit breakers or fuses in the LCE.

7.2 PERSONNEL

All assemblies are grounded to the equipment rack in which they are mounted.

4.2 PYROTECHNIC SUBSYSTEM

The displays and controls on this panel will interface with the pyrotechnic controllers through the launch complex cabling and through the Spacecraft umbilical connector.

5.0 PERFORMANCE PARAMETERS

The performance parameters of this panel will depend on the voltage and currents required to operate the pyrotechnic controllers.

6.0 PHYSICAL CHARACTERISTICS

The panel will be 19 inches wide by 7 inches high.

7.0 SAFETY CONSIDERATIONS

Special safety procedures for the use of this panel should be developed for the pad safety plan. The design of this equipment is required to assure that connections established in the OSE, and signals generated by the OSE, can not actuate pyrotechnics. This includes OSE-to-OSE short circuits and OSE abnormal overvoltage conditions or surges.

CII - VB280FD107

LAUNCH COMPLEX EQUIPMENT SCIENCE

PAYLOAD PROVISIONS

Index

1.0 SCOPE

This document is an estimated functional description of the science subsystem LCE for purposes of defining the interface with the rest of the LCE. The provisions which are to be made to accommodate Science OSE in the LCE are defined.

2.0 APPLICABLE DOCUMENTS

Specifications

VB280SR101	1971 Voyager Spacecraft LCE Test Objectives and Design Criteria
VB280SR102	1971 Voyager Spacecraft LCE Design Characteristics and Restraints

Drawings

SKS-6152-558	Functional Block Diagram LCE, Voyager 1971

3.0 FUNCTIONAL DESCRIPTION

This equipment will be a part of the Launch Complex Equipment. It will monitor and provide limited control of the major subassemblies in the Science payload. The functions to be performed are:

a. Monitor and control the Data Automation System.

b. Monitor experiments as required prior to launch.

c. Provide status signals as required to the launch conductor's console.

While standard items of test equipment may be included for display, measurement and signal generation, the Science Subsystem LCE will not be used for detailed testing. The LCE will, with the support of the STC, be capable of isolating malfunctions to the science subsystem. Any trouble analysis extending into the science subsystem will depend on information obtained through the telemetry system and decommutated in the STC.

4.0 INTERFACE DEFINITION

The following are anticipated interfaces based on examinations of Mariner "C" interfaces and comparison of weight provisions of Mariner C Science to Voyager 71 Science. All inputs to and outputs from the science LCE will be through the spacecraft LCE. The interface will be as listed.

4.1 LAUNCH COMPLEX

a. Space for a two-bay cabinet in blockhouse.

b. 120 VAC, 60-cycle, single-phase, 3-wire. Two 20-ampere circuits will be required.

c. Controlled atmosphere, but no special cooling requirements.

d. Cable connection from blockhouse to umbilical. Approximately 24 wires.

e. Isolation amplifiers in umbilical J-Box. Approximately eight amplifiers.

5.0 PERFORMANCE PARAMETERS

(To be determined)

6.0 PHYSICAL CHARACTERISTICS

a. General - The Science LCE will be mounted in two standard racks. The racks will be identical to the racks used for other LCE.

b. Data Automation Panel - Contains the switches, indicators and circuits necessary to interface with the Data Automation System and perform prelaunch and countdown checks on it.

c. Experiment Panels - Most experiment monitor points will be monitored through the telemetry system, but it can be safely assumed that there will be functions or subsystems requiring hardwire monitor or stimulation during prelaunch. Control panels for these functions will be located in the science LCE.

d. Power Supply - An adequate power supply will be required.

7.0 SAFETY PROVISIONS

7.1 EQUIPMENT SAFETY

The Science LCE is required to have design features such that Science LCE failure shall not directly or by interaction with the rest of the LCE, prejudice the integrity of the spacecraft or its payloads.

7.2 PERSONNEL SAFETY

Normal precautions associated with the use of electrical equipment will be exercised with this equipment. No special safety precautions are required.

CII - VB280FD108

LAUNCH COMPLEX EQUIPMENT

CAPSULE PROVISIONS

Index

1.0 SCOPE

This document is an estimated functional description of the Capsule LCE for purposes of defining the interface with the rest of the LCE. The provisions which are to be made to accommodate capsule OSE in the LCE are defined.

2.0 APPLICABLE DOCUMENTS

VB280SR101	1971 Voyager Spacecraft LCE Test Objectives and Design Criteria
VB280SR102	1971 Voyager Spacecraft LCE Design Characteristics and Restraints Drawings

Drawings

SK5612-558	Functional Block Diagram LCE, Voyager 1971

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

At the launch complex the Capsule will already have been installed in the Spacecraft and will be completely inaccessible except for the umbilical interface between the capsule and the Spacecraft. Any external power required by the capsule will come from the S/C power bus. The only connection between the Capsule and the LCE will be the S/C umbilical.

For purposes of defining the interface at the pad, the following conditions are assumed:

a. Some lines from capsule umbilical go to S/C umbilical Control and Monitor lines.

b. Capsule signals go to S/C Telemetry.

Further, it is assumed that confidence tests will be made on the capsule. In order to perform these confidence tests, some capsule LCE will be required.

3.2 BLOCK DIAGRAM

Figure 3-1 shows the effect of the anticipated capsule/Spacecraft interface on the LCE.

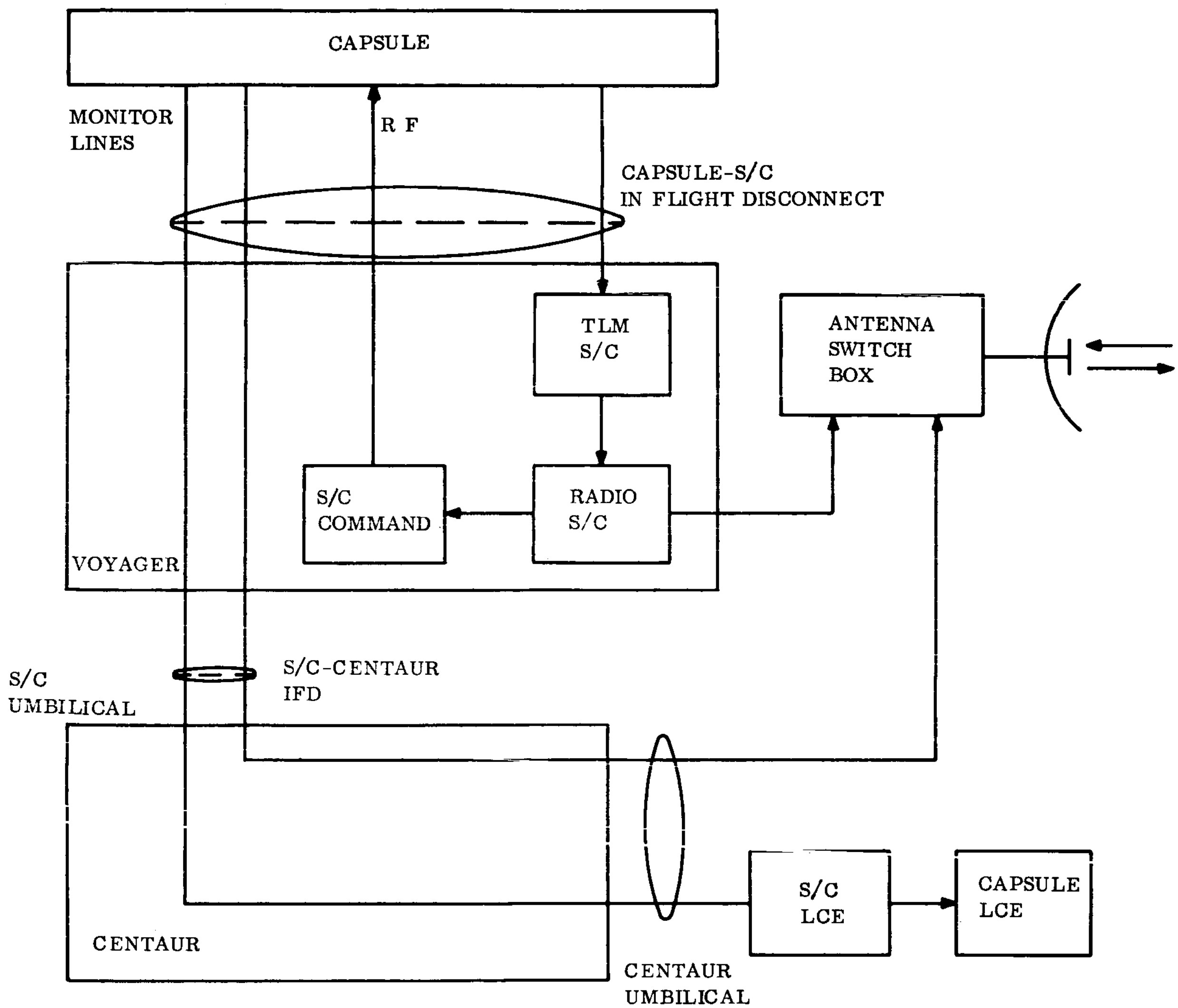


Figure 3-1. Capsule Interface with LCE, Block Diagram

4.0 INTERFACE DEFINITION

The interface between capsule OSE and LCE is not yet defined. To anticipate this interface, the following assumptions are made.

a. Two racks of test and monitor equipment will be required, located in the blockhouse.

b. All signals into and out of the capsule LCE will go to the S/C LCE which will provide the connecting link with the capsule.

c. Power required will be two circuits, 120-VAC, 60-cycle, 1-phase, 3-wire.

d. Any Telemetry reduction of capsule information will be done at the STC.

5.0 PERFORMANCE PARAMETERS

(To be determined.)

6.0 PHYSICAL CHARACTERISTICS

Two standard racks, with no special cooling requirements other than circulation of ambient air.

7.0 SAFETY

(To be determined.)

CII - VB280FD108

LAUNCH COMPLEX EQUIPMENT

CAPSULE PROVISIONS

Index

1.0 SCOPE

This document is an estimated functional description of the Capsule LCE for purposes of defining the interface with the rest of the LCE. The provisions which are to be made to accommodate capsule OSE in the LCE are defined.

2.0 APPLICABLE DOCUMENTS

VB280SR101	1971 Voyager Spacecraft LCE Test Objectives and Design Criteria
VB280SR102	1971 Voyager Spacecraft LCE Design Characteristics and Restraints Drawings

Drawings

SK5612-558	Functional Block Diagram LCE, Voyager 1971

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

At the launch complex the Capsule will already have been installed in the Spacecraft and will be completely inaccessible except for the umbilical interface between the capsule and the Spacecraft. Any external power required by the capsule will come from the S/C power bus. The only connection between the Capsule and the LCE will be the S/C umbilical.

For purposes of defining the interface at the pad, the following conditions are assumed:

a. Some lines from capsule umbilical go to S/C umbilical Control and Monitor lines.

b. Capsule signals go to S/C Telemetry.

Further, it is assumed that confidence tests will be made on the capsule. In order to perform these confidence tests, some capsule LCE will be required.

3.2 BLOCK DIAGRAM

Figure 3-1 shows the effect of the anticipated capsule/Spacecraft interface on the LCE.

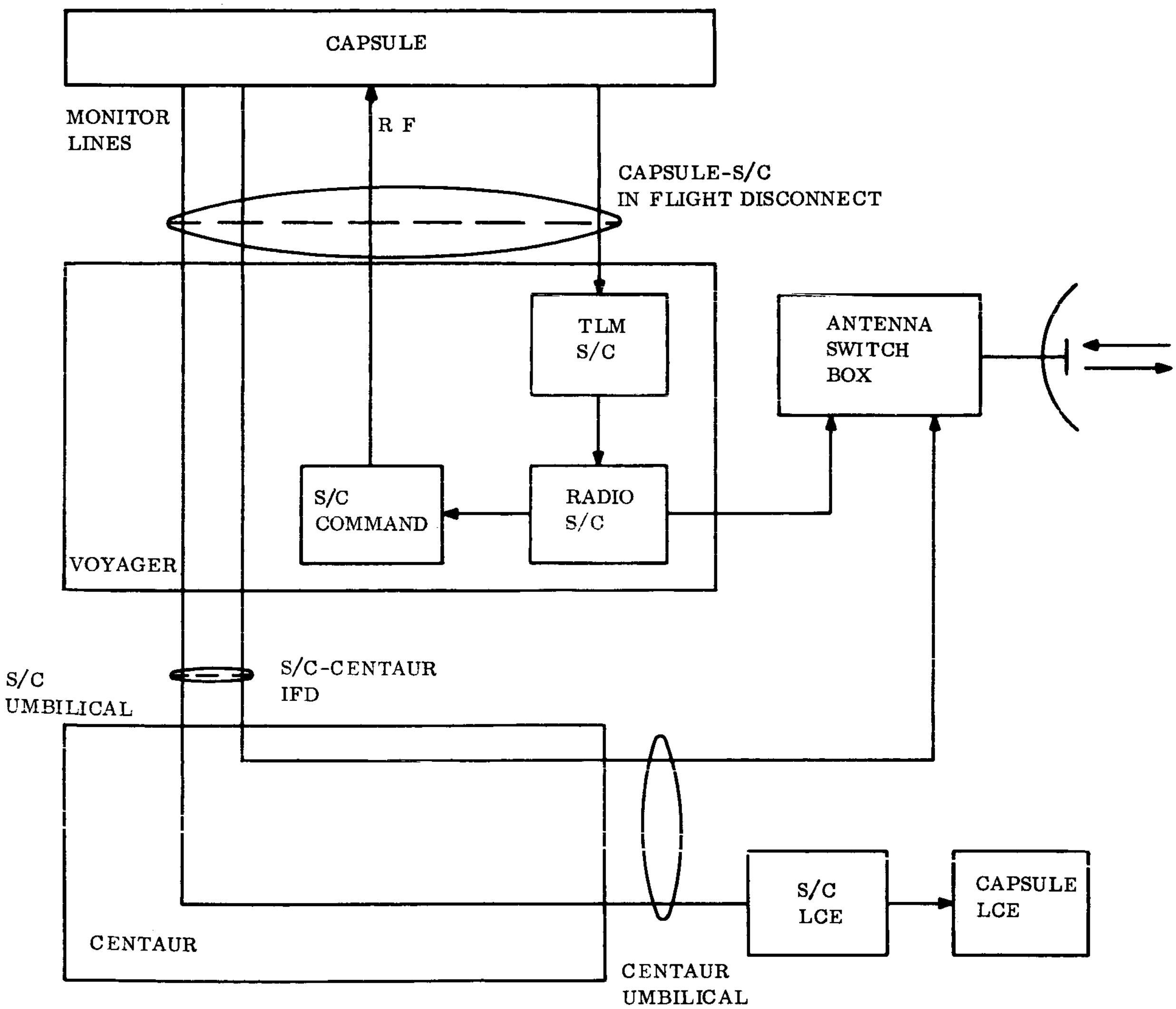


Figure 3-1. Capsule Interface with LCE, Block Diagram

4.0 INTERFACE DEFINITION

The interface between capsule OSE and LCE is not yet defined. To anticipate this interface, the following assumptions are made.

a. Two racks of test and monitor equipment will be required, located in the blockhouse.

b. All signals into and out of the capsule LCE will go to the S/C LCE which will provide the connecting link with the capsule.

c. Power required will be two circuits, 120-VAC, 60-cycle, 1-phase, 3-wire.

d. Any Telemetry reduction of capsule information will be done at the STC.

5.0 PERFORMANCE PARAMETERS

(To be determined.)

6.0 PHYSICAL CHARACTERISTICS

Two standard racks, with no special cooling requirements other than circulation of ambient air.

7.0 SAFETY

(To be determined.)

CII-VB280FD109

LAUNCH COMPLEX EQUIPMENT

PROPELLANT AND GAS LOADING EQUIPMENT

Index

1.0 SCOPE

This document is the functional description of propellant and pressurant servicing equipment for the Orbiter Separation Propulsion Assembly, and the Retropropulsion, Midcourse Propulsion and Attitude Control Subsystems. The use of this equipment is confined to the Explosive Safe Area. The equipment forms a part of the Launch Complex Equipment.

2.0 APPLICABLE DOCUMENTS

GE

VB238FD101	Voyager Propulsion Subsystem, Functional Description
VB280SR101	Voyager LCE Test Objectives and Design Criteria
VB280SR102	Voyager LCE Design Characteristics and Constraints

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

There are three Propellant Loading Units in the Propellant Loading OSE; a Retropropulsion Fuel Loading Unit, a Retropropulsion Oxidizer Loading Unit, and a Midcourse Propulsion Fuel Loading Unit.

Each Propellant Loading Unit consists of three physically separate items: A Propellant Transfer Assembly, a Control Console and a Pressurant Supply Cart. Block diagrams of this equipment are shown in Figures 3-1, 3-2, 3-3, and 3-4. Figure 3-5 is a schematic diagram of the fluid system of a Propellant Transfer Assembly.

3.2 RETROPROPULSION FUEL LOADING UNIT

The equipment will enable a UDMH-hydrazine fuel to be transferred from a Transfer Assembly, while under control of a Control Console in the vicinity of the spacecraft. Fluid lines and electrical cables are required to enable the Retropropulsion Fuel Loading Unit to transfer the UDMH-hydrazine fuel to the spacecraft retropropulsion tanks and to transfer pressurant gas to the spacecraft pressurant tanks. The operational functions which the equipment performs are:

a. Storage of UDMH-hydrazine fuel in self-contained tanks, and the delivery of accurate weight quantities of this fuel to the spacecraft Retropropulsion Subsystem.

b. Application of controlled gas pressure and vacuum to the spacecraft Retropropulsion Subsystem via a network of transfer lines for the purposes of purging, drying and pressurization of the Retropropulsion Subsystem, incident to the fuel and pressurant transfers.

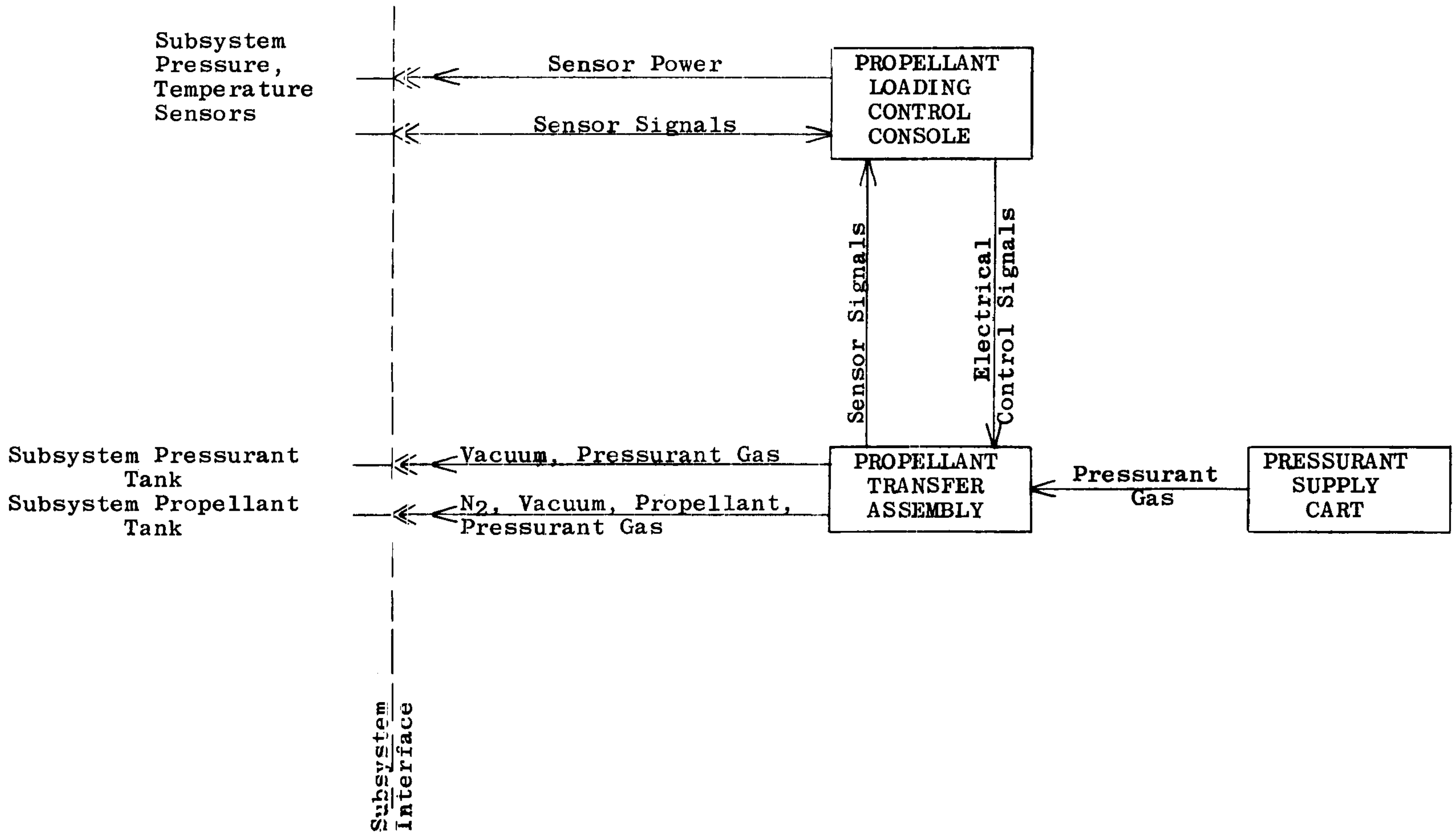


Figure 3-1. Functional Interfaces Between the Three Separable Items of a Propellant Loading OSE Unit and the Propulsion Subsystem

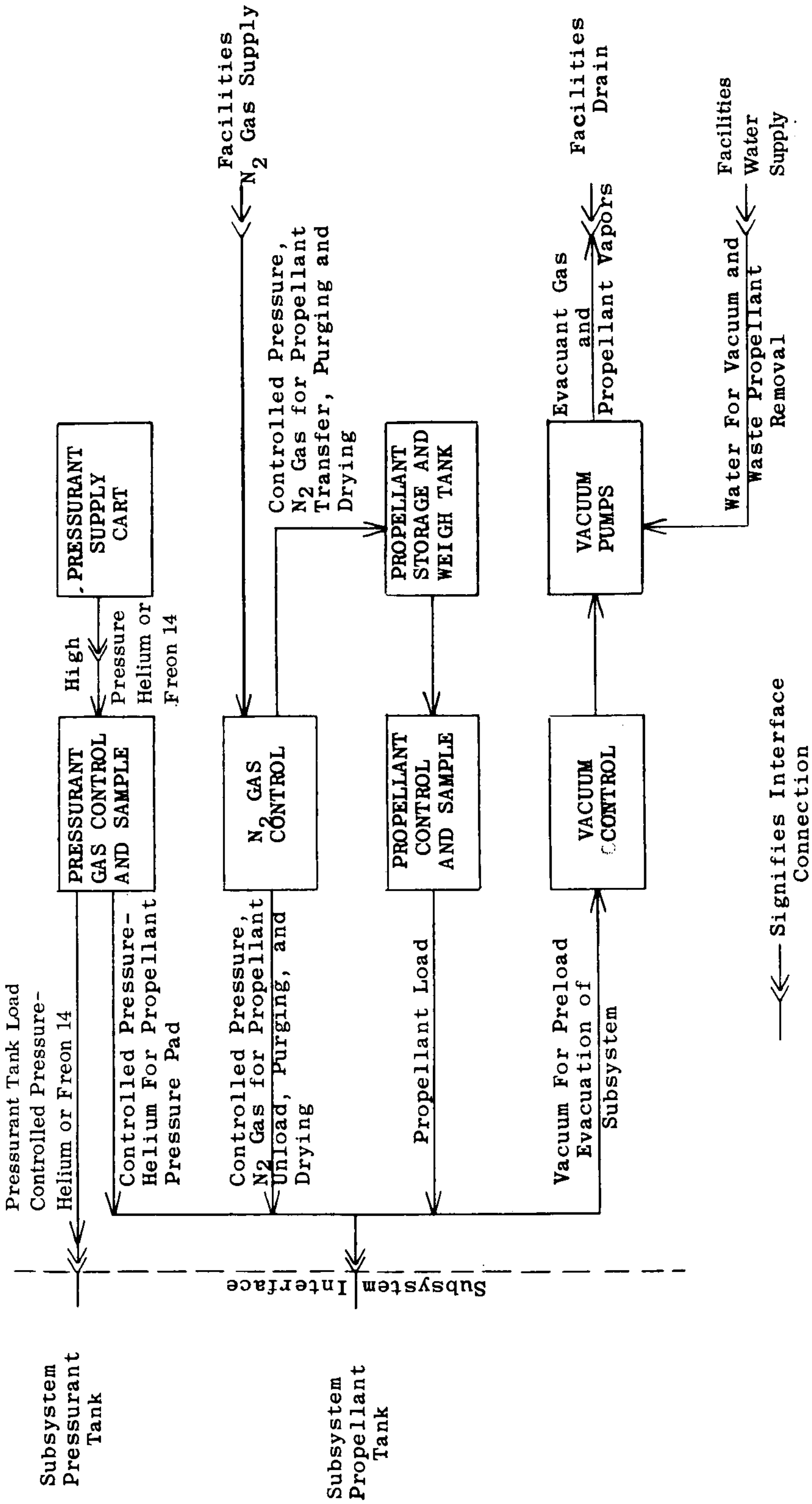


Figure 3-2. Functional Diagram Propellant Transfer Assembly

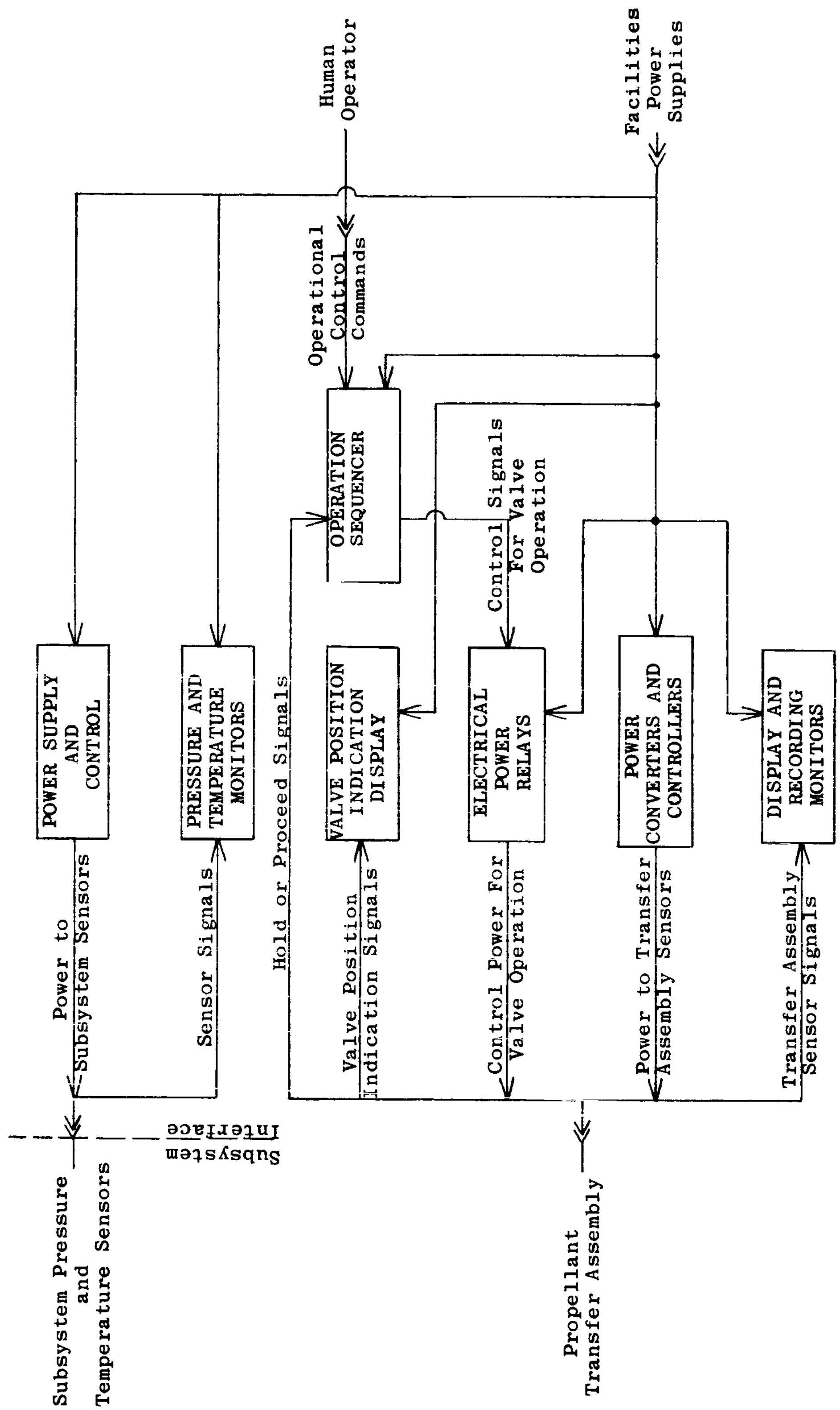


Figure 3-3. Propellant Transfer Control Console, Functional Diagram

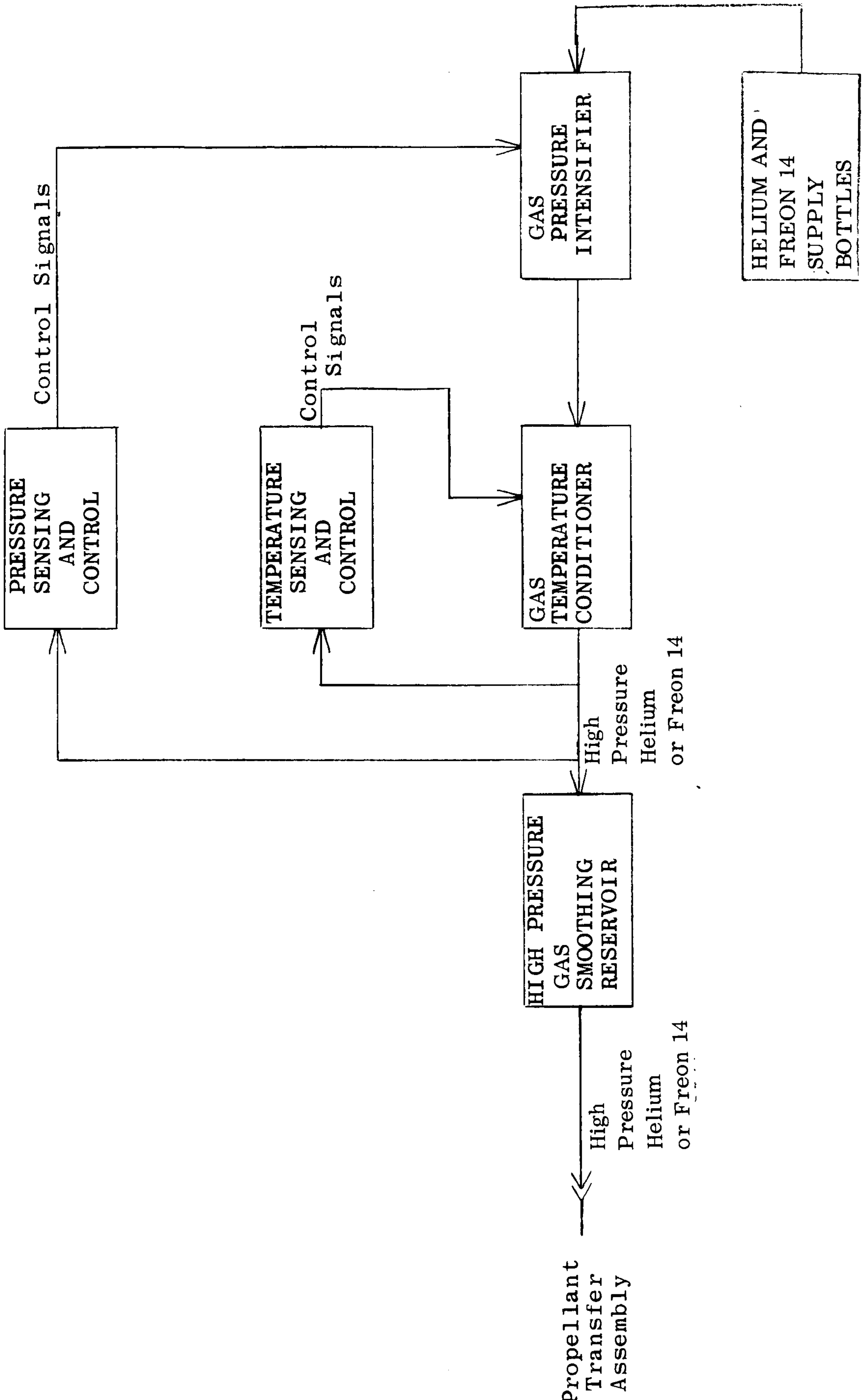


Figure 3-4. Pressurant Supply Cart, Functional Diagram

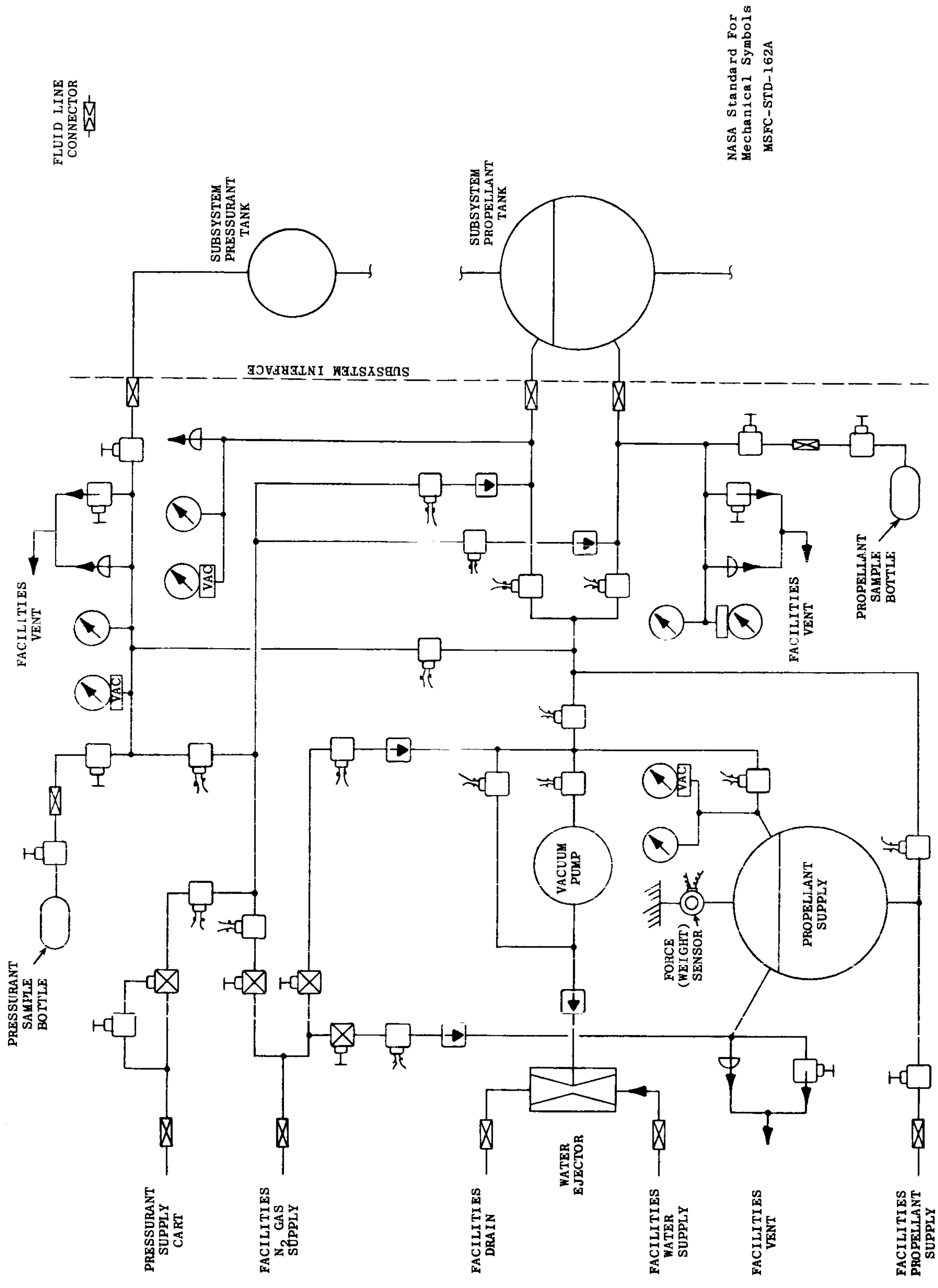


Figure 3-5. Propellant Transfer Assembly, Schematic Diagram

c. Sampling the transferred hydrazine-blend fuel and pressurant gas, for analysis.

d. Control and display of the above, so organized that it can be accomplished from a single location.

3.3 RETROPROPULSION OXIDIZER LOADING UNIT

The equipment is physically and functionally similar to the Retropropulsion Fuel Loading Unit, but it will service the Retropropulsion Subsystem with N_2O_4 oxidizer instead of UDMH-hydrazine fuel.

3.4 MIDCOURSE PROPULSION FUEL LOADING UNIT

The equipment is physically and functionally similar to the Retropropulsion Fuel Loading Unit, but it will service the Midcourse Propulsion Subsystem with unblended N_2H_4 instead of UDMH-hydrazine blend fuel.

3.5 ATTITUDE CONTROL CF_4 LOADING EQUIPMENT

This equipment consists of any one of the three Propellant Loading Units, supplemented by bottles of pressurized Freon 14 gas.

The equipment will enable Freon 14 gas to be transferred from a Transfer Assembly to the tanks of the spacecraft Attitude Control Subsystem.

3.6 ORBITER SEPARATION PROPULSION ASSEMBLY-N_2 LOADING EQUIPMENT

This equipment consists of any one of the three Propellant Loading Units, modified by the addition of pressure relief capability for protection of the Separation Assembly tanks. Also, no pressurant supply cart is required. This equipment will enable N_2 gas to be transferred from a Transfer Assembly to the tanks of the subject spacecraft assembly.

4.0 INTERFACE DEFINITION

The following interface definitions apply equally to each of the three Propellant Loading Units:

a. Liquid and Gas Transfer Interfaces

1. Propellant Transfer Assembly

Component	Purpose	Characteristics
Facilities N_2 supply	N_2 gas replenishment	Flexible hose

1. Propellant Transfer Assembly (Continued)

Component	Purpose	Characteristics
Facilities H_2O supply	Produce vacuum; Entrain waste propellant	Flexible hose
Facilities H_2O Drain	Dispose of water and waste propellant	Flexible hose
Midcourse and Retro-propulsion Propellant tanks	Propellant loading of subsystems	Flexible hose
Midcourse and Retro-propulsion Pressurant tanks	Pressurant loading of subsystems	Flexible hose
Pressurant Supply Cart	Pressurant supply to Transfer Assembly	Flexible hose
Facilities propellant supply tank	Obtain propellant used for subsystem load	Flexible hose
Facilitics gas vents	Dispose vented gas	Flexible hose
Propulsion Subsystem	Exterior surface of tanks and lines for leak testing	Fleible plastic tubing
Attitude Control and Orbiter Separation Propulsion Assembly Cold gas tanks	Gas loading of subsystems	Flexible hose

2. Pressurant Supply Cart

Component	Purpose	Characteristics
Facilities helium supply bottles	Replace depleted helium supply	Eight cylindrical gas tanks
Facilities Freon 14 supply bottles	Replace depleted Freon 14 supply	Four cylindrical gas tanks

2. Pressurant Supply Cart (Continued)

Component	Purpose	Characteristics
Facilities N_2 gas supply line	N_2 replenishment	Flexible hose
Facilities gas vents	Dispose vented gas	Flexible hose

b. Electrical Interfaces

1. Control Console

Component	Purpose	Characteristics
Facilities power	120 VAC 60C 1 Φ supply	Armored flexible cable
Facilities power	28 VDC supply	Flexible cable
Facilities ground	Electrical ground access	Flexible braided strap
Propellant Transfer Assembly	28 VDC Control power signals	Flexible cable
Propellant Transfer Assembly	Sensor output signals	Flexible cable of shielded conductors
Propellant Transfer Assembly	120 VAC 60C 1 Φ Control power signals	Armored flexible cable

2. Propellant Transfer Assembly

Component	Purpose	Characteristics
Facilities power	120 VAC 60C 1 Φ supply	Armored flexible cable
Facilities power	240 VAC 60C 3 Φ supply	Armored flexible cable
Facilities power	28 VDC supply	Flexible cable
Facilities ground	Electrical ground access	Flexible braided strap

3. Pressurant Supply Cart

Component	Purpose	Characteristics
Facilities power	120 VAC 60C 1Φ supply	Armored flexible cable
Facilities power	250 VAC 60C 3 Φ supply	Armored flexible cable
Facilities power	Electrical ground access	Flexible braided strap

5.0 PERFORMANCE PARAMETERS

5.1 PROPELLANT LOAD PARAMETERS

Performance parameters of propellant loading with this equipment are summarized in Table 5-1 of this document.

5.1.1 PROPELLANT LOADING ACCURACY

The propellant loading OSE must be capable of high degree of weight accuracy in loading propellants, with capacity for supplying and measuring a wide load range. The accuracy is important in minimizing the imbalance between the two tanks, thus reducing TVC requirements. Confidence of high loading accuracy also permits lowest S/C weight because the specified propellant loads can approach the ideal mission requirements. With a high accuracy in propellant loading, the center of gravity of the launch-ready S/C need not be relocated by test after propellant loading, which relieves the program of that difficult task and its statistical risk to facilities, personnel and the S/C. A high accuracy in propellant loading also may be the manner in which the highest total S/C weight accuracy can be obtained. Again, this avoids the risk and difficulty of weighing the S/C after propellant loading.

The high loading accuracy capability of this equipment depends primarily on the use of a linear, sensitive and stable force sensor (or load cell) and its associated electronics for weight measurement. One hundred-twenty percent of the maximum retropropulsion subsystem oxidizer load, plus sufficient expulsion gas for the oxidizer transfer, can be contained in one 4-foot diameter spherical tank in the Retropropulsion Oxidizer Transfer Assembly. The weight of tank plus oxidizer is 2350 pounds. The variable inductance load cell supporting the tank has an inherent error in repeatibility of force measurement of 0.04% of full scale. Deadweight calibration of the load cell plus its excitation and readout electronics can be held to 0.03% full scale error for any preselected propellant load up to 110% of the maximum mission load. Volume uncertainties in the transfer line will not exceed 0.02% of the load. The total oxidizer loading error is then (approximately) equal to (0.0009)(2350), or ± 2.1 pounds. Fuel loading error is similarly determined at ± 1.5 pounds for the retropropulsion subsystem and ± 0.4 pound for the MC propulsion subsystem. The total possible combined propellant load error is thus ± 4 pounds. It can be reduced to little more than the inherent error of the load cell or ± 1.8 pounds, by using Class P calibrating weights, by adjusting the

Table 5-1. Propellant Load Parameters

Applicable Subsystem	Propellant	Maximum Subsystem Propellant Load Required (LB)	OSE Propellant Load Range Capability (LB)	OSE Propellant Loading Error (±LB)	OSE Propellant Verification Sample (LB)	OSE Pre-Load Vacuum Pump Displacement (ft^3/min)	OSE Pre-Load Vacuum Shutoff Capability (TORR)	OSE Post-Load Pressurant	OSE Post-Load Pressurizing Error (±PSIG)	OSE Propellant Temperature Measurement Accuracy (±° f)	OSE Propellant Expulsion Gas Pressure (PSIG)	Propellant Total Loading Time Per Subsystem Tank (Hours)
Retro-Propulsion Fuel	50-50 UDMH N_2H_4	1100	0. to 1250	1.5	.4	17	1.0	N_2 Or He	1.0	1.0	100 at START 40 at FINISH	6
Retro-Propulsion Oxidizer	N_2O_4	1620	0. to 1800	2.1	.6	17	1.0	N_2 Or He	1.0	1.0	100 at START 40 at FINISH	6
Mid Course Propulsion Fuel	N_2H_4	375	0. to 425	.4	.4	17	1.0	N_2 Or He	1.0	1.0	100 at START 40 at FINISH	5

OSE supply of propellant to ambient temperature before transferring to the S/C so that line volumes will remain constant, and by special selection and training of the operating crew so that these men have a history of working together, of familiarity with the OSE, and several successful transfers of mission equivalent loads of pressurants and propellants into a Voyager equivalent tankage system.

Error of load of the midcourse propulsion fuel is (by total weight error) 10% of the total weight error of loading the retropropulsion propellants, or ± 0.4 pound. If exceptional accuracy is designed for, the error can be reduced to ± 0.2 pound.

Immediately after propellant transfer and under the identical conditions, standardized dead weights are applied in steps to the load cell. Its response to weight changes is recorded and compared to the pre-transfer calibration for verification of the accuracy of the transferred propellant load.

The load cell readout is given a resolution of approximately 0.01% of fuel load with an electronics counter. The counter reading would be repeated on digital tape printer for verification record.

5.2 PRESSURANT LOAD PARAMETERS

Performance parameters of this equipment in loading gases for propellant pressurizing, Attitude Control and Orbiter Separation are summarized in Table 5-2.

5.2.1 PRESSURANT LOAD ACCURACY

The total weight of propellant pressurants of the Voyager propulsion subsystem will be approximately 12.5 pounds. This gas will be continuously monitored during loading for pressure and temperature, using the output of the propulsion subsystem pressurant tank sensors to drive OSE instruments. The combination of these subsystem sensors and OSE readouts will have been calibrated together with calibration accuracy of ± 5°F and ± 10 psig, and will yield a total potential loading error (in predetermined volume Voyager pressurant tanks) of approximately ± 0.15 pound by weight of pressurant gas.

Error in loading the Freon 14 gas (CF_4) for attitude control is similarly determined to be approximately ± 0.3 pound per tank for each of the two subsystem storage tanks. This error has approximately the same significance in shifting of the spacecraft center of gravity as the error in loading midcourse propulsion fuel. The error is primarily due to the expected use during gas transfer of the subsystems tank temperature transducers for a measurement of the transferred gas temperature. These subsystem transducers are expected to have readout errors of about ± 5°F.

An alternate approach would be to use OSE sensors instead of subsystem sensors for measuring subsystem pressurant tank pressure and temperature. The higher accuracy possible would result in a reduction in pressurant loading error to approximately 0.2%

Table 5-2. Pressurant Loading Parameters

Applicable Subsystem	Pressurant	Maximum Subsystem Pressurant Load Required (LB)	OSE Pressurant Storage Capacity (LB)	OSE Pressurant Loading Error (±LB)	OSE Pre-Load Vacuum Shutoff Capability (TORR)	OSE Pressurant Load Pressure Range Capability (PSIG)	Pressurant Load Pressure Error (±PSIG)	Pressurant Verification Sample (Bottle Volume-ft^3)	OSE Temperature Control Range of Pressurant (°F)	Assumed Accuracy of Subsystem Pressurant Tank Temperature Measurement (±°F)	Pressurant Total Loading Time Per Subsystem Tank (Hrs)
Retro-Propulsion	Helium	11.3	15+	.15	1.0	100 to 4500	10	.01	65 to 100	5	2
Mid-Course Propulsion	Helium	1.25	15+	.02	1.0	100 to 4500	10	.01	65 to 100	5	1.5
Guidance and Control (Attitude Control)	CF_4	56	70	.6	1.0	100 to 4500	10	.01	65 to 100	5	2
Orbiter Separation Propulsion Assembly	N_2	.52	–	.007	1.0	20 to 1000	2	.01	None	5	2

of load from approximately 1% of load, but requires that the subsystem pressurant tank be accessible for the temporary surface attachment (by contact adhesive) of an OSE temperature sensor. Such accessibility is not presently considered practical.

5.3 OSE VACUUM

The vacuum system of each Propellant Transfer Assembly consists of a water ejector, backing up two stages of rotary dry mechanical pumps, all materials compatible with N_2O_4 and the hydrazines. This vacuum system has been operationally proven for use in equipment similar to the Transfer Assemblies recommended for Voyager.

Vacuum is important for purging, creating differential pressures to cause propellant flow, and for drying after an unscheduled unloading of propulsion subystem propellant. Its primary need, however, is in exhausting the subsystem propellant tanks prior to propellant loading. This makes the loading possible without a vent, avoids high differential pressures on the Midcourse Subsystem tank bladders, and reduces trapped gas to a minimum in the propellant section of a Midcourse Subsystem tank. The vacuum capability of this equipment will reduce the propellant tank pressure to two torr before propellant loading, and will result in approximately five cubic inches of gas at 45 psia trapped with the propellant in each tank after loading. This amount of trapped gas is not considered excessive in this case, but it could be reduced to about 0.5 cubic inch by a 90% collapse of the subsystem tank bladder prior to transfer of propellant if it should become a requirement.

5.4 PROPELLANT AND PRESSURANT LOAD TIME

Each of the four Retropropulsion Subsystem propellant tanks will require approximately six hours for propellant loading by this OSE, including the time for fluid and electrical lines connections, OSE pre-load weighing calibration, OSE vacuum evacuation of the subsystem tank, transferring the propellant, applying gas pressure to the loaded propellant, OSE post-load weighing calibration, sealing of subsystem lines and disconnection from the OSE.

Fuel transfer for the midcourse propulsion subsystem will require approximately five hours per tank for the same operations, due to the smaller tank capacities.

Pressurant tanks of the spacecraft need not be evacuated more than ten minutes per tank prior to loading with pressurant gas. Pressurant transfer time per Retropropulsion subsystem tank will require about two hours, because the gas pressure must be intensified from OSE storage pressure of less than 1000 psi to the subsystem pressurant tank pressure, and the increased temperature due to this compression reduced to ambient by OSE cooling. As the desired subsystem tank load is approached, pressurant transfer will be reduced to a low flow rate in order to allow time for the subsystem tank walls and transferred pressurant gas to reach a nearly equal temperature. This is necessary, because the measured tank wall temperature is used as transferred pressurant temperature in calculating transferred weight of pressurant. The total time required for pressurant loading of the Midcourse and Retropropulsion Subsystems is not expected to exceed 18 hours. Time required for loading the two Attitude Control Freon 14 tanks will probably not exceed eight hours, and about four hours will be needed for loading the N_2 gas tanks of the Orbiter Separation Propulsion Assembly.

The sum of propellant and pressurant loading time requirements for the entire propulsion subsystem and attitude control gas is, therefore, approximately 74 hours. This might be reduced to a minimum time limit of approximately 50 hours by moderate changes in the OSE and in the load procedure, which normally limits fluid line connections to one subsystem tank at any given time. The OSE operators would also require more loading rehearsals to obtain an efficiency necessary to support the shorter loading time cycle.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

Each of the Retropropulsion Subsystem Propellant Transfer Assemblies will weigh approximately 4500 pounds, including propellant supply. Dimensions will be confined to an envelope five feet wide, seven feet high, and eight feet long. Casters, brakes and built-in jacks will allow towing, turning, and level positioning of the assembly. The four side faces of the assembly will be accessible for servicing and making connections. Fork lift slots in the base will be provided for handling ease during transport.

The Midcourse Propulsion Propellant Transfer Assembly will weigh approximately 1500 pounds, including propellant supply, and will be confined to an envelope four feet wide, 6-1/2 feet high and eight feet long. The propellant fuel can be stored in a three-foot diameter service tank. Except for the smaller noted dimensions, this Midcourse Propulsion Propellant Transfer Assembly will be essentially identical to the two Retropropulsion Propellant Transfer Assemblies.

Each of the control consoles will weigh approximately 1200 pounds complete, and will be confined to a closed envelope 3-1/2 feet wide, 6-1/2 feet high and eight feet long. Casters and fork lift base slots will be provided for easy handling and positioning. Service access to the sides and rear will be provided by removable panels. All propellant and pressurant transfer controls, indicators and monitoring instruments will be mounted on the front panels of the enclosure, and arrayed conveniently for the operator's use.

The pressurant supply cart will weigh about 3000 pounds. The design should include space for pressure bottles for containment of 25 pounds, by weight, of helium gas at an initial pressure of 2000 psig, or for pressure bottles containing 100 pounds of CF_4 at an initial pressure of 2000 psig.

An American Instrument Company pressure intensifier pump accepts the output flow from these bottles and (even when the bottled gas pressure decays to below 1000 psi) the pump output capability will be two pounds of helium gas per hour at 5000 psig pressure, or 35 pounds/hour of CF_4 at 3000 psig. The pump output will be stored in a 0.5-cubic foot volume tank in the cart. A line from the 0.5-ft^3 volume storage tank is fitted with a shutoff valve and terminated with a connector for attaching a connecting line between the Pressurant Supply Cart and the Propellant Transfer Assembly. An electric heater and also a refrigeration unit will allow an adjustment of the temperature of the pressurant gas in the line leaving the 0.5-ft^3 tank to a predetermined temperature level, and will be controlled by a temperature controller.

A bulkhead connector will permit connection of a facilities pressurized N_2 gas supply line to the Pressurant Supply Cart. The N_2 gas can be valved into the intensifier pump instead of helium or CF_4 whenever desired by hand operated valves, but the N_2 supply line to the cart is disconnected when helium or CF_4 are used to prevent any possibility of mixing.

Facilities requirements of the Propellant Loading OSE are as follows:

a. Propellant Transfer Assembly

1. 110 VAC 60C 1 Φ 20a
2. 240 VAC 60C 3 Φ 40a
3. 28 VDC 40a
4. 80 gpm water at 100 psig and 60°F max. temperature, filtered to 100 microns
5. Drain for item (4) water
6. Facilities ground
7. Facilities vent stack

b. Control Console

1. 110 VAC 60C 1 Φ 30a
2. 28 VDC 60a
3. Facilities ground

c. Pressurant Supply Cart

1. 110 VAC 60C 1 Φ 30a
2. 240 VAC 60C 3 Φ 20a
3. Facilities ground

4. Facilities N_2 gas, 100 SCFM at 1500 psig

5. Facilities vent stack

7.0 SAFETY PROVISIONS

7.1 PRESSURE SAFETY

Each section of the subsystem would be protected against overpressurization by the OSE with OSE burst discs rated for burst at 120% of the working pressure of the subsystem section, and a relief venting valve which is adjustable to any desired relief pressure, but nominally set for operation at 110% of normal working pressure.

At pressures above 80% of final load pressure in a Voyager pressurant tank being loaded, an OSE safety device sensitive to the differential temperature between the transferred pressurant gas and ambient air will cause an OSE valve to close and stop the transfer if the Voyager pressurant temperature falls to 10°F less than ambient. This is to prevent the possibility of overpressures after loading, as the cold pressurant adjusts to ambient temperature.

All pressurized components of the OSE should be designed with a safety factor of at least four, between maximum working pressure and design burst pressure. The OSE should be proof tested to 2.5 times the maximum working pressure.

All mechanical pressure gauges should be of closed front design, where the case casting of the instrument prevents blowout in the observer's direction. A thin blowout plug or patch on the rear of the case would allow gas to escape in case of failure.

All tankage of the OSE should be fitted with burst discs and relief valves for operator safety. Each combination of a burst disc and relief valve would be connected to a facilities vent stack for safe removal of released gases from the area.

7.2 TOXICITY SAFETY FOR PERSONNEL

Operators using the Propellant Loading OSE to load propellants will require the following facilities equipment:

a. Propellant handler's safety suits and breathing apparatus.

b. Toxicity level indicators for propellant vapor in the room air.

c. Room air circulating and purifying system.

d. Safety showers for operating personnel.

e. Water deluge system to reduce fire hazard, in case of propellant spill.

CII - VB280FD110

LAUNCH COMPLEX EQUIPMENT

LEAK TEST EQUIPMENT

Index

1.0 SCOPE

This document describes the equipment used at the Explosive Safe Area for leak testing the propellant and pressurized gas portions of the spacecraft subsystems. The subsystems to be leak tested are the Retro-propulsion, Midcourse, Attitude Control, and the Orbiter Separation Propulsion Assembly. The equipment includes a portable enclosure containing a gas fluid system and a console face equipped for control and monitoring.

2.0 APPLICABLE DOCUMENTS

GE

VB280SR101 1971 Voyager S/C Launch Complex Equipment Design Characteristic and Restraints

3.0 FUNCTIONAL DESCRIPTION

The Leak Test OSE will be housed in a portable cabinet with control capability available at one console face. The operational functions are:

a. Proof pressure test of spacecraft subsystem liquid propellant and gas tankage by internal gas pressurization, to satisfy that these tanks will not rupture under normal working pressures.

b. Testing of in-line valves, regulators, diaphragms and bladders for through-leaks.

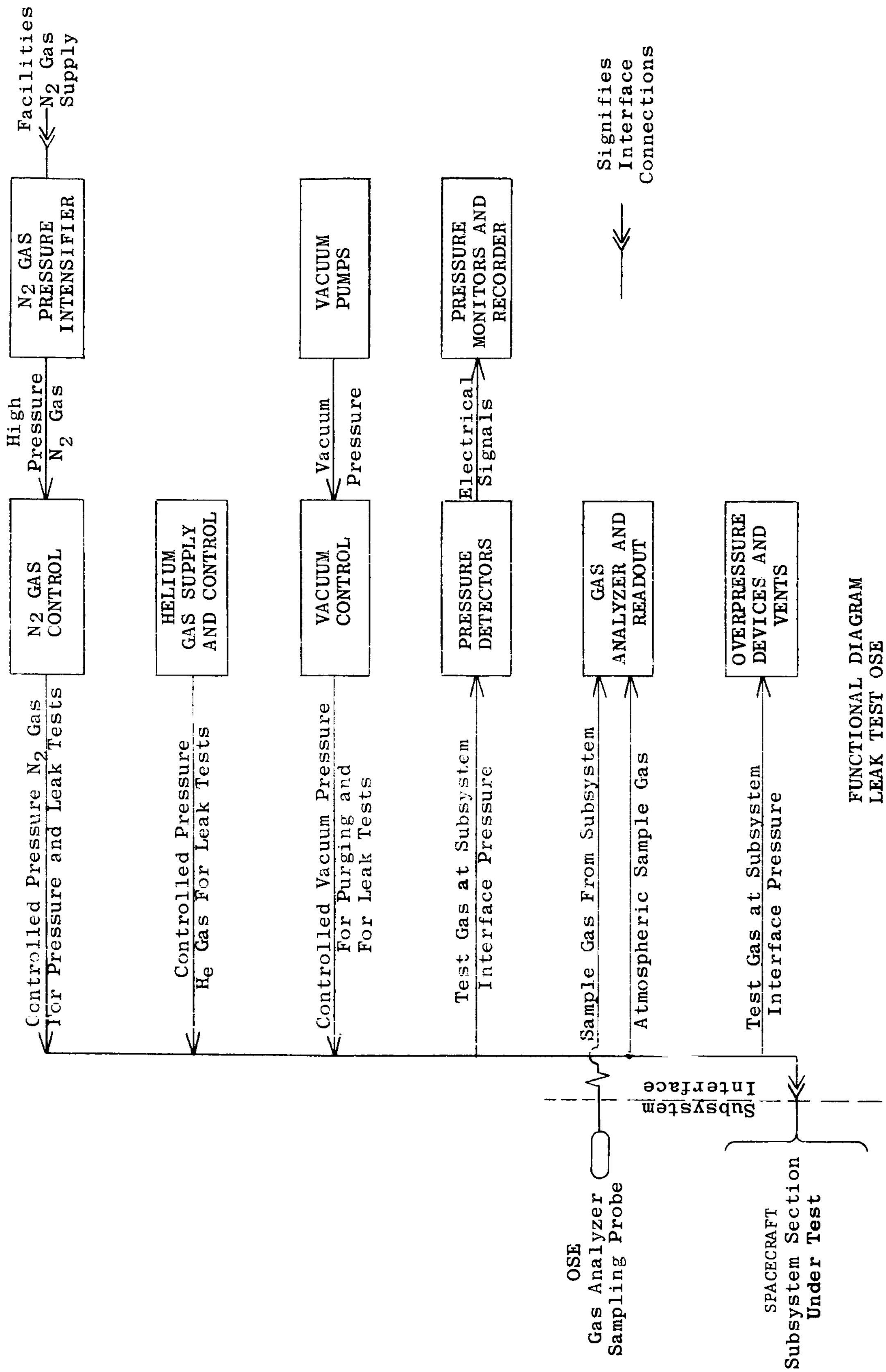
c. Testing for external leaks over all of the pressurized portions of the subsystems before propellant and pressurant loading. The OSE will also have limited monitoring capability for sensing external propellant leaks during storage of the spacecraft with propellants and pressurants loaded.

d. The leak test capability of this equipment applies to the Midcourse Propulsion, Retropropulsion, Attitude Control and Orbiter Separation Propulsion subassemblies.

e. This OSE would have a self-contained helium supply and control of facilities N_2 gas supply for independence from other OSE in performance of its leak testing of the spacecraft subsystems.

f. Response measurements of spacecraft subsystem pressure transducers may be performed by this OSE with readout error no greater than ± 1% over the range of measurement.

A functional block diagram of the Leak Test equipment is given in Figure 3-1. Figure 3-2 is a schematic diagram of the fluid system developed to satisfy the functional service requirements of this equipment.



FUNCTIONAL DIAGRAM
LEAK TEST OSE

FIGURE 1

Figure 3-1. Leak Test OSE, Functional Diagram

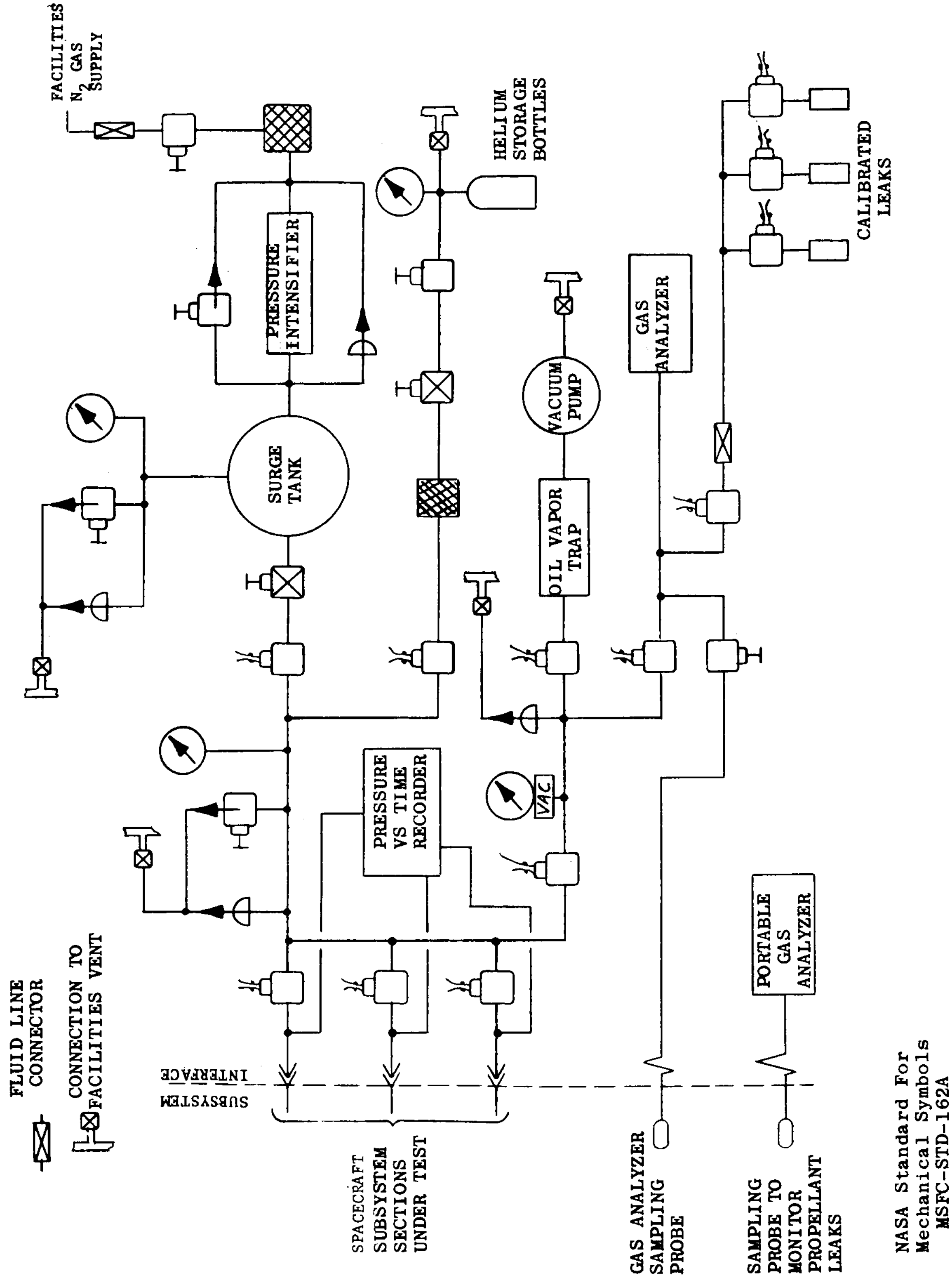


Figure 3-2. Leak Test OSE Fluid System, Schematic Diagram

4.0 INTERFACE DEFINITION

4.1 PNEUMATIC

Pneumatic line interfaces exist between this OSE and each of the applicable spacecraft subsystems. Subsystem connectors must allow application of vacuum and pressurant gas to all subsystem tanks, and on both sides of all diaphragms, bladders, valves or valve matrices, pressure regulators and in-line burst seals.

4.2 ELECTRICAL

Electrical interfaces exist between this OSE and each of the applicable spacecraft subsystems, since hardwire connections will be required between subsystem pressure and temperature sensors and this OSE, for use in calibrating and monitoring the sensors.

4.3 MECHANICAL

Interface must exist for the contact or near approach of OSE leak detection sensors to the external surfaces of spacecraft subsystem lines and tanks.

5.0 PERFORMANCE PARAMETERS

The leak test OSE will have the following performance capabilities:

a. PROOF PRESSURE - Filters, driers, pressure regulators, an intensifier pump, valves and instrumentation of the OSE will condition facilities N_2 gas received at 1500 psig and deliver it with separate hoses to any three connectors of a given spacecraft subsystem interface. Pressure in the delivery hoses may be separately preset and automatically controlled to any value from 100 psig to 4500 psig. After establishing the desired proof pressure in sections of the subsystem under test, these sections will be isolated from further gas input by OSE valves. Small ruptures will be detected by pressure decay in the isolated sections as indicated on OSE pressure gauges.

b. Through-leak test capability by this OSE is primarily for diagnostic purposes and to differentiate between external leaks and leaks through normally gas tight in-line components of the subsystems. It is performed by evacuating the normal low pressure side of the subsystem compoent under test to a pressure of 10^{-3} torr or lower with vacuum pumps of this OSE. Helium enriched nitrogen is then introduced to the high pressure side of the tested subsystem component until normal working differential pressure is established, and the vacuum pump inlet sampled with an OSE gas analyzer to detect leaking helium. Leak rate is determined by isolating the evacuated section with an OSE valve and measuring the rate-of-rise of pressure in the isolated section with the use of an OSE timer and vacuum gauge.

c. Small external leaks of a tested subsystem will be detected, located and measured with OSE gas analyzer probes held against or near the outside surfaces of the subsystem tanks and lines after internal pressurization of the subsystem to normal working pressure with helium enriched nitrogen, supplied by and controlled from the leak test OSE console cabinet. Leak rates lower than 10 scc/hr may be determined by comparing the gas analyzer response to the leak with the response to a known (or calibrated) leak.

External leakage rate from small volume sections of the subsystems may also be determined by pressure decay rate measured with OSE instruments after isolating the leaking section, but determining the rate of external leakage from large volumes such as the retropropulsion pressurant or propellant tanks is not practical by pressure decay measurement, because the test time requirement is too great for this type of test at the ESA.

The spacecraft subsystems will be comprehensively proof-pressure and leak tested at least three times, the final test to be performed in the ESA at the Eastern Test Range. All of the leak testing will be performed without the use of liquids, because gases allow easier, faster and more accurate analysis of small leaks. Also, liquids can plug small leaks so that they might not be detectable. Records from this series of tests will be compared to obtain a knowledge of the leak characteristics of the subsystems which would not otherwise be possible, such as the incidence and size of new leaks, and the trend or rate of increase of existant leaks. Time extrapolation of this data may be made to forecast the integrity of the subsystems during the mission.

During and after propellant loading of the propulsion subsystem, gas analyzers of this OSE will be used to "sniff" for traces of propellant issuing from external subsystem leaks, with the use of a probe and flexible tube.

d. Electrical and/or electronic circuits, power supplies and instrumentation within the OSE console will be used in conjunction with the gas pressurizing capability to test the response of subsystem pressure transducers. The output of these subsystem pressure sensors will also be utilized by pressure monitors of this OSE where practical during leak testing.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Leak Test OSE cabinet will weigh approximately 1200 pounds. Its dimensions may be confined to four-foot width, six-foot height and eight-foot length. Casters allow hand towing and turning the console cabinet. Transverse slots in the base are convenient for handling by fork truck. The cabinet will have internal access from three sides, but all control will be performed from the console face.

a. 110 VAC, 60cps, 80 ampere, and 28 VDC, 40 ampere regulated power inputs to the OSE are required.

b. Facilities N_2 gas input to the OSE at 1500 psig is required.

c. Helium supply bottles within the OSE cabinet require recharging before use from facilities supplies at 2200 psig.

d. All pressure monitoring instruments on the OSE console face will have a maximum repeatability error of 1/750th of full scale. The OSE cabinet may not be used in contact with a heavily vibrating floor or other structure during use.

e. This OSE will be made for operation in areas shaded from the sun, in air atmosphere below 95% relative humidity, and temperature range of 40°F to 100°F.

f. All probes, electrical cabling and hoses will be storable within the OSE cabinet. A portable gas analyzer with probe for propellant leak detection should be provided, with storage space alloted in the OSE cabinet.

g. Calibrated leaks will be utilized as a part of the OSE to determine that analyzer probes and equipment are functioning properly, and for comparison with detected spacecraft subsystem leaks as a primary method of establishing external leak rates.

7.0 SAFETY CONSIDERATIONS

The proof-pressure testing with gas incurs the possibility of explosive fragmentation of subsystem tanks. Proof-pressure testing is necessary for assurance that the subsystems will not fail during subsequent propellant and pressurant loading, but hydrostatic proof testing is to be avoided because of difficulty in unloading the subsystem tanks and the inevitable plugging of small gas leaks by the test liquid. Where possible during this critical test, one tank of the subsystem will be pressurized at a time in order to hold explosive energy to a minimum. Also, a steel or concrete baffle must be placed so as to protect personnel from the concussion and flying fragments of a bursting tank.

Safety burst discs and pressure relief valves will be placed on all OSE lines used for pressurizing the spacecraft subsystems, and will be sized to prevent inadvertant overpressures. Similar safety devices will also be connected to the OSE helium bottle manifold to prevent overpressurizing these bottles.

The subsystems will be proof-pressure tested prior to final assembly to the S/C. After proof-pressure test, subsystem tankage is considered safe for personnel to work around when pressurized to normal working pressures. However, facilities propellant safety suits are required for leak test personnel during propellant loading, and the air toxicity level must be determined from facilities indicators before approaching the loaded S/C for leak tests at any later time.

CII - VB280FD111

LAUNCH COMPLEX EQUIPMENT

EXPLOSIVE SAFE AREA

PYROTECHNICS TEST KIT

Index

1.0 SCOPE

This document is a functional description of the requirements of the pyrotechnic subsystem OSE to support operations in the Explosive Safe Area. This equipment will be used to prepare the pyrotechnic subsystem for installation of explosive devices, verify its safety, and check the subsystem after pyrotechnics are installed.

2.0 APPLICABLE DOCUMENTS

GE	
VB235FD104	Functional Description of the Pyrotechnic Subsystem
VB280SR101	1971 Voyager Spacecraft LCE Test Objectives and Design Criteria
VB280SR102	1971 Voyager Spacecraft LCE Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

The Pyrotechnic Subsystem support equipment provided for the ESA should be a self-powered portable unit. It will be used to condition the spacecraft for installation of pyros and to check the spacecraft after pyros are installed. It is required to perform self tests, test the pyro-controller and test the spacecraft harness.

These tests consist of the following:

a. Pyro-controller Tests

- Measure Voltage on Capacitors
- Discharge Capacitor Bank
- Check for Stray Voltage on Firing Lines
- Set Safe/Arm Relays
- Provide Required Preset Signals
- Indicate Relay Positions

b. Harness Resistance Tests

- Determine if squib firing lines are shorted or grounded.
- Measure resistance of each squib and squib firing line after installation. The measurement will be made through direct access connectors.

Figure 3-1 is a functional block diagram which illustrates the functional configuration required for these tests. Self tests are accomplished by using internal connection capability to determine if the Test Kit is operating properly and is safe.

4.0 INTERFACES

Interfaces with the portable test unit should be effected through cables and harnesses provided with the test unit.

4.1 PYRO CONTROLLER INTERFACE

Function	Characteristics
Capacitor	High Impedance Voltmeter
Stray Voltage	High Impedance Voltmeter
Capacitor Discharge	10 ohms resistance
Relay Position Indication	Continuity Measurement
Reset Signals	As required
Preset Signals	As required

4.2 SPACECRAFT HARNESS INTERFACE

Function	Characteristics
Short Test	Current limited to 10 ma.
Grounding Test	Current limited to 10 ma.
Squib Resistance	Current limited to 10 ma.

5.0 PERFORMANCE PARAMETERS

Voltage Measurement	• ± 2% • 1 megohm input impedance
Ohmeter Circuit	• ± 0.1 ohm on 1.0 ohm resistances • 10 ma maximum current through load

6.0 PHYSICAL CHARACTERISTICS

6.1 SIZE AND WEIGHT

The portable test unit will be packaged in a suitcase sufficiently small and light in weight that one man can carry it conveniently.

6.2 POWER REQUIREMENT

The test unit will use self contained dry cell batteries.

7.0 SAFETY CONSIDERATIONS

7.1 SPACECRAFT AND PYROTECHNICS PROTECTION

Particular care must be taken in the design of the resistance measuring circuits to limit the currents applied to any squib circuit to values which cannot cause premature operation of the pyrotechnics. There also can be no possibility that an internal short in the equipment could result in enough current to fire a squib.

7.2 FACILITIES

The facility safety criteria will cover the safety details in the use of this piece of equipment.

7.3 PERSONNEL

The Portable Test Unit is not dangerous in itself but the use of the equipment must be controlled to insure an adequate level of safety in the ESA.

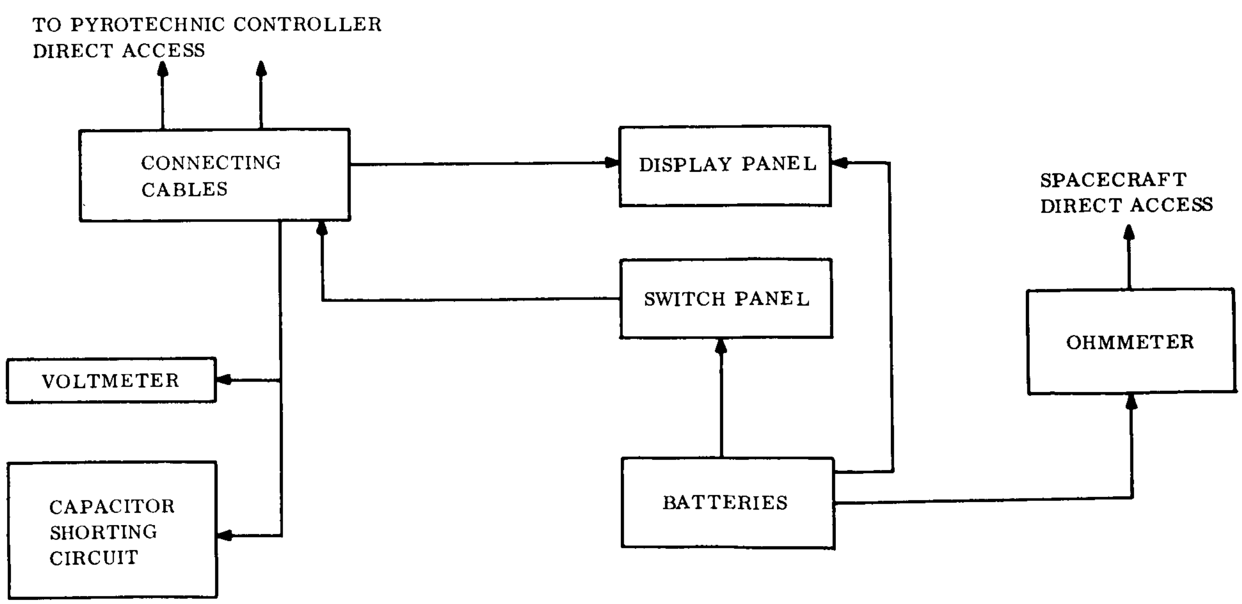


Figure 3-1 Pyrotechnic Test Kit, Functional Block Diagram

CII - VB280SR103

MISSION DEPENDENT EQUIPMENT

OBJECTIVES AND DESIGN CRITERIA

Index

1.0 SCOPE

The intent of this document is to define the objectives and design criteria which should be applied to the Mission Dependent Equipment (MDE) necessary at the DSIF and SFOF to support the launch and post-launch operations of the Voyager 1971 Spacecraft. MDE is interpreted to include both hardware and software.

2.0 APPLICABLE DOCUMENTS

The MDE shall conform to the applicable sections of the following documents:

V-MA 004 001 14 03 Voyager 1971 Mission Specification (Preliminary)
V-MA 004 002 14 03 Voyager 1971 Mission Guidelines
EPD-283 The Deep Space Network
GMG 50109 DSNA Design Specification Telecomm Development,
GSDS Command System

3.0 OBJECTIVES

3.1 DEFINITION

Any item required by the DSN or SFOF to complete the functional requirements of a particular program which is not required on any other program is considered to be MDE. The item may be categorized as either software or hardware.

3.2 FUNCTIONS

MDE may be required to perform the following functions:

a. Encoding of commands.

b. Demodulation and decommutation of telemetry data.

c. Supplying appropriate spacecraft telemetry data as inputs to other equipments.

This data includes:

1. Spacecraft receiver and command detector parameters used for acquisition prior to commanding the spacecraft.

2. Other selected spacecraft engineering data, for quick look purposes at the DSIF or for transmission to the SFOF in near real time.

3. Selected spacecraft non-engineering data, for quick look purposes at the DSIF station or for transmission to the SFOF in near real time.

d. Recording requirements beyond the capacity of the DSIF or SFOF recording facilities.

e. Display requirements beyond the capacity of the DSIF or SFOF data display facilities.

f. Interfacing between data processing and data transmission facilities.

g. Tracking and orbit determination requirements beyond the existing facilities of the DSIF and SFOF.

3.3 MINIMIZATION OF MDE

Wherever possible, existing DSIF and SFOF non-mission dependent equipment (hardware and software) shall be used in order to minimize the requirement for MDE.

4.0 DESIGN CRITERIA

4.1 DSIF MDE

The primary criterion in the design of MDE is reliability of operation.

The MDE designated for use at the DSIF sites must be designed to conform to standard DSIF specifications in order to insure mechanical, electrical, and functional compatibility with existing DSIF facilities. The MDE must provide isolation between itself and the existing DSIF facilities, sufficient to preclude any degradation of performance of the existing facilities due to the addition of the MDE. The MDE shall be designed to afford ease of operation and of maintenance. MDE shall be designed to minimize the requirements for spare parts by using standard circuits and parts. The MDE shall be compatible with the spacecraft and capsule telecommunication subsystems. The MDE shall be compatible with the mission operational requirements, such as being capable of supporting multiple spacecraft. DSIF MDE software shall be compatible with DSIF existing computer facilities and with mission requirements, and shall make maximum use of existing routines and sub routines.

4.2 SFOF MDE

The MDE for the SFOF shall be compatible with existing SFOF mission independent hardware and software. The MDE hardware shall be designed to meet the same requirements as the DSIF MDE for isolation, ease of operation and maintenance, and compatibility with the spacecraft and capsule telecommunication subsystems and with the mission operational requirements. SFOF MDE software shall be compatible with SFOF existing computer facilities and with mission requirements, and shall make maximum use of existing routines and sub-routines.

MISSION DEPENDENT EQUIPMENT

DESIGN CHARACTERISTICS AND RESTRAINTS

Index

1.0 SCOPE

It is the intent of this document to define the design characteristics with restraints to be applied to the Mission Dependent Equipment (MDE) necessary at the DSIF and SFOF to support the launch and post-launch operations of the Voyager 1971 Spacecraft. MDE is interpreted to include both hardware and software.

2.0 APPLICABLE DOCUMENTS

V-MA-004-001-14-03	Voyager 1971 Mission Specification (Preliminary)
V-MA-004-002-14-03	Voyager 1971 Mission Guidelines
EPD-283	The Deep Space Network
GMG-50109-DSN-A	Design Specification, Telecommunications Development, GSDS Command System, Ground Subsystem, Command Verification Equipment
JPL-8907A-DSIF	General Specification, General Requirements for DSIF Electronic Equipment
JPL-8900	Environmental Specification DSIF Ground Equipment Assembly Level Last Requirements
JPL-8902	DSIF General Specification, Documentation Requirements
JPL-8905	Preferred Parts List for DSIF RF Equipment
JPL-8906	General Requirements for GSDS Standard Modules
DOO-1022-GEN	General Specification, DSIF Drawing Documentation Requirements

3.0 DESIGN CHARACTERISTICS AND RESTRAINTS

3.1 MDE HARDWARE FOR DSIF AND SFOF

a. Reliability shall be of prime importance in the design of MDE hardware. Materials, parts, and processes used in design shall conform to JPL specification 8907A. The selection of parts, where applicable, shall adhere to JPL specification 8905.

b. MDE shall operate from primary input power in accordance with that listed in JPL specification 8907A.

c. MDE shall be sufficiently isolated from DSIF and SFOF facility hardware so that failure of the MDE hardware will produce no degradation of the facility hardware.

d. MDE hardware shall be designed to assure interface compatibility with DSIF and SFOF equipment as described in EPD-283 and GMG-50109-DSN-A.

3.2 MDE SOFTWARE FOR DSIF AND SFOF

MDE software shall be compatible with existing operational equipment and shall make efficient use of the elements of the existing system, as described in EPD-283.

The programs developed must:

a. Be designed for control by the existing Monitor.

b. Use compatible conventions; e.g., tape assignments, interrupt priorities.

c. Make efficient use of existing system library.

The programs shall be generated making maximum use of existing compilers, and the documentation generated shall conform to existing standards.

Programs shall be thoroughly checked out prior to delivery, making maximum use of all debugging, tracing, and diagnostic routines available.

CII - VB280FD115

MISSION DEPENDENT EQUIPMENT

DSIF COMMAND MODULATOR

Index

1.0 SCOPE

This document describes the mission dependent command modulator required at the DSIF to support the Voyager Program.

2.0 APPLICABLE DOCUMENTS

GE

VB280SR103	MDE Objectives and Design Criteria
VB280SR104	MDE Design Characteristics and Restraints
VB233FD101	Telecommunication Subsystem
VB233FD103	Flight Command Subsystem
GMG50109DSN-A	Design Specification, Command Verification Equipment

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The DSIF Command Verification Equipment provides the necessary hardware to encode commands at specific standard bit rates. A command modulator is required as Mission Dependent Equipment for those projects using non-standard bit rates. A 30-bit per second command modulator is required for the Voyager project.

3.2 FUNCTION

The basic command modulator receives command data bits from the CVE general purpose computer and a subcarrier frequency from a voltage-controlled oscillator. The modulator generates a pseudo-noise(PN) code and performs the logic operations required to combine the data bits, the PN code and the subcarrier to produce the command subcarrier spectrum, and presents this spectrum to the DSIF transmitter.

3.3 BLOCK DIAGRAM

A block diagram showing the functional relationship of the command modulator to other DSIF command components is given in Figure 3-1. The block diagram of the modulator is that shown in Figure 4 of GMG-50109-DSN-A.

4.0 INTERFACE DEFINITIONS

Interfaces with the command modulator are given in Table 4-1.

5.0 PERFORMANCE PARAMETERS

The command modulator generates a two-channel command signal compatible with the Voyager Spacecraft Command Subsystem as described in VB233FD103.

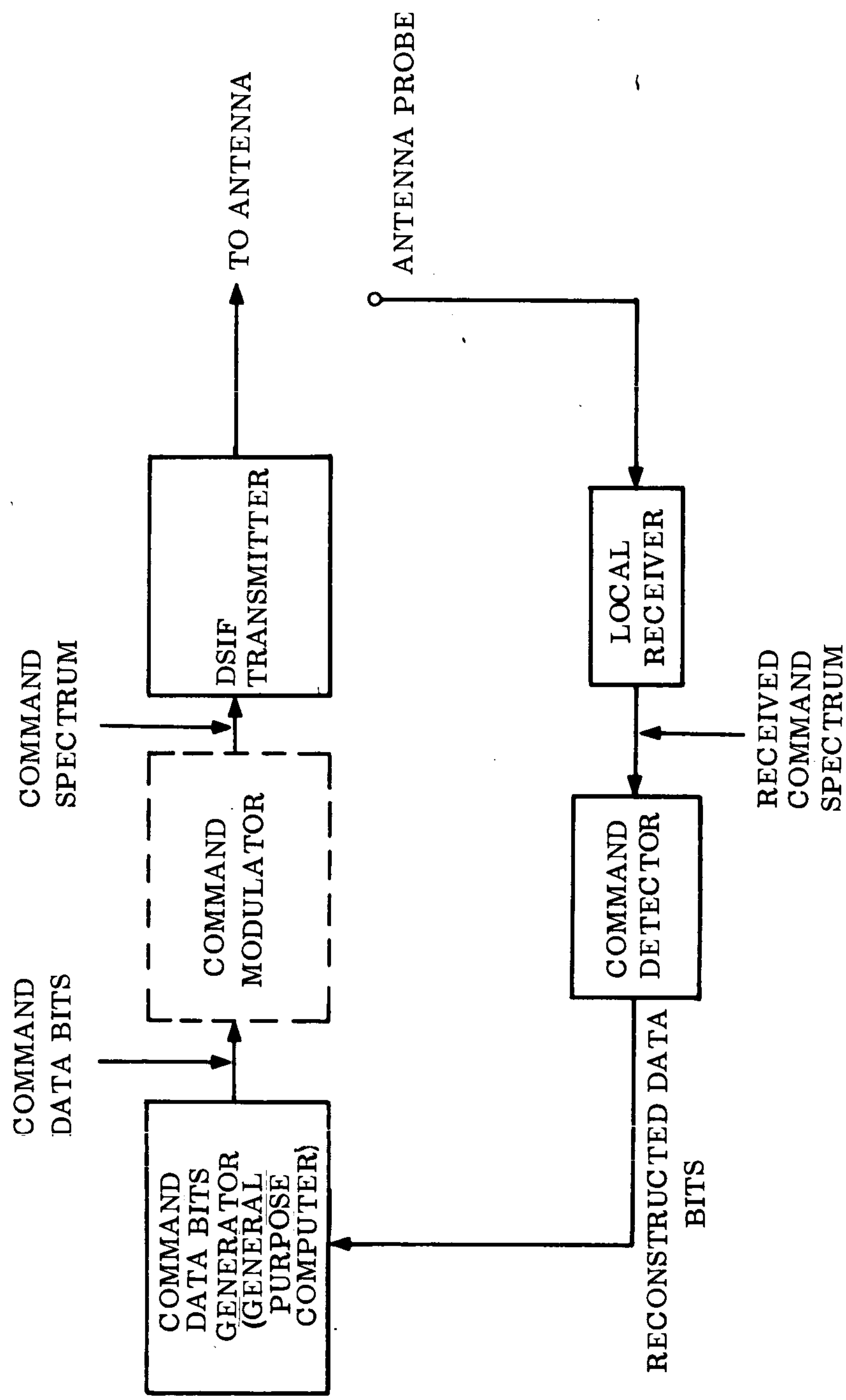


Figure 3-1. Functional Location of Command Modulator at the DSIF

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The Voyager command modulator is physically similar to the existing CVE command modulators, and is constructed to fit within the space provided in the CVE for MDE. It is built to conform to the design constraints given in VB280SR104.

7.0 SAFETY CONSIDERATIONS

7.1 SPACECRAFT INTEGRITY

Accuracy of command processing and reliability of operation is of primary concern in order to minimize the possibility of jeopardizing the mission success.

7.2 FACILITIES

The command modulator is designed to provide interface isolation so that its malfunction will not cause degradation of the performance of DSIF facility equipment.

7.3 PERSONNEL

The command modulator is designed to present no danger to personnel.

Table 4.1. Command Modulator Interfaces

Interfacing Component	Function	Input/Output	Characteristic
General Purpose Computer	Command Data	Input	Digital command bits
	Acquisition Control	Input	Digital frequency control
	Bit Sync	Output	Command bit rate
DSIF Transmitter	Command Modulation	Output	Encoded command spectrum

CII - VB280FD116

MISSION DEPENDENT EQUIPMENT

DSIF COMMAND DETECTOR

Index

FUNCTIONAL DESCRIPTION
1971 VOYAGER SPACECRAFT MISSION DEPENDENT EQUIPMENT
DSIF COMMAND DETECTOR

1.0 SCOPE

This document describes the mission dependent command detector required at the DSIF to support the Voyager Project.

2.0 APPLICABLE DOCUMENTS

VB280SR103	MDE Objectives and Design Criteria
VB280SR104	MDE Design Characteristics and Restraints
VB233FD103	Voyager Flight Command Subsystem
VB233FD101	Voyager Telecommunication Subsystem
GMG-50109-DSN A	Design Specification, Command Verification Equipment

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The DSIF Command Verification Equipment (CVE) provides the necessary hardware to encode commands, and to verify their accurate transmission, at specific standard bit rates. A command detector is required as Mission Dependent Equipment (MDE) for those projects using non-standard bit rates. A 30-bit per second command detector is required for the Voyager Project.

3.2 FUNCTIONS

The command detector receives the command subcarrier spectrum from an S-band monitor receiver at the DSIF site. The function of the detector is to reconstruct the command data bit sequence so that the computer may use it for comparison.

3.3 BLOCK DIAGRAM

A block diagram showing the functional relationships of the command detector to other DSIF command components is given in Figure 3-1.

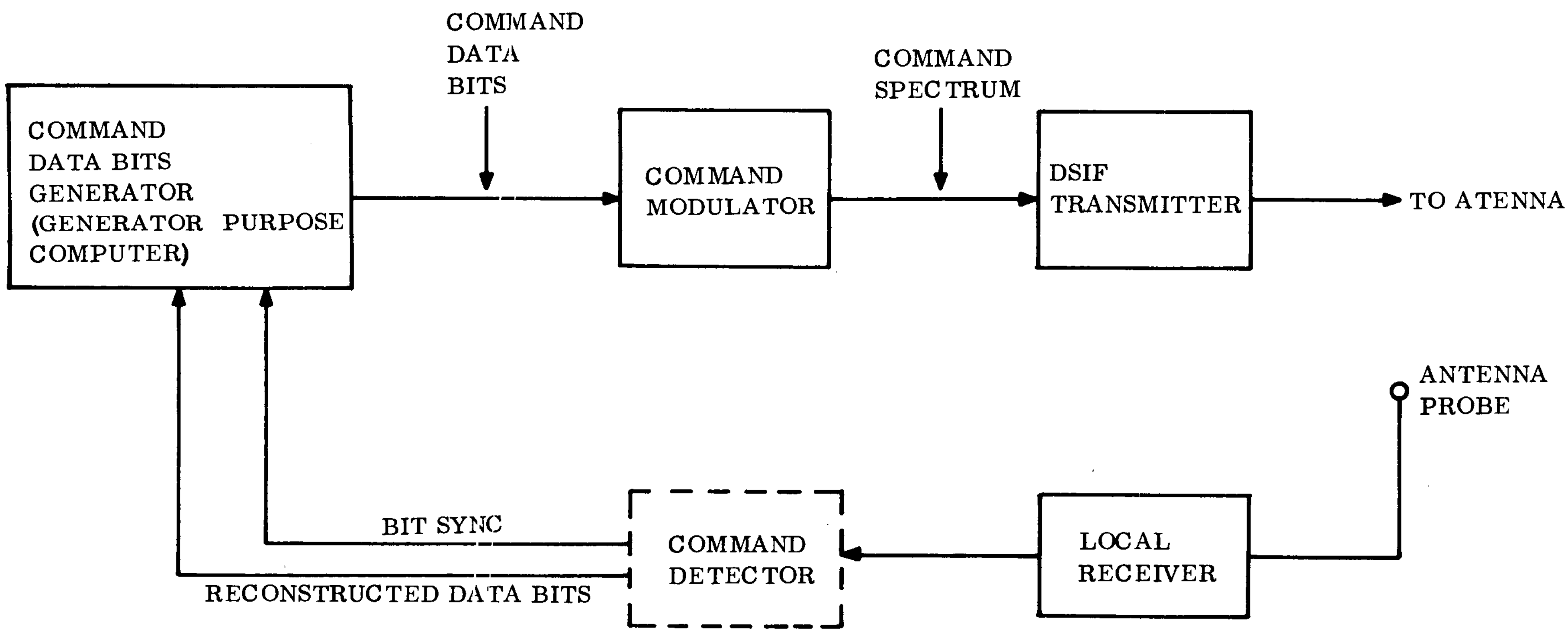


Figure 3-1. Functional Location of the Command Detector at the DSIF

4.0 INTERFACE DEFINITION

Interfaces with the command detector are as follows:

Interfacing Component	Function	Input/ Output	Characteristic
General Purpose Computer	Command Data	Output	Digital
	Bit Sync	Output	Digital
Local Receiver	Command Signal	Input	Command Spectrum

5.0 PERFORMANCE PARAMETERS

The command detector is capable of detecting the 30-bit per second Voyager command signal, and is functionally identical with the Voyager flight command detector.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The command detector is physically similar to the existing CVE command detectors, and is constructed to fit within the space provided in the CVE for MDE, and is built to conform to the design constraints given in VB280SR104.

7.0 SAFETY CONSIDERATIONS

7.1 SPACECRAFT INTEGRITY

Accuracy of command processing and reliability of operation is of prime concern in order to minimize the possibility of jeopardizing the mission success.

7.2 FACILITIES

The command detector is designed to provide interface isolation so that its malfunction will not cause degradation of the performance of DSIF facility equipment.

7.3 PERSONNEL

The command detector is designed to present no danger to personnel.

CII - VB280FD117

MISSION DEPENDENT EQUIPMENT

DSIF TELEMETRY DEMODULATOR

Index

1.0 SCOPE

This specification describes the function of the Voyager telemetry demodulator. The mission dependent demodulator consists of that equipment necessary to demodulate the noise corrupted spacecraft telemetry signal provided by the ground RF receiver at the Deep Space Instrumentation Facility (DSIF) stations.

2.0 APPLICABLE DOCUMENTS

GE

VB280SR103 — 1971 Voyager Spacecraft MDE Objectives & Design Criteria

VB280SR104 — 1971 Voyager Spacecraft MDE Design Characteristics & Restraints

JPL

EPD-283 — The Deep Space Network. "A Planned Capability for the 1965-1980 Period."

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The telemetry demodulator has the task of detecting and decoding a noise corrupted bi-phase modulated telemetry signal from ground telemetry receivers, magnetic tape recorders, or data encoder simulator. The telemetry signal is a composite signal containing both data and synchronization information on a single subcarrier. The demodulator output is a serial or parallel binary train of data pulses, bit sync, word sync, preamble sync, and demodulator sync condition indication (SCI). The output data and sync information will be sent to the Telemetry, Command and Data Handling (TCD) system for processing and to the tape recorder system for storing. Bit sync, word sync and data are time synchronous. The demodulator will be required to operate at the following six different bit rates (R) during the Voyager mission:

R_1 = 8,533 1/3 bps

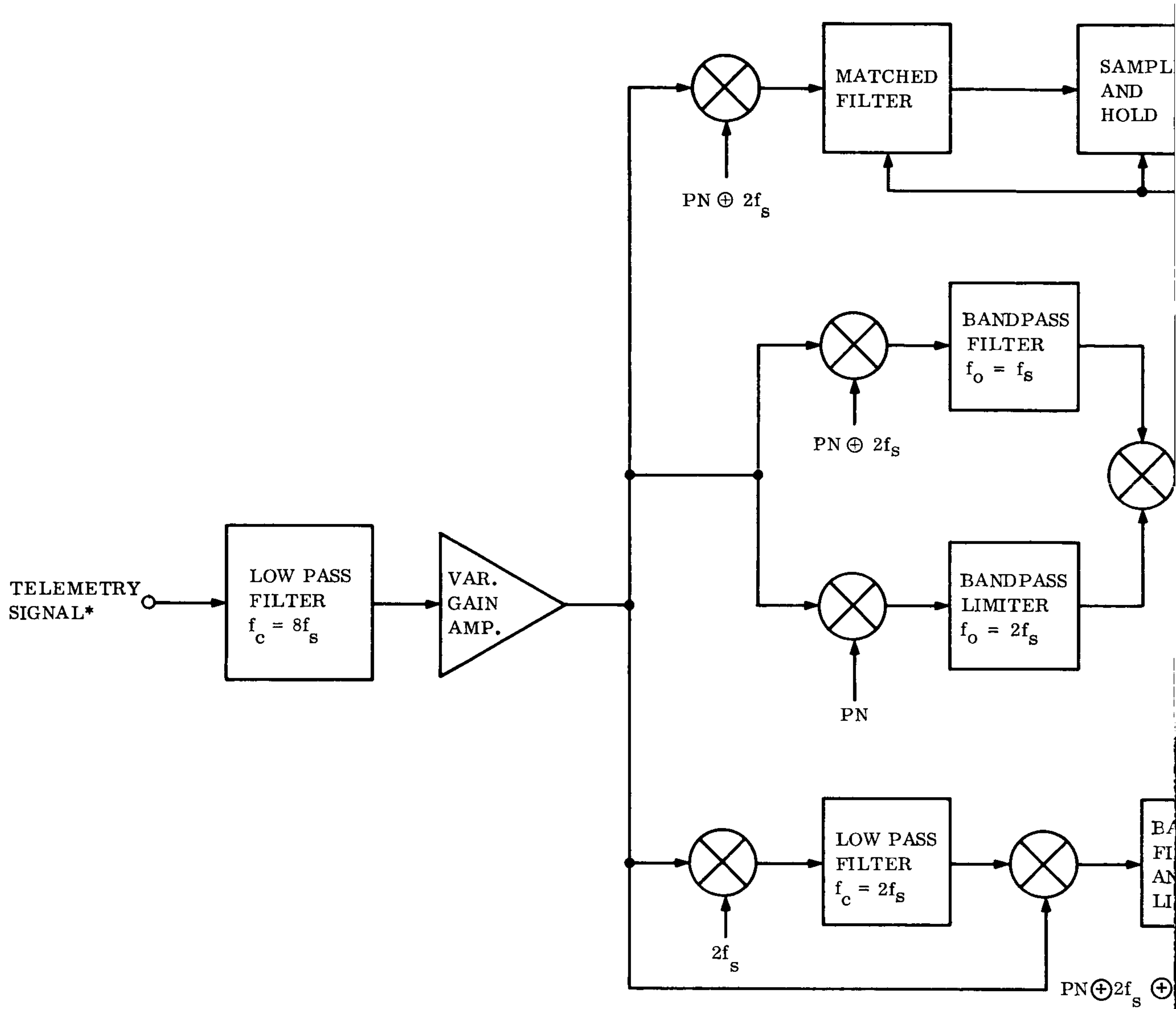
R_2 = 4,266 2/3 bps

R_3 = 2,133 1/3 bps

R_4 = 533 2/3 bps

R_5 = 106 2/3 bps

R_6 = 3 1/3 bps



* TELEMETRY SIGNAL = $2f_s$ ⊕ PN ⊕ DATA

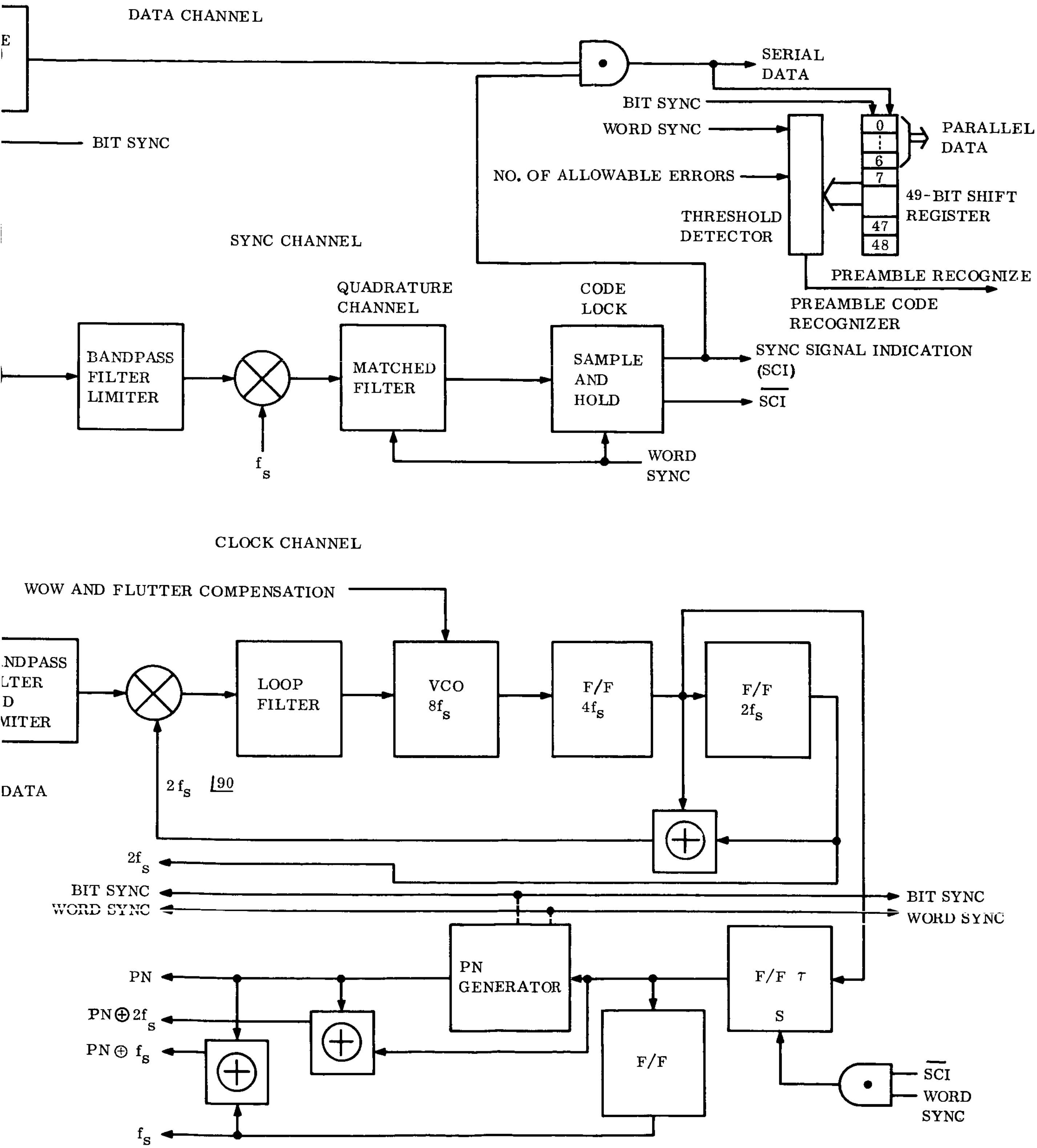


Figure 3-1. Telemetry Demodulator

2

Upon receipt of the noise corrupted telemetry signal the signal is sent through a low pass filter ($f_c = 8\ f_s$) and amplifier as shown in Figure 3-1. The low pass filter masks out a portion of the unwanted noise while the amplifier serves as an isolation device and amplitude compensator. The amplitude compensation allows closer control of the signal level entering the demodulator. The filter amplifier presents the resulting signal to three detector filter branches; clock, data and sync.

Demodulation of data is accomplished by first acquiring clock and then synchronizing the sync channel. Clock is acquired by locking a phase lock loop on the incoming clock after all modulation has been removed by mixing and squaring. The noise free clock from the PLL is used for driving the logic circuits including the pseudo-noise (PN) code generator. To acquire word sync the demodulator PN code generator must align or correlate with the received PN code. Thus, synchronization is determined by matched filter detection of the quadrature channel as the internally generated code is correlated with the received code. When the two codes are aligned, synchronization is accomplished, thus emitting code lock or sync condition indication (SCI), data, bit sync, word sync, and termination of the acquisition process.

Code slide-by is accomplished by adding one 4-f_s clock pulse every 63 PN bits until acquisition is complete. When acquisition is complete, the addition of extra clock pulses is stopped by inhibiting the add-in process with $\overline{\text{SCI}}$.

After acquisition is completed the loop filter time constants may be decreased to give better tracking performance by reducing rms phase error.

Data is matched-filter detected in the data channel after removing PN code and 2-f_s clock from the bi-phase modulated signal.

The Serial-to-Parallel Converter and Preamble Recognizer receives the serial data and word sync and converts the serial data to seven-bit parallel data. It also scans incoming data, seven words at a time, in search of the unique 49-bit preamble sync that precedes every new data type. The logic used permits recognition of the preamble sync with up to three errors. When the preamble sync is recognized, a Preamble Recognizer pulse is generated and sent to the decommutator for enabling interrogation of the data type word.

3.2 SELF CHECK

A self check capability is provided by an encoder simulator. The output of the encoder simulator is a composite single channel telemetry signal equivalent to that of the Voyager spacecraft. The simulator can operate at each of the six Voyager data rates, and generates data including preamble sync and data type word. An output of binary NRZ data is also available for bit error tests.

4.0 INTERFACE DEFINITION

The telemetry demodulator interfaces are shown in the interface tables as listed below:

Table 4-1 Telemetry Demodulator and DSIF Receiver

Table 4-2 Telemetry Demodulator and TCD

Table 4-3 Telemetry Demodulator and Tape Recorder Storage

Table 4-4 Telemetry Demodulator and Data Encoder Simulator

Table 4-1. Telemetry Demodulator and DSIF Receiver Interface

Function	Characteristic	Impedance (OHMS)	Comments
S/C telemetry Signal	0.3 to 3.0-v rms	10k	Single Subcarrier bi-phase modulated with data and sync.

Table 4-2. Telemetry Demodulator and TCD Interface

Function	Characteristic	Comments
Telemetry Data	Serial or Parallel Binary, NRZ	
Telemetry Word Sync	Binary, RZ	Synchronous with data/7
Telemetry Bit Sync	Binary, RZ	Synchronous with data
SCI	Binary, NRZ	Continuous during phase lock.
Preamble Recognize	Single Pulse	Indicates recognition of preamble.

Table 4-3. Telemetry Demodulator and Tape Recorder Storage

Function	Characteristic	Impedance (OHMS)	Comments
Input Signal			
Stored Telemetry Signal	0.3 v to 3.0 v rms	10k	Single Subcarrier bi-phase modulated with data and sync.
Output Signals			
Telemetry Data	Serial Binary, NRZ		
Telemetry Word Sync	Binary, RZ		Synchronous with data/7
Telemetry Bit Sync	Binary, RZ		Synchronous with data
SCI	Binary, NRZ		Continuous during phase lock

Table 4-4. Telemetry Demodulator and Data Encoder Simulator Interface

Function	Characteristic	Impedance (OHMS)	Comments
Simulated S/C Telemetry Signal	0.3 to 3.0 v rms	10k	Single Subcarrier bi-phase modulated with data and sync.

5.0 PERFORMANCE PARAMETERS

5.1 BIT RATES

The telemetry demodulator operates over six different bit rates as shown in Table 5-1. The bit rates have an accuracy of 0.01% or better. The bit rate is one ninth of the subcarrier frequency, 2 f_s (ie $2f_s = 9R$).

5.2 DYNAMIC RANGE

The demodulator operates over an input dynamic range of 0.3 volt to 3.0 volt rms signal plus noise in an 8 f_s bandwidth.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 MECHANICAL

The Voyager telemetry demodulator is a rack-mounted subsystem. All panel assemblies are designed to mount in 19-inch wide racks that are 15 inches or less in height. The weight of the demodulator is 80 pounds or less.

Visual displays and monitors are supplied sufficient to determine the status of the subsystem.

Frequency dependent components will be built on plug in assemblies where more than one frequency of operation is required. To change frequency or bit rates, modules are interchanged.

The Data Encoder simulator is rack mounted in a standard 19-inch rack and is approximately five inches in height.

6.2 ELECTRICAL

D-C power supplies are used to supply the required internal voltages for the telemetry demodulator subsystem operation. The required input voltage is 105-125 volts, 60 cps, single phase. The power consumed by the demodulator is less than 50 watts average.

The Data Encoder simulator requires 10 watts.

7.0 SAFETY CONSIDERATIONS

The telemetry demodulator MDE conforms to good engineering practice providing circuit protection and safety devices so as to minimize and negate internal damage caused by faulty equipment or performance. The telemetry demodulator is designed to interface safely with the spacecraft operations support equipment and DSN equipment and to eliminate any possibilities of damage to the spacecraft or ground equipment. Self checks are on a non interferring test basis. The telemeter demodulator presents no hazards to personnel. The demodulator does not cause erroneous operation to other equipment so as to alter their operational procedures nor prejudice the mission.

CII - VB280FD118

MISSION DEPENDENT EQUIPMENT

DEEP SPACE INSTRUMENTATION FACILITY (DSIF) SOFTWARE

Index

1.0 SCOPE

This document describes the Mission Dependent Software required at the DSIF sites in support of the 1971 Voyager mission.

2.0 APPLICABLE DOCUMENTS

JPL

GMG-50109-DSN-A		Design Specification, Command Verification Equipment
EPD-283		The Deep Space Network, A Planned Capability for the 1965-1980 Period

GE

VB280SR103	1971 Voyager S/C	MDE Objectives and Design Criteria
VB280SR104	1971 Voyager S/C	MDE Design Characteristics and Restraints
VB233FD106	1971 Voyager S/C	Data Encoder Subsystem
VB233FD107	1971 Voyager S/C	Data Storage Subsystem
VB263FD106	1971 Voyager S/C	Data Encoder Subsystem OSE.

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

It is the purpose of the Telemetry and Command Data Handling Subsystem (TCD) of the Deep Space Instrument Facility (DSIF) to provide a general purpose facility to process telemetry data and to prepare that data for distribution over the Ground Communications System (GCS) (Telemetry function). The TCD must also receive command messages from the Space Flight Operational Facility (SFOF), store the commands until the Time of Execution, initiate transmission of commands to the spacecraft, and verify the proper transmission of the commands (command function). The mission dependent software required to accomplish the TCD function is described in the following sections.

3.2 TELEMETRY DATA PROCESSING

Telemetry data is processed as received from the demodulator, from the Tape Storage System, or from a Telemetry Signal Simulator.

3.2.1 INPUT INFORMATION

The TCD receives two types of data: Spacecraft telemetry data and station data. The telemetry data is received as a parallel eight-bit binary word consisting of seven data bits and Sync Condition Indicator (SCI), plus word sync and preamble sync.

3.2.2 DECOMMUTATION PROGRAM

The data processing portion of the decommutation program is exactly the same as the program used in the Data Encoder Operational Support Equipment, and is described in detail in VB263FD106.

3.2.3 DATA OUTPUT

The output instructions of the seven data types route the data in proper form to the appropriate user. The output data is presented in 24-bit parallel form in either binary or BCD as determined by the user. Four users require the output data: Teletype, High-Speed Data Lines, Tape Storage, and Telemetry Panel Displays.

In modes I, II, IV, and VI, engineering data is stripped out and stored if necessary, for transmission at an average rate within the data rate range of the high speed data transmission lines. Mode III and Mode V data require special treatment. During Mode III, planet scan data (Data Type C) and engineering and non-scan science data (Data Type D) are received in alternating blocks at a bit rate of 8533 1/3 bits per second. Each block of Type D data, approximately 7000 words, is stored temporarily in the computer. As the Type C data is received, it is formatted and stored on magnetic tape. Simultaneously, as the Type C data is received, the Type D data in storage is formatted for transmission at an average bit rate of 426 bits per second. Mode V data is received at 8533 1/3 bits per second. The data is edited for real time display and sampled at a rate suitable for high-speed data transmission.

3.3 COMMAND DATA

The transfer of command messages from the SFOF to the DSIF stations is specified in GMG-50109-DSN-A. The length of the Voyager command word is 50 bits (100 sub-bits) maximum. Program modification is required to permit Manchester coding and to effect the command word synchronization code of six sub-bits at the beginning of the command word. The modification may be obtained by increasing the maximum permissible command word bit length to a number greater than the present 63.

4.0 INTERFACE

The telemetry software is designed to accept the data inputs from the telemetry demodulator, and to format data for storage or transmission.

5.0 PERFORMANCE PARAMETERS

The telemetry software is designed to process data at the six Voyager data rates, and to process data received in the seven data types and six data modes. Processing priorities are established to permit data input, processing, and output simultaneously.

6.0 CHARACTERISTICS AND CONSTRAINTS

The telemetry programmer is constrained to the use of existing compilers and assemblers available for the SDS 920 computer. The processing and input/output of data is constrained to the requirements of the spacecraft and the DSIF.

7.0 SAFETY

There are no special requirements applicable to the software described that will damage or jeopardize the success of the flight equipment or the mission.

CII - VB280FD119

MISSION DEPENDENT EQUIPMENT

SFOF TELEMETRY DEMODULATOR

Index

1.0 SCOPE

This document describes the function of the Voyager telemetry demodulator at the SFOF. The mission dependent demodulator consists of that equipment necessary to demodulate the noise corrupted spacecraft telemetry signal which has been recorded on magnetic tape at the DSIF stations. The tapes are later delivered to the Space Flight Operations Facility (SFOF). The SFOF telemetry demodulator is a functional duplicate (identical in every respect) of the DSIF MDE telemetry demodulator.

2.0 APPLICABLE DOCUMENTS

VB280SR103	MDE Objectives and Design Criteria
VB280SR104	MDE Design Characteristics and Restraints
VB280FD117	Functional Description, DSIF Telemetry Demodulator

3.0 FUNCTIONAL DESCRIPTION

The SFOF Telemetry Demodulator is identical to the DSIF Telemetry Demodulator described in VB280FD117.

4.0 INTERFACE DEFINITION

The SFOF telemetry demodulator interfaces with the Magnetic Tape Recorder and the General Purpose Station. Their interfaces are listed in Tables 4-1 and 4-2.

5.0 PERFORMANCE PARAMETERS

Identical in every respect to the DSIF telemetry demodulator description, VB280FD117.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

Identical in every respect to the DSIF telemetry demodulator description, VB280FD117.

7.0 SAFETY CONSIDERATIONS

Identical to the DSIF telemetry demodulator description, VB280FD117.

Table 4-1. Telemetry Demodulator and SFOF Magnetic Tape Recorder

Function	Characteristic	Impedance	Comments
S/C Telemetry Signal	0.3 to 3.0 v rms	10k	Single subcarrier bi-phase modulated with data and sync.
Wow and Flutter Compensation	Analog		Discriminator output for wow and flutter compensation

Table 4-2. Telemetry Demodulator and General Purpose Station Interface

Function	Characteristic	Comments
Telemetry Data	Serial or Parallel Binary, NRZ	
Telemetry Word Sync	Binary, RZ	Synchronous with data
Telemetry Bit Sync	Binary, RZ	Synchronous with data
SCI	Binary, NRZ	Continuous during phase lock.
Preamble Recognize	Single Pulse	Indicates recognition of preamble

CII - VB280FD120

MISSION DEPENDENT EQUIPMENT

SFOF SOFTWARE

Index

1.0 SCOPE

This document contains a functional description and interface definition for the Mission Dependent Software required at the Space Flight Operations Facility for the Voyager Spacecraft 1971 Mission.

2.0 APPLICABLE DOCUMENTS

JPL

Engineering Planning Document No. 283 (Preliminary)

"The Deep Space Network - A Planned Capability for the 1965-1980 Period"

GE

VB280SR103	1971 Voyager Mission Dependent Equipment Objectives and Design Criteria
VB280SR104	1971 Voyager Mission Dependent Equipment Design Characteristics and Constraints

3.0 FUNCTIONAL DESCRIPTION

3.1 GENERAL

The requirements for mission dependent software at the SFOF for the 1971 Voyager mission can be grouped in three different categories; i. e., event generation, telemetry data handling and command handling. Not all software required for the mission is mission dependent; however, all required software is given in paragraph 4.0 to demonstrate the flow of data through the SFOF Data Processing System. The computer programs which are mission independent will not be treated in this specification even though there may be some minor modifications required to these programs to handle this specific vehicle.

3.2 EVENT GENERATION SOFTWARE

3.2.1 GENERAL

There are five different categories of events which must be considered in this area; i. e., tracking and orbit determination, spacecraft maneuvers, DSN station contacts, capsule lander events and payload operations. Of these five areas only spacecraft maneuver requires treatment in this specification since orbit determination and station contact programs are already available and the programs for the capsule and payload, while being mission dependent, are the responsibility of the respective contractors.

3.2.2 SPACECRAFT MANEUVER PROGRAM

The spacecraft maneuver program is required to calculate mid-course maneuver requirements, determine the engine burn parameters to place the spacecraft in orbit around Mars and alter the orbit once it has been attained. Each of the three requirements will be treated as a separate problem; however, they have many similar computational requirements. The midcourse maneuver will be calculated using as inputs the present flight path parameters and the desired point of approach to Mars. Using these values a series of parameters will be calculated which will provide the capability to select the best time to execute the manuever consistent with other operational requirements.

The requirement to place the spacecraft in orbit around Mars will be satisfied by determining the spacecraft maneuver needed to transfer from the interplanetary travel flight path to the orbital path around Mars. The subroutine required to perform this transfer will use as inputs the present flight path parameters and the desired orbital parameters. The output will be the necessary spacecraft maneuver and engine burn parameters to accomplish the transfer.

Once the spacecraft has been placed into orbit around Mars it may be necessary to alter the orbit to better accomplish the requirements for this phase of the mission. The subroutine to perform this orbit adjustment will use as inputs the existing orbital parameters and the required parameters and will calculate the required spacecraft maneuver and engine burn parameters to attain the desired orbit.

3.3 TELEMETRY DATA HANDLING SOFTWARE

3.3.1 GENERAL

There are separate divisions of software required in the telemetry data handling area, which are, format conversion and data record at the SFOF, raw data editing, display of real time data and telemetry data processing. The format conversion and raw data editing are mission independent; however, a subroutine will be added to the raw data editing program to segregate the data according to type--tracking, messages, command verification, telemetry.

3.3.2 DISPLAY OF REAL TIME DATA

As telemetry data is received at the SFOF specific critical items can be extracted from the raw telemetry data and displayed at the user area printer and plotter. This permits the user rapid access to critical items to evaluate the performance of the spacecraft. The data will be presented at the user area in either engineering units or per cent full scale as requested. This program will use numerous subroutines which are also required as part of the Telemetry Data Processing. The subroutines used by both programs will be made availabe as general subroutines and therefore will be programmed only once.

3.3.3 TELEMETRY DATA PROCESSING

The raw telemetry data received from the Raw Data Editing program will be processed in real time or delayed time using the same mission-dependent computer program. The data will be separated according to type of data--engineering, scientific, or capsule lander. The latter two types will be made available to the payload and capsule lander contractors for their evaluation. The engineering data will be grouped as:

- Event occurrence and command verification
- Digital conversion, display and printout
- Automated analysis plotting

A subroutine will be prepared for each of these modes to provide the required capability. The output of this program will be displayed at the user area printer and plotter. The format for the output will be designed so as to provide the user with the data grouped according to subsystems.

3.4 COMMAND HANDLING SOFTWARE

3.4.1 GENERAL

There are four categories of programs required for the commanding area which are, command preparation, tape preparation, tape verification and vehicle storage. The purpose of these four programs is to provide the capability to prepare a group of commands for transmission to the spacecraft, verify proper transmission and maintain a current list of commands stored in the vehicle. Of the programs necesssary only two are mission dependent. A program exists which will prepare the tape for transmission and verify that the tape transmitted to the DSIF has been correctly transmitted.

3.4.2 COMMAND PREPARATION PROGRAM

The purpose of this program is to assemble the commands required to perform the events as established in the sequence of events. In addition, the program will be capable of preparing other commands as requested as part of the list. The list of available commands and sequences of commands will be stored on the disc prior to the mission. The use of sequence of commands permits the ability to input one reference number and obtain a group of commands. This reduces the input required to the program. Commands from the sequences or individual commands will be assembled with time tags, where appropriate. The output of this program will be a list of commands for the spacecraft which will be displayed on the printer in the user area in addition to placing it on the disc for later use.

3.4.3 VEHICLE STORAGE PROGRAM

This program will use as input the data from the DSIF pertaining to the success of the transmission of the command message to the spacecraft. The data will include the number of the commands which were not successfully transmitted. This information is used by the program to prepare a current list of commands actually stored in the spacecraft. This list is displayed on the printer in the user area and stored on the disc for later use.

4.0 INTERFACE DEFINITION

The following table depicts the data interface requirements for the Voyager mission.

Program	Input Requirements	Output Requirements
Spacecraft Maneuvers	Flight path parameters, Desired parameters	Engine burn parameters
Display of Real Time Data	Telemetry raw data	
Telemetry Data Handling	Telemetry raw data	
Command Preparation	Engine burn parameters, DSN station contact, Payload operations, Capsule lander data, Vehicle storage list	Command list
Vehicle Storage	Command list Verification data	Vehicle Storage List

5.0 PERFORMANCE

The programs described in this document will be supplied a position in the priority scheme relative to similar programs in the scheme. Programs which have no similarity will be assigned a priority position relative to other programs in this specificiation. All the programs specified herein will be handled in the computer subsystem so that processing interrruptions will be held to a minimum. This will be accomplished by utilizing the switch-over capability inherent in the computer subsystem.

6.0 CHARACTERISTICS AND CONSTRAINTS

The programmer will be constrained to the use of the existing compilers, assemblers, etc., available for the IBM 7040 and 7094.

7.0 SAFETY

There are no special requirements applicable to the software described to avoid damage to flight equipment or to the mission.

CII - VB270SR101

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT (AHSE)

OBJECTIVES AND DESIGN CRITERIA

Index

1.0 SCOPE

This document delineates the functional objectives and design criteria for the 1971 Voyager Spacecraft Assembly, Handling and Shipping Equipment (AHSE).

2.0 APPLICABLE DOCUMENTS

JPL

V-MA-004-002-14-03	Voyager 1971 Mission Guidelines
V-MA-004-001-14-03	Voyager 1971 Mission Specification (Preliminary)

General Electric

VR270SR102	Voyager AHSE Design Characteristics and Constraints

3.0 OBJECTIVES

3.1 ASSEMBLY EQUIPMENT

The assembly equipment supplied for field use as part of the OSE will be required to support the reinstallation and realignment of all components removed for shipping, handling, or testing purposes.

To implement the design requirement that malfunctions be isolated to the subsystem level and the faulty subsystem or component replaced from the spare spacecraft, the assembly equipment must provide assembly and alignment services for all spacecraft components identified as spares.

The OSE subsystem test equipment will have the additional capability of testing flight subsystems and isolating faults to the level of module groups. Accordingly, the assembly equipment objectives include the assembly support required by such replacements.

3.2 HANDLING EQUIPMENT

The objective of the spacecraft and component handling equipment is to implement the mobility needs of the Voyager spacecraft and its subsystems and parts.

3.3 SHIPPING EQUIPMENT

The major objective of the shipping equipment is to contain and/or transport the spacecraft, its major components, and all tools and equipment to the field site.

Another objective is protection of the spacecraft or spacecraft parts during transit between factory and using sites.

4.0 DESIGN CRITERIA

4.1 ASSEMBLY EQUIPMENT

Reinstallation of major spacecraft components (e.g. solar arrays and antennas) at the SCF should be accomplished through assembly tooling and equipment designed and supplied specifically for reassembly purposes. This assembly equipment should be compatible in function and performance with analogous assembly equipment used in development and fabrication.

The assembly equipment will incorporate integral alignment provisions such that major components reassembled to the spacecraft body will be aligned within required limits as part of the assembly process. These provisions will consist of integral mounts for auto collimation targets or telescopes, reference points on the assembly tooling defined by bolt-on plates and holding fixtures, leveling points permanently attached to tooling, gage and tool mounting points to permit assessment and adjustment of alignment accuracies obtained.

The assembly tooling integral alignment design provisions should permit themselves to be located and checked by optical methods from an externally installed alignment facility. This is a permanent alignment dock used to set up and check the assembly and alignment tooling, not used to check vehicle alignments.

A major design criteria for the field assembly equipment is that the holding and handling fixtures supplied with the major spacecraft components be physically compatible with the permanently installed assembly equipment. Design of the two families of hardware must be coordinated such that the installation of the retropropulsion package, for instance, can be accomplished without interference with the tooling, or the solar arrays can be reinstalled to the spacecraft bus without the array's handling structure interfering with the assembly tooling. These assembly operations will all be physically checked out at GE-SD before the assembly equipment is supplied for field use.

Continuing use of assembly equipment for succeeding launch opportunities will be a design criterion.

4.2 HANDLING EQUIPMENT

The prime consideration in Voyager handling equipment design is to provide safe, efficient movement of the spacecraft and/or its major components. These spacecraft components may be handled in any possible state of assembly, placing stability limits on the larger items.

Commercially available equipment shall be preferred where its use does not limit the accomplishment of the desired operations.

The handling equipment shall be completely compatible with all environmental conditions to be encountered by the spacecraft in prelaunch operations and must provide a controlled environment for the spacecraft, where required, such as cooling and heating, dust protection, etc.

Coordinated design shall assure compatibility between items of handling equipment which must be used together, as in the case of lift slings and ground transporters, test equipment, and handling dollies, etc. Handling procedures will be written to support the assembly and flow operations defined in the field operations plan and to meet the ETR safety requirements.

Economical design procedures and materials shall be used to assure minimum cost handling equipment without compromising the design functional objectives.

Use of the handling equipment for successive launch opportunities shall be a design consideration.

4.3 SHIPPING EQUIPMENT

The basic service provided by the Voyager shipping equipment is protection, both mechanical (i.e., shock and vibration attenuation) and environmental (temperature, humidity, dust, etc.), during movement from GE-SD into field use.

Consideration of commercial shipping restrictions on package size and weight shall be a design condition in defining shipping equipment. Meeting all pertinent local and interstate travel restrictions and regulations is mandatory. Out-size items must be seriously studied as much with an eye to reducing the shipping size as to meeting or securing exceptions to shipping restrictions.

The Voyager spacecraft and many of its components will require individual handling fixtures which will go with the various items into the field in the shipping containers. The containers themselves must then be designed to mate with these fixtures and allow the contents of the containers to be removed efficiently at their destinations.

Economical designs and materials shall be used to assure accomplishment of all functional design objectives at minimum cost.

Utilization of the shipping equipment for transportation of spacecraft in succeeding launch opportunities shall be a design criterion.

CII - VB270SR102

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT (AHSE)

DESIGN CHARACTERISTICS AND CONSTRAINTS

Index

1.0 SCOPE

This document delineates the design characteristics and constraints to which the AHSE for 1971 Voyager test and launch operations support must conform. This document, therefore, controls the design of requirements of AHSE to be delivered to AFETR, and to JPL. Identical AHSE is anticipated to be part of the AHSE used in the factory to support development fabrication and developmental tests.

2.0 APPLICABLE DOCUMENTS

JPL

V-MA-004-002-14-03	Voyager 1971 Mission Guideline
V-MA-004-001-14-03	Voyager 1971 Mission Specification (Preliminary

General Electric

VB270SR101	1971 Voyager AHSE Objectives and Design Criteria

3.0 DESIGN CHARACTERISTICS

3.1 ASSEMBLY EQUIPMENT

The assembly equipment supplied for field use as part of the Voyager OSE will be used to facilitate reassembly to the spacecraft of those items removed for shipment to ETR, replaced at ETR, or for movement about the ETR facility during test, checkout and launch preparation.

3.1.1 SPACECRAFT INCOMING CONFIDENCE CHECK AND TRIAL FITS

During the receiving inspection, incoming confidence check, and appendage trial fit periods for spacecraft and capsule, the assembly equipment will be set up and aligned to provide complete assessment of physical condition of the major mechanical components. This equipment will provide assembly points, alignment check instrumentation and support for spacecraft and capsule during inspection procedures.

3.1.2 SPACECRAFT ASSEMBLY IN ESF

During both prelaunch dry run and preparation for the actual launch, a set of assembly equipment will be required at the ESF to facilitate remating of capsule to spacecraft after sterilization and reassembly of appendages: solar arrays, antenna, PSP, etc. This assembly equipment is also used to facilitate post-assembly mechanical confidence checks. To discharge both these duties, the assembly equipment is required to be provided with a complete set of points to establish spacecraft component alignments and to check them mechanically to establish accuracies of assembly alignment.

The equipment should include an accountability kit to log non-flight equipment carried on spacecraft up to this point for purposes of cleanliness, safety tie-downs, squib shorts, etc.

3.1.3 DISASSEMBLY OF SPACECRAFT AFTER LAUNCH DRY RUN

At the Spacecraft Checkout Facility, the assembly equipment used for receiving inspection and assembly operations must also be designed to provide means for disassembly after the launch dry-run procedure. The equipment need only be easily disassembled to clear working space for removal of major spacecraft items, such as capsule, arrays, PSP, etc.

3.2 HANDLING EQUIPMENT

The handling equipment supplied for field use as part of the Voyager OSE should be designed to meet all mobility requirements of the spacecraft system and its components.

3.2.1 MOVEMENT WITHIN SCF

All movement of the assembled spacecraft within the SCF will be done by overhead crane. Major subassemblies will be moved by both castered dolly and overhead crane. Dollies require special design to handle particular components and these dollies will be reserved for use in the SCF. The spacecraft movement will be accomplished using the handling fixture which picks up the fittings mated to the spacecraft lower ring.

3.2.2 MOVEMENT WITHIN ESF

As with the SCF, only overhead cranes are used for movement of the assembled spacecraft within the ESF and the spacecraft will be on its handling fixture at all such times. Special precautions are required for safety within the ESF, these are discussed below.

3.2.3 MOVEMENT BETWEEN SCF, ESF and LC

The special ground transporter designed for moving the spacecraft between major physical facilities at the ETR is described in VB270FD104. This item provides all ground mobility over the paved surfaces at ETR and protects the spacecraft from ambient conditions and undesirable mechanical environments. The transporter can be reduced in size for shipment by partial disassembly. It provides protection from the ambient environment by mating with a weatherproof cover and also will mate with the fairing in the encapsulated configuration.

3.3 SHIPPING EQUIPMENT

The shipping equipment supplied as OSE provides protected transportation environments for all spacecraft components from the manufacturing site to the launch site.

Basically, the equipment design must only provide a housing for the item being transported and attenuate all environmental inputs to levels tolerable for the spacecraft subsystems.

4.0 DESIGN CONSTRAINTS

4.1 GENERAL AHSE DESIGN CONSTRAINTS

4.1.1 MECHANICAL

a. All AHSE shall be constructed of non-magnetic materials and be worked with tools both non-ferrous themselves and that have never been used on ferrous materials, so as not to influence the spacecraft's magnetic characteristics or magnetic test results.

b. Standard parts and materials and proven processes shall be used throughout the AHSE development cycle as long as they do not degrade the use of the equipment.

c. All operating mechanisms shall have self-acting brakes or mechanical locks, which will prevent undesirable movement or travel and will provide maximum safety to the spacecraft and operating personnel.

d. Alignment requirements and assembly accuracies to be achieved require coordinated tooling and mutual or common assembly and alignment points between differing, but sometimes interacting hardware.

e. Satisfactory performance of all AHSE shall be demonstrated in all cases by a physical walk-through of the spacecraft and support equipment.

4.1.2 ELECTRICAL

a. Power required by an item of AHSE shall be supplied by standard cable and connector details, which must be coordinated with point-of-use facility power equipment.

b. All electrical services and mechanisms shall be shielded so as to introduce no magnetic or stray field effects in the spacecraft operation or test results and to prevent interference with RF test operations.

c. All items of AHSE shall carry an appropriate overload protective device and shall be adequately grounded to provide maximum protection to equipment, spacecraft, and personnel.

4.1.3 ENVIRONMENTAL

a. Voyager AHSE will use JPL spacecraft environmental specifications for ground support equipment as guides.

b. All AHSE shall have the capability of operating without difficulties in the temperature and humidity conditions expected for the spacecraft.

c. AHSE subject to an outdoor environment at the launch site shall be capable of complying with the sand and dust, salt spray, rain and fungus requirements of the JPL specification noted in a, above.

d. Vibration, shock, and other mechanical environments shall not degrade the AHSE operations and the AHSE must be capable of attenuating these load inputs to the spacecraft such that they are lower than any flight loadings anticipated.

4.2 SPECIFIC ASSEMBLY EQUIPMENT DESIGN CONSTRAINTS

Hardware items supplied for field assembly work will provide adequate access to the spacecraft with good protection of the deployable elements and means to remove the completed spacecraft for transportation. In addition, all subsystem handling fixtures shall be designed to fit the assembly tooling without interferences.

Assembly equipment will contain all alignment points as integral design and not require any clamp or bolt on mounts. Optical alignment equipment will be locked into the mount points provided on the equipment.

4.3 SPECIFIC HANDLING EQUIPMENT DESIGN CONSTRAINTS

Weight control of handling equipment will be a design consideration to control costs of shipping, raw materials, fabrication, etc.

The ground transporter must be capable of disassembly into a package compatible with available shipping restrictions.

All dollies, slings, handling fixtures, tools, etc., to be used in the SCF shall be designed to be cleaned in compliance with SCF cleanroom restrictions. Some of the design restrictions are: no unsealed oil or grease lubricated joints will be allowed, no dust catching pockets will be permitted, paint surfaces will not chip, peel, crack or flake, pneumatic rubber tires will not be used.

All handling equipment to be used in the ESF will be designed to be neutral or inactive in the presence of all the propellant fluids to be used in the Voyager program. In addition, it shall be possible to flush the equipment with water without degrading performance.

The ground transporter shall be designed in agreement with the over-the-road regulations in existence at all potential use sites. Such items as parking brakes, stop and running lights, reflectors, maximum weight, turning ratios shall be considered.

4.4 SPECIFIC SHIPPING DESIGN CONSTRAINTS

Design of shipping equipment shall be such as to aid in attenuating mechanical environment inputs to the spacecraft and/or its components in conjunction with the associated handling equipment.

Compliance with regulations of available transportation methods with respect to size, weight, CG location, methods of securing identification, etc., shall be a design consideration.

Shipping of all explosive, pyrotechnic or other hazardous components shall be strictly in accordance with ICC and local regulations.

CII - VB270FD101

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

SUBSYSTEM ASSEMBLY AND HANDLING EQUIPMENT

Index

1.0 SCOPE

This document is a functional description of the assembly fixtures and handling equipment required to remove and reassemble a flight subsystem with the spacecraft bus. This replacement and re-assembly equipment would be used at the Eastern Test Range or at the Jet Propulsion Laboratories to replace a subsystem found to be sub-marginal for flight. This equipment is identical to fixtures and handling equipment required in the fabrication process.

2.0 APPLICABLE DOCUMENTS

GE

VB270FD102 - 1971 Voyager Spacecraft Interface Fixtures; Functional Description

VB270FD103 - 1971 Voyager Spacecraft Alignment Equipment; Functional Description

VB270FD104 - 1971 Voyager Spacecraft Handling Fixture; Functional Description

VB270SR101 - 1971 Voyager Spacecraft Test Objective and Design Criteria

VB270SR102 - 1971 Voyager Spacecraft Design Characteristics and Constraints

3.0 FUNCTIONAL DESCRIPTION

3.1 S/C STRUCTURE

The Spacecraft structure will be built up from the S/C support cone, equipment ring, and capsule support cone. The S/C support cone will be held by an OSE support fixture simulating a Centaur interface. The equipment ring will be lifted and placed on the S/C support cone. This assembly comprises the spacecraft bus.

3.2 PROPULSION SUBSYSTEM

The propulsion subsystem assembly equipment will have the function of positioning the R/P unit under the S/C and will have the capability of raising or lowering the R/P unit to mate with the S/C mounting interface. The assembly fixture will be held by a mobile handling dolly. The handling fixture will be required to be lifted by an overhead crane. Assembly fixtures of this type will be required in-house and in the field.

3.3 GUIDANCE AND CONTROL SUBSYSTEM

The G&C subsystem will be integrated with the main spacecraft structure. The main component bay will be panel-mounted in the equipment ring. No assembly fixtures will be required. Alignment fixtures will be used.

3.4 POWER SUBSYSTEM

The stationary solar panels will be light in weight (less than 10 pounds each) and will be stored in a protective container before assembly. No assembly fixture is required; however, a protective plastic cover will be placed over each solar panel to avoid damage from contamination and impact. Since one of the solar panels will be deployable, a deployment check fixture will be used to hold the stationary solar array while the deployment mechanism is exercised.

3.5 COMMUNICATION SUBSYSTEM

The high-gain antenna will be held in a suitable,assembly fixture prior to assembly. The fixture will be so designed that lifting loads will be taken by tie points on the fixture. An appropriate lifting sling will be used for positioning at assembly. An antenna lifting sling will be made available at ETR and JPL.

The mid-gain antenna will be handled in the same manner. A handling dolly will be used to move the antenna and fixture about the assembly area.

3.6 SCIENCE SUBSYSTEM

The science subsystem main component is the planet scan package (PSP). It will be mounted prior to assembly, on a fixture which will have lift points for a PSP Lifting Sling. An appropriate dolly similar in design to the antenna dollies will hold the PSP for movement about the work area. A set of handling fixtures similar to those used in house will be made available in the field.

3.7 CAPSULE AND BIO-BARRIER SUBSYSTEM

The Bio-barrier will be assembled to the capsule prior to assembly to S/C.

The bio-barrier will be placed in a handling fixture using a bio-barrier lifting sling. The fixture in turn will rest on a dolly for maneuvering about the assembly area. A capsule lifting sling will have the capability of lifting the capsule out of the shipping container and placing it in the bio-barrier for sterilizing. It will also be capable of lifting the entire assembly off the bio-barrier fixture and lowering it on the S/C mounting interface for in-house checkout. Since it is likely that capsule and bio-barrier will be removed for shipment to Field, a matching set of fixtures and dollies will be required in the Field.

4.0 INTERFACE DEFINITION

4.1 MECHANICAL

The following mechanical interfaces exist between the subsystems equipment to be replaced and the assembly and handling equipment:

a. R/P Unit and R/P Unit Assembly Fixture

b. R/P Unit and R/P Unit Lifting Sling

c. High-gain Antenna and Antenna Assembly Fixture

d. Mid-gain Antenna and Antenna Assembly Fixture

e. PSP and PSP Assembly Fixture

f. Capsule and Capsule Lifting Sling

g. Bio-Barrier and Bio-Barrier Assembly Fixture

h. Bio-Barrier and Bio-Barrier Lifting Fixture

i. Solar Panels and Solar Panel Protective Cover

4.2 ELECTRICAL

In all fixtures, clearance will be maintained for all electrical connectors on the S/C.

4.3 OPTICAL

All solar panel protective covers will be of translucent material to allow light to impinge on solar cells.

5.0 PERFORMANCE PARAMETERS

5.1 TEMPERATURE EXTREMES

The Voyager subsystem to Spacecraft Handling Equipment shall be operable from +35°F to +125°F.

5.2 HUMIDITY LIMITS

The Voyager Subsystem to Spacecraft Handling Equipment shall be operable for any value of relative humidity for the temperature extremes in paragraph 5.1.

5.3 DYNAMIC LOADS

The Subsystem to Spacecraft Handling Equipment can be expected to see maximum load factors of 3.

5.4 STRUCTURAL STRENGTH

The Subsystem to Spacecraft Handling Equipment shall be designed to a factor of safety of 5 as compared with yield strength.

$$\frac{\text{Yield Stress}}{\text{Working Stress}} \geq 5$$

5.5 CLEANLINESS PROVISIONS

The Subsystem to Spacecraft Handling Equipment shall be designed for easy cleaning and shall have finishes that will not chip, peel, crack, or flake.

5.6 CONTAMINATION

Protective barriers will be provided to keep particles from contaminating the Spacecraft.

6.0 <u>PHYSICAL CHARACTERISTICS AND CONSTRAINTS</u>

6.1 HANDLING PROVISIONS

a. <u>Tie Down</u> — All dollies shall have eye bolts strategically located for lifting and tie down.

b. <u>Towing</u> — The dolly shall be provided with a removable tow bar ending in a standard ordnance lunette.

c. <u>Shipping</u> — Large fixtures shall be capable of disassembly for shipment.

6.2 SERVICES REQUIRED

a. Overhead Crane — An overhead crane of 25-ton capacity is required in the areas where the handling equipment and S/C components are to be lifted.

b. Hydraset — A 25-ton hydraset is required to go in series with the crane hook to provide minute height adjustments.

6.3 OPERATING CONDITIONS

a. Mobility — The mobile handling equipment shall be operable over improved surfaces and steerable from either end.

b. Spacecraft transport dolly will be steerable from either end.

The magnetic interactions between S/S components, spacecraft and handling/test equipment shall be a minimum. No magnetic material is to be permitted to contact the Spacecraft.

RF interference from the use of handling equipment in conjunction with spacecraft test procedures shall not occur.

Materials used in handling/test equipment fabrication shall not be susceptible to radioactive degradation.

7.0 SAFETY CONSIDERATIONS

7.1 S/C AND COMPONENT SAFETY

a. AHSE shall be designed to avoid interference with spacecraft during assembly operations.

b. Protective covers will be used on solar arrays to prevent damage due to impact.

c. Coverings used in assembly processes shall not be subject to electrostatic discharge.

7.2 PERSONNEL SAFETY

a. AHSE shall be provided with appropriate hand holds to avoid injury to personnel using the fixtures.

b. The dollies shall have adequate braking capability to prevent accidental movement.

Both design and procedures should be evaluated continuously from the safety viewpoint and periodic refinement of each will result in the techniques to be evaluated in the first mating and walk-through of the S/C - AHSE combination.

CII - VB270FD102

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT (AHSE)

INTERFACE FIXTURES (CAPSULE, SCIENCE AND BOOSTER)

Index

1.0 SCOPE

This document is a functional description of the major interface fixtures required during assembly and checkout of the overall spacecraft of the AFETR.

2.0 APPLICABLE DOCUMENTS

GENERAL ELECTRIC

VR270SR101	1971 Voyager AHSE Objectives and Design Criteria
VR270SR102	1971 Voyager AHSE Design Characteristics and Restraints
VB270FD101	Assembly Fixtures - Functional Description
VB270FD103	S/C Alignment Equipment - Functional Description
VB270FD104	S/C Handling Fixture and Dolly - Functional Description
VB235FD105	Weight, c.g. and mass properties equipment - Functional Description

3.0 FUNCTIONAL DESCRIPTION

3.1 CAPSULE INTERFACE FIXTURE

To properly exercise the spacecraft where lander/spacecraft interactions and interfaces exist, a dummy lander interface fixture will provide, to the spacecraft, all the physical, thermal, optical, mechanical influences to be expected of the flight capsule.

The capsule interface fixture should be utilized to evaluate handling techniques and procedures where the spacecraft is affected and to check out mating provisions as a dry run for mating the actual flight capsule.

3.2 SCIENCE INTERFACE FIXTURE

To exercise the spacecraft properly with respect to the science payload, a simulated science interface fixture is required. The fixture provides two areas of simulation: the planet scan package (PSP) simulation and the spacecraft body-mounted scientific instruments. The fixture should provide functions such as the following:

a. To properly assess the influence of the science package on the spacecraft before the actual flight PSP is available, a science interface fixture will be provided.

b. The science interface fixture should present, to the spacecraft, all the physical, thermal, optical, etc. interfaces and influences to be expected of the flight PSP.

c. In addition a set of dummy spacecraft-mounted science packages are required for evaluation of their physical and thermal influences on the spacecraft.

3.3 BOOSTER INTERFACE FIXTURE

A complete in-house "walk-through" of the spacecraft-booster-shroud mating sequence will be required to check out all mechanical interfaces involved and to evaluate the mating procedures developed. To accomplish these purposes, the booster interface fixture must perform the following functions:

a. Provide, to the spacecraft, a mating surface identical to that expected on the actual booster.

b. Locate the spacecraft in the proper orientation with respect to the shroud.

c. Permit checkout of telemetry, air conditioning, power and other required connections between booster and spacecraft to evaluate mating problems and develop optimum procedures.

d. Simulate shroud support points to permit trial run of shroud tie techniques as a part of evaluating spacecraft design for launch pad compatibility.

e. Provide shroud envelope to evaluate shroud to spacecraft clearances.

f. Permit trial test of mechanical buildup of spacecraft, shroud and composite handling fixture for trial installation to booster.

4.0 INTERFACE DEFINITION

4.1 CAPSULE INTERFACE FIXTURE

4.1.1 MECHANICAL

a. The mechanical interface between spacecraft and capsule interface fixture must be the same as the capsule/spacecraft interface, utilizing the same attach points and mechanical mating procedures.

b. No mechanical interface exists between the capsule interface fixture and the shroud.

c. Mechanical provisions must be made for lifting the capsule interface fixture.

4.1.2 THERMAL

a. The capsule interface fixture must present to the spacecraft, a thermal environment identical to that possessed by the actual capsule.

b. Thermal coatings on the fixture side facing the spacecraft must be identical to those on the actual capsule.

c. Heat flows and temperature differentials along existing physical (not necessarily structural) ties between capsule and spacecraft must be simulated.

4.1.3 INSTRUMENTATION

a. Signal requirements between spacecraft and capsule should be simulated such that systems performances can be evaluated completely.

b. Wiring provisions in spacecraft/capsule mating should be duplicated to evaluate mechanical behavior of cabling and disconnect designs.

c. Diagnostic instrumentation will be needed to monitor the behavior of the capsule interface fixture.

4.2 SCIENCE INTERFACE FIXTURE

4.2.1 MECHANICAL

a. The primary mechanical interface between the dummy PSP and the spacecraft is the same as that existing for the actual PSP/Spacecraft combination.

b. The only other mechanical interface foreseen at this time is the method of latching during launch which must be duplicated on the spacecraft for the dummy PSP package.

c. The cabling, both electrical and instrumental, between PSP and spacecraft presents a mechanical interface to be duplicated.

4.2.2 THERMAL

a. Thermal energy interchange between PSP and spacecraft under operating conditions presents a thermal interface to be duplicated where necessary.

b. A thermal interface also exists for certain tests between the PSP and the surrounding environment.

4.2.3 OPTICAL

It may be required to duplicate an optical interface between PSP and spacecraft such as would be required to allow PSP instrument calibration check. (This is not a firm need since some instruments may be either internally calibrated or use targets within the PSP itself.)

4.2.4 ELECTRICAL

The power requirements of the PSP during operation present an electrical interface requirement between spacecraft and PSP.

4.3 BOOSTER INTERFACE FIXTURE

4.3.1 MECHANICAL

a. The primary mechanical interface for the dummy booster interface fixture is the spacecraft mating surface which must be an exact duplicate of the Centaur forward mounting bulkhead.

b. Mating of the shroud requires a mechanical provision on the booster fixture.

c. Handling and base tie-down points constitute a mechanical interface on the booster fixture.

d. A mechanical simulation of all cabling and connector requirements represents a mechanical interface; this covers not only cabling to spacecraft/ Lander, but also all diagnostic instrumentation.

4.3.2 THERMAL

a. Simulation of cooling air flow from the Saturn/Centaur combination may be required.

b. Evaluation of spacecraft environmental conditions within the shroud may require a thermal input to the shroud outer surface.

4.3.3 ELECTRICAL

Simulation of electrical power and signals to spacecraft/Lander may be required.

5.0 PERFORMANCE PARAMETERS

See paragraphs 3.0 and 4.0.

6.0 PHYSICAL CHARACTERISTICS AND RESTRAINTS

6.1 CAPSULE INTERFACE FIXTURE

a. The capsule interface fixture will provide to the spacecraft during pertinent tests, the weight, CG location and mass properties possessed by the actual lander capsule.

b. The capsule fixture need not be the actual shape and size of the flight capsule except in the area of minimum fairing--capsule clearance. A light, temporary framework simulating the external envelope of the capsule will be available to attach to the interface fixture for evaluating clearances and mating interferences.

c. The capsule fixture should radiate the same radiant energy, in the same manner, as expected from the flight capsule.

d. The capsule fixture will utilize the same lifting devices as the flight capsule to evaluate capsule - spacecraft mating techniques.

6.2 SCIENCE INTERFACE FIXTURE

a. The science interface fixture will provide to the spacecraft, the identical weight, CG, mass properties and volume occupied characteristics of the actual science package.

b. The science fixture will radiate to the spacecraft through the same thermal coating as the actual science package, the radiant energy to be expected from the flight science package.

c. The science fixture will utilize the same handling equipment and procedures as the flight science package and be mounted on the spacecraft using the same mounts.

d. The science interface fixture will also include simulations of all spacecraft body mounted instrumentation.

6.3 BOOSTER INTERFACE FIXTURE

a. Basically the dummy booster interface fixture must provide to the spacecraft/ Lander combination, the mechanical, electrical, etc., inputs the actual booster would possess.

b. The physical size of the fixture need be no longer than necessary to adequately simulate the booster interface characteristics, i.e., it is not necessary that an entire dummy Centaur be obtained.

7.0 SAFETY CONSIDERATIONS

A complete checkout of mating and interface procedures has to be accomplished, using the PTM, before flight hardware is committed to an interface test or a system check using interface simulators. This is required to avoid inflicting damage or otherwise jeopardizing the quality of the flight hardware.

CII - VB270FD103

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

ASSEMBLY EQUIPMENT

ALIGNMENT EQUIPMENT

Index

1.0 SCOPE

This document is a functional description of the recommended mechanical alignment equipment, procedures, and tolerances for alignment of the 1971 Voyager Spacecraft (S/C) and its subsystems, at the AFETR.

2.0 APPLICABLE DOCUMENTS

General Electric

VB270FD101 — Assembly Fixtures; Functional Description

VB270FD102 — Interface Fixtures; Functional Description

VB270FD106 — OSE Weight and Balance Equipment; Functional Description

VB235FD105 — Weight, C.G., and Mass Properties Equipment; Functional Description

VB270FD104 — S/C Handling Fixture and Dolly; Functional Description

VR270SR101 — 1971 Voyager Spacecraft AHSE Test Objectives and Design Criteria

VB220FD113 — Table 2-3a Mechanical Alignment Tolerances

VR270SR102 — 1971 Voyager Spacecraft AHSE Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

The following are the principle alignment functions required in assembly of components to the spacecraft. These functions are analagous to alignment functions performed in the factory.

3.1 S/C TO PROPULSION UNIT

The nozzle thrust vector, which is coincident with the center line of the nozzle, is to be aligned coincident with the center line of the spacecraft within tolerance specified in paragraph 5.0 of this document. The propulsion unit mounting surface tolerances determine the pointing error, and displacement. Both errors can be measured by autocollimation techniques, and can be brought into specification by shimming at the mounting surface.

3.2 S/C TO CAPSULE

The capsule displacement and pointing angle are found in relation to the engine centerline. This can be done also by autocollimation and autoreflection techniques, using an appropriate reference fixture which will be applied to the capsule mounting surface. The tolerances on the flatness of the capsule mounting surfaces will be held closely to minimize adjustment, and keep stress levels low. Adjustments must be made by means other than shims as this is a separation surface.

3.3 ATTITUDE CONTROL

Nozzles to S/C — Each nozzle will be aligned with reference to the spacecraft. Plug targets are to be used with theodolites to determine pointing error between each nozzle and main S/C centerline fixture, in three planes. They can also be checked by optical means for parallelism and coincidence.

3.4 PSP ALIGHMENT TO SPACECRAFT

The PSP is to be aligned to the spacecraft in the deployed position and its position related to the S/C axes by optical methods. The PSP mounting interface with S/C will be checked for manufacturing accuracy. Design of the interface should be of such a nature as to allow adjustment by shimming or other means.

3.5 ANTENNA ALIGNMENT

The antenna mounting interface is checked by appropriate fixturing and optical methods, utilizing reflecting target mirrors and theodolites. The relationship of the antenna mounting surface to the S/C axes will be established to within tolerances given in paragraph 5.0.

3.6 S/C TO LAUNCH VEHICLE ALIGNMENT

The alignment of the S/C-to-launch vehicle is checked using an accurate interface fixture and optical equipment. The S/C-to-launch vehicle interface will provide the reference surface to measure tilt angle and displacement of the S/C from the launch vehicle vertical centerline.

3.7 SOLAR PANELS TO S/C

The alignment of fixed solar panels to the S/C vertical axis will be checked by a mechanical fixture and telescope relative to the engine nozzle centerline. The tolerances required to measure relationship of panel plane angle to nozzle C/L are not severe.

4.0 INTERFACE DEFINITION — ALIGNMENT EQUIPMENT TO SPACECRAFT

4.1 OPTICAL

All optical interfaces will be between OSE scopes and collimators and OSE alignment fixtures.

4.2 MECHANICAL

Mechanical interfaces exist between alignment fixtures and vehicle in the following areas:

4.2.1 PROPULSION UNIT ALIGNMENT INTERFACES

a. Engine nozzle to plug target fixture.

b. Propulsion unit mounting surface of S/C to propulsion unit alignment simulator.

4.2.2 SPACECRAFT ALIGNMENT INTERFACES

a. S/C-to-S/C handling and alignment fixture.

b. S/C attitude control nozzle mounting surface to alignment fixtures.

c. S/C antenna mounting surface to antenna alignment target fixture.

d. S/C Solar panel supports to Solar Panel Alignment fixture.

4.3 ELECTRICAL

An electrical interface exists between in-house electrical power and all OSE collimating devices. A power source of 115V AC 60 cps 1 Φ will be required.

The alignment equipment will have a direct interface with the assembly and test building. Floors and structure should be adequate to support alignment equipment with minimum outside vibration.

5.0 PERFORMANCE PARAMETERS

The overall alignment accuracies of each item listed in paragraph 3.0 are given below:

a. S/C-to-Propulsion Unit

Pointing error	±0.5 deg.
Displacement	±0.188 in.

b. S/C-to-Capsule

Pointing error	±10 min.
Surface flatness	±0.020 in.
Displacement	±0.10 in.

c. Attitude Control Nozzles to S/C

Parallelism	±1 deg.
Indexing	±1 deg.

d. PSP Alignment to S/C — The PSP mounting surface to be aligned to S/C axes within ±20 min.

e. Antenna Alignment to S/C

Antenna beam axis to antenna mechanical axis	±9 min.
Antenna mechanical axis to S/C axes	±6 min.

f. S/C to Launch Vehicle

Pointing angle	±5 min.
Displacement	±0.15 in.

g. Solar Panels to Spacecraft

Deployable panel plane perpendicular to S/C roll axis	±1.0 deg.
Fixed panel plane perpendicular to S/C roll axis	±0.50 deg.

A detailed list of alignment tolerances is given in VB220FD113. Items a through g, above, represent gross tolerances of major subsystem alignment.

Errors of telescopes and Collimating equipment should be ±3 seconds of arc or less.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The alignment fixtures required will be rigid in construction and as light as possible, consistent with accuracy. No permeable materials will be used in the construction of the fixtures.

Scopes and target devices shall be of high quality to ensure accuracy and reliability. Commercial theodolites with accuracies of position of ±3 arc seconds and commonly available and will satisfy any requirement found necessary for Voyager alignment. Commercially available target mirrors will be adequate for all Voyager needs.

7.0 SAFETY CONSIDERATIONS

Alignment equipment shall be designed such that no loads are placed on parts of the S/C structure which are not load-bearing members.

Lens caps and covers will be provided to keep contaminants out of optical equipment.

Closed-off alignment areas are required to preserve the accuracy and integrity of alignments.

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

SPACECRAFT HANDLING FIXTURE

GROUND TRANSPORTERS AND DOLLIES

Index

1.0 SCOPE

This document provides a functional description of the handling fixture, ground transporter, and the dollies used to perform assembly, test and prelaunch operations on the Spacecraft Bus and Overall Flight Spacecraft at the Air Force Eastern Test Range and at Jet Propulsion Laboratories.

2.0 APPLICABLE DOCUMENTS

GE

VR270SR101	1971 Voyager Spacecraft AHSE Test Objectives and Design Criteria
VR270SR102	1971 Voyager Spacecraft AHSE Design Characteristics and Restraints

3.0 FUNCTIONAL DESCRIPTION

3.1 SPACECRAFT HANDLING FIXTURE

The spacecraft handling fixture is the primary means of moving the Spacecraft Bus and the Overall Flight Spacecraft as required. The fixture is lifted by overhead crane and matches the Spacecraft's shipping container and ground transporter. A functional sketch of the handling fixture is shown in Figure 3-1.

3.2 SPACECRAFT GROUND TRANSPORTER

The Spacecraft Bus and Overall Flight Spacecraft are provided with ground mobility between assembly, checkout, and launch facilities by the ground transporter. A functional sketch of the ground transporter is shown in Figure 3-2. The ground transporter is capable of rotating the spacecraft around its Z axis by electrical power or hand crank. A functional flow diagram is shown in Figure 3-3.

3.3 SPACECRAFT HANDLING DOLLIES

What ground mobility is required within laboratories and assembly buildings will be provided by dollies designed to be compatible with the clean internal conditions in those facilities.

4.0 INTERFACE DEFINITIONS

4.1 SPACECRAFT HANDLING FIXTURE WITH SPACECRAFT AND OTHER AHSE

4.1.1 MECHANICAL

The primary mechanical interface of the spacecraft handling fixture is the set of points at which the fixture contacts and supports the spacecraft. The fixture shall also be capable of matching the spacecraft support sling assembly.

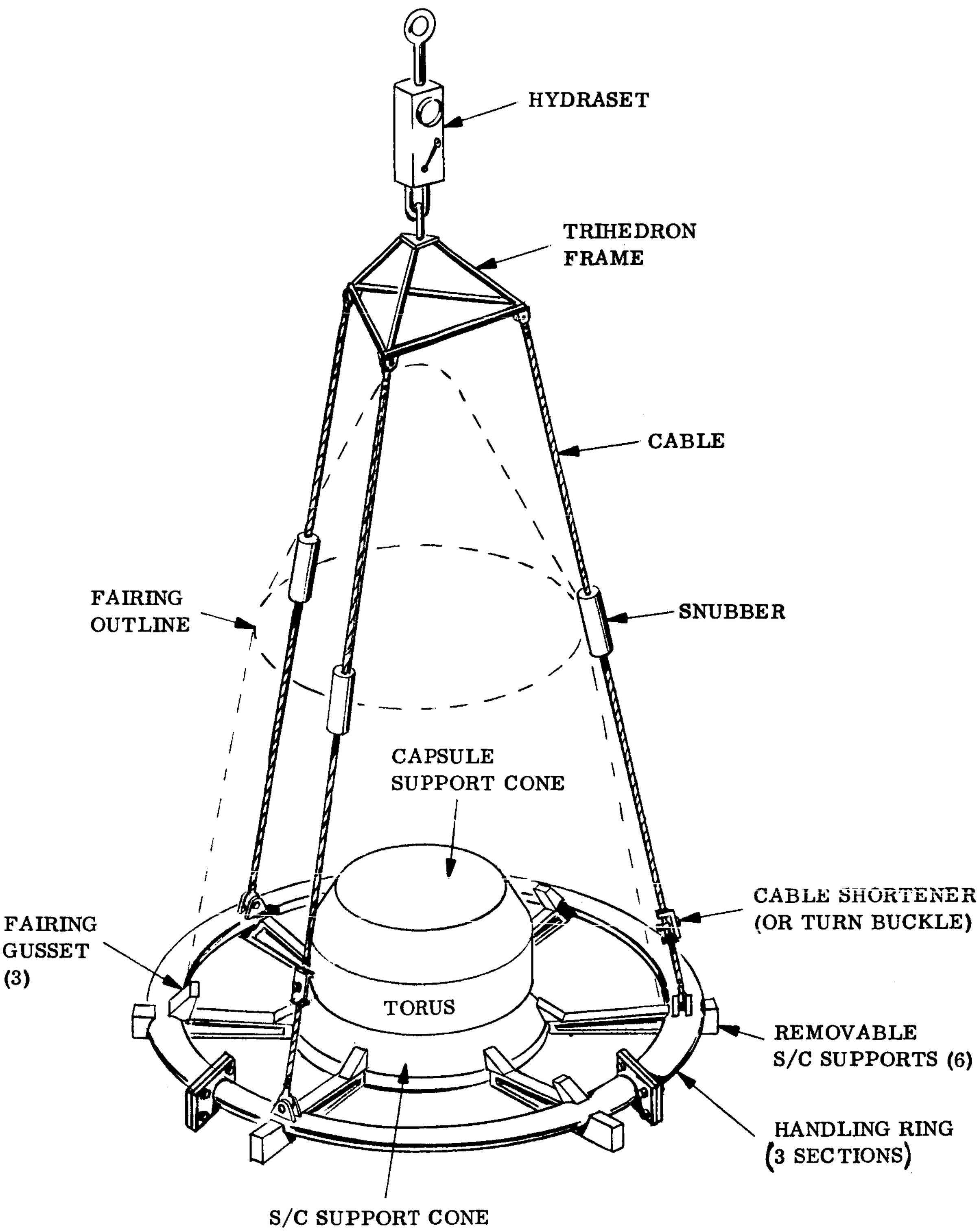


Figure 3-1. S/C Lifting and Handling Fixture

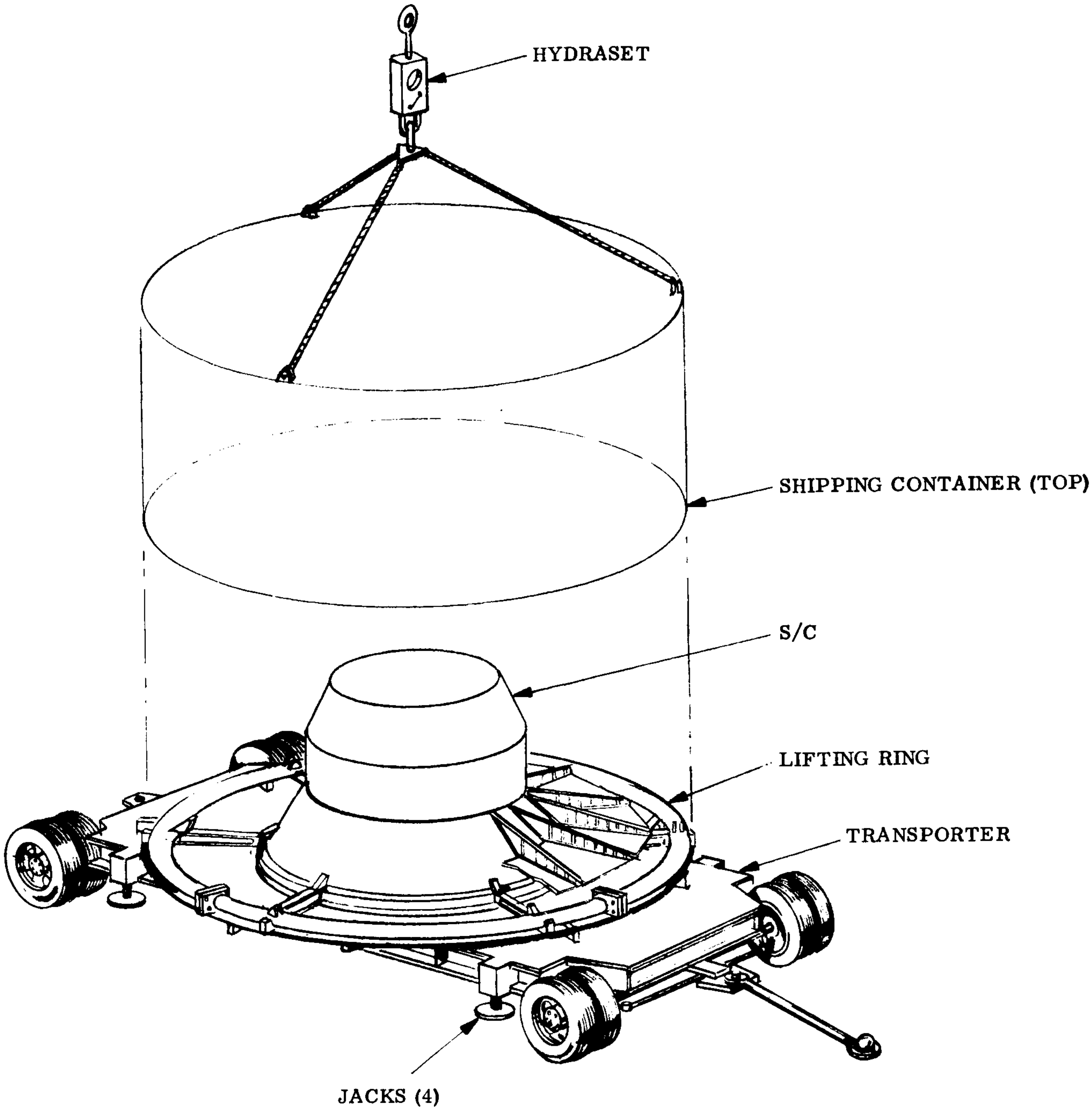
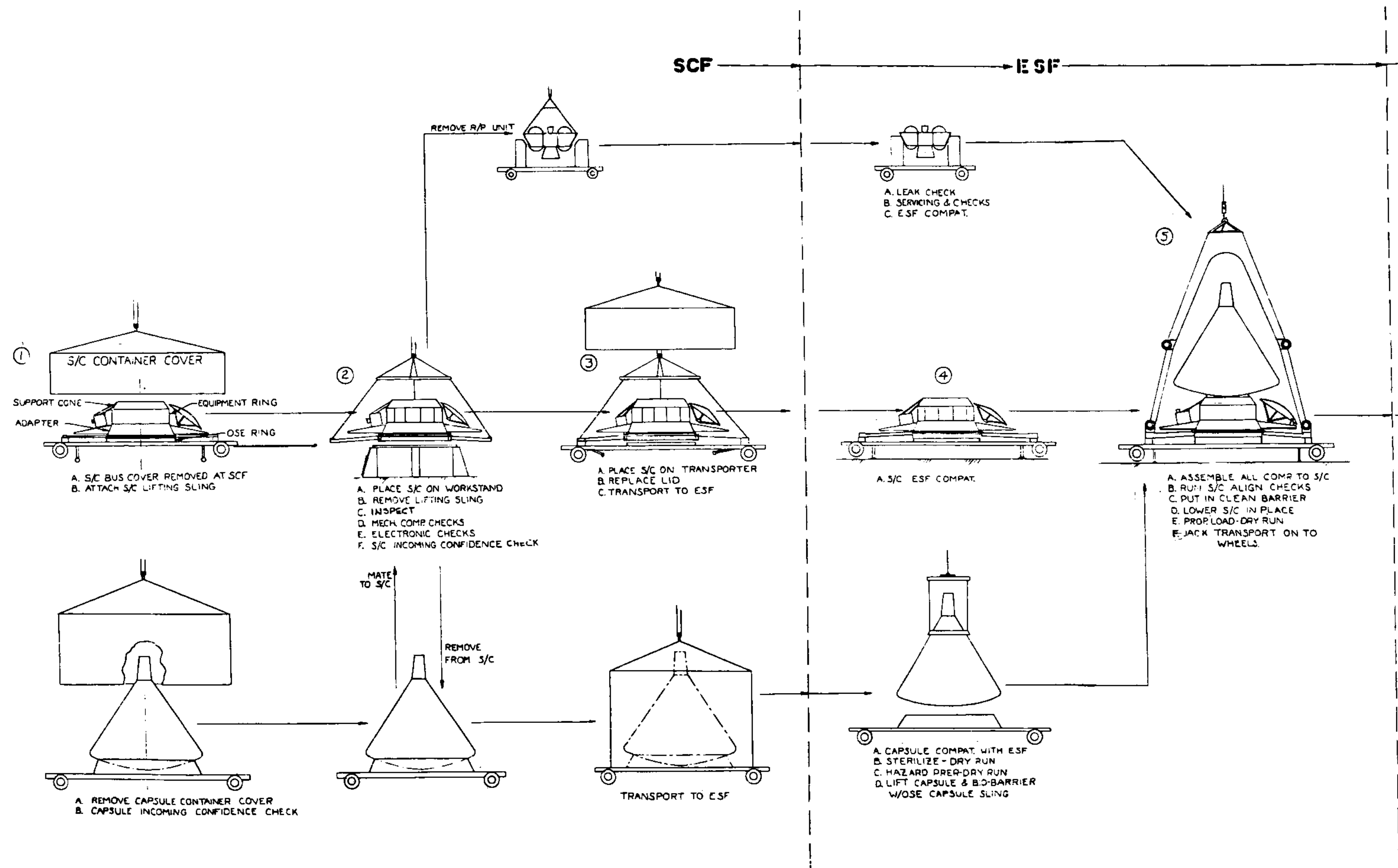


Figure 3-2. Voyager S/C Transporter



SCF
ESF
REMOVE R/P UNIT
A. LEAK CHECK
B. SERVICING & CHECKS
C. ESF COMPAT.
1
S/C CONTAINER COVER
SUPPORT CONE
EQUIPMENT RING
ADAPTER
OSE RING
A. S/C BUS COVER REMOVED AT SCF
B. ATTACH S/C LIFTING SLING
2
A. PLACE S/C ON WORKSTAND
B. REMOVE LIFTING SLING
C. INSPECT
D. MECH. COMP. CHECKS
E. ELECTRONIC CHECKS
F. S/C INCOMING CONFIDENCE CHECK
3
A. PLACE S/C ON TRANSPORTER
B. REPLACE LID
C. TRANSPORT TO ESF
4
A. S/C ESF COMPAT.
5
A. ASSEMBLE ALL COMP. TO S/C
B. RUN S/C ALIGN CHECKS
C. PUT IN CLEAN BARRIER
D. LOWER S/C IN PLACE
E. PROP. LOAD-DRY RUN
F. JACK TRANSPORT ON TO WHEELS.
MATE TO S/C
REMOVE FROM S/C
A. REMOVE CAPSULE CONTAINER COVER
B. CAPSULE INCOMING CONFIDENCE CHECK
TRANSPORT TO ESF
A. CAPSULE COMPAT. WITH ESF
B. STERILIZE - DRY RUN
C. HAZARD PREP-DRY RUN
D. LIFT CAPSULE & BIO-BARRIER W/OSE CAPSULE SLING

1

5

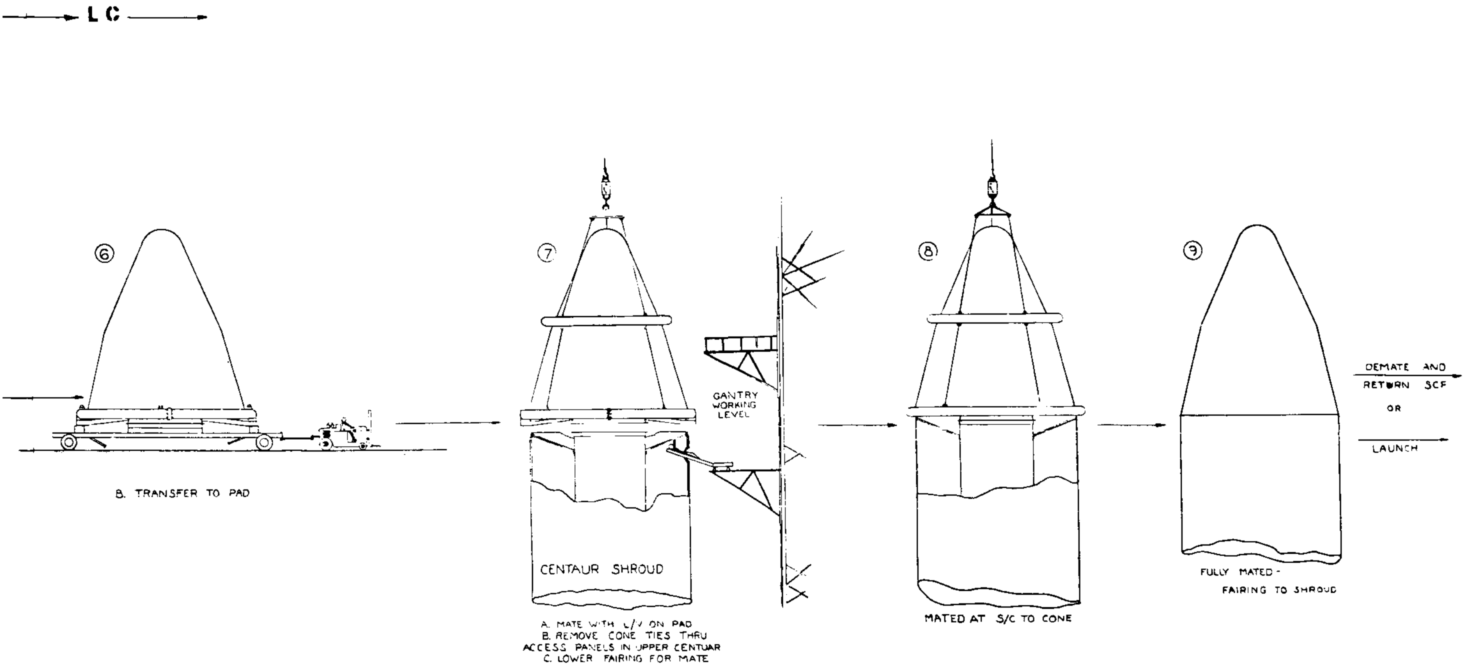


Figure 3-3. Typical Voyager Field Handling Flow

2

For movement of the spacecraft encapsulated within the fairing, the spacecraft handling fixture must mechanically mate with the fairing handling ring.

Assembly of the spacecraft handling fixture on the ground transporter is required. Matching of the spacecraft handling fixture to the bus handling dolly for handling of the bus subassembly is another mechanical interface.

4.1.2 ELECTRICAL

An electrical power supply connection on the handling fixture is required to power the spacecraft positioning motors. During shipping of the spacecraft, the handling fixture must mechanically match the shipping container.

4.2 SPACECRAFT GROUND TRANSPORTER WITH SPACECRAFT HANDLING FIXTURE

4.2.1 MECHANICAL

The spacecraft ground transporter mechanically mates with the spacecraft handling fixture for transporting the bus subassembly with or without the capsule, antennae, solar arrays, etc.

The ground transporter carries a cover sized to enclose the entire spacecraft assembly, except for the capsule, and makes an airtight joint with it.

4.3 SPACECRAFT HANDLING DOLLIES WITH OTHER AHSE

4.3.1 MECHANICAL

The spacecraft handling dolly mechanically mates to the spacecraft handling fixture which supports the actual spacecraft. The handling dollies shall have anchoring devices to allow them to be secured firmly to the floor.

5.0 PERFORMANCE PARAMETERS

5.1 SPACECRAFT HANDLING FIXTURE

a. The handling fixture shall withstand temperature extremes of -35° F to +125° F without degrading performance.

b. Relative humidity of 50% for the temperature ranges noted in a, above, shall not degrade performance of the handling fixture.

c. The handling fixture shall not experience significant deformation when fully loaded under load factors up to 3 "g's".

5.2 GROUND TRANSPORTER

a. The ground transporter shall be roadable at a maximum speed of 5 mph over good, paved surfaces.

b. The ground transporter shall be designed to withstand temperature extremes of -35° F to +125° F without degrading performance.

c. A relative humidity of 50% for the temperature ranges noted in b, above, shall not degrade performance of the ground transporter.

d. The ground transporter shall not experience significant deflections when fully loaded under load factors of up to 3 "g's".

5.3 SPACECRAFT HANDLING DOLLIES

a. The handling dollies will be capable of being moved about on a smooth, paved surface on standard casters.

b. The handling dollies shall not experience significant deflections when loaded under load factors of up to 3 "g's".

6.0 PHYSICAL CHARACTERISTICS AND RESTRAINTS

6.1 SPACECRAFT HANDLING FIXTURE

a. The handling fixture shall be designed for use in the clean room atmosphere of the SCF and ESA. This involves painted surfaces that do not chip, peel or flake, no lubricated joints, no dust pockets, easily cleaned configuration, etc.

b. All construction materials shall be non-magnetic and the fixture shall be formed and assembled with non-magnetic tools.

6.2 SPACECRAFT GROUND TRANSPORTER

a. The ground transporter is not intended to transport the spacecraft over public highways at any time.

b. The transporter must be so designed as to be easily disassembled to reduce its shipping width.

c. Non-magnetic construction materials must be used throughout the transporter and all electrical wires, leads, motors, switches, etc. must be shielded to prevent magnetic inferferences.

d. The transporter shall be capable of moving the spacecraft when encapsulated in its fairing and loaded with all propellants, gases, etc.

6.3 SPACECRAFT HANDLING DOLLIES

a. The handling dollies must be designed for use in the clean room atmosphere of the SCF and ESA. Such design constraints must be implemented as the following: painted surfaces that do not chip, peel or crack; no lubricated joints; no pneumatic rubber tires for movement; no pockets for collecting dust; an easily cleaned configuration; etc.

b. All construction materials must be non-magnetic and the fixture shall be formed and assembled with non-magnetic tools.

c. The dollies are not intended for rolling about the floor when loaded. Wheels should be equipped with footbrakes.

7.0 <u>SAFETY CONSIDERATIONS</u>

7.1 SPACECRAFT HANDLING FIXTURE

The handling fixture materials shall be inert in the presence of fluids to be handled in the ESA or be shielded from them.

7.2 GROUND TRANSPORTER

a. The ground transporter shall be equipped with both manual and power brakes.

b. The transporter shall be equipped with a static ground plus a ground strap to the spacecraft and fairing when installed.

c. For use in the ESA, all construction materials shall be either inert in the presence of the fluids being handled or be shielded from contact with them.

d. The transporter shall be provided with lifting eyes and tie down points to restrain its movements physically, if required.

7.3 SPACECRAFT HANDLING DOLLIES

a. The dollies shall be equipped with jacks and tie bolts to anchor them firmly on the floor when loaded in use.

b. For dollies used in the ESA, ground straps shall be provided.

c. All dollies for use in the ESA shall be constructed of materials inert in the presence of the fluids being handled or else be protected from contact with them.

CII - VB270FD105

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

MAGNETIC MAPPING EQUIPMENT

Index

1.0 SCOPE

This document provides a functional description of the magnetic mapping equipment and facilities necessary to perform magnetic mapping of the 1971 Voyager Spacecraft during checkout and calibration operations at JPL, ETR, and the place of manufacture. It forms a part of the AHSE,

2.0 APPLICABLE DOCUMENTATION

GE

VB270SR101	1971 Voyager Spacecraft AHSE Objectives and Design Criteria
VB270SR102	1971 Voyager Spacecraft AHSE Design Characteristics and Constraints
VB270FD104	1971 Voyager Spacecraft Handling Fixture; Functional Description

3.0 DESCRIPTION

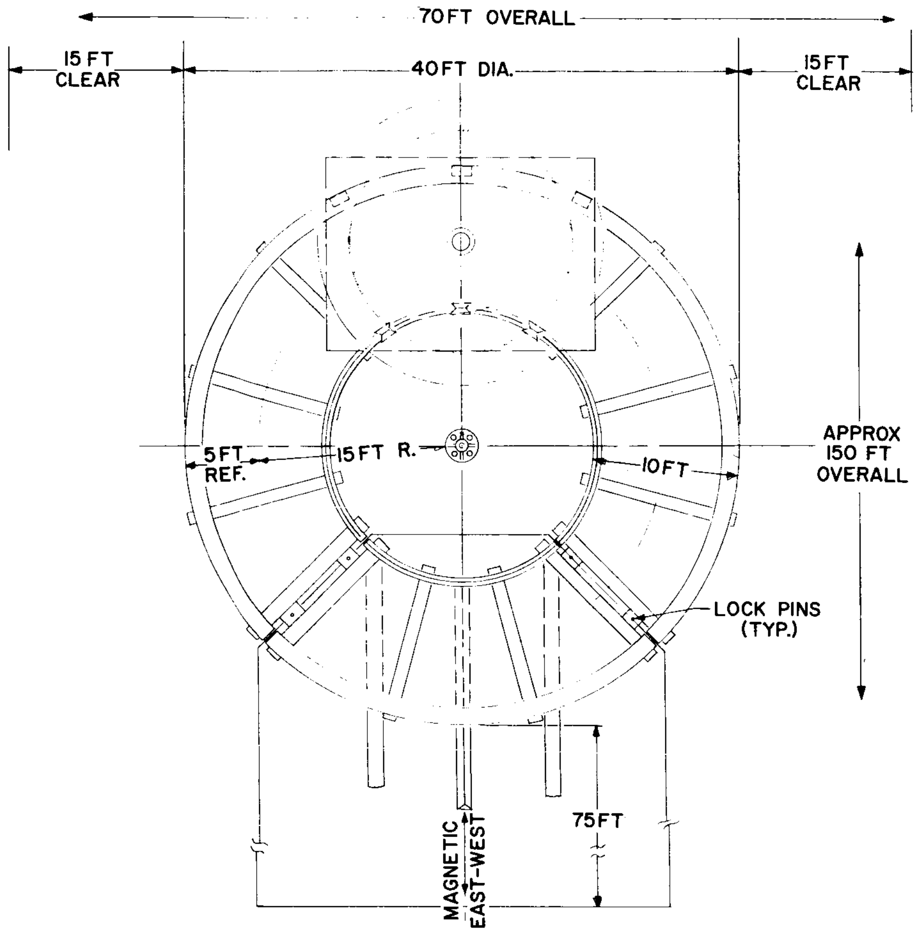
The magnetic mapping equipment is required to perform field mapping and deperming operations on the spacecraft. These operations would probably be required to be performed at several locations. The equipment described is intended for use at any such location.

3.1 PERMANENT MAGNETIC FIELD MAPPING

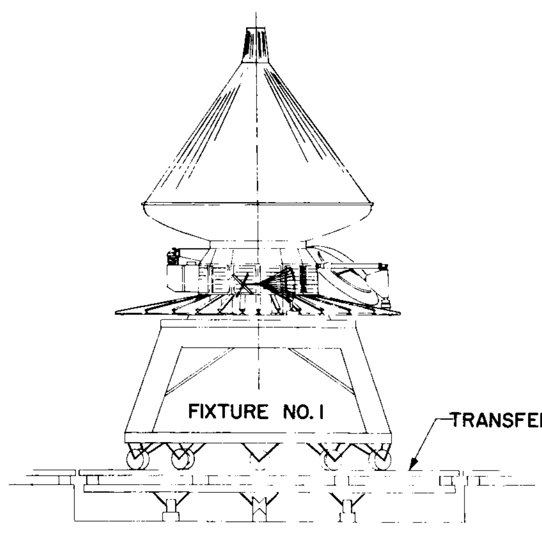
The object of this test is to determine components of the permanent magnetic field of the overall S/C and coefficient of induction for an external magnetic field at the location of the magnetometer sensor. In this test, the inactive S/C is used without external power cables, and without the flight magnetometer.

Figure 3-1 is a layout sketch of the Magnetic Mapping equipment. It shows the track, dolly, transfer carriage, magnetometer support and the two special spacecraft holding fixtures which are required for this operation.

The nonmagnetic fixture is mounted on a dolly traversing an approximate 40-foot O.D. circular track, at the center of which is mounted the Test Magnetometer Sensor. The Test Magnetometer Sensor will be rigidly mounted so that when the spacecraft is mounted on the dolly, the test magnetometer sensor occupies the position, relative to the spacecraft, that is occupied by the deployed flight magnetometer sensor. The two fixtures will allow the complete spacecraft (and capsule) to be rotated through 360 degrees about the axis parallel to the Z axis and passing through the position of the center of the magnetometer sensor. Using the second fixture (see Figure 3-1), the spacecraft (only) will be rotated through 360 degrees about a vertical axis through the position of the magnetometer sensor with the spacecraft Z axis horizontal.



PLAN VIEW

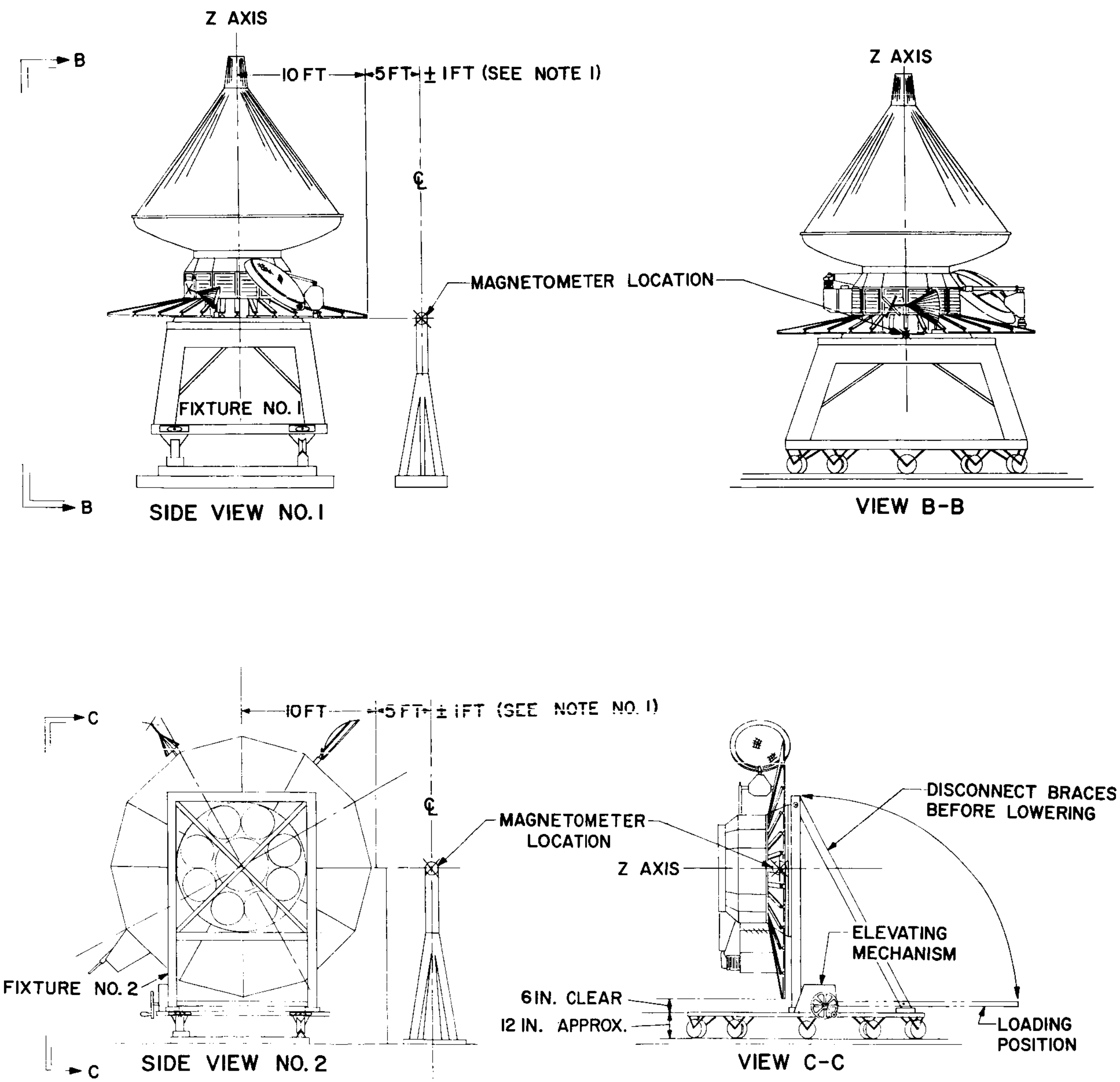


ELEVATION VIEW

NOTES:
1.- S/C SUPPORT PLATFORM LATERALLY ADJUSTABLE

1

B



2

Figure 3-1. Magnetic Mapping OSE

Seventy-five feet of accurately reproducible travel away from the test magnetometer sensor in the direction of magnetic East-West is required to be accomplished with the use of a transfer carriage.

The magnetic mapping fixture and facility should be calibrated so that the relative orientation and position of the spacecraft assembly and the test magnetometer sensor is reproducible within 1.0 degree and 1/8 inch, respectively.

All of the equipment should be set up in an available closed area, including the tracks. The detailed design of this equipment should permit set up and adjustment to be accomplished within one month.

3.2 VARIABLE MAGNETIC FIELDS

In mapping the spacecraft, the effect of movable appendages (e.g., antennas, etc.) must be evaluated. The following will be accomplished:

a. Mapping to determine the effect of deploying and articulating the antennas, etc. The fixtures and facility should permit complete freedom of motion of all such movable appendages.

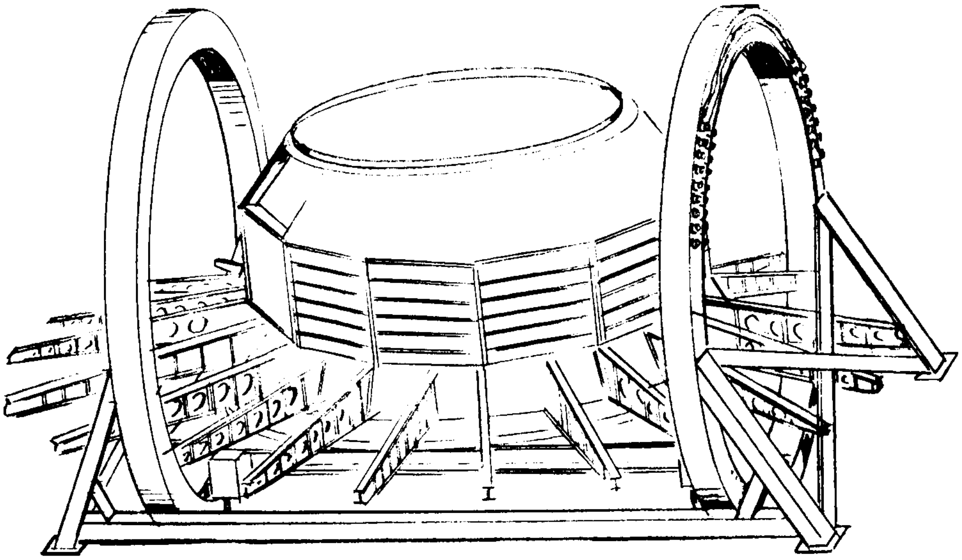
b. For the planet scan package (PSP), the mapping will be accomplished in the stowed position and in the deployed position. In addition, motions of the PSP in the deployed position will be evaluated.

Mapping of current fields is accomplished by using a magnetometer, situated so as to simulate the flight magnetometer, to sense the effects of current fields. The equipment illustrated in figure 3-1 is not required for this type of mapping. This mapping can be accomplished within the Spacecraft Checkout Facility. All normal modes of operation will be evaluated as well as principal failure modes. The current loop effect of motions of both the antenna and PSP will also be evaluated.

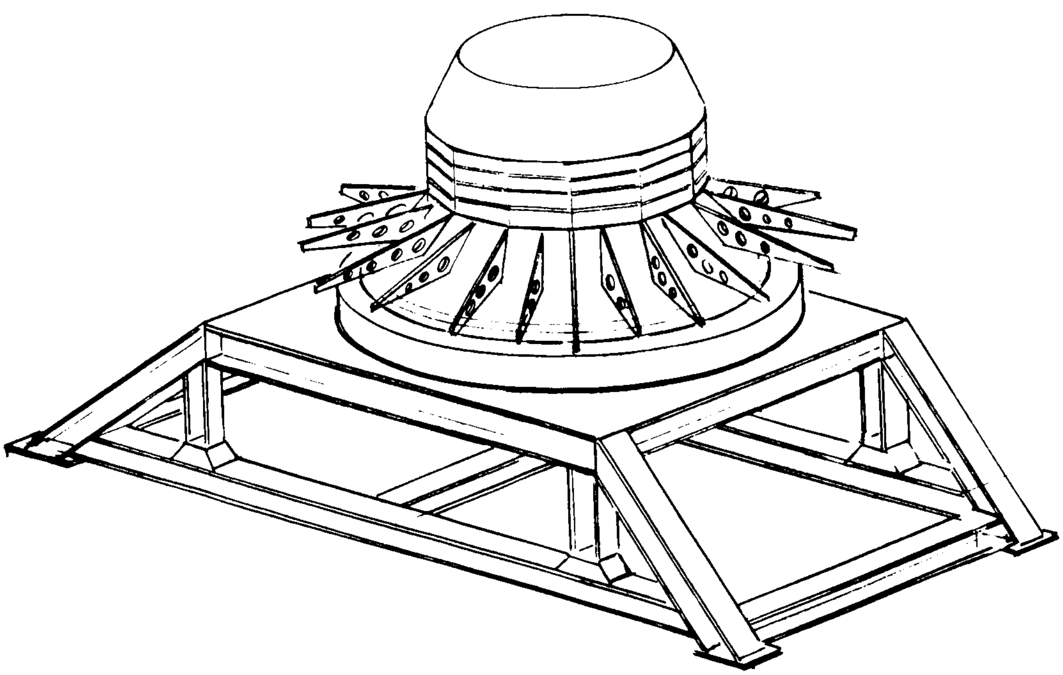
The magnetic mapping of the Voyager Spacecraft will be accomplished by rotation of the spacecraft about two of the three orthogonal coordinate axes parallel to the spacecraft coordinates and passing through the position of the flight magnetometer sensor. Spacecraft magnetic fields can be derived by analysis and reduction of the data obtained in these magnetic mapping operations.

3.3 DEPERMING

A deperming operation is required to remove residual magnetism from the spacecraft bus structure and components. This will be accomplished by utilizing a pair of 7-foot (nominal) diameter coils, similar to the JPL Mariner C Installation, mounted parallel to each other in a vertical position and spaced approximately 10 feet apart (see Figure 3-2).



A. Spacecraft Being Depermed About X and Y Axes



B. Spacecraft Being Depermed About Z Axis

Figure 3-2. Deperming

The Voyager Spacecraft Bus Assembly, less the solar panels, will be placed on a positioning fixture between the coils, with the Z axis in the vertical position and parallel to the plane of the coils. Deperming about the X and Y axes will be accomplished by positioning the spacecraft in prescribed increments and simultaneously exercising control of the coil current during a pre-programmed cycle.

In addition to the above operations, deperming about the Z axis will be accomplished using a single coil placed in the horizontal plane, with the spacecraft supported in a fixed position above and parallel to the plane of the coil with the Z axis vertical as shown in Figure 3-2. Programmed cycling of the coil current with sufficiently precise control, will result in a successful deperming operation.

It is intended that the Spacecraft Handling Fixture, VB270FD104, and the Ground Transporter, be utilized during deperming.

4.0 INTERFACE DEFINITIONS

4.1 MECHANICAL

The magnetic mapping mechanical OSE will interface with the spacecraft handling fixture and must provide the necessary mechanical interfaces for handling and protecting the Voyager Spacecraft during all operations involving, and related to, Magnetic Mapping.

4.2 ELECTRICAL

Facility and OSE power sources and power conditioning are necessary for all required vehicle inputs (i.e., flight simulation with STC, etc.) during all current loop magnetic mapping operations with the Voyager Spacecraft and/or the related subsystems and components.

4.3 THERMAL

Because the spacecraft systems are required to be operating during the current loop test portions of the magnetic mapping operation at the STC, thermal conditioning may be necessary to provide the Voyager Spacecraft and/or the related subsystems with the required optimum thermal characteristics.

4.4 MAGNETIC

The Magnetic Mapping OSE and Facilities will necessitate a maximum effort during Design and Operating Phases to minimize magnetic contamination of the Voyager Spacecraft and to prevent the development of interfering or spurious magnetic fields during magnetic mapping operation. This will involve the use of non-magnetic materials in tools and equipment; magnetic mapping of background fields in OSE equipment and facilities; the use of deperming techniques on the facilities, equipment, and tools. It may be necessary, in addition, to perform magnetic mapping operations during periods of minimum activity in the facility area, such as at night, on weekends, etc.

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

WEIGHT, BALANCE, AND MASS PROPERTIES EQUIPMENT

Index

1.0 SCOPE

This document describes the equipment to be used in determining the weight and center of gravity of the Voyager S/C. C.G. determination will be made both along the Z axis and in the XY plane. This equipment is required at the manufacurer's plant, at the AF Eastern Test Range, and at the Jet Propulsion Labs.

2.0 APPLICABLE DOCUMENTS

GE

VB220FD113	Table 2-3a Mechanical Alignment Tolerances
VB270FD101	Assembly Fixtures; Functional Description
VB270FD102	Interface Fixtures; Functional Description
VB270FD103	S/C Alignment Equipment; Functional Description
VR270SR101	Voyager AHSE Test Objectives and Design Criteria
VR270SR102	AHSE Design Characteristics and Restraints
VB235FD105	Determination of Weight, Center of Gravity and Mass Properties

3.0 FUNCTIONAL DESCRIPTION

The weight and center of gravity of the S/C will be found for three phases of flight:

a. Earth orbit (entire S/C, including Capsule)

b. Transit (entire S/C, Capsule, appendages deployed)

c. Mars Orbit (S/C Less Capsule)

Weights and centers of gravity will be found using a multiple load and cell system. A fixture will be fabricated which will have a load point on each load cell 120 degrees apart. The weight and center of gravity of the fixture will be accounted for and the fixture leveled. When the S/C is placed on the fixture with the field joint interface parallel to the fixture, the amount of difference in cell readouts along with a known moment arm to the center of gravity of the fixture, is used to compute location of the center of gravity in the X-Y plane.

The S/C is then positioned with the X axis vertical and the center of gravity is found in the X-Z plane. The fixture will have the capability to allow the S/C to be rolled 90 degrees and pitched 90 degrees at the same time. Centers of gravity will be checked against the known center of gravity of the fixture.

The sum of the weight indications of the load cells, less the known weight of the fixture will be the total weight of the S/C. The method described above will be used for all three conditions of the flight S/C.

4.0 INTERFACE DEFINITION

4.1 OPTICAL

An optical interface exists between the S/C three-cell fixture, and leveling transit.

4.2 MECHANICAL

a. Weighing load cells interface with S/C through a three-legged fixture capable of rotating the S/C through 90 degrees in the X-Z or Y-Z plane.

b. Load cell structure must interface with facility floor and have leveling capability.

c. Tie points on top of weighing fixtures allow them to interface with lifting sling.

d. Load cells must be calibrated before each use, with a calibrating mass of accurately known weight.

4.3 ELECTRICAL

a. An electrical interface shall exist between load cells and readout device.

b. Power (110V, 60 cycle, single phase) must be provided to drive force indicator.

c. Power must be provided to rotate item to be weighed about two axes.

5.0 PERFORMANCE PARAMETERS

a.	Load cell accuracy (Full Capacity)	±0.01%
b.	Tolerance between balance points	±0.003"
c.	Fixture leveling accuracy	±10 arc sec.
d.	Center of gravity accuracy X, Y, Z	±0.05"

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

a. Load cells used shall be three in number each with a 5000-pound capacity (full deflection) and having leveling features.

b. Fixtures for holding S/C and other components shall be as rigid as possible to keep deflections to a minimum.

c. Fixtures will have three pickup points for the lifting sling.

d. Turn over portion of the fixture shall not allow any part of S/C or subassemblies to take loads in excess of S/C design loads.

7.0 SAFETY CONSIDERATIONS

a. Weight and center of gravity equipment shall be designed such that no loads are placed on the S/C structure which are not load-bearing members.

b. Covers will be provided to keep contaminants out of critical parts of Weight and Balance equipment.

c. All Weight and Balance fixtures shall be provided with appropriate hand holds to avoid injury to personnel using the equipment.

d. Both design and procedures will be evaluated continuously from the safety aspect, and periodic refinement of each will result in the techniques to be evaluated in the first "walk-through" and mating of the S/C and Weight and Balance equipment.

ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

SHIPPING EQUIPMENT

Index

1.0 SCOPE

This document describes the functional characteristics of the shipping equipment required to transport the spacecraft from the manufacturer's plant to using sites.

2.0 APPLICABLE DOCUMENTS

GE

VR270SR101	1971 Voyager Spacecraft AHSE, Objectives and Design Criteria
VR270SR102	1971 Voyager Spacecraft AHSE, Design Characteristics and Constraints

3.0 FUNCTIONAL DESCRIPTION

The equipment described here is that required to facilitate movement of the flight spacecraft. The spacecraft shipping container is the prime protection for the spacecraft bus during shipment. As such, it is required to completely protect its contents from all harmful environmental factors and still be easily handled, transported and secured. The container houses the complete spacecraft with all subsystem components installed, except for the capsule and payload fairing.

4.0 INTERFACE DEFINITION

4.1 MECHANICAL

The mechanical interfaces between the shipping container and its contents will consist of tie-down points on the assembly being shipped and the container with the required integral shock attenuating mechanisms. The spacecraft container will contain tie-down points for securing to the transportation mediums used. These mediums include highway and/or barge movement, air transportation and ground transport dolly movement at ETR.

4.2 THERMAL

A thermal interface exists between shipping container and spacecraft to provide the controlled thermal environment required by the spacecraft. The interface between the spacecraft and the shipping equipment is a heat exchanger which is integral with the shipping equipment. The heat exchanger has a thermal and electrical/fluid interface with ancillary heating and cooling equipment.

4.3 ELECTRICAL

When shipping the spacecraft assembly, it may be necessary to provide signals or power during movement. Such services will be provided if further studies show such a necessity.

4.4 INSTRUMENTATION

An instrumentation interface between the shipping container and the spacecraft exists to provide monitoring and recording of environmental conditions during transportation.

5.0 PERFORMANCE PARAMETERS

5.1 TEMPERATURE EXTREMES

The shipping containers shall not allow their contents to experience temperature extremes greater than +35° F to +125° F. Sensors to control ancillary heating and cooling equipment are to be incorporated into the shipping equipment and are to assure maintenance of this temperature range.

5.2 HUMIDITY

Humidity limits within the shipping containers shall be maintained at 50% relative humidity regardless of ambient conditions.

5.3 PRESSURE

The various shipping containers shall be capable of containing an internal pressure of 1.0 psig maximum regardless of ambient conditions.

5.4 CLEANLINESS

The containers shall be designed as to be sealed against particulate intrusion and also be capable of cleaning before installation of their contents.

5.5 LOADS ENVIRONMENT

The maximum load inputs to the containers' contents shall in all cases be attenuated to levels below those expected during flight.

6.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

a. The container will require lifting eyes and tie-down points for safe efficient handling.

b. The container will not require integral wheels, but will mate to a truck or ground transporter for ground movement.

c. Within the configuration constraints imposed by the spacecraft, the container will be designed to fit within the planned shipping method volumes.

d. Construction materials of the container will be non-magnetic.

7.0 SAFETY CONSIDERATIONS

a. A primary safety provision within the spacecraft shipping container will be the provision of shock attenuating mounts to prevent load inputs to the spacecraft greater than those expected during launch.

b. Shock indicators mounted on the spacecraft will determine inputs in excess of the design allowables.

c. Humidity and temperature recorders will keep permanent records of environmental conditions during shipment.

d. All environmental sampling and recording equipment shall have external indicators to facilitate assessing the imposed conditions during shipment and detect any out-of-tolerance inputs.

e. Design criteria for handling provisions shall be high enough to assure safe margins for all handling conditions.

CII - VB600VP

OPERATIONAL SUPPORT EQUIPMENT

IMPLEMENTATION PLAN

Index

1.0 INTRODUCTION

Primary objectives of the OSE Implementation Plan are:

a. Ensure that OSE is available to satisfy in-house requirements for handling and testing spacecraft subsystems and the assembled spacecraft.

b. Efficient and timely integration of OSE-spacecraft and OSE-DSN interfaces.

c. Development, to the extent feasible, of OSE items that are suitable for multiple use, e.g., test equipment which can be used initially at principle vendor plants, later at GE-SD, and subsequently in the field, and for both '69 and '71 flights.

d. Provide for easy integration of the OSE with the overall Voyager system so as to allow sufficient time for verification of operating procedures and training of personnel.

e. Support type approval and life testing of the PTM Spacecraft.

f. Supply certified system and subsystem OSE to support three flight spacecraft assigned to the 1971 Voyager mission.

2.0 REFERENCE PLANNING DOCUMENTS

Voyager OSE will, in general, be designed, fabricated, checked out, utilized, and controlled in a manner similar to prime flight equipment. Consequently, the detailed implementation planning of Section 5, Volume A, is largely applicable to OSE. The following plans from Volume A are considered to be sufficiently descriptive of (or non-applicable to) OSE planning as to warrant no further discussion in this volume.

VB110VP003	Design and Development Plan
VB110VP005	Assembly and Checkout
VB110VP006	Launch Operations
VB110VP007	Space Flight Operations
VB110VP008	Special Test Plans - Life
VB110VP009	Special Test Plans - Interface
VB110VP010	Reliability
VB110VP012	Safety

VB110VP014	Procurement and Fabrication
VB110VP015	Magnetic Cleanliness
VB110VP016	EMI
VB110VP017	Pasadena Engineering Office
VB110VP018	Facilities
VB120VP	Project Control Plan

Portions of the Project Plan other than those above contain significant differences or require expansion when applied to OSE. These differences are discussed in the following sections. Where specific differences are not so noted, the equivalent Volume A planning document should be assumed to apply.

3.0 ORGANIZATION

Shown in Figure 3-1 are those functions of the overall Voyager Project Organization directly involved in the design of OSE. OSE functional requirements will be established in detail by OSE Systems Engineering, working in close consort with JPL. These system requirements are translated to subsystem OSE requirements by OSE design, adding sufficient constraints to assure OSE subsystem interface compatibility. The various subsystem design functions then create specifications, design, and develop dual-purpose required OSE which is capable of testing their equipment both at the subsystem and system level. OSE design specifies or designs portions of the overall OSE system not unique to a particular subsystem, and provides a continued integration function between the subsystem OSE design groups.

Certain special test equipment for testing of components below the subassembly level will be designed by Quality Assurance, since they will utilize this equipment during in-process testing. OSE Design will also, however, monitor and approve these designs to assure hardware compatibility to prevent in-process degradation due to test equipment.

In a like manner, some special tools and handling equipment will be designed within Manufacturing and used for processes below the system assembly level. Such equipment which interfaces with OSE or spacecraft hardware will also be subject to engineering approval and integration.

This organization should result in versatile and efficiently designed equipment since, in each case, its design is in the hands of the ultimate user, through the level of subassembly testing. Further, the integrating efforts of OSE System Engineering and OSE Design will ensure the evolution from these individual items of an optimum overall system test complex and other system level OSE.

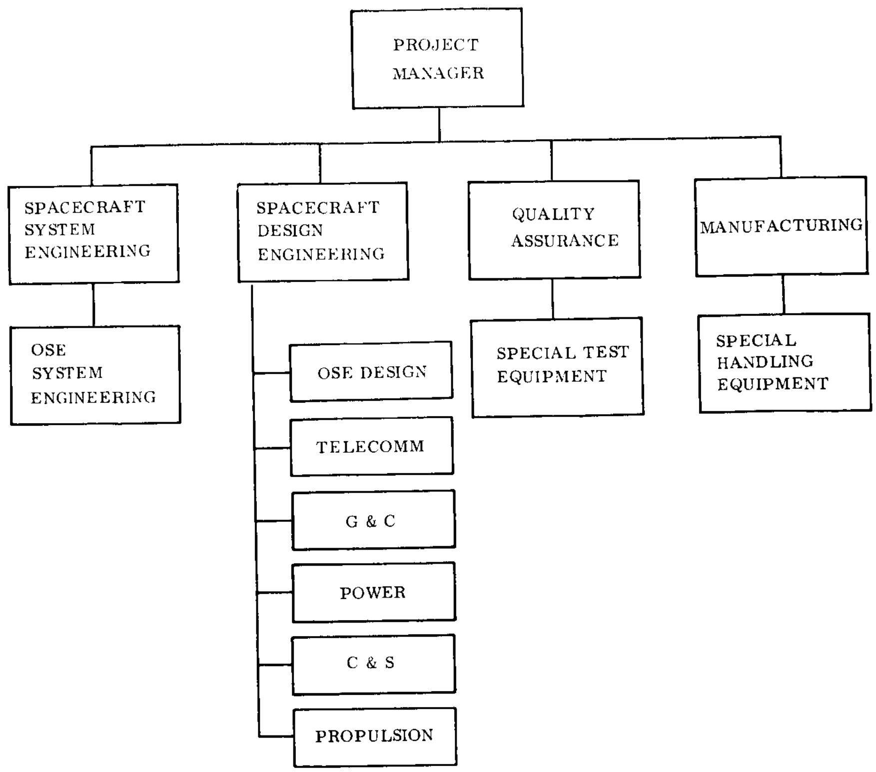


Figure 3-1. OSE Design Organization

4.0 SCHEDULE

As shown in Table 4-1, Voyager OSE deployment requirements are extensive. The OSE schedule (Figure 4-1) to provide this deployment properly phased to the availability of spacecraft hardware and with a safe minimum of OSE duplication must conform to the following constraints and requirements.

4.1 STC SCHEDULE

Factors which influence the STC schedule are:

a. At least one STC will be required for each of the flight, PTM, and space spacecraft.

b. Each STC will be shipped with its associated spacecraft for use at one of several destinations.

c. While in-house at GE-SD, each STC will be configured for testing isolated spacecraft subsystems to satisfy Integrated Test Program requirements. This configuration includes auxiliary STC equipment which enables subsystem testing on the assembled spacecraft.

Table 4-1. OSE Deployment Requirements

	Location					
Requirements	GE	GE	AFETR	GE-AFETR	GE-JPL	AFETR
Spacecraft	Structural Test Model and Thermal Control Model	Development Test Model	1969 Flight Spacecraft and FATMO (Backup)	PTM No. 1	PTM No. 2	1971 Flight Spacecraft (2) and Backup Spacecraft
OSE	Engineering Model AHSE	Stage III Information AHSE 2-STC (Engineering Models-Subsystem) 1-LCE (Engineering Model) 1-MDE (Hardware and Software)	2-AHSE 1-STC (System) 1-STC (Subsystem) 2-LCE (Launch Pad) 1-LCE (ESF) 1-MDE (Hardware) 2-MDE (Software	1-AHSE 1-STC (System) 1-LCE (Launch Pad) 1-LCE (ESF) 1-MDE (Software and Hardware)	1-AHSE 1-STC (Subsystem) 1-LCE 1-MDE (Software and Hardware)	3-AHSE 2-STC (System) 1-STC (Subsystem) 2-LCE (Launch Pads) 1-LCE (ESF) 1-MDE (Hardware) 3-MDE (Software)

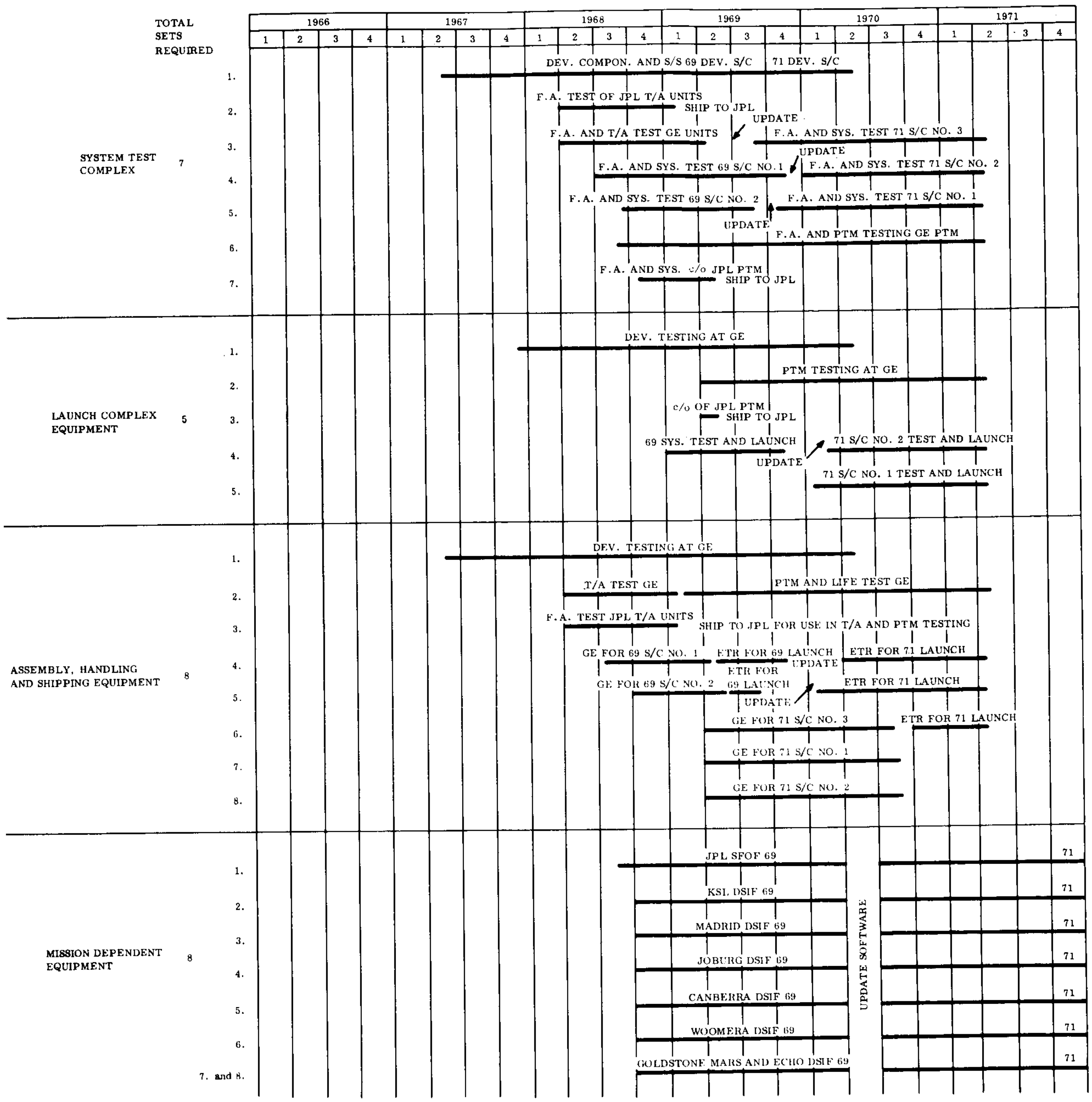


Figure 4-1. Voyager OSE Utilization Schedule

d. The STC for the spare S/C and the STC for the PTM which goes to JPL will also be configured for isolated spacecraft subsystem testing.

e. The STC will have the inherent capability of flexible allocation within the Spacecraft Checkout Facility (SCF), but this capability will not be utilized in the GE-SD preliminary design and functional descriptions. For example, Computer Data System No. 1 can be wired to STC No. 2, etc., if needed.

f. STC at ETR Spacecraft Checkout Facility will be used to support spacecraft on the launch pad by augmenting the specialized minimal blockhouse equipment.

4.2 LCE SCHEDULE

Factors which influence the LCE schedule are:

a. A total of three LCE's are required at ETR.

b. Logistic support, compatible with the effects a failed LCE could have on the launch window, is planned for the LCE. Substitution between the LCE at the Explosive Safe Area (ESA) and the LCE at the launch Complex (LC) in the event of a failed assembly, augments this logistic support.

c. One LCE, or at least the major portion of one, will be required at GE-SD for OSE/spacecraft compatibility testing.

4.3 MDE SCHEDULE

Implementation of the MDE subsystem will be accomplished so as to permit timely delivery of these items to the DSIF for scheduled calibration and verification of compatibility. Factors which influence the MDE schedule are:

a. Each Deep Space Instrumentation Facility (DSIF) will include a general purpose digital computer for processing of telemetry and command data (TCD).

b. MDE hardware requirement will be minimal (probably modulation and demodulation equipment only), however, MDE software requirement will be heavy.

c. MDE hardware items should be sole source, obtained from the same sources as Command Flight Equipment.

4.4 AHSE SUBSYSTEM SCHEDULE

Project phasing of the AHSE subsystems will be scheduled to assure that design, development, fabrication, and assembly of these items will be completed to satisfy in-house requirements for such items to support spacecraft development, assembly, test, checkout, and shipment. AHSE will be used throughout the entire factory sequence;

those items which are deliverable to ETR and JPL becoming operational AHSE, the remainder being Factory Support Equipment (FSE).

5.0 QUALITY ASSURANCE

The Quality Assurance plan of Section 5, Volume A VB110VP011 will be applied, for the most part, directly to OSE. Certain controls, however, may be eliminated or modified with respect to OSE in the interest of cost or schedule savings without jeopardizing the mission. The major changes of this nature foreseen are as given in paragraphs 5.1 through 5.4.

5.1 PROCUREMENT

A special procurement quality document for OSE will be prepared during Phase IB. Vendors will be required to maintain a working logbook that will identify an up-to-date record of lists of materials, break of inspections (items removed or interchanged for some reason), list of deviations from planned tests or inspections, completed inspection and test planning, discrepancy reports, and performance data sheets or engineering test reports. OSE quality plans will identify the degree of control/acceptance to be accomplished by the company and GE-SD. The quality plans will further be addressed to the interfacing required for preventive maintenance and calibration by subcontractors retaining a set of OSE for flight article testing. Surveillance, Receiving Inspection, and in-house testing will be used in varying degrees to provide the proper balance of cost with vendors capabilities to provide assurance of required quality levels. Spare parts for OSE will be procured to VR130TC002, "Quality Assurance Requirements for Suppliers." All OSE, including spare parts, will receive some level of acceptance testing performed upon receipt in-house. Commercial equipment purchased for installation in OSE will be routed to the Instrument Calibration Lab for verification tests to catalog specifications.

5.2 CONTROL OF GE-SD FABRICATED OSE

OSE will be fabricated in a different area from flight hardware using the same methods of manufacturing and inspection planning. It is planned that piece part serialization will not be required for OSE, and the parts will be stored in a segregated bonded stock area. In general, photographic records of OSE will not be maintained. Acceptance tests will be defined by an engineering specification, with test procedures detailing the instructions for performing the required tests. Acceptance tests of end items will not include the environmental tests required for flight hardware. Logistic spare parts for OSE will be acceptance tested in a manner similar to end usage and will be operated some minimum time analogous to the green line limits of flight articles.

A workbook will be maintained containing the same information as described in Paragraph 5.1. It will be the responsibility of the personnel working directly on the hardware to keep the book current and accurate. A final historical logbook will be prepared including a signature page for responsible GE-SD and JPL representatives signifying acceptance of the end item. The concept of team transfer from system test to the launch site will assure a maintenance of the logbook, configuration and adequate planning for the incorporation of changes.

5.3 PRESERVATION, PACKAGING, HANDLING, STORAGE AND SHIPPING

The specific engineering OSE specifications will identify the requirements for preservation, packing, and packaging.

5.4 OSE TESTS

OSE tests will be performed at various levels of OSE development starting with the lowest level and working toward the higher. The approach, plans, scheduling, and means for monitoring the progress of these tests will be as defined in the Integrated Test Plan. This plan provides individual test plans for parts and materials development, subassembly and subsystem development, system development, and OSE certification. MDE (hardware/software) to DSIF interface tests will be performed to demonstrate that the MDE hardware and associated procedures and programs are functionally adequate and compatible with the DSIF. This test will be performed initially in conjunction with the STC to Spacecraft Assembly Facility No. 2 (SAF No. 2) and DSN interface test. This is necessary to demonstrate the capability of the bi-directional pathway over which signals are fed into the STC by the spacecraft and received from the STC by the spacecraft. The test will be performed initially using the PTM and its associated STC. MDE to SFOF interface tests will be performed to verify the adequacy of installed MDE and to verify its compatibility in the SFOF.

Compatibility between items of AHSE will be demonstrated by mechanical walk through of AHSE/spacecraft interfaces as early as practical in the development cycle. Specific objectives of the interface tests will be:

a. Verification that spacecraft assembly equipment will provide for accurate assembly of all spacecraft components to the basic spacecraft structure.

b. Verification of AHSE capability to provide correct indications of spacecraft component alignment.

c. Verification that the AHSE will provide for safe, efficient movement of the spacecraft and/or its major components.

d. Verification of AHSE capability of positioning and stabilizing the spacecraft in any attitude required for servicing, adjustment, component replacement, testing, and personnel access.

6.0 OSE UTILIZATION

The OSE for the 1969 mission has been configured for minimum updating requirements in order to recycle it for 1971 use. Table 6-1, shows the degree of modification required by each kind of 1969 OSE, to utilize it for 1971.

Most of the major items in the 1969 STC are usable for 1971, as is. Some of these might well be GFE to General Electric for 1969.

The LCE also shows little need for specific 1971 design.

The AHSE shows that substantial 1969 peculiar design is needed. However, these AHSE items lend themselves to salvage in the recycling through 1971 OSE development. Several specific major hardware items can be salvaged by redesign and rework at GE-SD and emerge suitable for 1971.

MDE hardware for 1969 will serve 1971 needs, and can stay at the DSN.

Table 6-1. Utilization Matrix of Equipment for the Voyager Operational Support Equipment

	Equipment Common to 1969 and 1971	Equipment Capable of Conversion 1969 to 1971	Equipment with Unique 1969 Configuration	Equipment with Unique 1971 Configuration
SYSTEM TEST COMPLEX EQUIPMENT	Computer Data System Central Recorder Central Time and Synch (T. I. U.) Command Verification Equipment Radio Subsystem OSE Command Subsystem OSE Relay Radio Subsystem OSE Data Encoder Subsystem OSE Bulk Storage Subsystem OSE Controller and Sequencer S. S. OSE	Test Conductor's Console Ground Power Distribution Computer Driven Printers and Displays Power Subsystem OSE Pyro Test OSE Science Package Provisions	Spacecraft Umbilical Simulator	Spacecraft Umbilical Simulator Capsule Provisions
LAUNCH COMPLEX EQUIPMENT — Launch Area	Telecommunications S. S. OSE Controller and Sequencer OSE Guidance and Control OSE	Power Subsystem OSE		
LAUNCH COMPLEX EQUIPMENT — Launch Area and Explosive Safe Area	Propellant and Gas Service Equipment Leak Test Equipment Pyrotechnics Instruction and Test Equipment	Pyrotechnics Subsystem OSE		
ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT	Magnetic Mapping Equipment Weight and Balance Equipment Shipping Container (Engine)	Assembly Fixtures Interface Fixtures Alignment Equipment S/C Handling Equipment	Shipping Containers (S/C and Arrays)	Shipping Containers (S/C and Arrays)
MISSION DEPENDENT EQUIPMENT	DSIF Modulators and Demodulators SFOF Demodulator	DSIF Software SFOF Software	Capsule Radio Simulator	

APPENDIX A

ALTERNATE APPROACHES

The following documents summarize the major trade offs made to establish the approach to be taken to implement the OSE-STC. In many cases these trade studies were made early in the study phase. As such they establish the philosophical approach to be taken. The subsequent work may have modified the details of this approach but the basic intent has been maintained.

The following documents are included:

VB260AA001	Basic System Test Approach and Philosophy
VB260AA003	Propellant Loading Alternatives
VB260AA004	Leak Test Alternatives
VB260AA005	Deployable Element Testing
VB260AA006	Computer Sizing Analysis

CII - VB260AA001

ALTERNATE APPROACHES

BASIC SYSTEM TEST APPROACH AND PHILOSOPHY

Index

1.0 PURPOSE

The purpose of this document is to provide the basis and rationale used to establish the configuration proposed for the System Test Complex (STC) for the Voyager System.

2.0 GENERAL

The STC is to support the following vehicle tests:

a. Final assembly steps required to assemble subsystems (previously checked out individually using their OSE consoles) into a system. This will include the following:

   1. Ground loop checks.

   2. Initial power application.

   3. Subsystem tests using vehicle power and cabling.

   4. Inter-subsystem tests using vehicle power, command and TLM interfaces, etc.

   5. Mission Profile Tests.

b. LCE compatibility tests.

c. EMI tests.

d. Pre and Post Vibration Tests.

e. Thermal-Vacuum Tests.

f. Final Factory Tests.

g. Field Reacceptance Test.

h. Support Explosive Safe Area Activities.

i. Support LCE.

These tests will be performed in several areas and will be supported by STC's in those areas.

3.0 SYSTEM TEST PHILOSOPHY

The following statements summarizes the philosophy used in establishing the STC configuration.

a. The primary purpose of the Vehicle Acceptance Test is to verify that the workmanship and materials used are of a high order.

b. A secondary purpose is to verify that the flight units are similar enough to the PTM and subsystem development units to allow correlating the results of those tests with the results of the Acceptance Test.

c. Due to the nature of the Voyager Program (long life - short launch windows several years apart - experimental nature - three vehicles built per window, etc.) the following aspects are to be emphasized.

   1. Vehicles are to be protected against damage induced by the OSE and their personnel.

   2. A low risk approach to OSE should be taken in order to insure that it is not a source of problems during the tests.

   3. It is desirable to be able to identify which vehicle is the best one.

   4. Overtesting (within reason) is to be preferred to undertesting.

   5. Tests will be performed by highly knowledgeable engineers.

   6. Development type tests will be required to a limited degree.

d. The Systems Tests are performed following installation of the subsystems into the vehicle and their preliminary tests (2.a.1 thru 2.a.3 above). After these tests no flight cabling will be broken until a fault has been detected. All hardwire points shall be provided properly buffered and isolated at separate connectors.

e. Repair will be done in the STC by replacement of bays. Faults need not be isolated any further than this level.

f. Test sequences and results will be recorded for post test evaluation and for test result verification.

g. The STC will be made up of the OSE used to check out the individual subsystems. These subsystem OSE consoles will be tied together and coordinated into system test sets by the use of additional STC peculiar equipment.

h. Mission Dependent Equipment (MDE), which includes hardware and software used in the DSIF, will be a part of the STC.

i. Redundancy is not a reason for reducing the level of testing, in fact it indicates that two or three well working systems are required at lift off. Provisions must be made in the vehicle to enable the OSE to detect the operation of parallel path, to stimulate fault detecting circuits (that switch in back-up circuitry) in such a manner that the transfer occurs, and to be able to operate majority logic circuits in such a manner as to verify that decisions are being made unanimously.

j. The degree to which the vehicle's design is made cooperative to depth of tests will be made on the basis of total system tradeoffs. The result must be that the confidence that the vehicle will perform its mission has been improved by increasing or decreasing the depth of tests. The general philosophy will be that end-to-end testing will be performed.

k. The STC's support of the LCE will be limited to receiving and analyzing RF TLM data and in sending RF Commands to the vehicle.

l. Advantage will be taken of the vehicle's TLM and Command capabilities to reduce the number of hardline leads. The only reasons for duplication will be inadequate availability, accuracy, or time to execute.

m. Most of the Systems Tests will be performed using simulated solar panel power and simulated battery power. Final verification of Panel-Vehicle interface will include the capability to transfer full power.

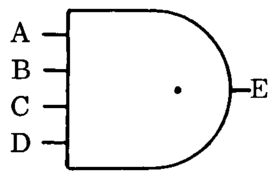
4.0 SYSTEMS TEST

The above require essentially two types of test capabilities - tests performed on individual subsystems disconnected from the other subsystems (assembly tests) and tests performed on the system (subsystems connected to other subsystems via the vehicle cabling). The basic OSE is designed to be capable of testing the individual subsystems. The main subject of this document is how to interconnect the individual subsystem OSE and to establish what additional equipment is desired to test the vehicle in the system configuration. This system test can be accomplished by performing two basic types of test - a performance test and a mission profile test as described below.

a. Performance Test

This test operates all elements of the system in the modes best suited to detect malfunctions and to establish how well they are operating. This may lead to operating the vehicle in a manner that could never occur in normal operation, but this is to be encouraged if it is the best way for finding faults or for identifying how well the element is working. An example of this is that rate and position errors during an A/C subsystem test need not be coordinated (vehicle rates can exist with fixed position errors) if this is the optimum way of testing for defects or evaluating performance.

It should be noted that all elements are to be tested. An example of this is an "and" gate as shown below:



It is not sufficient to demonstrate that the output is a "one" for A • B • C • D, but four of the fifteen "zero" states must also be verified. These are:

$\bar{E} = \bar{A} \cdot B \cdot C \cdot D$ $\qquad$ $\bar{E} = A \cdot B \cdot \bar{C} \cdot D$

$\bar{E} = A \cdot \bar{B} \cdot C \cdot D$ $\qquad$ $\bar{E} = A \cdot B \cdot C \cdot \bar{D}$

The Performance Test must guarantee that those four "zero" and the one "one" state are verified; any other combinations are practically superfluous as far as that "and" gate is concerned. In many cases the verification of these "zero" states will require the OSE to operate (stimulate) the vehicle in a manner that is not normal to operational usage.

To establish how well an element is operating is difficult. Methods used in the past have included operating the element beyond its design limit and detecting its margin of operation. This must obviously be a non-injurious type of operation. A possible example of this would be in the phase locking circuits of the Radio Subsystem. In this, phase lock is established by having the ground transmitter operating at a different frequency than the expected airborne frequency. This Δf causes the phase angle between the signal received by the vehicle and the vehicle's self-generated frequency to vary from 0 to 360 degrees. When the phase angle passes thru 0° the airborne frequency is to be altered to maintain the phase at 0° (i.e., phase lock). This will occur if the phase angle rate is within some limits. The lowest rates (+ and -) at which phase lock does not occur gives an indication of the margin of operation of this circuitry. It is felt that the change of these points as a function of life or environment is significant as to its capability to withstand the mission. It should be noted that the above test operates the element in a manner not to find that it operates within specified limits but to find its limits of operation and the more quantitative the test results the better its changes can be observed. It should also be noted that this is an end-to-end test, (i.e., the stimulation is varied and its effect observed by watching the vehicle's normal output).

To apply this philosophy to elements further down the chain of a subsystem (i.e., further away from the front end or normal stimulation point) becomes more and more difficult. The sensor output signal while it may be marginal is eventually, in most cases, reshaped, amplified, etc., to become a nominal signal; thus, the elements that follow this cannot be tested with marginal signals using end-to-end test philosophy. While adjusting the voltage settings or temperatures, etc., may give a method of accomplishing this to a degree, it is felt that this is asking for trouble, especially in a system configuration. Therefore, the evaluation at these points would at most be by observing wave shapes, time delays, etc., as a function of life or environment or where possible by varying an external load.

The Performance Test is a rigid pre-planned sequence. When faults are detected certain other sequences may be required (the most common of which will be to stop). The test progresses sequentially through a particular subsystem starting by verifying the front end and progressively adding more and more elements and different combinations until the entire subsystem is verified. This same approach is applied to the system and eventually all interfaces will be verified completely. Since elements within a subsystem (or system) are verified in the sequence in which they are interconnected the reason for a failure of a test can be deduced to be the new element added to the chain for that sequence. This type of test therefore identifies the faulty element in almost every case directly.

b. Mission Profile Test

The Mission Profile Test is, to a large degree, a special case of the Performance Test in which the inputs to the system are the nominal inputs programmed to occur as in flight. This sequence would include all commands and operate all of the subsystems in all of their modes. During this test the inputs to the system should be as realistic as possible. This could be accomplished by:

1. Using the CVE and its flight software for commanding the vehicle.

2. Operating the G & C in a simulated closed loop as far as possible.

3. Use the TLM as a primary source of data - with greatly reduced hardwire monitoring and control.

4. Use flight batteries.

5. Some considerable operation with no test connections at all.

The Mission Profile Test is still a rigid fixed routine such that the test and monitoring equipment knows the mode and state of the vehicle at any instant. The OSE also knows when and what changes are to be made to what subsystem. It is primarily these changes that are then to be verified as having taken place as well as the fact that no other changes occurred. This then at that instant becomes essentially a subsystem test to verify that the change expected did in fact occur. This is true primarily due to the nature of the Voyager (i.e., every sequence, command etc., is a definite fixed routine effecting a very small part of a particular subsystem and only one command is executed at one time).

It is well to note the differences between the Performance Tests and the Mission Profile Test as follows:

1. Performance Test

   (a) Operating system in optimum manner to detect faults.

   (b) Operates system in optimum manner to detect margins of operation.

2. Mission Profile Test

   (a) Operates system in a mission sequence.

   (b) Operates system with nominal inputs.

From this it can be seen that if the vehicle passes its Performance Test it should pass the Mission Profile Test provided there are no vehicle design errors (i.e., margins of operation are not exceeded during a mission sequence), that there were no crosstalk or interference problems that were not anticipated, and that the Performance Test does all it was meant to. The Mission Profile Test is therefore a confidence test to assure that the vehicle will accomplish a simulated mission.

While the above may give the impression that the two types of tests are separate and distinct, it must be recognized that they can each incorporate as many aspects of the other as desired. This combination is mainly a matter of incorporating the desired sequence of controls/commands into the test. It should also be realized that reduced levels of testing will be accomplished as desired at any test location primarily by the elimination of steps.

From the above discussion of tests it can be seen that all System Tests are in reality a fixed series of short test steps. Each step being made up of one command (or some other single stimulation occurrence) that is verified by observing a few TLM or hardwire values change to certain new values. The following test steps may further verify this command's operation by stimulating the vehicle and verifying once again by testing several data values. The sequence of these steps are fixed and for a given sequence (several steps) they should always occur in that manner. The tests would generally be terminated if the response is incorrect. This termination may be just to stop or it could entail performing some of the following steps or it could require generating some back-up sequences in order to stop a function that should not be allowed to continue.

5.0 SYSTEM TEST CONFIGURATION ALTERNATIVES

a. General

Three STC configurations were investigated as to their adequacy to perform the tests as outlined above. The three approaches were:

1. The Mariner C approach.

2. The Mariner C approach modified such that the computer data system (CDS) provides sequencing functions to the vehicle by remotely controlling certain aspects of the Subsystem OSE.

3. Adapting the Mariner C approach to an ACE (Acceptance Checkout Equipment - developed by NASA and GE-ASD for Apollo).

The differences in these approaches is primarily in the manner in which the computer is used. Each are described in more detail below with the emphasis on the computer application.

b. Mariner C Approach

This approach is shown in Figure 5-1. The system is basically a manual one with the computer serving the function of data acquisition, processing, and alarm generation in parallel with the subsystem engineers. The primary computer functions are the following:

1. Obtains TLM data from TCD.

2. Controls the selection of Hardwire Monitor groups and obtains its digital output.

3. Monitors OSE switch operations to establish inputs to vehicle.

4. Generates alarms (out-of-limit) indications based on a-b-c above.

5. Prints out data (engineering units if desired) as requested.

6. Controls X-Y plotter.

7. Records data for off-line analysis by an IBM 7094 computer.

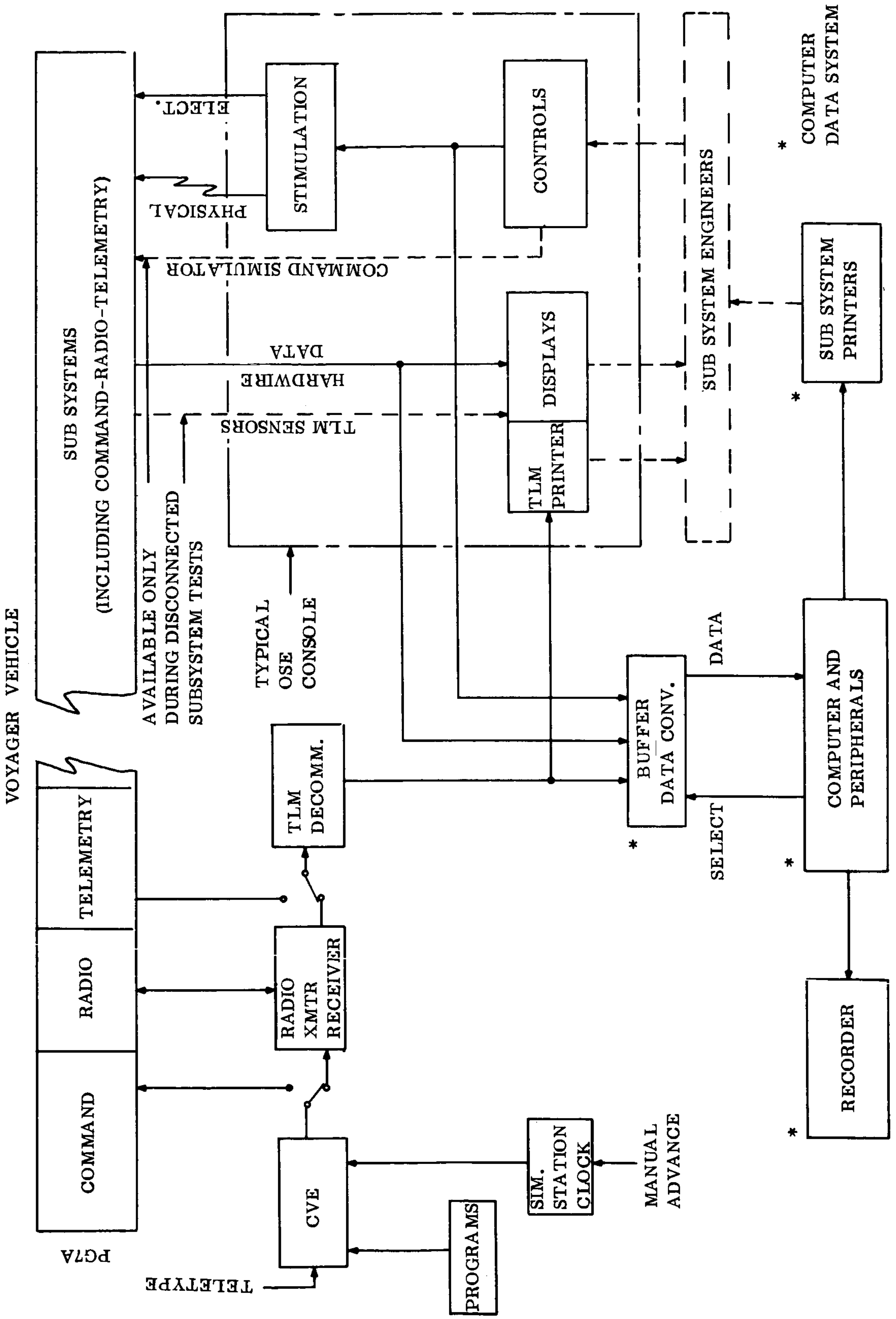


Figure 5-1. Mariner C Approach Applied to Voyager

In the Voyager configuration the Command Verification Equipment (CVE) or the Command OSE will be used to send commands to the vehicle. Any of the following modes could be used:

1. Load blocks of pre-programmed command sequences into the CVE prior to start of test sequences. The commands would be released by supplying the simulated time code associated with each block of commands. This simulated time code should probably be controlled manually (manually, advance a counter) since using real time might cause the test sequence to proceed faster than the other subsystem engineers could keep up with, especially if problems arise.

2. Load commands into CVE in real time (i.e., send when inserted) by means of typing codes (apparent random characters at best) in via the CVE's teletype input.

3. Send commands via the Command OSE by manually setting in commands (setting $\approx$ 45 switches in apparent random manner).

   Note: In order to protect the vehicle from having the wrong command inserted at the wrong time, the manual techniques (2-3) should have a check function performed by the Test Conductor before sending them to the vehicle.

Due to the large number (over 1000) of commands (good and bad) to be sent to the vehicle during System Tests, the pre-programmed or pre-punched, Method 1 above, is the most practical method of accomplishing this task. Methods 2 and 3 are primarily reserved for troubleshooting types of activities.

In concert with the commands, the other OSE subsystem engineers must set up their equipment to operate in a specified manner in order to evaluate the commands and the result of them on their subsystem. The monitoring and evaluating functions are performed by these operators in parallel with the CDS.

There are basically two ways of programming the computer to correlate the received data as a function of the commands and the subsystem stimulation. These are:

1. Program the computer such that it can establish the mode of the vehicle and the status of its stimulation. Then, by using this data, calculate the limits of all the other data for correct operation. This type of programming can allow the system to operate (as far as the computer analysis is concerned) with random inputs. It, however, requires that the computer be able to ascertain the status of all inputs into the vehicle and the sequence of them where important.

2. Program the computer knowing what the sequence of commands and stimulation is going to be. Then by detecting when these changes occur, the limits for the analysis are changed by the program to those calculated by the programmer.

Method 1 has great flexibility but is difficult to program and to insure that it can monitor all of the significant inputs into the vehicle. Method 2 is easier to program (essentially mechanizing the analyzing function of a manual test sequence), but it cannot handle tests performed incorrectly or out of sequence. (Note: It can detect these conditions and so inform the operators, etc.).

The programming technique that would be eventually used would be a combination of the two but leaning more towards the second than the first.

The advantages of the Mariner C approach are:

1. Minimum interface with Subsystem Test Consoles.

2. Makes effective use of subsystem engineers. Generates alarms or out-of-tolerance conditions for the operators to respond to.

3. Used successfully on Mariner C and can probably be directly applied to Voyager including software techniques.

4. Requires an operator who knows the vehicle subsystem and the OSE. This is considered an advantage to the overall system especially on a program like Voyager (i.e., two vehicles every two years that function properly).

The disadvantages of this approach are:

1. It is impossible to repeat tests exactly, if timing is considered.

2. Tests must proceed at manual rate which may be excessively long if tests are large in number.

3. Coordination of complex test sequences are difficult to accomplish.

c. Modified Mariner C Approach

This approach is shown in Figure 5-2. The difference between this approach and the above is that the computer in addition to the above functions also closes the loop by providing control functions to sequence the commands and the stimuli based on pre-programmed routines, and as a result of the data received from the vehicle resulting from that stimulation as required. It must be emphasized that this control is in addition to the manual control available which would be maintained at the Mariner C level.

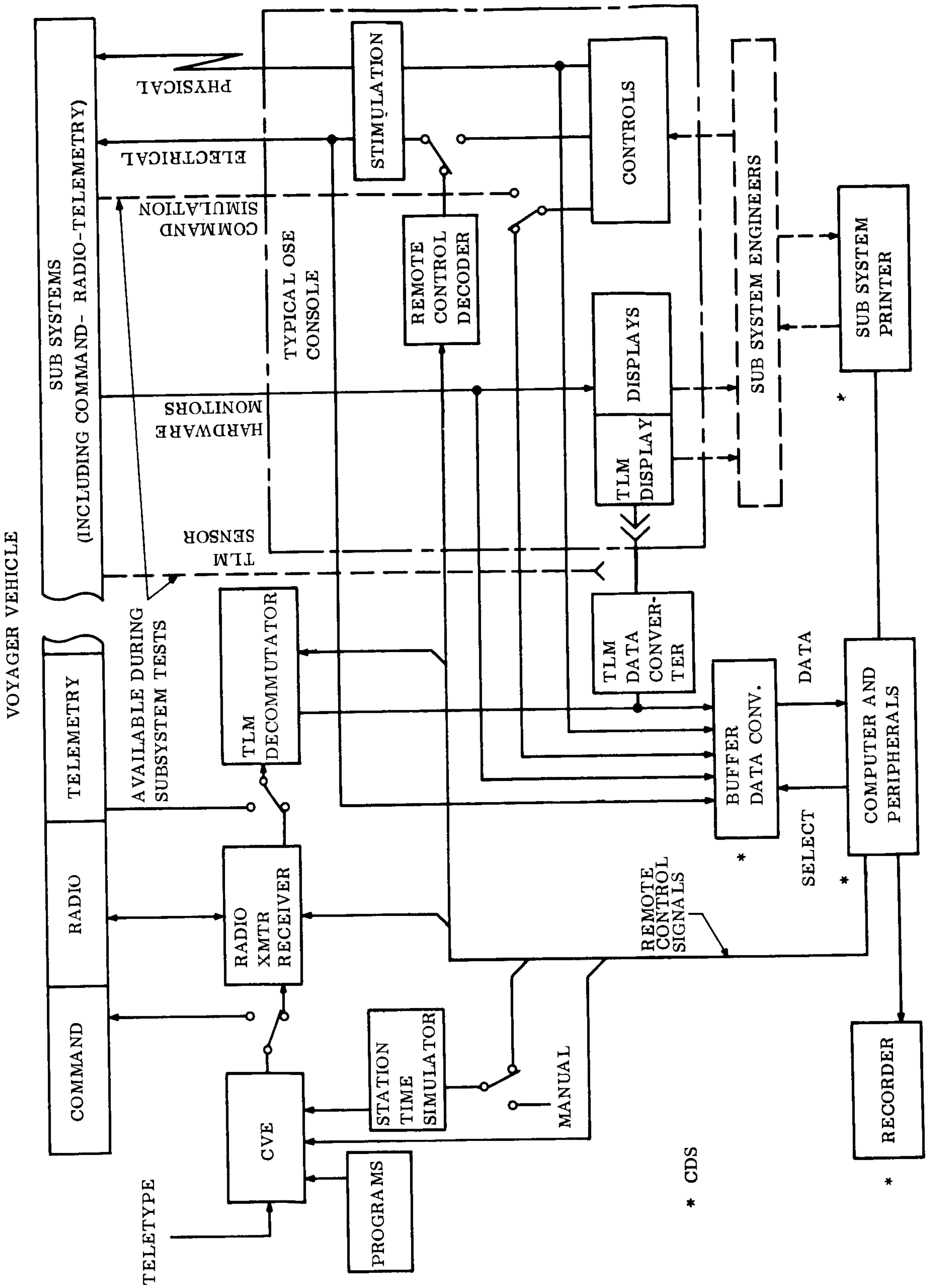


Figure 5-2. Modified Mariner C Approach

This approach allows automatic or manual testing to be completed, whichever is appropriate at the time. The automatic aspects can be approached gradually starting with the functions that will help the system most. Even the ultimate would not automate all functions and the troubleshooting type sequences would be among the last that would be attempted.

The desirability of this approach depends greatly on the types of test to be run. The automatic features are desirable where:

1. Routine tests are run for prolonged periods.

2. Tests require coordination of operators or operator actions beyond their capabilities.

3. Tests are to be exactly duplicated.

4. Tests are large in number.

All of these factors exist in the Voyager System Test to some degree and "automation" therefore offers some attraction. Any method proposed, however, must retain the capability of performing the tests manually such that subsystems can be tested using this equipment at other locations (where a CDS will not be present), for troubleshooting or investigatory tests and as a low risk backup mode to the automated tests.

The functions to be controlled by the computer are not new or different because of the computer. These functions are required regardless of the approach to be used. The computer will have to set up the same circuitry remotely in the same manner that it was to be set up manually. The tests to be run are the same and the test programs required are, for all intents and purposes, identical. The computer would set up the OSE and the vehicle using the same techniques and in parallel to the switching or controlling circuits that would be set up manually. The evaluation of the data would be as done on Mariner C except the operator would back the CDS up to the extent desired.

The additional load imposed on the computer should not be large. The output rate of the control can be as slow as a manual response and still be adequate for most of the sequences foreseen. This then allows the computer to output the control signals almost at its leisure. While it is not expected that this will be that slow, it can be. The computer does not have to store all of the sequences in its core. Only those sequences that are required on short notice and logical groups of test steps need be stored at any one time. The sequences can be read in from tape or some such device as they are requested by the Test Conductor.

The computer also has an easier task in evaluating the data obtained from the vehicle and the OSE. This is obtained by the fact that the same sequence that generates the command can alter the computer's evaluation criteria directly. The computer no longer has to determine what is being done to the vehicle, because it is the program source. The command functions OSE results will be fed back to the computer (via its hardwire conversion circuitry) to verify that the OSE responded properly as well as obtaining the vehicle data to verify its response to the OSE. The computer will not reduce the amount of in-process shelf testing circuits designed into the OSE consoles. These functions will be performed independently of the CDS.

The primary functions to be controlled remotely are those things done repeatedly or that may require close coordination with other functions. These primary functions would include the following:

1. Vehicle commands (good and bad formats).

2. Vehicle clock control (set-hold-rate control).

3. TLM data rate control.

4. TLM multiplexer control (homing).

5. Control and Guidance Controls.

    (a) Gyro torque start points.

    (b) Gyro torque stop points.

    (c) Gyro torque rate points.

    (d) Other sensor stimulator or simulator controls as required.

6. Radio Controls as accomplished normally by CVE.

Other functions may also be remotely controlled, not because great benefits will be obtained, but because they will be easy to do once the above capability has been added and because some benefit will be obtained. This would be such functions as:

1. Hardwire mode selection of the Radio (i.e., antenna-receiver combinations).

2. Hardwire mode selection of the TLM.

3. Guidance mode control.

4. Other simple functions - switch closures, etc.

It should be obvious that all the System Test control functions and analysis will not be automated. There are functions to be performed and analysis to be made that can not be easily automated. From the test procedures outlined, these tests are in the minority and they would remain manual. If these tests are required in the middle of an automated sequence, this can easily be handled by programming the computer to pause and allow the manual functions to be accomplished and resume when they are finished as indicated by the Test Conductor's instructions to the CDS.

The main advantages of this approach are:

1. Test times will be long. The use of the computer should materially reduce this.

2. The test will be repeatable - making comparison of like data obtained from many tests on same system possible.

3. Human error reduced.

4. Coordination of the STC is accomplished.

5. Large number of commands to be sent requires some automated means of accomplishing this. (This method is not too different from the approach using pre-stored commands in the CVE's.)

6. Computer tasks have not been increased appreciably - programs are nearly alike.

7. Logical extension of Mariner C.

8. Automated tests are backed up by being able to be performed manually.

9. Availability of OSE can be improved over manual by having computer programmed self checks.

The main disadvantages of this approach are:

1. Not all tests can be remotely controlled or monitored. If these predominates, it may not make sense to control the remainder.

2. Subsystem consoles have to contain provisions for remote control.

3. Operators may lose feel of test.

4. Program changes are harder to make than in a manual setup.

5. Equipment reliability has been decreased.

d. Modified ACE/SC Approach

ACE/SC (Acceptance Checkout Equipment/Spacecraft) was developed by NASA and GE-ASD for testing the Apollo vehicle. This equipment is in production and could be applied to the Voyager program. The basic concept is shown in Figure 5-3.

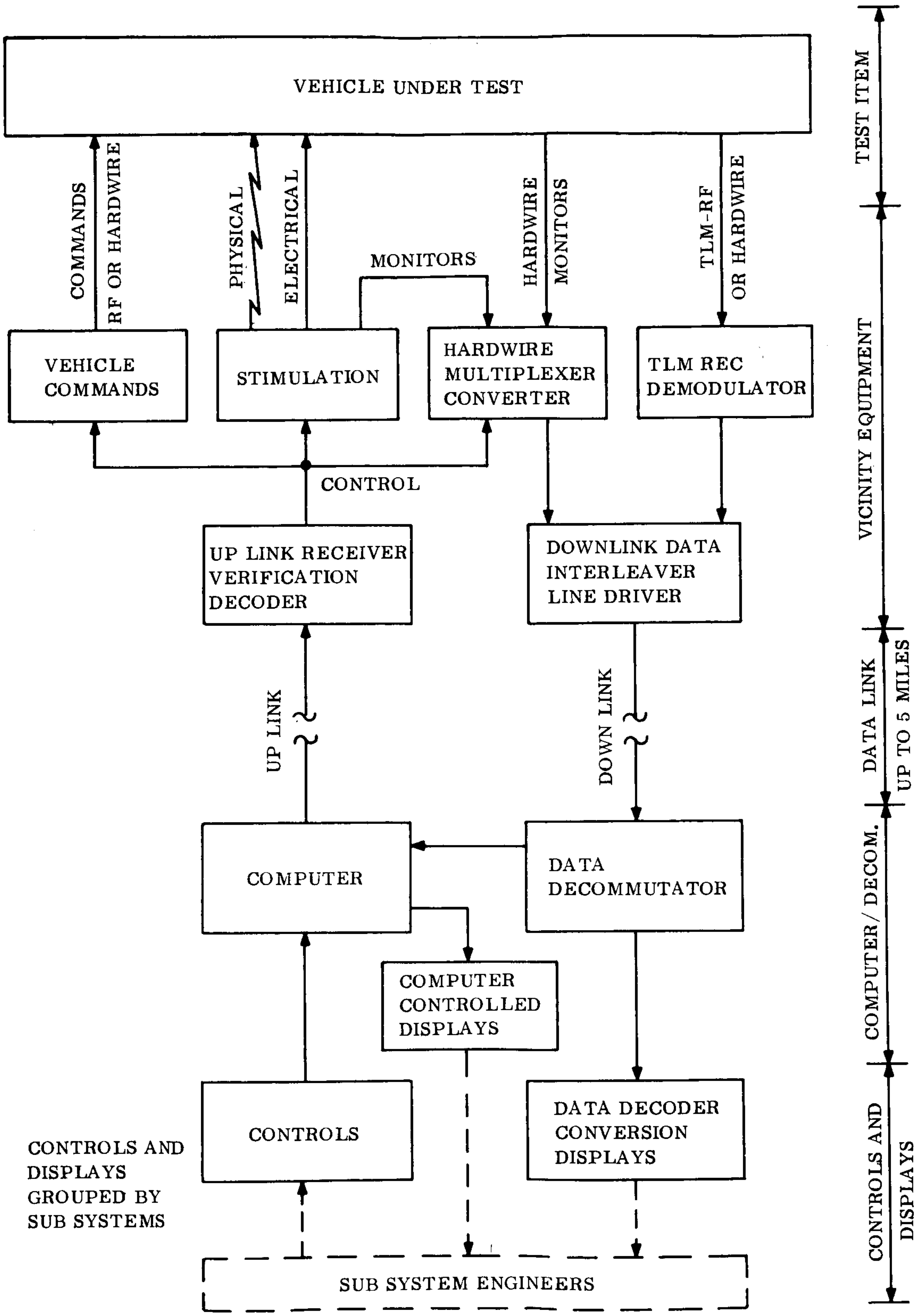


Figure 5-3. ACE (SC) Simplified Block Diagram

The computer/decom in this approach is placed in series between the controls and stimulation equipment and between the data sources and the displays. The displays, controls and computer/decom are normally located remotely (up to five miles) from the stimulation equipment, data sources and vehicle under test. The interconnection between these two locations is ideally two data links (a control or up link - to the vehicle - and a data or down link - from the vehicle). Serial digital data (up to 1 megabit/second) is transmitted up/down these lines and is received, decoded and distributed to the proper destination at the receiving end. This is described in more detail as follows:

1. Control - The computer senses/detects manual switch operations from the control area. It then generates and transmits a unique serial code of digital data (depending on which switch was operated) up the control or up link data line. This serial code is received at the stimulation area and is decoded and distributed to the proper stimulation equipment where a function (set a relay - set a D/A converter - set a function generator, etc.) is performed.

2. Data - The data (in the vehicle vicinity) is obtained from the vehicle PCM Telemetry System and several ground (or vicinity) PCM Telemetry Systems. These serial digital data sources are interleaved to form one data stream which is transmitted down the data or down link. The decommutator separates the serial data into data words and distributes this along with identification bits to the display area. These devices, using the identification bits, store the data for their displays and convert (via D/A converters or lamp drivers) to drive the proper display.

The computer also receives this data and can be used to limit compare, convert to engineering units, generate additional displays, etc..

This system can be as manual or as automatic as desired, depending on how the computer is programmed. The manual capability is implemented by programming the stimulation area on a one-to-one basis for the switches in the control area and to respond only to control switch operation. The automatic is implemented by causing a sequence to occur as a result of a switch operation with the sequence altered as a result of the data received. The diagram as shown is idealized since some manual local tests will have to be performed at the stimulation location. These should be few in number and done at a time when this area can be manned.

The advantages of this system are:

1. The control and display area can be remote from the test area. This makes the approach almost mandatory for an ITL (Integrated Transfer and Launch) approach, where the hangar equipment is used for launch pad operations.

2. Special Launch Control Equipment (LCE) is not required. (Hangar test vehicle vicinity equipment moved to pad area.)

3. Flexibility and growth capability large.

4. Equipment is in production.

The disadvantages are:

1. The display and control area require the computer to be on line.

2. The consoles (stimulation plus control and displays) cannot support subsystem tests in other than system test areas. This requires the stimulation equipment to have the display and control capabilities that exist on today's OSE. This makes the remote control and displays equipment completely additional equipment.

3. Item 2 makes this system inefficient unless the remote control is required.

4. LCE cannot be eliminated entirely since vicinity equipment (ground power, vicinity A/D conversion equipment, etc.) are always required.

This concept, however, can be modified such that it can be applied to the Mariner C configuration without such radical departure from existing techniques. Figure 5-4 shows this approach as used for hangar tests and pad operations. The hangar configuration is practically identical to the modified Mariner C approach discussed above (Section 5c). The main difference is that the computer interface with the STC area equipment is over serial digital data links rather than parallel leads.

The launch pad configuration uses the STC controls and displays in place of the blockhouse console. Operation of these controls, causes the computer to command the vehicle vicinity equipment just as though the blockhouse console was in control. Data from the vehicle umbilical is digitized at the pad and transmitted serial to the STC. This data (along with the existing TLM data) would be decommutated and distributed to the STC consoles via their umbilical interface connectors, as well as to the computer where it can be analyzed.

This approaches main advantage is the elimination of the blockhouse equipment. This requires a major alteration in launch operation philosophy from that used in Mariner C; and it, therefore, may not be acceptable. However, if future launches from Merritt Island or Titan IIIC are anticipated, this mode of operation may prove very desirable.

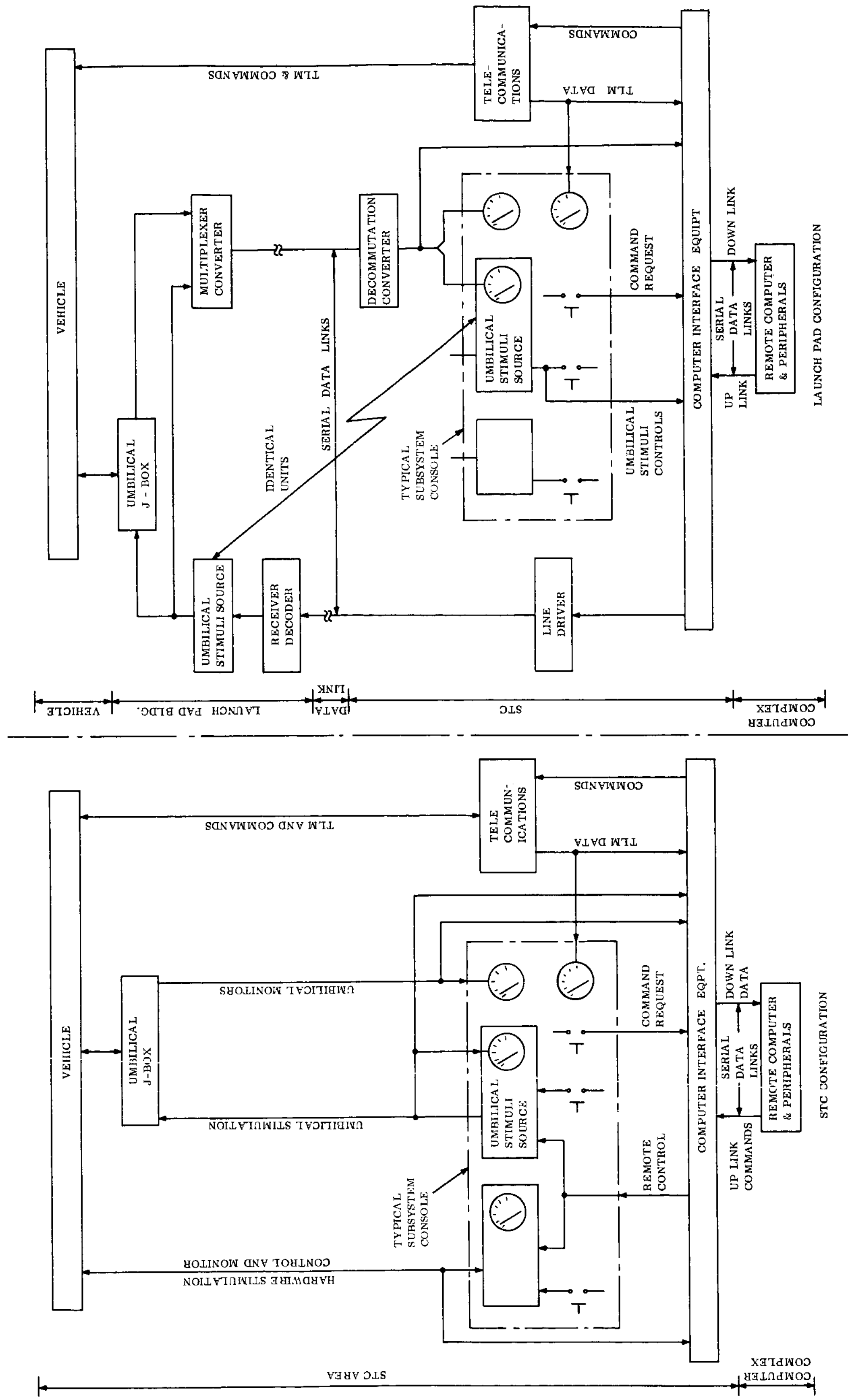


Figure 5-4. Modified ACE, STC and Pad Configurations

6.1 RECOMMENDED STC CONFIGURATION

a. General

The recommended STC configuration is in reality a combination of all three of the above schemes. This configuration is shown in Figure 6-1 and 6-2 and the main features and differences from Mariner C are described below.

b. Basic Configuration

The basic configuration is very similar to the Mariner C STC configuration in that it consists of subsystem oriented OSE consoles tied together by STC peculiar equipment in order to perform system type tests. The major pieces of subsystem OSE will be:

1. Power.
2. Guidance and Control.
3. TLM.
4. Command.
5. Radio.
6. Control and Sequencer.
7. Pyro.
8. Thermal.
9. Capsule.
10. Capsule Simulator (for tests without Capsule).
11. Experiment.

These consoles will be used to test their subsystems prior to installation in the vehicle and provide these functions as well as the STC peculiar functions required during System Tests.

While the hardware associated with the CVE and TCD is not Mission Dependent, the software required by them will be. These pieces of hardware are therefore required in order to verify the Mission Dependent Software during the System Tests.

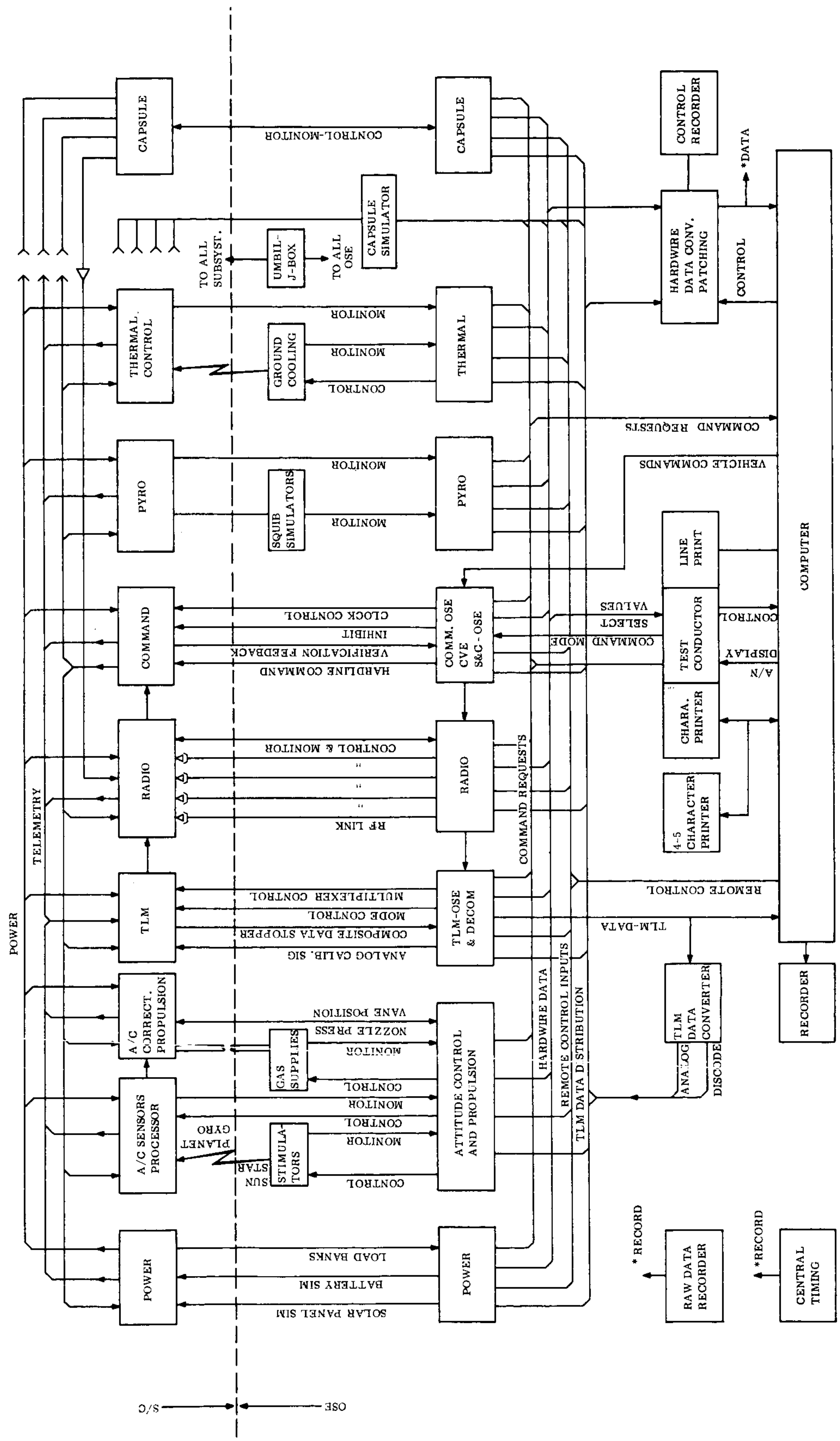


Figure 6-1. STC Block Diagram

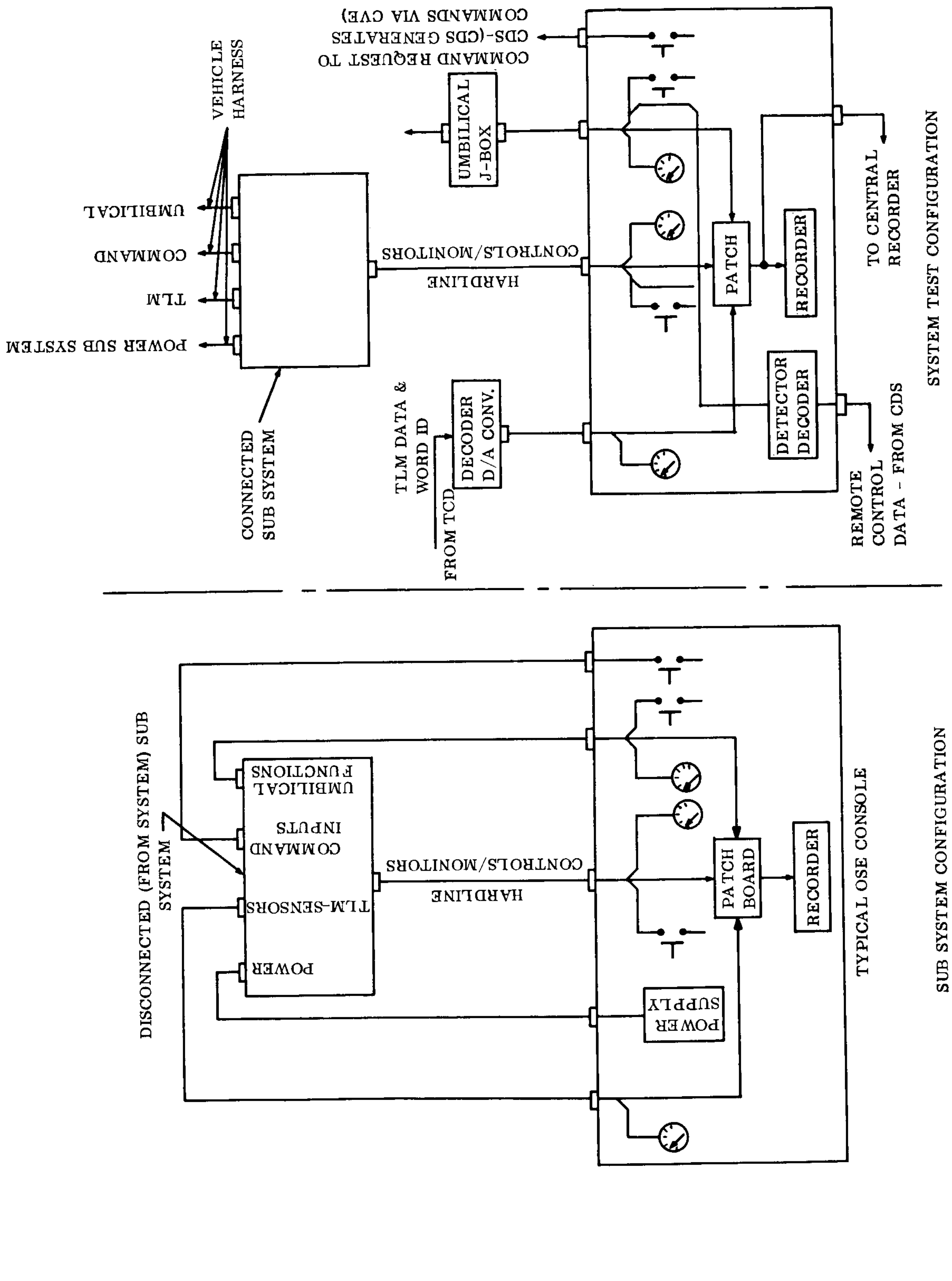


Figure 6-2. STC Subsystem OSE Interconnection Modes

c. Computer Data System

The usage of the CDS as outlines in Section 5c, Modified Mariner C approach, is recommended. The primary reason for this is that the Systems Test will require so many routines (even though each is extremely simple the total quantity makes it complex) that the time involved, the coordination required over long test periods, etc., the operators, even though they will be high level and highly motivated, will not be able to perform as reliably and repeatably as the computer. (Note: The type of operator failures expected would not cause damage to the vehicle, but would require rerunning the tests, etc.)

This remote control capability should be incorporated now while it can be done inexpensively. If later studies indicate that it is not required, then it does not have to be used. However, if these later studies verify this preliminary feel, it would be too late (schedule and design wise) to incorporate it then. This remote control capability will be provided for by the OSE console providing circuits to allow their circuits to be set up remotely as well as locally. It is felt that at this stage of development, greater capability for this control should be provided than just the minimum. This capability should include at least the following functions:

1. Command data.
2. Command format control - modify preamble - parity errors - bit counts, etc.
3. Control vehicle master timer - set - hold - hi/lo rate select, etc.
4. Control TLM mode - data rate - homing controls, etc.
5. Control Guidance and Control so as to introduce errors (rate-position) into all channels and sensors.
6. Control Guidance and Control's OSE Stimulation equipment.
7. Control Radio modes.

The computer will also control printers throughout the STC. These will include a high speed line printer located adjacent to the Test Conductor's console. This printer will be used to record the official results of the test sequences performed and will become an integral part of the buyoff documentation. Character printers (teletype) will be distributed around the STC such that adjacent subsystem engineers will have access to one as well as one for the Test Conductor. These units will be used as scratch pad type of data interchange between the computer and the operators. The operators will be able to call up data from the computer as provided in its program. The computer will also present instructions or data to the subsystem engineers during the test sequences in order to keep them aware of the progress or their function during the test.

The computer will obtain data from the Telemetry Decommutator. This data will contain at least the data (in binary) and an identification code.

The CDS will also contain means for converting selected hardware (vehicle or OSE) analog signals to digital as well as detecting switch operations of the OSE. It is expected that equipment similar to that used on Mariner C for this purpose would be directly applicable to the Voyager Project, if it can be expanded by a factor of two (or use two separate units).

During subsystem tests (or troubleshooting, etc.) on an assembled vehicle, the OSE subsystem operators will require the capability to send commands to sequence their components. Their OSE consoles contain switches that were used for this purpose during their disconnected subsystem tests performed prior to assembly into the vehicle. By having the CDS sense these switches, the computer can cause commands to be sent via the CVE or the Command OSE to the vehicle command decoder and to the desired subsystem. This gives an extremely convenient method of generating these commands as opposed to punching new tapes/cards, typing apparent random characters, etc., to accomplish this function. It should be expected, however, that the Test Conductor will have to approve these commands when used in this mode. He will, therefore, have control functions to inhibit these commands or prevent their being transmitted until he approves.

The computer will record the test results. This should be a relatively small amount of data (10-15 K values) that represent the results of the tests and not the data from which the results could be obtained. This data will be used to plot the trends of the test results for a detail comparison of test results from test to test. This latter function will be done off line and probably at a centralized data processing center. The data should be recorded in a format suitable for that purpose.

The computer will be remote from the STC. It should also be able to control/monitor several STC areas. The interconnection between the computer and STC is planned to be parallel links although serial transfer of data can be accomplished if later studies indicate that this is desirable.

d. Data Recording

The data listed below is to be recorded unprocessed and independently of the computer. This data is to be used to reconstruct test sequences to investigate failure, to provide historical data such that all data leading up to a failure is retained, and to retain data some of which later tests may reveal as being significant. The data to be recorded is:

1. Continuous hardwire Engineering TLM data.
2. Composite hardwire or recovered RF TLM data.
3. Converted hardwire monitored data.
4. Vehicle command bits.

5. OSE control signals.

6. Time.

7. Voice.

The STC will have the capability of playing these signals back into the computer and the OSE consoles displays and recorders.

In addition, the STC will contain a oentralized recorder for the direct recording of selected analog signals. These signals will be selected by means of a patch board from the analog signals supplied to the CDS. The signals to be recorded are those signals from several subsystems that may require time correlation. Each subsystem OSE will contain recorders required to monitor and correlate the signals within that subsystem.

e. TLM Data Distribution

During the time the OSE consoles are operating with their respective disconnected subsystems, they are monitoring their TLM sensors directly (or through suitable conditioning/conversion equipment). When the subsystem is connected into the system, these connections are lost to the console. The only way to recover this data is to obtain it from the TLM system. This will be done by providing STC peculiar equipment that will operate on the output of the TLM decommutator (word value and word count) so as to gate the desired values into flip flop storage to drive D/A converters or lamp drivers which can then be connected to the meters or lights that had been connected to the TLM sensors. (It should be noted that this logic, etc., could be provided in each OSE console. Future study may indicate that this is the preferred way.)

This method allows the removal of the TLM OSE printers used on Mariner C. This method is also preferred over the printer method for the following reasons:

1. Allows subsystem engineers to use same displays as were used in subsystem tests.

2. Meters can be calibrated in units desired, where as TLM units only were available from the printers.

3. Voyager data rate will be much higher (8 K bits second) making dynamic displays more meaningful and desirable.

4. CDS character printers will be available to print data as requested.

It is not expected that all TLM data will be converted in this manner continuously. The significant data will be, while the other data would be obtained by calling up special displays (controlled by patch boards or selector switches) or request printouts from the CDS.

f. Test Conductor's Console

The Test Conductor will be responsible for the vehicle while it is in the STC. His console should therefore provide him with data from which he can quickly assess the status of the vehicle, the OSE and the test sequence. His control capabilities, however, will be quite limited. He will have to depend on discipline to insure that the subsystem engineers perform their functions when and in the proper manner that the agreed upon procedure requires.

In addition to the printers (part of CDS), the Test Conductor should have the capability to monitor the system on a more selective or dynamic basis. To provide this capability an Alpha Numeric Display presented on a Cathode Ray Tube is recommended. The Test Conductor will be able to select the display group to be displayed from many groups. These groups will be formatted by the CDS such that values, limits, and comments can be presented. Another mode should allow the display of groupings of TLM data (in TLM units) as a backup for the CDS displays.

g. Central Timing

A Central Timing source will be provided essentially identical to that used in the Mariner C STC.

7.0 REMOTE CONTROL IMPLEMENTATION

The same ground rules used for incorporating the CDS into the Mariner C STC can be applied to providing the remote control capability for the Voyager STC. That is, the satisfactory operation of the subsystem OSE during System Tests should not be dependent on the remote control capability. This not only requires the OSE to provide its own controls but requires the System Tests to be planned to be able to be carried out without this feature.

The remote control feature will be applied first to those areas in which the greatest gains to the system can be obtained. It is not expected that troubleshooting procedures would be attempted.

Considerable effort should be expended in developing mnemonic coding for the computer that is system oriented rather than computer oriented. This should be accomplished such that the subsystem engineers can write programs directly (to be converted to computer codes by compiler) without an extensive course in computer programming, etc. If this is done, the programs will be superior and their usage more easily incorporated into the system.

While the circuit details have not been designed for the Computer OSE console interface for this remote control function, it should not be a complex problem. There should be considerable care in establishing the self check (in process) criteria of this data interchange such that improper codes are not accepted. It is anticipated that the vehicle will not be damaged if incorrect data is received. This should at the most require the tests to be rerun.

CII - VB260AA003

PROPELLANT LOADING ALTERNATIVES

Index

1.0 PROPELLANT LOADING ALTERNATIVES

The recommended propellant loading OSE, controls the propellant load by continuously monitoring the weight reduction of a supply tank as the propellant is transferred from it to the subsystem by the pressure of an isolated quantity of expulsion gas. The selection of method was made after considering several. Brief discussions of three of the other approaches considered follow in the succeeding sections of this document.

2.0 MECHANICAL SCALES

A mechanical weight scale could be substituted for the variable inductance load cell of the recommended OSE at some cost advantage.

However, a scale of sufficiently high resolution and repeatability for the application would be clumsy to implement into the design. It would introduce difficulty in transporting the OSE so as not to damage the very delicate scale mechanism, and it could not be monitored from the Control Console of the OSE. Finally, a mechanical scale would not allow any greater sophistication than presently planned in load control, such as a warning indication when the selected load is nearly complete, or an automatic stop-load when the selected load is transferred.

3.0 SPECIFIC VOLUME

Several variations of specific volume loads were considered. In all of these, a tank in the OSE of known volume is loaded full of propellant, and then completely discharged into the subsystem tank.

This method is the least costly, fastest and most foolproof of all considered, requiring only a consideration for the propellant temperature by the transfer control operators before loading, in order to determine the weight actually loaded.

It was not favored for Voyager propellant loading OSE because off-loads could not be accomplished in any simple manner. Also, it appeared that temperature stratification of propellant in the OSE tank could take place and thus prevent an accurate determination of the propellant density by temperature measurement.

4.0 OVERFILL

If subsystem propellant tanks are used with butyl bladders, or without bladders or diaphragms, the tank can be filled completely with propellant of measured density and then a measured weight quantity removed to reduce to the desired load. The technique applies and does not require an exceptionally accurate weight sensor where the subsystem tank volume is known.

Disadvantages are that some Voyager tanks will have diaphragms or bladders which may not allow complete filling, and that the desired load may not result in the subsystem tank being nearly full. Also, propellant density calculation from temperature measurements may not be accurate due to the possible tank temperature stratification.

CII - VB260AA004

LEAK TEST ALTERNATIVES

Index

1	General
2	Reverse Pressure Differential Leak - Atmospheric Pressure
3	Normal Forward Pressure Differential - Captured Leak
4	Vacuum Chamber - Normal Pressure Forward Leak
5	Pressure Drop - Normal Forward Pressure
6	Acoustic Leak Testing
7	Foam or Bubble Tests

1.0 GENERAL

The recommended primary method of leak testing the pressurized liquid and gas sections of the spacecraft subsystems is by internal pressurization with helium enriched nitrogen, and then detecting and measuring individual external leaks by means of a gas analyzer and probe. This primary method is backed up by capability to measure pressure rise and pressure drop in isolated sections of the subsystems.

Major advantages of these methods are that the subsystems are tested at or near working pressure and they remain clean and dry. Also, testing is reasonably rapid and does not require major support facilities except for burst safety during proof testing.

Several other approaches to leak testing were considered. These techniques are briefly discussed in the following sections of this document.

2.0 REVERSE PRESSURE DIFFERENTIAL LEAK - ATMOSPHERIC PRESSURE

This method is attractive because it can measure the overall leak behavior of a system, requires no more equipment than the selected method, and does not contaminate the subsystem.

The technique consists of purging the subsystem with pure N_2 and evacuating to approximately 10^{-2} torr. Atmospheric oxygen enters the subsystem through existing leak paths under the influence of atmospheric pressure, and is detected by mass spectrometer.

If the subsystem has an allowable leak of 8 scc/hr from a tank pressurized to 4000 psig, the reverse leak rate of atmospheric 0_2 at one atmospheric differential pressure is in the order of 1×10^{-5} sec/hr. Leaking into a 4 cubic foot tankage volume which has been evacuated to 10^{-2} torr, the oxygen concentration would build up to a measureable level within about 2 hours.

Capability for utilizing this technique could be added to the recommended equipment by using a residual gas mass spectometer as a gas analyzer instead of the less sensitive analyzer which would otherwise be used.

The method was not incorporated in the recommended design since, for the type of system to be tested, leak paths which exist under normal working conditions may close under test conditions, or at least diminish in size so that a clear correlation is not possible.

3.0 NORMAL FORWARD PRESSURE DIFFERENTIAL - CAPTURED LEAK

Using this method the subsystem is pressurized with helium containing nitrogen, as with the selected method. However, instead of "sniffing" directly over the exterior subsystem surface for helium, the escaping helium is collected within a plastic film envelope or a tent surrounding the subsystem. When the air within the plastic envelope has collected sufficient helium, the helium concentration can be measured by mass spectrometer.

For each 100 cubic feet of tent volume about 3 to 4 hours would be required for a sufficient helium concentration to build up (at an 8 scc/hr leak rate) to enable measurement by mass spectrometer.

The technique is applicable to overall leak determination of Voyager subsystem modules, except that the probable difficulty in bagging a subsystem with low volume tenting, the long test time, and the lack of capability for pinpointing leaks limit the relative attractiveness.

The recommended leak test equipment would be given the capability of this leak test technique merely by addition of the tent, should it become a desirable feature.

Variations of the tent are to be employed with the selected method, in that the gas analyzer probe will at times be fitted with a small rubber cup, or a small probe tip will be enclosed within a section of subsystem line wrapped with plastic kitchen wrap to obtain a high concentration of helium for quick detection and accurate measurement of small leaks.

4.0 VACUUM CHAMBER - NORMAL PRESSURE FORWARD LEAK

A variation of section 3.0, the method utilizes a vacuum chamber in which to place the subsystem instead of a tent. The particular advantage is that test time is reduced to about one minute per 100 cubic feet of vacuum chamber volume for a leak rate of 8 scc/hr. However, the cost, awkward aspects of handling the subsystem and the fact that leaks cannot be pinpointed prevent serious consideration of the use of a vacuum chamber for leak testing.

5.0 PRESSURE DROP - NORMAL FORWARD PRESSURE

In this old method, a section of the subsystem would be pressurized to normal working pressure and leak rate determined by the length of time required for the subsystem pressure to drop some measurable amount. In the case of subsystem tankage, even 20 scc/hr leak rates would require several weeks to detect. Very small volumes such as a short line section can be tested by this method, and the recommended OSE has this capability.

6.0 ACOUSTIC LEAK TESTING

For acoustic leak testing, a small sensitive microphone is used to detect the sound of leaking gas escaping from the pressurized subsystem. The method is useful for detecting and locating leak rates of interest. The disadvantages are that the background sound level must be controlled, and that the method does not measure leakage rate.

7.0 FOAM OR BUBBLE TESTS

This is leak detection by causing the escaping gas to produce bubbles in a thick liquid solution which is applied to the suspected surfaces of the pressurized subsection. Leak detection, location and measurement of leak rate can all be obtained for leaks of the minimum sizes of interest in Voyager tanks by bubble test.

However, such solutions, or other liquids, can plug small leak paths by capillary action, and then possibly several of these could evaporate or blow out during the mission to yield a higher total leak for the subsystem than is allowable. Also, liquid plugged leaks can make the results of a series of tests confusing. (For instance, as initially plugged leaks begin to flow clear, successive leak tests could falsely indicate that welds in subsystem may be stress cracking.)

It is considered important to avoid all liquid or grease contact with the subsystem up to the time of propellant loading in order that the initial leak evaluation of the subsystems will have greatest probability of remaining stable over successive tests, and during the mission.

CII - VB260AA005

DEPLOYABLE ELEMENT TESTING

Index

1.0 SCOPE

This document will discuss the considerations involved in testing and evaluating the performance in a one g environment of elements essentially designed to operate in zero g.

2.0 REQUIREMENTS FOR TESTING

This evaluation of operating functions of a mechanism or mechanical component or electronic package can only be done by actual test operation. Additional complications are introduced when the test cannot exactly duplicate operating conditions. Such a case is that under discussion here - zero g conditions cannot be duplicated on Earth except for very short periods yet the mechanisms to be used in these conditions must be evaluated in some manner. Analysis must then be used to demonstrate that the 1 g test will indeed evaluate zero g behavior of the item being tested.

3.0 VOYAGER DEPLOYABLE ELEMENTS

Several deploying mechanisms have been identified on the Voyager Spacecraft thus far. They are as follows:

a. High gain antenna.

b. Deployable portions of the solar arrays.

c. Magnetometer on its boom.

d. Planet scan package.

All of these mechanisms must withstand very similar erecting and operating conditions. The environment is the same for all of them with respect to vacuum, temperatures, radiation, etc. Therefore, the mechanical testing conditions will be discussed as a group rather than part by part. Such subjects as the following will be covered:

a. Required movements.

b. Environmental conditions.

c. Required life.

d. Consequences of failure.

e. Simulation of conditions for test.

f. Validity of test results.

g. Test on spacecraft vs on a test stand.

4.0 OPERATIONAL MOVEMENTS

The four mechanisms listed above have in general two movements: the high gain antenna and the planet scan package, once deployed, continually perform scanning movements of varying magnitude while the magnetometer and movable portions of the solar arrays are deployed once and have no further movements.

5.0 ENVIRONMENTAL CONDITIONS

All the mechanisms must operate in space. The antenna and planet scan package mechanisms will be sealed and pressurized with a cold gas to preserve more nearly an Earth environment to bypass problems of mechanisms operating in a vacuum.

The solar array and magnetometer mechanisms will be exposed to deep space conditions directly but since they only operate once early in the mission they will not be protected.

During mid course correction and retropropulsion engine firings, some heating of the mechanisms may possibly occur and approximately one g forces will be imposed on all mechanisms, though only the antenna and scan package may be seriously affected.

6.0 DESIGN CRITERIA

To survive the imposed environments and physical loadings, certain design criteria must be set as follows:

a. Once deployed into operating position, no appendages will be restowed.

b. No deployed members nor any operating mechanisms shall experience significant distortion, permanent or transient, under the loads imposed by the engine firings or any separation sequence shocks.

c. All one g testing will give results which can be interpreted from the standpoint of zero g performance.

d. During boost conditions, all deployable mechanisms will be restrained against damaging movements.

7.0 REQUIRED LIFE

The high gain antenna and planet scan package have the most severe operating life conditions. The antenna must operate continuously throughout the mission after deployment. The scan package is deployed just prior to the Mars approach phase and is operated in a planet scan mode during Mars orbit. The operating life of both items will be a limiting factor on mission life. If the planet package fails, no planet science data from it will be obtainable; if the high gain antenna drive fails, no data will be transmitted in either direction unless the spacecraft itself can be controlled to point the antenna to earth.

Failure of the solar arrays to deploy will reduce power generation capacity of the spacecraft. Jammed panels will not receive sunlight but they will not shadow the fixed panels. Thermal control of the fixed panels may be seriously degraded by the presence of the jammed deployable panel.

8.0 SIMULATION OF CONDITIONS FOR TEST

Zero g operating conditions will not be duplicated for the deployment tests. The mechanisms will be oriented for test such that the force of gravity will not affect the test results (required motion in the horizontal plane). Hard vacuum and thermal conditions can be simulated adequately with facilities existing at GE VFSTC such that an exact mission profile can be run to evaluate affects of exposure to space conditions before operation of the mechanisms.

9.0 VALIDITY OF TESTS AND RESULTS

The problem in justifying one g testing as representative of zero - g conditions is one of arranging the test itself such that gravity does not add or subtract from the forces being evaluated. By mounting such that the movement is in the horizontal plane, gravity essentially adds only friction forces in the bearings and does not affect the magnitude of the actuating forces.

Performing the test such that gravity tends to help or retard the motion being evaluated, still poses the question of adequecy of the test, even if successful. With gravity helping, a successful test does not prove a satisfactory arrangement since the gravity attraction may be necessary for complete deployment. With gravity opposing the action, an unsuccessful test does not prove that the arrangement could not operate satisfactorily in a zero g environment.

Counteracting the friction possibly created in a joint presents a design problem. A counter balancing of the weight involved would be one answer, another method would be the use of very low friction materials in the joints. For a sealed and pressurized joint, outgassing of materials would not pose a serious problem so that materials could be chosen for low friction abilities alone.

The high gain antenna operates in its primary mode during both midcourse and retro engine firings, which apply approximately one g to the arrangement. This one-g applied load will probably be the design condition for the extended arrangement and as such will be identical with the test conditions.

10.0 COMPONENT TESTING ON INTERFACE FIXTURES VS TESTING WHILE MOUNTED ON SPACECRAFT

The question of testing deployable elements while mounted on the actual spacecraft vs mounted on an interface or test fixture is primarily one of determining the validity of the interface fixture test. It can be argued that conducting a test under any but completely operational conditions fails to prove operational capability. On the other hand, developmental testing of a deployable appendage on a spacecraft may be impossible from a scheduling standpoint.

Evaluation of concepts and developmental tests, at least, should be done on interface fixtures and only a final systems check should be made on the actual spacecraft. These evaluation tests are mechanical in nature only; thermal vacuum testing, where an interface other than purely mechanical may exist with the spacecraft, would be quite difficult to simulate. In such a case, test on a spacecraft model (PTM) may be mandatory.

The advantages in terms of schedule flexibility, lower costs, easier handling, make component testing on interface fixtures much more attractive than on a complete spacecraft, and such a plan will be the preferred one.

CII - VB260AA002

ALTERNATE GUIDANCE APPROACHES

STATIC VERSUS DYNAMIC AND CONTROL TESTS

Index

1.0 SCOPE

The Voyager Guidance and Control (G&C) Subsystem must be thoroughly tested in an earth-ambient environment to ensure a high degree of confidence that it will successfully fulfill its designed mission. This document discusses some of the test criteria and trade-off parameters for optimum testing in the System Test Complex for Acceptance Tests.

2.0 APPROACHES

The major problem to be resolved in how to test the A/C in a System Acceptance Test environment is how to stimulate the vehicle such that the sensors, the following electronics, and the control forces can be verified in all of their modes. There are basically three ways of accomplishing this that have been used on past programs. They are:

a. Approach No. 1 - Place the vehicle on an air bearing and stimulate the IR or sun sensors with high precision fixed stimulation devices. The S/C control forces will maintain vehicle attitude in a closed loop manner.

b. Approach No. 2 - Provide precision stimulation devices that can be positioned and moved about the fixed vehicle in a manner suitable to test the sensors and the following electronics. The control operation will be observed (i.e., gas jets operating or flywheel speed changing or jet vanes moving) as occurring at proper input conditions and in proper amounts and directions. The tests can be performed in a simulated closed loop manner by altering the position or motion of the simulation device as a function of the detected and assumed control force. This control force effect can be idealized (i.e., a gas jet causes constant acceleration) or it can be adjusted to give varying feed back. This capability requires a motorized stimulation source. The tests can also be performed in an open loop manner by not altering the stimulation position/rate as a function of the control action. (Note: Driving the vehicle about fixed stimuli is considered an alternate of this basic approach to be used when the S/C contains predominantly motion sensing devices - gyros accelerometers.)

c. Approach No. 3 - Perform the tests in two steps by (1) using sensor stimulators that are optimized for testing the sensor, and (2) inserting a ground generated electrical signal at the output of the sensor (sensor signal simulator) that is optimized to test the electronics. The control action is noted as in (2) above and can be used to alter the electrical signal to operate the electronics in a simulated close loop manner if desired.

3.0 COMPARISON

Methods given in paragraphs 2.0 a and c, are the extremes to the approaches and can be compared as follows:

a. Air Bearing Approach

Advantages	Disadvantages
1. Verifies Design	1. Complex Test Equipment
2. Mission oriented people understand test	2. Voyager Attitude Control subsystem is not new in concept
3. Requirements can be established early in design sequence	3. Minimum Test connections allowed
	4. Trend type data can not be obtained without additional equipment
	5. Requires extra equipment/capability to perform Manufacturing Acceptance Test
	6. Comparatively costly
	7. Risk of test failure
	8. Data difficult to analyze

b. Static Test Approach

Advantages	Disadvantages
1. Verifies proper operation as designed	1. Requires access to sensitive signal leads
2. Simple test equipment	2. Mission oriented people generally do not understand test
3. Data simple to analyze	3. Requirements can not be established until design details are known
4. Trend type data directly available	
5. A manufacturing acceptance test	
6. Comparatively inexpensive	
7. Low risk of test failure	

The air bearing (Approach No. 1) is basically a development tool ("pure" closed loop operation) and does not lend itself to acceptance testing which requires some open loop testing. To provide this capability this approach must also contain means for operating as in one of the other approaches. Because of this and because this is a high OSE risk approach, this approach was discarded.

Approach No. 3 is the optimum approach from a test equipment point of view (e.g., simplest, lowest risk, best flexibility). It also tests the vehicle components in an optimum manner. Its major drawback is that it requires the introduction of an OSE generated electrical signal into the vehicle system at a critical circuit location. If this approach is to be used, the vehicle design must provide a reliable method (from a flight configuration point of view) of introducing this signal. This must be done without destroying the capability of also verifying that the sensor signal will also be processed properly when the ground signal is removed. This can be accomplished by inserting signals that bias the sensor signal (as stimulated by fixed steady stimuli) in such a manner that the net signal is the one desired to test the following electronics. This approach is further enhanced when the sensor can be tested with a fixed stimulation device.

The No. 2 approach is a compromise between 1 and 3. Compared to Approach No. 3, it is a more optimum approach from a vehicle point of view (i.e., runs end-to-end tests, does not require sensor simulation lead, etc.). However, this approach has a higher OSE risk in direct proportion to the amount and degree of drive capability required to position the stimulation and the degree to which the stimulation has to simulate the real source. If the vehicle design is such that these problems can be minimized without adversely affecting the depth or degree of test, then this becomes the optimum system approach.

4.0 CONCLUSIONS

The following statements summarize the conclusion reached as to generalized approaches:

a. Cost, complexity, and masking problems associated with dynamic G&C test equipment can quickly pass a point of diminishing returns when combined with space environment simulators, dynamic space motion simulators and the earth's gravity, and magnetic fields.

b. State-of-the-art and new concept loops should be feasibility proven on dynamic test fixtures under ambient conditions, and flight performance analytically predicted from these test results.

c. Once design feasibility has been satisfactorily demonstrated, it is not of further use to exercise principles on complicated dynamic test fixtures.

d. Complicated test fixtures and test equipment, beyond what is reasonably required to prove flight design, also require maintenance and operational personnel which do not contribute directly to the spacecraft success.

e. Due to test fixture alignment maintenance and troubleshooting being in-line with subsystem tests, real cost and time savings can be realized by using the simplest adequate test procedures.

This criteria can be applied to the Voyager on a preliminary basis as indicated below. Any final decision must ideally be made only when the vehicle design details are reasonably well defined. This detail will not be available until the end of the Phase IA study. However, for the purposes of this study the following assumptions can be made:

a. The star and sun sensors can be adequately tested with fixed stimuli, or at most with stimuli manually positioned to one of several discrete positions while the electronics that operate on their generated signals require signals containing known and varying position and rate components. To stimulate the sensors to generate those types of signals with the required accuracy, would complicate the stimuli unnecessarily, while it is a relatively easy task to generate the required electrical signal electronically. Because of this, these sensors and circuitry will be tested using Approach No. 3.

b. The planet sensor and its electronics generate an output pulse to move the planet package platform when the sensed position error is greater than a fixed amount. This type of system can be tested by slowly sweeping an IR target in front of the sensor and noting the position and S/C sensed error signal at which the step pulses are obtained and detecting the platform movement by noting the drop in error signal. This sweep can be controlled manually or a simple one-speed motorized drive could be used. This circuit and sensor would be tested, therefore, by Approach No. 2.

c. The gyros and their associated electronics can be tested adequately by using an OSE signal lead into their torque motor circuitry. By controlling this signal, simulated rates can be introduced into the system. These rates can be altered in a closed loop manner by detecting and integrating nozzle operations, etc. It may be desired to demonstrate proper polarity of gyros and control action by physically moving the vehicle on its checkout stand. To accomplish the intent of this test, this motion can be gross and the mechanism to accomplish it should be simple.

CII - VB260AA006

ALTERNATE APPROACHES

USE OF COMPUTERS IN THE SYSTEM TEST COMPLEX

Index

1.0 INTRODUCTION

1.1 THE SYSTEM TEST COMPLEX

The System Test Complex (STC) is the basic tool for testing and evaluating the Voyager Spacecraft. Its primary objective is to serve as a tool which can control, stimulate and evaluate the Spacecraft. To accomplish this objective, it must have the following capabilities:

a. Power the S/C by a ground power supply.

b. Monitor, charge and test the S/C flight battery.

c. Power individual subsystems by simulated S/C power.

d. Monitor and record operation time placed on a S/C subsystem.

e. Command the S/C through all its normal and back up flight modes.

f. Verify the S/C reactions to all commands.

g. Monitor and evaluate the operation of all subsystems via their direct access and umbilical functions.

h. Monitor all transducer outputs through telemetry to permit effective calibration.

i. Exercise all subsystems through their entire operating range.

j. Monitor all commands, control, and stimuli applied to the S/C from the OSE.

k. Record all major system parameters.

l. Verify the compatibility of the S/C with the DSIF in the areas of telemetry and command.

1.2 COMPUTER ORIENTED FUNCTIONS

The System Test Complex being developed in support of the Voyager Program will be required to perform many functions which lend themselves to implementation by a general purpose digital computer in conjunction with some special purpose peripheral equipment. This document is a comparative evaluation of several configurations of equipment for accomplishing these computer oriented functions. It also includes an evaluation of the specific pieces of hardware to be used in each system configuration.

The functions of the System Test Complex which are considered adaptable to a general purpose digital computer are as follows:

a. Telemetry Functions

1. Decommutation
2. Limit Checking of Telemetry
3. Selective Editing of Telemetry
4. Formatting and Distribution
5. Recording of Telemetry

b. Command Functions

1. Origination of Commands
2. Storage of Command Requests until needed
3. Encoding and Formatting Command Messages
4. Verification of Command Transfer Integrity
5. Decoding and Monitoring Command Transmission
6. Recording Command Transmissions

c. Test Control Functions

1. Control of the test sequence in a prearranged manner
2. Selection of alternate test sequences as a result of telemetry and/or test data
3. Display of test data and test status

d. Test Data Processing Functions

1. Collection of test data (analog, digital and time)
2. Limit Checking of Test Data
3. Calculation of desired data
4. Recording a complete history of all tests

Since the program requirements call for the near simultaneous launch of two Spacecraft and the availability of a spare Spacecraft, it is necessary to have the capability to perform all of the above mentioned functions for two Spacecraft simultaneously.

1.3 SPECIAL CONSIDERATIONS

The selection of the final configuration of equipment will be influenced by the following facts:

a. An automatic data system known as the Computer Data System (CDS) has been developed for supporting testing of assembled spacecraft and subsystems. The system consists of a medium sized, high-speed digital computer to process both telemetry and hardwire data. The hardwire data are of several types: analog, digital events, a-c voltage or pulse inputs for counting, and serial or parallel digital inputs. The system was planned for use on the Mariner Mars 1964 Project and the Ranger Project and is currently being used on the Surveyor Project. The computer programs developed for the CDS are of two classifications: those programs that are not affected by the peculiarities of the Spacecraft under test are called "Mission Independent" programs, and those that are affected are called "Mission Dependent" programs.

b. Another automatic data system known as the Telemetry and Command Data Handling Subsystem (TCD) is under development. The TCD will consist of one or more general purpose digital computers and peripheral equipment, such as that contained in the Digital Instrumentation Subsystem current installed at the DSIF stations. The TCD is designed for use in the DSIF stations where it will perform the following functions:

    1. Selective editing of S/C telemetry data

    2. Decommutation

    3. Generation of alarms

    4. Telemetry monitoring

    5. Processing of selected station instrumentation data

    6. Formatting telemetry data messages

    7. Verification of Spacecraft command data both as received and as transmitted

    8. Generation and verification of command tapes

    9. Permissive and limit checking of commands

A secondary use of the TCD is during system testing where it will perform functions similar to those listed. The use of the TCD in a system test complex has been considered in its basic design and appropriate input and output channels are incorporated into the system. The system is planned to be available during the Voyager program. The functions listed were performed by special purpose equipment on the Mariner, Ranger and Surveyor projects.

NOTE

> The GSDS Command System, Ground Subsystem known as the Command Verification Equipment (CVE) is considered to be that part of the TCD which is used to perform functions 7, 8, and 9, shown on the previous page. Therefore, the term TCD as used in this document refers to the equipment used for both the telemetry and command functions; whereas, CVE refers to the command portion of the TCD.

2.0 CONFIGURATION STUDY

2.1 INTRODUCTION

For the purposes of this report the functions of the STC are grouped in five main classes as follows:

a. Telemetry Function

b. Command Function

c. Test Data Processing Function

d. Test Control Function

e. Non-computer Oriented Functions

Functions a through d are those defined in paragraph 1.2 as computer oriented functions, while function e is all other functions performed by the STC. Each of these five functions can be considered for implementation either by a general purpose digital computer or by special purpose hardware. This gives rise to thirty-two (2^5) possible systems configurations.

A matrix of functions versus equipment type is shown in Table 2-1. Listed across the top of the table are the five functions while each row represents a different configuration. In the columns of each row are the letters S or C where S represents implementation of the function by special purpose hardware and C represents a computer implementation. Most of the configurations have been crossed out and the last column on the right gives the number of the reason why the configuration was eliminated from consideration.

Table 2-1. Function Versus Equipment Types, Matrix

	Configuration Number	Telemetry Function	Command Function	Test Data Processing	Test Control	Non-Computer Oriented Functions	Reason for Elimination
	1	S	S	S	S	S	2
	2	C	S	S			
	3	S	C	S			
	4	C	C	S			
	5	S	S	C			
	6	C	S	C			
	7	S	C	C			
	8	C	C	C			
	9	S	S	S	C		3
	10	C	S	S			
	11	S	C	S			
	12	C	C	S			
Configuration A	13	S	S	C	C	S	
	14	C	S	C			4
	15	S	C	C			4
Configuration B	16	C	C	C			
	17	S	S	S	S	C	1
	18	C	S	S			
	19	S	C	S			
	20	C	C	S			
	21	S	S	C			
	22	C	S	C			
	23	S	C	C			
	24	C	C	C	C		
	25	S	S	S			
	26	C	S	S			
	27	S	C	S			
	28	C	C	S			
	29	S	S	C			
	30	C	S	C			
	31	S	C	C			
	32	C	C	C			

The reasons are as follows:

1. Configurations 17 through 32 require the non-computer oriented functions to be accomplished by a digital computer. They are impossible configurations by definition and, therefore, removed from consideration.

2. Configurations 1 through 8 all require the use of special purpose hardware for the Test Control Function. It will be shown elsewhere that this function should be accomplished by a computer.

3. Configurations 9 through 12 accomplish the data processing function by means of special purpose hardware. This function does not lend itself to special purpose hardware in an economic fashion and is, therefore, eliminated.

4. Configurations 14 and 15 require the command and telemetry functions to be performed in different types of equipment. Either one is performed in a computer, while the other is performed by special purpose hardware. These two functions are essentially the inverse of each other and what rationale is used on one applies equally to the other.

This elimination process leaves two configurations as follows:

Configuration A — The telemetry and command functions are performed in special purpose hardware while the data processing/control functions are performed in a digital computer. This will be recognized as the system used on Mariner Mars 1964, Ranger and Surveyor.

Configuration B — All four computer oriented functions are performed in a general purpose digital computer. This configuration can be broken down further into:

Configuration B 1 — In this configuration, all four of the functions (command, telemetry, data processing and test control) are performed in a single digital computer.

Configuration B 2 — In this configuration, at least two and possibly three computers will be used. One computer will be used to perform the data processing test control function. The telemetry and command functions may each use a separate machine or it may be possible to accomplish both functions in the same computer.

The remainder of this report will be used to expand the definition of each of the last three configurations and to make a comparative evaluation of the three.

2.2 CONFIGURATION A

2.2.1 GENERAL

This configuration is essentially the same as that used for the Mariner Mars 1964 Vehicle. A general purpose digital computer is used to perform the test data processing function and the test control function. The addition of the latter function to the computers task list is the only change from the Mariner configuration. All other functions are performed by special purpose hardware designed specifically for each program. A Block Diagram of Configuration A is shown in Figure 2-1.

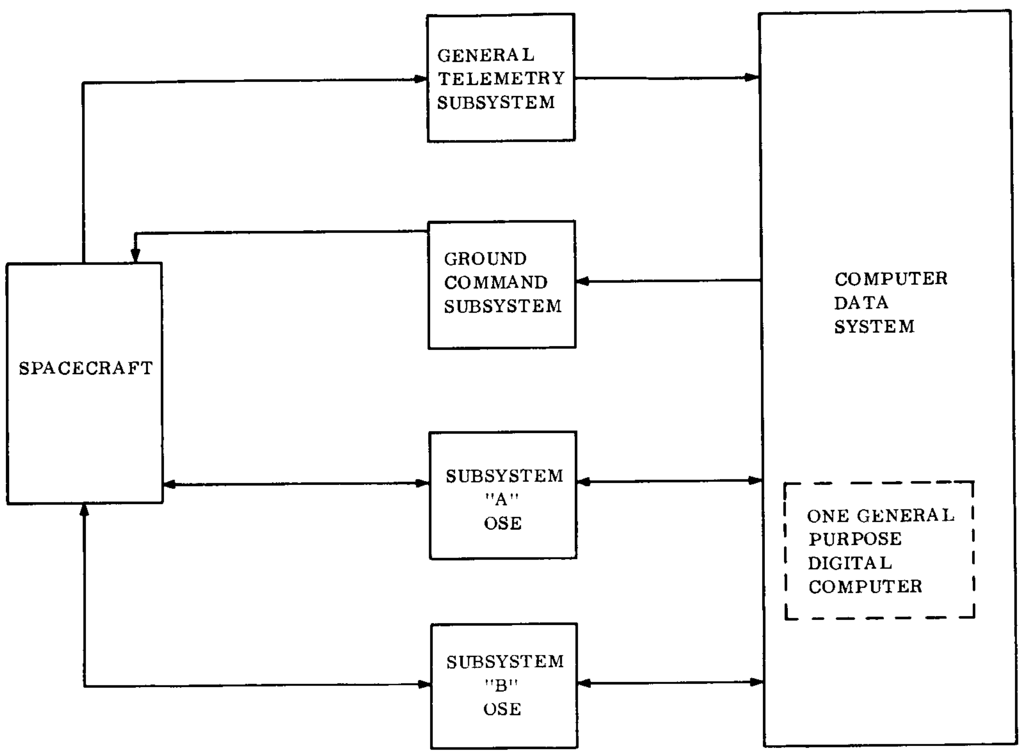


Figure 2-1. Configuration A

2.2.2 GROUND TELEMETRY SUBSYSTEM

The general telemetry subsystem is used to demodulate, decommutate and decode the noise corrupted Spacecraft telemetry signal and to distribute this data throughout the STC. It does not check or process that data. It is made up entirely of special purpose hardware, designed for this specific program.

2.2.3 GROUND COMMAND SUBSYSTEM

The ground command subsystem is used to receive command instructions transform the data into a form suitable for transmission, read the transmitted signal and verify its correctness and notify the originator of the command transmission. It is made up entirely of special purpose hardware, designed for this specific program.

2.2.4 COMPUTER DATA SYSTEM

The computer data system is designed to acquire, process and display data from the spacecraft system test and to control the test sequence. Data is acquired from the ground telemetry subsystem and the subsystem OSE and control is exercised by way of the ground command subsystem and the subsystem OSE.

Data Processing and control functions are performed by an on-line digital computer in the data system. All data acquired is digitally recorded on magnetic tape for subsequent detailed analysis. Data and control parameters are displayed for real time use by the Test Director's team and subsystem engineers.

2.2.5 CONFIGURATION IMPLEMENTATION

The heart of the system is the Computer Data System as designed for the Mariner Program. It is made up of the computer subsystem which is a Univac 1218 Computer with its associated printers, tape units, etc. and the data input subsystem which contains the telemetry input unit, the analog input unit and various digital input units. In addition, there would be a ground telemetry subsystem and a ground command subsystem, both of which would be special purpose devices developed specially for the Voyager program.

2.3 CONFIGURATION B 1

2.3.1 GENERAL

This configuration is essentially one general purpose digital computer to perform all of the functions with special purpose input/output equipment to get the data into the computer from the STC. The CDS interfaces with the subsystem OSE where it collects the data. It exercises control over the test by sending digital messages to the subsystem OSE when the control is implemented. A Block Diagram of Configuration B1 is shown in Figure 2-2.

2.3.2 GROUND TELEMETRY FUNCTIONS

Most of the telemetry function is performed in the computer, however, a special input device is needed to demodulate the signal and to provide bit sync and word sync. The output of this device would be telemetry words in parallel to the computer.

2.3.3 GROUND COMMAND FUNCTIONS

All of the command functions except the subcarrier modulation and sync code generator could be done by the computer. A special output device would be necessary to perform the aforementioned tasks.

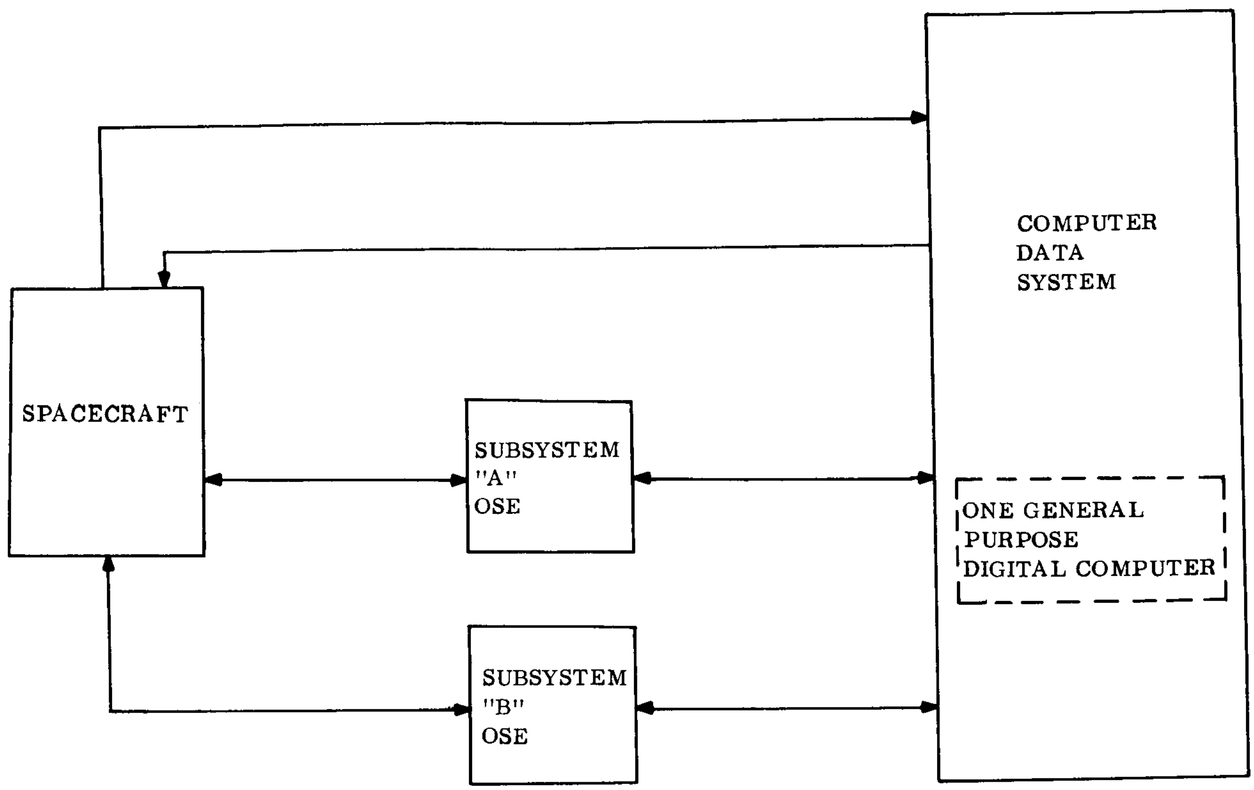


Figure 2-2. Configuration B1

2.3.4 COMPUTER DATA SYSTEM

a. General — The computer data system would consist of one general purpose digital computer with the telemetry and command input/output device plus the analog input devices associated with Configuration A.

b. Characteristics of the Computer — The requirement imposed upon the Computer Data System should reflect themselves as certain basic characteristics in the digital Computer Subsystem. This section discusses those aspects of hardware, software and other factors that are necessary and/or desirable, based upon the functions of the CDS.

1. Hardware

(a) Speed — The real-time functions require a fast machine in order to receive the data as it is generated and to maintain control over the sequence. In the case where the computer is used to service only one STC a speed of under four microseconds is desirable. To service more than one STC on a simultaneous basis, the speed should be increased to at least two microseconds.

(b) Core Size — The amount of memory in the computer is based on the size of the real time program, the executive control program and the type of non-real-time programs which must be assembled and run in the real time mode. The word size also affects the size of the core. The memory size thus becomes a trade-off between software complications of a small memory (job partitioning, executive program sophistication, increased I/O complexity) and the additional cost of a large memory. From the standpoint of system efficiency it is felt that the executive control program and the real time program for a complete test sequence should remain in core at all times. Thus, the core size chosen should be large enough to keep these programs in core and have sufficient room left to handle the non-real time program. It is felt that a 32K core is needed.

(c) Word Size — Available computers provide word sizes of twelve, eighteen, twenty-four, thirty-six, and forty-eight bits. The latter two are usually present in large scale computers and allow for power that is not needed in this application. A twelve-bit word presents addressing problems and is usually available on small computers. A twenty-four bit word seems ideal from the standpoint of speed and programming ease while eighteen bits is probably adequate.

(d) Instruction Repertoire — The programs require the normal arithmetic and control instructions and simple, fast input/output instructions. Conditional transfer and skip instructions are required to provide easy program looping and test control. This type of instruction repertoire is fairly standard.

(e) Interrupts — A priority interrupt system is the very heart of a real time system and, thus, is a prime requirement of this system.

(f) Input/Output — The real time function consists largely of inputting data and test conditions and outputting signals to display the test status and to control its sequence. Along with the priority interrupt system, this is the critical part of the computer. There should be provisions for buffered transfer of blocks of data into and out of memory as well as for normal transfers of single words from memory to the peripheral equipment. It is highly desirable to have the ability to input and output single bits of data under program control. Without this capability it will be necessary to assemble bits into a word and distribute the bits of an output word externally. The flexibility of the I/O hardware is important for future expansion considerations

(g) Other Features — Features such as indirect addressing and index registers are important aids to programming and efficiency. Since the non-real time programs may be run with the real time program still in core, a safety device is desirable to lock out the write capability in those areas of core occupied by the real-time program in order to prevent the latter from being destroyed during a test. A power fail-safe feature is also desirable to detect a power failure in the making and to allow the core contents to be saved.

2. Software — The software necessary for implementation of the real time function is special purpose and cannot be provided by the computer manufacturer. However, the non-real time functions require extensive system software. This would include a symbolic assembly program, a Fortran compiler, a complete subroutine library and an operating system for this application.

3. Other Factors

   (a) Reliability — The continuing real time nature of the testing program requires a high degree of reliability from the system. This means that the computer must have a high up time rating with minimum maintenance requirements.

   (b) Ease of Operation — The system cannot be so big and complex that, to be used effectively, it requires constant monitoring and much preliminary training. Usage of the system must be simple for both the operator and the programmer.

   (c) Type of Circuitry — Most of the computers available today are constructed using models containing discrete circuit elements. However, all of the major manufacturers have introduced computers using microcircuit logic modules. These modules potentially offer price, size and reliability advantages over the discrete logic modules. These advantages are only potential since they have not been proven by field use in large quantities. The aforementioned reliability requirements make it necessary to make a detailed assessment before using a microcircuit computer.

c. Computers Under Consideration — Based on the functions which it must perform, desirable characteristics for the CDS Computer Subsystem appear in the following machines:

Computers Using Standard Circuits:

CDC 3100

DDP 224

IBM 1800

SDS 9300

UNIVAC 1219

GE PAC 4000

Computers Using Microcircuits:

DDP 124

PDP 8

IBM 360

SDS 92

The matrix shown in Table 2-2 describes the important characteristics of these computers.

Of the machines listed, the SDS92, PDPS and the IBM 1800 must be ruled out because the word size is too small. The IBM 1800 and the IBM 360/MOD 30 have a multiply time which is considered too long for this application. The CDC 3100 and the Univac 1219 are difficult to use because they do not have single bit input/output capability. This reduces the choice to three conventional circuit machines, DDP224, SDS9300 and GE/PAC 4000 and one microcircuit computer DDP124. The choice among this machine would have to be made on the basis of a more detailed analysis of the requirements and the interface requirements.

2.4 CONFIGURATION B2

2.4.1 GENERAL

This configuration is shown in block diagram form in Figure 2-3. It is made up almost entirely of the data system already proposed or in use as described in paragraph 1.3. The telemetry and command functions are performed in the Telemetry and Command Data Handling System (TCD) while the test data reduction and test control functions are performed by the Computer Data System (CDS) as designed for the Mariner Program. The system can be configured to handle more than one spacecraft test by using the Univac 1219 in the CDS and using one TCD for each spacecraft

3.0 CONCLUSIONS

Each of the three configurations proposed is capable of performing all of the functions necessary to support the subsystem and system testing of the spacecraft. However, Configuration B 2 is recommended for this program for the following reasons.

Table 2-2. Computer Characteristics

		Word Size	Add Time	Multiply	Memory Cycle	Memory Size (Min/Max)	Inter-rupts	Indirect Add	Index Reg.	Input/Output		
										Words	Bits	
CDC	3100	24b	3.5 μs	14.5 μs	1.75 μs	4/32K	x	x	3	Yes	No	STD CKTS
DDP	224	24b	3.8 μs	6.46 μs	1.9 μs	4/65K	x	x	1	Yes	Yes	
IBM	1800	16b	12.5 μs	34 μs	2 μs	4/32K	x	x	3	Yes	Yes	
SDS	9300	24b	1.75 μs	7.0 μs	1.75 μs	4/32K	x	x	3	Yes	Yes	
Univac	1219	18b	4 μs	14 μs	2 μs	8/65K	x	x	8	Yes	No	
GE PAC	4000	24b	4 μs	20 μs	2 μs	4/16K	x	x	7	Yes	Yes	
DDP	124	24b	3.5 μs	14 μs	1.75 μs	4/32K	x	x	3	Yes	Yes	μ LOGIC CKTS
PDP	8	12b	3.2 μs	15.2 μs	1.6 μs	4/32K	x	x	8	Yes	Yes	
IBM	360	1a	39 μs	313 μs	2 μs	8/65K	x	x	x	Yes	No	
SDS	92	12b	3.5 μs	7 μs	1.75 μs	2/32K	x	x	1	Yes	Yes	

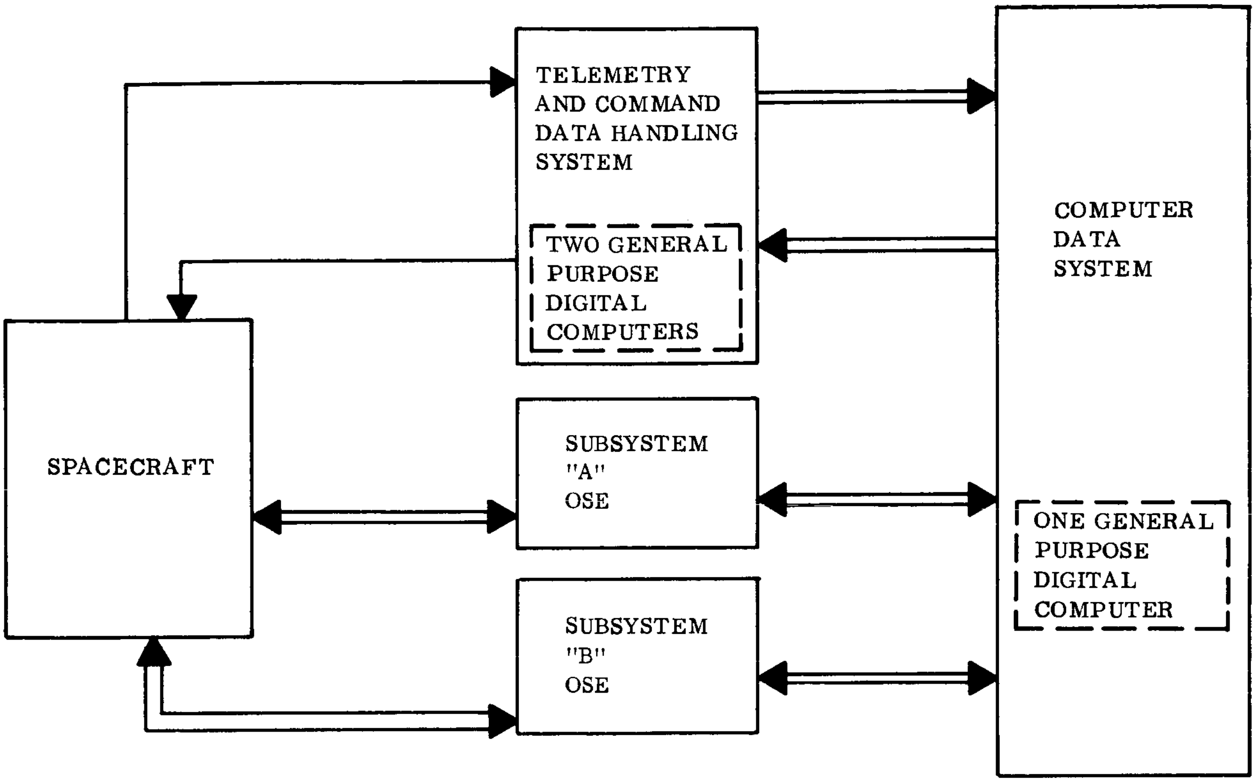


Figure 2-3. Configuration B2

a. It makes maximum use of the hardware already on hand and presently planned for performing the assigned tasks on other JPL programs.

b. It makes maximum use of the software systems and program subroutines already available and debugged.

c. Those functions which are common to the System Test Complex and the Deep Space Instrumentation Facility will be performed on identical pieces of equipment.

d. It is clear that the cost of the configuration is the smallest since it involves a minimum of new equipment and programs, whereas each of the other configurations involves designing new equipment and new programs.

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ALTERNATE APPROACHES

APPLICATION OF ACE TO VOYAGER

Index

1.0 INTRODUCTION

In the first section of this appendix, a configuration of the ACE system was discussed, in the context of a possible application of this design approach to the Voyager program.

This section of the appendix presents:

a. A functional description of the ACE system, as it was developed for and applied to the testing of the Apollo spacecraft, and

b. A functional description of the initial concept of how this ACE system could be configured to be applied to testing the Voyager spacecraft.

2.0 FUNCTIONAL DESCRIPTION - ACCEPTANCE CHECKOUT EQUIPMENT (ACE) APPLIED TO THE APOLLO SPACECRAFT

2.1 APOLLO APPLICATION - GENERAL DESCRIPTION

Several units of the ACE have been operational on the Apollo program, at spacecraft contractor's plants and at the Merritt Island Launch Area (MILA). Physically, the ACE consists of a ground station, remote from the spacecraft vicinity equipment. The ground station is shown in Figure 2-1. It performs the following tasks:

- Provides the control, display, data processing, and recording that is required to control spacecraft stimuli equipment.
- Receives, processes, displays, and records spacecraft parameter data derived from spacecraft ground and flight systems.
- Provides self-check and calibration capability for itself and related equipment.

The spacecraft vicinity equipment:

- Provides stimuli for the spacecraft, under the control of the engineer at the ground station console.
- Samples, interleaves and transmits test data to the ground station.

The ACE-S/C may be divided into two functional systems, the command system and the monitor system. The command system comprises those equipment groups which form the communication path over which all test commands and sequences are transmitted to the spacecraft. In addition, verification of receipt of valid commands is transmitted from the spacecraft vicinity back through the command system to the ground station. The monitor system comprises those equipment groups which form

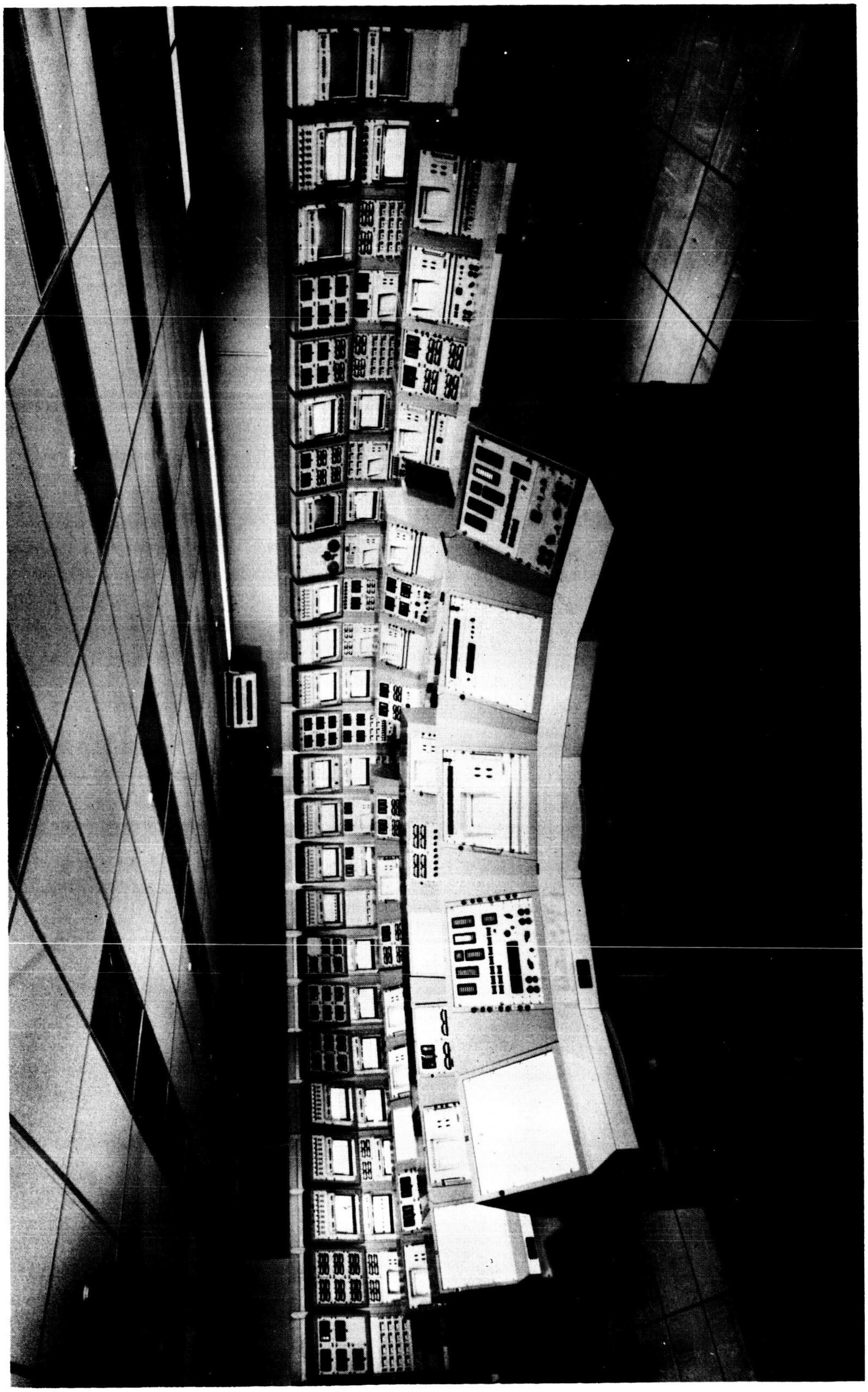


Figure 2-1. Typical ACE-S/C Control Room

the communication path over which spacecraft responses and test data are transmitted to ground station recorders and displays for evaluation. The block diagram, Figure 2-2, illustrates how these two functional systems operate.

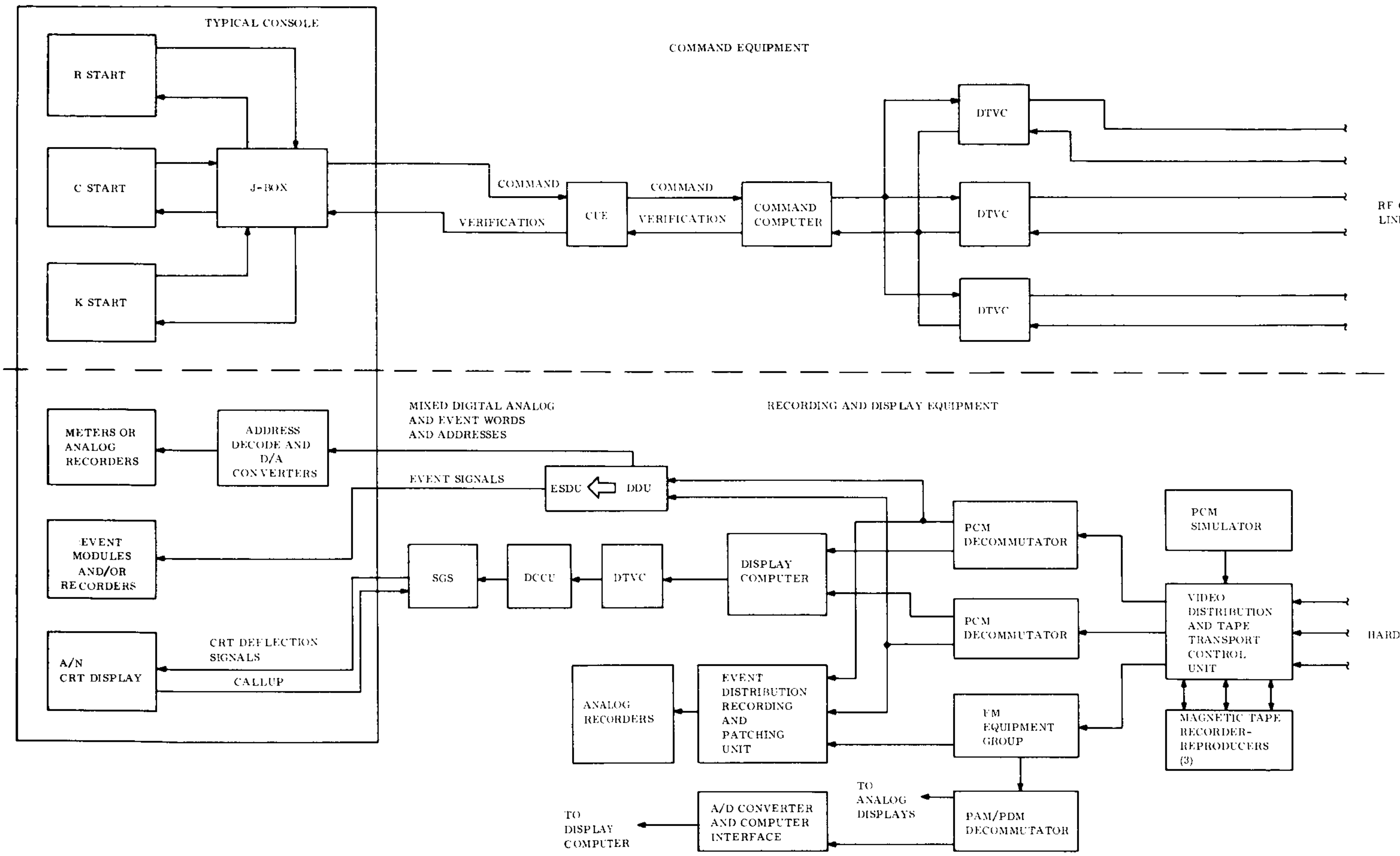
2.2 COMMAND

Control of command system operation, and thus, of the entire test sequence, can be manual or automatic depending upon the computer programming being utilized. In every case, tests are initiated at various system control consoles by setting switches on units called START modules. The setting of the switches provides for commands in digital form to the Command System computer (Uplink Computer). The types of commands vary from individual event functions (specific relay action) to the callup of prestored computer subroutines which control sequences of events and/or various analog operations at the spacecraft.

The testing of each functional spacecraft system is controlled from an associated group of system control consoles. Each system control console group operates simultaneously with and independently of the other system console groups and has a wide variety of test command capability necessary for complete checkout of a particular spacecraft system. In order that the computer may systematically process each of the many parallel inputs, a unit called Communications Unit Executor (CUE), operating essentially as a commutator in this respect, interrogates each START module on all of the system control consoles in sequence. The interrogations occur at a high rate, and to the console operators, there is no perceptible delay in their individual test procedures. When an interrogation determines that a command input exists at a particular START module, the sampling process ceases momentarily while the CUE transfers the digital command to the Uplink Computer.

The computer interprets and acts upon the command under program control. Some commands instruct the computer to modify memory while others require some action to occur in the spacecraft. In the latter case, the computer formulates a digital command message for transmission to the spacecraft. The digital message, which is in parallel format, is converted to a serial bit stream by a Data Transmission and Verification Converter (DTVC) and transmitted to the spacecraft vicinity over a hardline link. At the spacecraft location, the message is received and decoded, and the proper stimulus is generated and applied to the spacecraft.

Verification of command messages is accomplished by redundant transmission between the DTVC and a receiver/decoder at the spacecraft location, and a bit-by-bit comparison of the redundant words. In addition, checks are made of each message to determine whether it has legal addresses. Verification reply messages are transmitted redundantly from the spacecraft locations to the DTVC where verification of redundance is made. The fact that a message was delivered to the computer and the fact that verification of proper transmission to the spacecraft locations was obtained are indicated to the control console operator by appropriate lights on the START modules.



TYPICAL CONSOLE
R START
C START
K START
J-BOX
COMMAND EQUIPMENT
COMMAND
VERIFICATION
CUE
COMMAND
VERIFICATION
COMMAND COMPUTER
DTVC
DTVC
DTVC
RF
LIN
METERS OR ANALOG RECORDERS
ADDRESS DECODE AND D/A CONVERTERS
MIXED DIGITAL ANALOG AND EVENT WORDS AND ADDRESSES
RECORDING AND DISPLAY EQUIPMENT
EVENT SIGNALS
ESDU
DDU
EVENT MODULES AND/OR RECORDERS
SGS
DCCU
DTVC
DISPLAY COMPUTER
PCM DECOMMUTATOR
PCM DECOMMUTATOR
PCM SIMULATOR
VIDEO DISTRIBUTION AND TAPE TRANSPORT CONTROL UNIT
HARD
A/N CRT DISPLAY
CRT DEFLECTION SIGNALS
CALLUP
ANALOG RECORDERS
EVENT DISTRIBUTION RECORDING AND PATCHING UNIT
FM EQUIPMENT GROUP
MAGNETIC TAPE RECORDER-REPRODUCERS (3)
TO ANALOG DISPLAYS
TO DISPLAY COMPUTER
A/D CONVERTER AND COMPUTER INTERFACE
PAM/PDM DECOMMUTATOR

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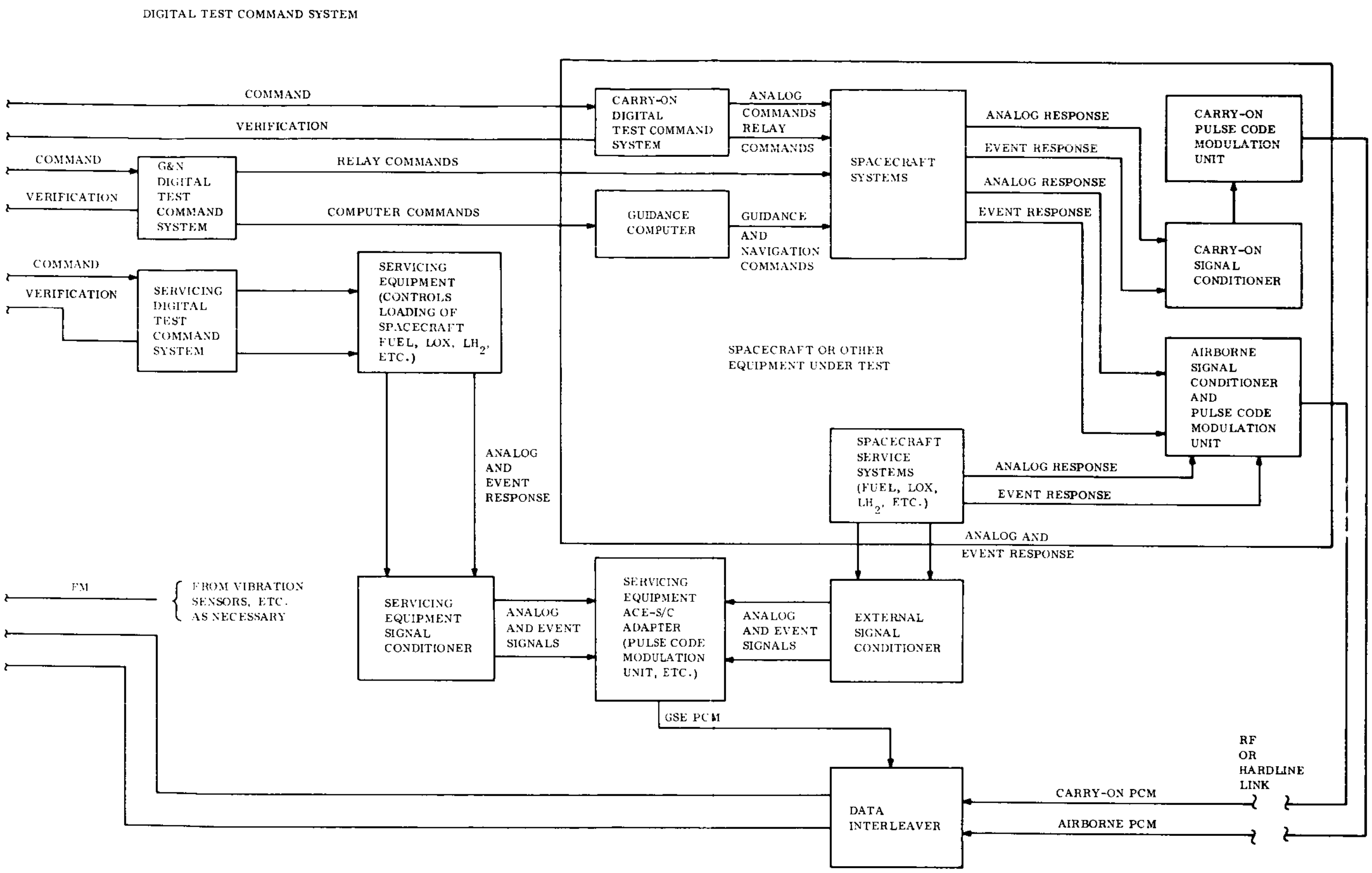


Figure 2-2. ACE-S/C Detailed Block Diagram

2

2.3 MONITOR

Spacecraft performance and status data are monitored by sensors coupled to the flight telemetry system, carry-on ground checkout equipment, and spacecraft service equipment. Most of the measurements are commutated, converted to digital form, interleaved, and transmitted in a serial pulse-code-modulated (PCM) format over a hardline link to the data acquisition equipment in the ground station. A small portion of the data is frequency modulated (FM) and is transmitted over a separate hardline link.

In the ground station the data is received by the data acquisition equipment. The FM data is recorded directly on wideband magnetic tape and, normally, no further distribution is made. For the PCM data (which is also recorded on wideband magnetic tape), the Digital Acquisition and Decommutation system (DADS) synchronizes on the incoming serial bit stream, decommutates, addresses each event and analog data word, and prepares these words for multiple parallel transmission. The PCM data follows three paths when it leaves the decommutator.

2.4 DOWNLINK COMPUTER

Selected portions of the decommutated PCM data are accepted by a computer, called the Downlink Computer, for processing. This processing includes comparison of analog functions with predetermined limits and the conversion of these functions into engineering units. A binary word representing the value of the function in engineering units is transferred into the core memory of a Symbol Generator and Storage (SGS) unit. From this memory the data is transferred on demand to any of the alphanumeric cathode ray tube (CRT) displays located on the system control consoles. The data is displayed on the CRT as a decimal number in engineering units in a "page" format. When a parameter has been determined to be out of limits, that specific portion of the CRT display can be caused to blink at a low rate. These displays are utilized by the engineers cognizant of the flight equipment being tested.

2.5 EVENT STORAGE AND DISTRIBUTION UNIT

A second data path transfers the digital data to an Event Storage and Distribution Unit (ESDU). By recognizing addresses, the unit accepts only event words and stores them in registers. Each bit of an event word represents the status of a discrete event. The outputs of the storage registers cause lights to illuminate and event recorders to indicate when the event occurs. The event lights and recorders are located on the system control consoles.

2.6 SYSTEM CONTROL CONSOLES

The third path transfers the binary data to the system control consoles. Each system console accepts appropriately addressed data and stores the data word in registers. Only analog data words are accepted by the system consoles. The outputs of the registers are transormed by digital-to-analog converters, and the analog signals are displayed on meters and analog recorders located on the console.

2.7 SUPPORT

Support systems which form an integral part of the ACE-S/C Ground Station used for Apollo include: (1) a timing system, (2) a personnel intercommunication system, and (3) a closed-circuit television system.

3.0 FUNCTIONAL DESCRIPTION - ACCEPTANCE CHECKOUT EQUIPMENT (ACE) APPLIED TO THE VOYAGER SPACECRAFT

The system considered would consist of a System Test Complex (STC), Spacecraft Vicinity Equipment (SCVE), and the Launch Complex Equipment (LCE). The STC is sized as half of an Apollo ACE station with one computer, one decommutator, standard major modules, and a simplified control room area with eight consoles. These consoles are standard ACE modules assembled to fit the needs of the engineer and test conductor of the Voyager system. The SCVE is that equipment required near the spacecraft during all testing. Its function is to interface the spacecraft with the system test complex. The LCE is the additional checkout equipment required to support the spacecraft at the ESA and the launch complex.

3.1 SYSTEM TEST COMPLEX

The system test complex equipment part of ACE has two functions, command and display.

3.1.1 COMMAND

The command function consists basically of test consoles, the computer, and transmitting equipment. With the command system, test operators may conduct a wide variety of tests from controls on their consoles. The test commands are received and interpreted by the computer which, in turn, sends appropriate instructions to the equipment under test. The command system, thus, initiates testing.

An additional command link consists of the ground command system and the ground radio system (two Voyager-unique systems that would be made functional parts of the System Test Complex equipment). This equipment allows commands to be entered into the spacecraft through the RF link. Commands for this link can originate in the System Test Complex, Launch Complex Equipment, or Space Flight Operations Facility (SFOF).

3.1.2 DISPLAY

Test results are monitored by the Display function which consists of the ground radio system, data acquisition and decommutation equipment (decom), the computer, recording equipment, and display devices located on the test consoles. Signal flow through the Display function begins with the transmission of test data to the decommutator. This data may come via RF through the ground radio system or via hardline from the PCM response system in the Spacecraft Vicinity Equipment. In either case, the data is acquired and decommutated. Unaddressed data is routed to the computer where it is processed for presentation on alphanumeric cathode ray tube (CRT) displays. Addressed data is sent to the Decommutator Distribution/Event Storage Distribution Unit (DD/ESDU) which is patched to output those measurands of interest to the system consoles for presentation on display devices incorporated in the consoles. This addressed data is also routed from the DD/ESDU to the central recording equipment for hard copy recording.

In a typical ACE-Voyager installation, the majority of equipment is located in one area, the System Test Complex area. This area contains the various consoles from which the test engineers initiate spacecraft testing and monitor the test results. The area also consists of the associated computer, data acquisition, and recording equipment.

Figure 3-1 is a functional block diagram of the STC equipment and Spacecraft Vicinity Equipment.

3.2 SPACECRAFT VICINTIY EQUIPMENT

The Spacecraft Vicinity Equipment consists of three subsets of equipment the Digital Test Command Equipment, PCM Response Equipment, and the Simulation Equipment. A brief discussion follows.

3.2.1 DIGITAL TEST COMMAND

The Digital Test Command Equipment is modular in concept and consists of the following:

a. Receiver/Decoder - The Receiver/Decoder receives input signals, checks for transmission errors, generates addresses, and routes information to the baseplates or the Guidance and Navigation unit. It also transmits verification messages to the System Test Complex computer.

b. Guidance and Navigation Module - This module receives parallel data from the Receiver/Decoder for the Spacecraft Guidance and Navigation system. It also checks and stores data until released to the spacecraft Guidance and Navigation system.

c. Baseplate Assembly - The baseplate contains digital circuits for checking addresses and detecting errors. It routes data to the proper baseplate module and transmits message error signals, detected within the baseplate, to the Receiver/Decoder. There are three types of modules mounted on baseplates as listed below:

   1. A digital-to-Analog Convertor Module provides spacecraft stimulus as a positive or negative voltage level.

   2. The Conventional Relay Module energizes the applicable relays in any one of four subgroups within the module in response to input data signal.

   3. The Latching Relay Module is functionally similar to the Conventional Relay Module.

3.2.3 PCM RESPONSE

The PCM Response System contains the signal conditioning, commutation, encoding, and digital multiplexing equipment necessary to convert the spacecraft sensor outputs, hardwire test point outputs, and Guidance and Navigation outputs into a serial PCM data stream for transmission to the System Test Complex.

3.2.3 SIMULATION

The Simulation System consists of equipment to simulate:

a. The spacecraft interface to a spacecraft system.

b. A spacecraft system to the rest of the spacecraft.

c. A sensor output; Sun, Canopus, etc.

d. Spacecraft power system outputs and loads.

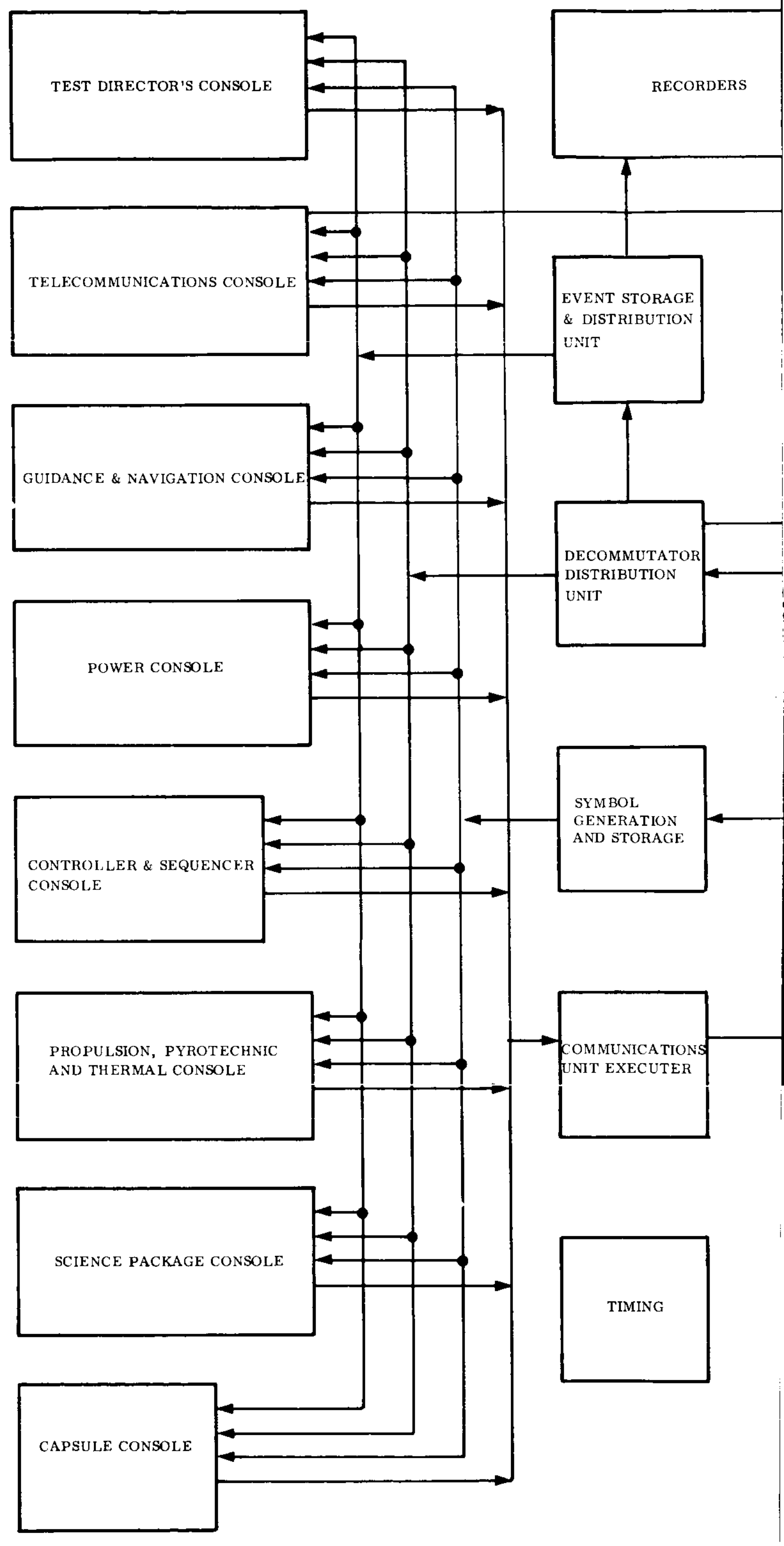
3.2.4 LAUNCH COMPLEX EQUIPMENT

The Launch Complex Equipment consists of equipment unique to the Launch area and the Explosive Safe area. This equipment includes:

a. Propellant loading system.

b. Pyrotechnic installation and test system.

c. Power switching devices (spacecraft vs ground power).


TEST DIRECTOR'S CONSOLE
RECORDERS
TELECOMMUNICATIONS CONSOLE
EVENT STORAGE & DISTRIBUTION UNIT
GUIDANCE & NAVIGATION CONSOLE
DECOMMUTATOR DISTRIBUTION UNIT
POWER CONSOLE
SYMBOL GENERATION AND STORAGE
CONTROLLER & SEQUENCER CONSOLE
PROPULSION, PYROTECHNIC AND THERMAL CONSOLE
COMMUNICATIONS UNIT EXECUTER
SCIENCE PACKAGE CONSOLE
TIMING
CAPSULE CONSOLE

11

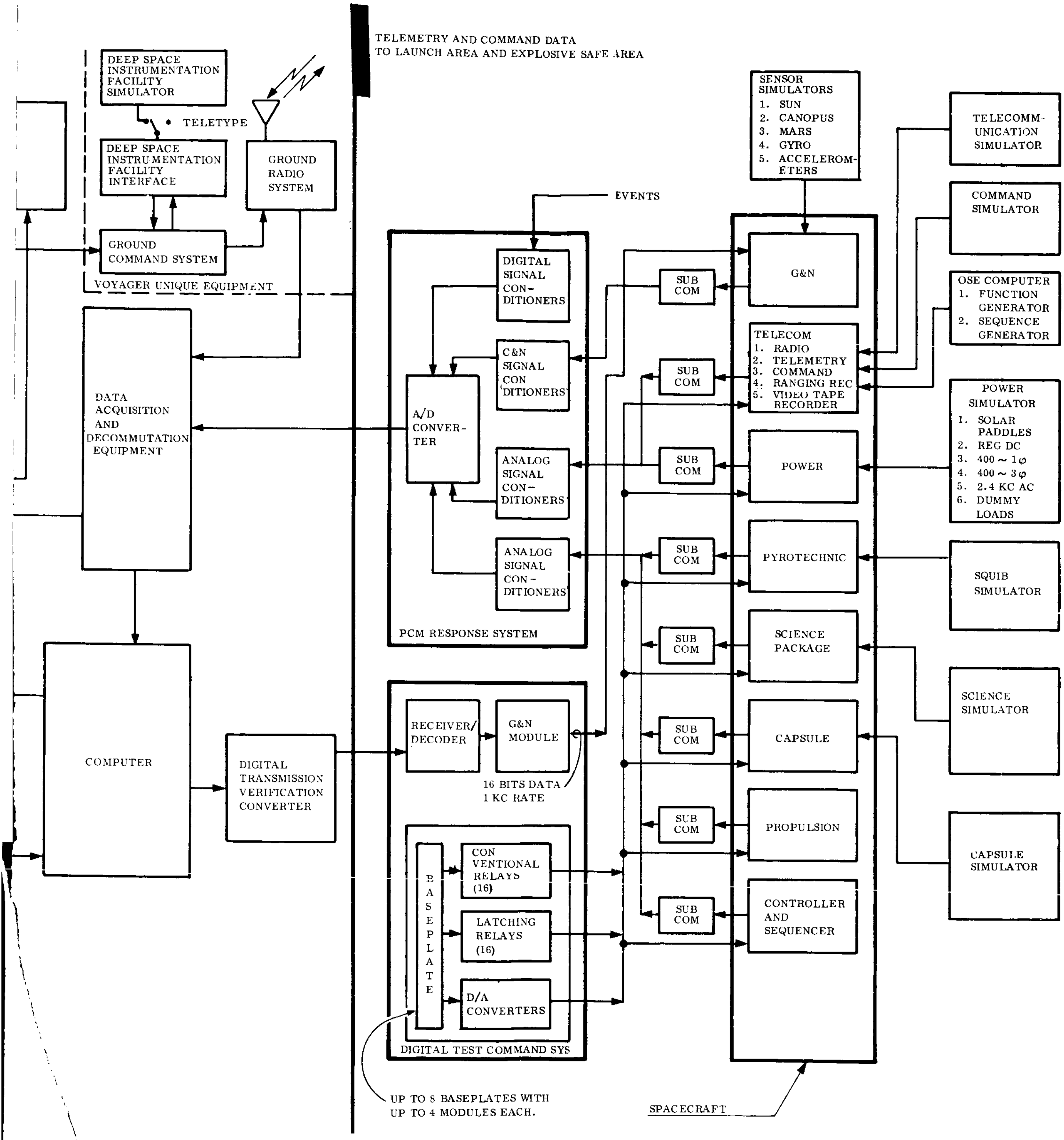


Figure 3-1. System Test Complex and Spacecraft Vicinity Equipment, Functional Block Diagram

2

d. Spacecraft simulator.

e. Communications to the System Test Complex.

Equipments (a) and (b) are not tied into the rest of the ACE system. Equipments (c) through (e) provide functions that are required at the Launch Area or Explosive Safe Area to enable the ACE ground station to implement the launch control and monitoring functions specified for the Voyager LCE.

VOLUME C

Insert attached Introduction to Volume C after Table of Contents

CII-VB260SR102

1. Page 5 of 32, Next to the last paragraph, Line 1. Change Depth or Fault to Depth of Fault

2. Page 6 of 32, Paragraph f., Line 2. Change estimated facilities to minimum facilities

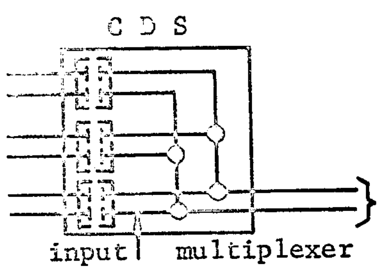
3. Page 7 of 32, Paragraph h, Line 4. Change transmitter commands to transmitted commands

4. Page 8 of 32, Paragraph 3.6, Line 4. Change simulator of the vehicle sources simulation of the vehicle sources,

5. Page 11 of 32, Figure 3.2. Delete approximate scale 1" = 4'

6. Page 14 of 32, Sketch of S/C Ground.

   a. SK should be changed to S/C
   b. CPS should be changed to CDS



Change sketch as shown

Input Signal to A/D

7. Page 17 of 32, Paragraph 5.4h. Delete into a 1000 ohm load

8. Page 18 of 32, Paragraph 5.6a. Delete entirely and add the following: "T" Boxes will be supplied for all spacecraft connectors. They will be used for monitoring, while troubleshooting, to provide access to all signals. The boxes must contain adequate buffering so the possibility of damaging the S/C is non-existent. A re-evaluation of the S/C will be required after using the "T" boxes.

9. Page 25 of 32, Table 5-1. Delete items 22 Helix Current, 26 Helix Current, and 30 Heater - 3V

10. Page 27 of 32, Table 5-1. Telemetry Subsystem. Change Data Storage to Data Storage and Data Encoder.

11. Page 30 of 32, Table 5-2. Controller and Sequencer. Item 8 change clean to clear.

12. Page 30 of 32, Table 5-2. Change item 9 from Recorders to launch mode to 2 fs, and delete item 10.

13. Page 31 of 32. Table 5-2. Data Storage Subsystem. Delete NONE. Add Recorders to launch mode.

CII-VB260FD105

1. Page 2 of 6, Paragraph 3.1, Line 4. Change page printers to character printers.

CII-VB260FD108

1. Page 4 of 7, Table 3-1. Radio Subsystem. Delete item 5 (coax) Precision Link

2. Page 5 of 7, Table 3-1. Controller and Sequencer. Item 8, change clean to clear.

3. Page 5 and 6 of 7, Table 3-1. Data Encoder.

   a. Delete item 9, Recorders to launch mode and replace with 2 fs
   b. Delete items 10 through 30

4. Page 6 of 7, Table 3-1, Data Storage Subsystem. Delete NONE and add the following:

Recorders to Launch Mode
Capsule
Telemetry Mode 3 Select
Telemetry Mode 4 Select
Telemetry Mode 5 Select
UHF Transmitter On/Off
UHF Power Amplifier On/Off
VHF Transmitter On/Off
Sequencer Enable
Sequencer Rate Select
Ground Telemetry Clocks
External/Internal Telemetry
Telemetry On/Off
Tape Recorder On/Off
Tape Recorder Record/Playback Select
Telemetry Bit Rate Output
Telemetry Frame Rate Output
Telemetry Encoder Output
Sensor Power Supply Monitor
Pyrotechnic Continuity
Telemetry Mode Monitor
Tape Recorder Reset

CII-VB260AA001

1. Appendix A - Alternate Approaches. Page A-1. Add VB260AA007 Application of ACE to Voyager

CII-VB262FD101

1. Page 3 of 4, Paragraph 6.1a. Insert word "Three" before standard electronic bays

2. Page 4 of 4, Paragraph 6.2a. Insert word "Six" before standard electronic bays

CII-VB263FD101

1. Page 23 of 31, Table 4-6, Item 18. Change Pow Amp No. 2 to Pow Amp No. 3

CII-VB266FD101

1. Page 11 of 17, Table 4-1. Add the following items:

a.	64. Stimulate fault sensor	Main Regulator
b.	65. Stimulate fault sensor	2.4 KC inverter
c.	66. Stimulate fault sensor	400 cps inverter

CII-VB270FD101

1. Page 5 of 7, Paragraph 6.1b, Second Line. Delete ordnance lunette and insert trailer hitch

CII-VB270VD103

1. Page 2 of 6, Paragraph 2.0, Delete "Table 2.3a" after VB220FD113.

CII-VB2[illegible]D106

1. Page 2 of 4, Paragraph 2.0, Delete "Table 2.3a" after VB220FD113

CII-VB280SR102

1. Page 6 of 12, Table 3-1. Controller and Sequencer, Item 8. Change "Clean" to "clear"

2. Page 6 of 12, Table 3-1. Data Encoder, Item 9. Delete "Recorders to Launch Mode" and insert 2 f_s. Delete item 10.

3. Page 6 of 12, Table 3-1. Data Storage Subsystem. Delete "None" and insert "Recorders to Launch Mode".

VOLUME C (continued)

CII-VB263FD106

1. Page 9 of 44, Figure 3-3. Format grouping should be as follows:

    Word 1 through 7 — Preamble Sync (49 bits)
    Word 8 — Data Type Word (7 bits)
    Word 9 through N — Data
    Word N is the last word in the vertical column

2. Page 22 of 44, Fourth Paragraph. Beginning with "The SF". Add the following sentence: "FS search is inhibited during Flare and Scan data or Data Type F and C, respectively."

3. Page 22 of 44, Paragraph 3(a). Change "   - out-of-limit indication" to read "   or out-of-limit indications".

4. Page 31 of 44, Figure 3-21A. Change "Flag word bit 0" to read "   or out-of-limit."

5. Page 33 of 44. Second Line. Change "(R   data rate)" to read (where R is data rate).

6. Page 37 of 44. Table 4-2, Item 8. Change "$2f_s$" to read "$2f_s$ Clock Signal".

CII-VB280FD101

1. Page 3 of 4, Figure 3-1. Change the block reading "Data Encoder Command and Control Unit" to read "Data Encoder OSE Command and Control Unit".

CII-VB600VP

1. Page 4 of 11, Paragraph 4.1.a. Change to read as follows: "At least one STC will be required for each of the flight and PTM Spacecraft."

VOLUME C

INTRODUCTION

This volume contains ~~functional~~ descriptions of the Operational Support Equipment recommended by the General Electric Spacecraft Department for the support of the 1971 Voyager Spacecraft. It is submitted as a part of the Phase IA study report, in accordance with the requirements of Contract JPL (95 1112).

In addition to these OSE functional descriptions, the report contains sections on "Test Objectives and Design Criteria" and on "Design Characteristics and Restraints" which apply to the System Test Complex; Launch Complex Equipment; Mission Dependent Equipment; and Assembly, Handling and Shipping Equipment; respectively. These have been included for the purpose of achieving clarity in the description of the concepts and approaches recommended for these four kinds of Voyager OSE.

The basic approach to testing of the Voyager Spacecraft, and the design philosophy adopted for the preliminary design of the STC, is given in the Appendix. The Appendix also contains brief discussions of concepts which were considered as alternates, during the Phase IA OSE study. Reading the appendix first is recommended to those wishing to quickly relate the selected design approach to the overall test approach for the System Test Complex. The sequence and description of the tests and procedures are more fully described in "Assembly and Checkout" (CII VB110VP005) and "Launch Operations" (CII VB110VP006) both in Volume A, Book 4A.

The guidelines used in selecting an approach and the recommended OSE design concept were as follows:

a. The utilization of the design, design approach and equipment which has been proven in the Mariner program. The functional descriptions of the OSE indicate, accordingly, the degree of applicability of the Mariner design or of the Mariner hardware itself, to the Voyager program.

b. The allocation of control of testing operations to the cognizant subsystem engineer. Accordingly, the Voyager OSE preliminary design reflects an OSE mechanization concept which gives the cognizant subsystem engineer the equipment features he needs to control testing of his subsystem.

In producing the functional descriptions of 1971 Voyager OSE, contained in this volume, two decisions were applied. The first of these decisions was concentration on functions performed by OSE in the field incident to launch preparation. The functions of OSE in supporting factory acceptance or development testing were deemphasized in writing the descriptions because the support at that phase of the program is developmental rather than operational. Equipments which support development and fabrication activities only have been substantially excluded from this volume of functional descriptions.

The second of these decisions was the integration of functional descriptions of common, often general purpose OSE, into generic categories. This decision to eliminate details of equipment, the functions of which are apparently not unique, was adapted to clarify those OSE functions which are uniquely required.

The basic aspects of the plan by which General Electric would implement the OSE portions of the 1971 Voyager program, are also contained in this volume.