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MISSILE AND SPACE DIVISION

VOYAGER SPACECRAFT SYSTEM

PHASE IA TASK B PRELIMINARY DESIGN

OSE FUNCTIONAL DESCRIPTION

VOLUME B

VOYAGER

GENERAL ELECTRIC

NASA CR71499

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VOLUME B

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CALIFORNIA INSTITUTE OF TECHNOLOGY

JET PROPULSION LABORATORY

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*This information provides a cross reference between Task A and Task B CII's to facilitate the location of similar information in the two reports.

CII VC260SR101

OPERATIONAL SUPPORT EQUIPMENT OBJECTIVES AND DESIGN CRITERIA

SECTION

OSE OBJECTIVES AND DESIGN CRITERIA

1.0 CONTENTS. This section presents functional descriptions of the Operational Support Equipment (OSE) recommended for the revised 1971 VOYAGER Spacecraft. Discussions of objectives and design criteria and of design characteristics and restraints are also presented.

The OSE recommended for the new spacecraft design has not, in general, been revised radically insofar as design philosophy is concerned. The differences between the OSE now recommended and that recommended in the Task A report are principally of function and implementation rather than approach.

One major change in approach, however, is the concept of Launch Complex Equipment (LCE). Since the launch area usage and design does not provide anything equivalent to a "block-house", the recommended LCE concept described here centers about remoting the LCE to the System Test Complex (STC). The confidence testing of the launch-ready Spacecraft, and the launch control is to be accomplished using unmanned LCE that is remotely controlled and monitored from the STC by means of data links.

The recommended OSE has changed in the degree to which flight counterparts have changed. For example, the recommended Assembly, Handling, and Shipping Equipment (AHSE), although functionally unchanged, would be different in most respects in order to accommodate a different Spacecraft, similarly the recommended subsystem OSE would also reflect the differences between the previous and current recommendations.

The depth of detail in the following OSE functional descriptions has been determined, in part, by the degree of change from recommendations made for Task A. All significant OSE has been described to some degree, however, regardless of changes in configuration and function.

To achieve brevity in this section and still preserve its scope, an abbreviated format has been used.

2.0 USE OF OSE. The OSE is required to provide the capability to test the subsystems at the subsystem's production facility. It is also to be integrated with other subsystem test sets and system test peculiar equipment into an STC to perform essentially identical subsystem tests, as well as a variety of coordinated systems tests. These tests are to be performed, with the subsystem or Spacecraft under ambient conditions, under environmental stresses, with variable input and output characteristics, and with significant internal parameter variations. This will be done in order to obtain assurance that the Spacecraft is ready to proceed to the next phase, to demonstrate its capability to withstand the mission environment, and to the degree possible, the mission duration. Identical OSE will be used during the vehicle type approval (TA) and flight acceptance (FA) tests. The STC, through data and control links to the LCE, will provide the test control and monitoring capability to perform pre-launch control and testing of the Spacecraft.

3.0 TEST PHILOSOPHY. The following summarizes the philosophy used to identify the functions to be performed by the OSE.

a. The primary purpose of the FA test is to verify that the Spacecraft was built to drawings and that the design, manufacturing processes, and materials used resulted in a vehicle with an adequate margin to withstand the mission environment and life requirements.

b. A secondary purpose is to verify that the flight units are similar enough to the Proof Test Model (PTM) and subsystem development units to allow correlating the results of those tests with the results of the FA Tests.

c. Due to the nature of the VOYAGER Program, the following aspects are to be emphasized.

   1. Spacecrafts are to be protected against damage induced by the OSE, their personnel, or operating procedures.

   2. A low-risk approach to the OSE design should be taken to ensure that it is ready and verified on time.

   3. The OSE cannot be economically designed to be more reliable than the Spacecraft. It must therefore contain a high degree of self-test (preferably on line but certainly off line) capability to indicate when it is malfunctioning.

   4. It is desirable to identify how good a vehicle is-as opposed to identifying that it does or does not meet the specification.

   5. In order to evaluate adequate margins of operation, overtesting (within reason) is to be preferred to undertesting.

   6. Test operations will be performed by highly knowledgeable and highly motivated engineers.

   7. Exploratory tests on flight vehicles will be required to a limited degree.

d. The systems tests are performed following installation of the subsystems into the vehicle and their preliminary installation tests. After these tests, no flight cabling will be broken until a fault has been detected, and then a preferred disconnection/troubleshooting scheme will be used to minimize the breaking of connections.

e. Fault isolation will be done to the maintenance level for that test location. Troubleshooting techniques can be used to isolate to that level as required. The maintenance level for the various areas is:

   1. Subsystem tests - subassembly
   2. System test - bays
   3. Pad - Spacecraft

f. Mission Dependent Equipment (MDE) including hardware and software used at the Deep Space Instrumentation Facility (DSIF) will be a part of the STC.

g. Redundancy is not a reason for reducing the level of testing. Provisions must be made in the vehicle design to enable the OSE to detect the operation of each path, to stimulate fault detecting circuits (that switch in backup circuitry) in such a manner that the transfer occurs, and to be able to operate or monitor the majority logic circuits in such a manner as to verify that decisions are being made unanimously and can be made by any majority configuration.

h. The degree to which the Spacecraft is made cooperative to depth of tests will be determined on the basis of total system tradeoffs. The result must be that the confidence that the vehicle will perform its mission has been increased by increasing (or decreasing) the depth of tests by providing or not providing extra test capability (i.e., additional capability to accommodate ground tests) into the vehicle's design. The general philosophy will be that end-to-end testing will be performed.

i. Advantage will be taken of the Command and TLM capabilities of the vehicle to reduce the number of hardline control and monitor leads. The only reason for duplicating with hardline leads will be inadequate availability, or time to execute via the Command links, or the inaccuracy or slowness of the TLM.

j. Most of the tests will be performed by using simulated solar panel power and simulated battery power. However, a free mode test will be run in which no OSE connections will be made to the vehicle and the Sun used to supply primary power in conjunction with installed batteries.

k. Two Planetary Vehicles will be launched by one Booster. In order to meet the schedule, the vehicles (plus one backup) will have to be processed almost side by side.

l. Where automatic capability exists, this should be backed up by identical manual capability.

m. All flight connections are to be verified by carrying full load after final mating wherever this is possible. (The primary areas where this is not possible are the squib connectors.)

n. Systems tests are to be able to be performed from a central source or locally from the Subsystem OSE. Suitable controls must be provided for the operator to assume control.

4.0 TEST FLOW. The TA and FA test requirements are given in Section 4.0 of JPL's Preliminary General Specification for the VOYAGER 1971 Spacecraft System. The following summarizes those that must be supported by the OSE:

4.1 Subsystem Tests

a. All functions shall be verified with ±15% variation in external supply voltages over a temperature range of 40°C (TA), 20°C (FA) above and 30°C (TA), 20°C (FA) below the envelope of the 95th percentile high- and low-predicted assembly flight temperature.

b. All functions shall be verified with nominal voltages and temperatures 15% in excess of those above.

c. Noncatastrophic performance in the presence of noise injection, power frequency variation, power overshoot, and power transients shall be 15% in excess of those specified in the appropriate detail specification.

d. FA vibration tests shall be conducted at the envelope of the 95th percentile of the predicted flight environment for 1.5 times the predicted total time duration where the wide-band vibration level is within 10 db of maximum or 60 seconds, whichever is greater.

e. TA vibration shall be 5 db above the FA level and shall be three times the FA duration.

f. Thermal vacuum

g. Magnetic evaluation for each operational mode in an ambient magnetic field of less than 100 gamma.

h. TA tests shall also include handling tests, ground environment tests, and launch environment tests.

4.2 System Tests. The following major tests are performed using the STC during TA (PTM) and FA tests at the vehicle assembly plant.

a. STC verification.

b. Ground integrity tests to verify that no unknown ground loops exist in the vehicle or when the vehicle is connected to the STC.

c. Initial power application.

d. Subsystem tests to evaluate each subsystem while operating with on-board power.

e. Intersubsystem tests performed on interfacing subsystems in pairs to verify all interfaces in such a manner as to minimize damage caused by improper interfacing.

f. Telemetry channel calibration by making end-to-end checks.

g. System test to establish functional integrity to the degree possible.

h. Parameter variation test in which selected system parameters are varied.

i. Magnetic evaluation.

j. Space simulator tests.

k. Vibration tests of acoustic and vibration equal for FA and in excess for TA to those levels expected during boost or orbit insertion.

l. Simulated countdown tests with LCE.

m. Spacecraft-MDE interface verification tests.

n. Free mode test in which system is run by using solar power and no OSE test connections.

o. Electromagnetic Interference (EMI) tests to determine susceptibility and contributions to the RF environment.

p. TA failure mode and logic tests.

4.3 FA Operations at the Eastern Test Range (ETR). The following tests will be performed at the ETR and supported by the STC or LCE as indicated.

a. Subsystem tests using STC.

b. Dummy countdown using STC/LCE.

c. Cooling tests in launch configuration.

d. Leak tests.

e. Magnetic mapping.

f. Spacecraft - capsule tests.

g. Spacecraft - Deep Space Network (DSN).

h. Spacecraft - mission operations system test.

i. Spacecraft - Launch Vehicle system test including J-FACT and simulated launch.

Since the TA and FA tests are to be performed by identical equipment, that equipment must be designed to provide support for the worst case of the tests that are common.

5.0 TEST OBJECTIVES AND APPROACH. The primary test objectives (not in order of importance) obtained from the above are:

a. Verify operation of all elements of the system.

b. Determine how good the vehicle is by obtaining operating points of significant points under ambient and significant environment extremes.

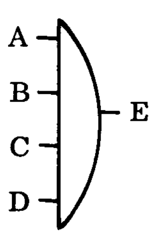
c. Attempt to demonstrate the capability to withstand the mission environment and duration.

d. Demonstrate compatibility of the vehicle with the entire system in all of its modes.

These test objectives can be achieved by performing two types of testing, e.g., performance testing and mission profile testing.

5.1 Performance Test. This test is an idealized sequence to obtain (a), (b), and (c). Objective (a) (verify operation of all elements of the system) is obtained by having the test operate all elements of the system in a manner best suited to detect malfunctions. This may lead to operating the vehicle in a manner that could never occur in flight (i.e., in the Attitude Control Subsystem tests, rate error signals should be able to exist with constant position error signals). This is to be encouraged if it is the best way for finding faults and does not impose restrictions on design of the subsystems.

An example of the type of test to find faults 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 4 of the 15 "zero" states must also be verified. These are:

$\bar{E} = \bar{A} \cdot B \cdot C \cdot D$

$\bar{E} = A \cdot \bar{B} \cdot C \cdot D$

$\bar{E} = A \cdot B \cdot \bar{C} \cdot D$

$\bar{E} = A \cdot B \cdot C \cdot \bar{D}$

The performance test must guarantee that those four "zero" states and the one "one" state are verified, any other combinations are 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 equipment and may require special sequences to perform the test.

Objective (b) (determine how good the vehicle is) is generally done by obtaining good measurements of the significant operating points and observing their margins in excess of those specified. In many cases this measurement is obtained directly by operating the system. Examples of this type of data are the noncoherent frequency of the transmitter-threshold amplifier input signal to cause turn on and turn off, etc. Other measurements (primarily those that are observed as a discrete change from one state to another) may have to be obtained by operating the system beyond specified limits until a misoperation is detected. Examples of this type of test are phase error rate at which lockup does not occur and command data modulation variation such that data is incorrectly detected. The test required to obtain this type of data is characterized by the statement that the test is to find the limits of operation rather than to verify that it operates within limits. All of these tests are to be performed with marginal power supplies, worst case input/output, under environmental stress, etc., in order to establish the system's sensitivity to those conditions and the "as built" margin of safety.

Objective (c) (demonstrate the capability to withstand the mission environment and duration) is accomplished largely by using the data obtained in (b) to obtain a trend of the measured parameters as a function of life and/or environment and this is compared with similar data obtained during development and TA tests. This trend, if excessive or unexplained, can be reason for rejection or for extending the testing. This objective is further enhanced by requiring the system or subsystem to operate under worse conditions (interface and environmental) than it will have to operate during the mission. The justification for this overstressing is basically that if a subsystem can operate with an out-of-specification (within reason) interface at the start of its life, it will probably be able to operate with an in-specification interface for a longer time than a subsystem which could not operate originally with the marginal signals.

A major characteristic of the performance test is that of operating the system in an abnormal way (i.e., marginal-to-out-of-specification interfaces, significant parameter variations, peculiar sequences required to uncover faults or marginal operations). As long as these tests can be run end-to-end, there is no problem, but to apply this requirement 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. While the subsystem input signal can be controlled by the stimulation, the subsystem takes this signal, in many cases, and reshapes and amplifies it so that it becomes a nominal signal and thus, the elements that follow cannot be tested with marginal signals using an end-to-end test philosophy. This limitation can only be overcome by making provisions for inserting marginal signals into these significant points. This generally entails spacecraft circuitry to allow this to be accomplished which detracts from the desirability of this procedure. These tradeoffs of "more elaborate tests versus vehicle requirements to accommodate these tests" are subjects of many studies. The major studies are discussed in Section 5.3.

5.2 Mission Profile Tests. While objective (d) (demonstrate compatibility of the vehicle with the entire system in all of its modes) is theoretically accomplished by the performance test, experience has shown that complex systems can have unexpected interactions between components, subsystems and assemblies. Because they are unexpected, special tests must be designed to allow the occurrence of the interaction, and special provisions must be made to detect them. It has been found that running mission profile test sequences (enough to verify all normal and backup modes in a reasonable number of combinations) places the system in the meaningful states under which this interaction may take place. The inputs during these tests are more nominal and are sequenced on a basis of expected mission sequences, as well as the backup modes. This test sequence is identical to the Dynamic Mission Equivalent (DME) tests described in the Reliability Plan Section.

An example of the difference between a mission profile test and a performance test is that during a mission profile test maneuver sequence, the solar power voltage output would be reduced and power eventually transferred to battery power (battery either high or low as desired) and the remaining tests carried out using batteries or simulated batteries which would be programmed to provide an expected power versus time profile. A performance test would check the maneuver functions with a group of calculated worst-case power profiles one of which may be similar to the above but is not limited to that. While the performance test sounds like a better test, a significant phrase above is "calculated worst case." The mission profile test will uncover those faults overlooked by the performance test. (When this occurs those tests should then be incorporated in the performance test.) The mission profile test also improves greatly the confidence of the personnel that a mission can in fact be executed.

The effect of mission profile tests on the OSE is primarily in the area of monitoring (stimulation/control signal characteristics for a performance test are more extensive than those required for a mission profile test but do require to be able to be sequenced as appropriate). Since the purpose of a mission profile test is largely to find unexpected interaction, the OSE must be able to detect whether interaction did or did not occur, whereas the performance test is used to check for only expected interaction. In order to detect the unexpected implies that the OSE must look at all of the data all of the time or enough of the time to detect potentially fault-inducing transients. Since this can never be done completely, the OSE and test sequences must be so designed to at least uncover unexpected interactions that cause improper output operations, mode changes, transient tripping of devices, etc.

It should be noted that, since the mission profile and spacecraft operating modes are known early, the mission profile test equipment requirements can generally be established early in the design stages. The performance test, however, requires detail vehicle circuit designs to be available in order to establish the test equipment requirements. This delay cannot be tolerated and therefore requires the performance test capabilities to be designed based on preliminary designs. In order to accommodate changes required by design changes, test experience, etc., the OSE design should include additional capability—primarily test sequence flexibility, variation of identified parameters beyond those foreseen, and capability to handle extra data points in a flexible manner.

While the above may give the impression that the two types of tests are separate and distinct, it must be recognized that the above discussion is idealized and that each test can incorporate many aspects of the other. This combination is mainly a matter of incorporating the desired sequence of controls/commands into the test and providing adequate monitoring. It should also be realized that reduced levels of testing can be accomplished at any test location primarily by the elimination of steps.

5.3 Spacecraft Test Limitations. The degree to which the vehicle can be tested without providing additional capabilities and whether this level is adequate or not is virtually impossible to define with the present level of design. This is also true for justification of providing additional capability in one subsystem to make other subsystems easier to test. While establishing the monitoring points is a problem, the major problem involves those areas where it is desirable to insert OSE generated signals in place of the vehicle signals. This generally involves switching out the vehicle signals and switching in the OSE signals. While this can be done reliably by use of redundant switching devices, there is always a penalty to be paid of at least weight if not a slight reduction in reliability. It is this penalty that must be evaluated against the gain to the system of tests in greater depth and detail. Various requirements/capabilities have been investigated and many preliminary decisions have been made at this time (the primary ones outlined below). In all cases they should be reviewed as a better understanding of the design becomes known.

5.3.1 TLM Homing. TLM Homing feature allows the OSE to advance the TLM commutator rapidly (OSE controlled) to a particular point, make continuous measurements (at the TLM conversion rate) until the OSE advances to another point or releases the homing control. This feature would allow the OSE to make maximum use of the TLM as a source of data and could significantly reduce the number of hardwire test points.

This reduction of test points is gained at the expense of additional circuitry to at least incorporating an "and/or" circuit in the TLM commutator control circuits. Since this is a most critical circuit and since the actual gain to the system (elimination of hardwire test points) cannot be measured at this time, the homing feature was not incorporated.

Essentially the same advantages obtained from homing can be obtained by speeding the TLM system up by a factor of 8 or 16. This also requires additional circuitry in the countdown chain and was therefore not incorporated.

5.3.2 Hardwire Command Loading. Since it is required to do limited testing at the pad during periods of silence and since it is felt that this limited testing should include the command loops, it is required that commands be loaded via hardline. There are three locations in which this command data can be inserted. These are: (a) at the input to the command detectors, (b) at the output of the command detectors, or (c) at the input to the Controller and Sequencer (C&S). The first location has the advantage of using all of the command subsystem while the last location has the advantage in that the C&S memory could be loaded at a high rate without causing any design perturbations on the command subsystem, however, this makes executions of the real-time commands impossible. The middle location is a compromise location in that it is felt that it could be implemented such as to allow high speed loading (if required) of the C&S memory. It also allows real-time commands to be executed, but it does not test the bit detection circuitry. The tradeoff as to the location of the hardwire command input could not be made, since it requires greater design detail of the Command and C&S subsystems.

5.3.3 Subsystem Power Supplies. In order to operate with significant internal parameter variations to establish as close to flight as possible that the subsystems have an adequate margin of operation (i.e., will be able to operate near the end of mission), a test with relatively wide limits of power supply outputs is desirable. By varying the power supply outputs, a major effect on the subsystem is caused by the control of a few leads. This, however, requires inserting ground power supplies in place of the subsystem TR's and running tests with various output levels under environmental extremes. The preferred method would be to provide relay switching in the vehicle (several in parallel to satisfy reliability rather than physically disconnecting flight connectors, etc.), to switch in OSE power supplies that could be adjusted in any desired manner.

It was decided not to incorporate this feature such that the tests could be run in a system configuration but this test is to be run on components, subassemblies, and possibly certain vehicle bays prior to being installed into the system.

5.3.4 Gyro Signals. Due to the size of the vehicle, it was decided not to test the Guidance and Control (G&C) subsystem by maneuvering the entire vehicle. Therefore, in order to insert error or control signals into the gyros and hence into the system, ground torquing signal capability is required and provided.

Tests to evaluate gyro motor and bearing performance have been attempted on other programs by measuring time and input power to reach synchronous speed during startup and by measuring time to coast down to some low speed following shut down. These require the incorporation of speed detectors within the gyro of some order of magnitude. These have not been included; however, the capability to measure the gyro motor current has been tentatively identified as being available. This will be used to obtain the start up current versus time characteristic, as well as steady-state currents. By having the measurement test point circuits on the motor side of the power contact, the self-generated emf can be monitored during shutdown and period measurements made periodically will produce the coast speed versus time characteristics.

CII VC260SR102

DESIGN CHARACTERISTICS AND RESTRAINTS

SECTION

DESIGN CHARACTERISTICS AND RESTRAINTS

1.0 SUBSYSTEM TEST CONFIGURATION (See Figure 1-1). The Operational Support Equipment (OSE) consoles will be used at the subsystem manufacturers' plant and again at the spacecraft assembly facility to test an independent subsystem or major section (e.g., the Guidance and Control subsystem may not have sensors available, or a subsystem may consist of several bays). The primary function of these tests is to demonstrate that the equipment is built according to drawings and that it has been designed and built with enough margin to be capable of operating with any in-specification interface.

1.1 Interface Criteria. The subsystem (or major section) is to be operated with, in effect, a spacecraft electrical simulator as seen by that subsystem's circuitry, or mechanical loading/position monitoring, etc., for that subsystem's mechanical drives or sensor stimulation for its sensors. The basic criteria for that simulation is to provide signals and loads that are required to test the subsystem and which need not represent the operational interface. This simulator (electrical, mechanical, physical, etc.) is a part of the subsystem OSE. The simulator must simulate marginal as well as nominal interface characteristics and must include at least the following capabilities:

a. Vehicle 2.4-kc power supply with varying amplitude, frequency, waveform, transients, etc., with ±15% variation in excess of limits specified for the subsystem.

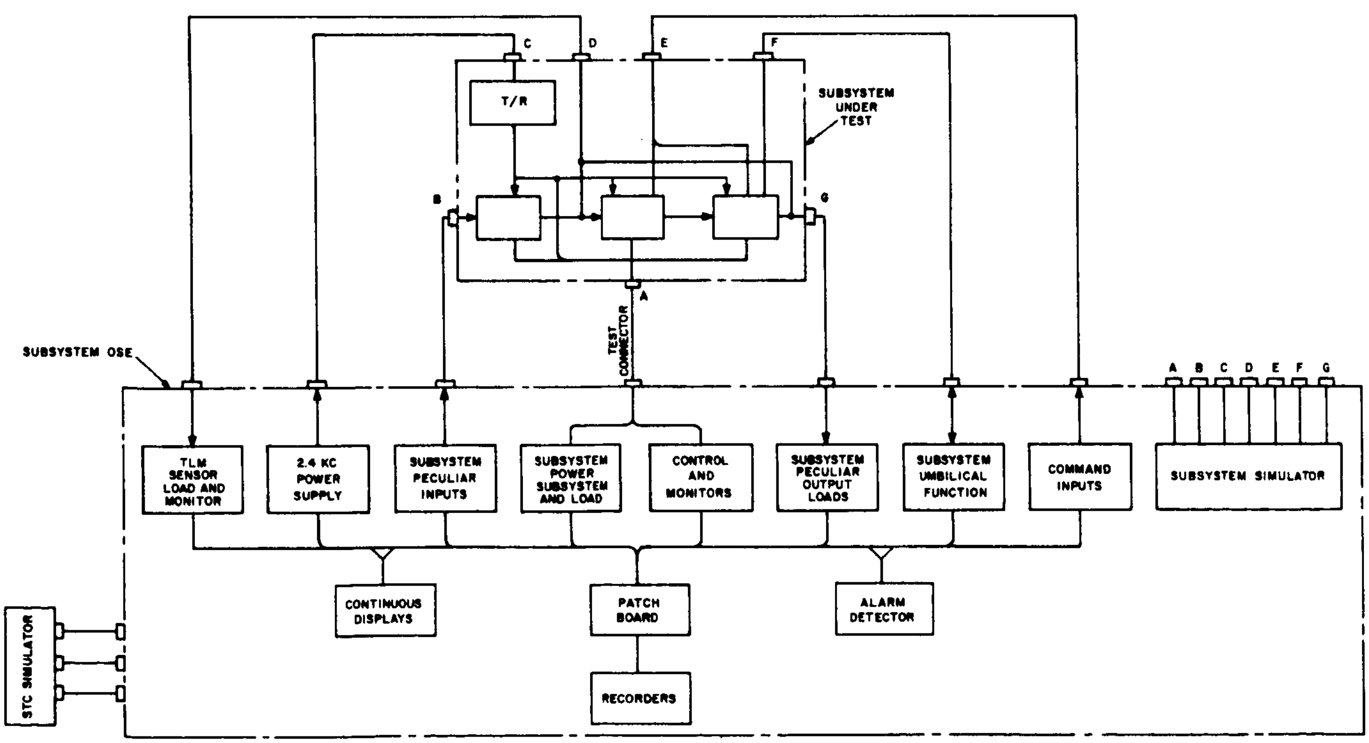


Figure 1-1 Typical Subsystem Test Configuration

b. Command subsystem inputs of discretes or quantitative data signal characteristics varied in amplitude, pulse widths, rise/fall time, etc., until misdetection by the subsystem under test is obtained. (Note: Normal control of mode, etc., of the subsystem should be accomplished by using the command subsystem interface with nominal signal characteristics.)

c. Output loading of the electrical signals should be accomplished with variable loads ranging from nominal to ± 15% of limits specified.

d. Output loading of mechanical drives should be loaded with friction inertia, force, back pressure, etc., to ± 15% of limits as specified.

e. Telemetry (TLM) sensor outputs should be capable of having their calibration checked without removal or disconnecting from the subsystem. (Suitable test points and stimulation capability must be provided for this purpose.) The effect of various loads (15% in excess of specified limits) on the output should be obtained. (Note: It is not necessary to duplicate the pulsing multiplexer loading effect.) After calibration, the TLM sensors should be used as a source of data.

f. Subsystem peculiar inputs (RF for radio-digital for Controller and Sequencer (C & S), physical stimulation, or sensor simulation if sensor is not present for the guidance and control, etc.) should be varied in all significant parameters and modes to establish the level of misoperation by the receiving subsystem.

g. The umbilical connections (those controls and hardwire monitors required during launch countdown) and test connections (those special points required to augment the umbilical, TLM and command for subsystem and system test) will also be available and should be verified with worst-case interface characteristics.

1.2 Subsystem Test. The tests performed on the subsystem are essentially performance tests. While each subsystem will have its own peculiarities, the following is typical of the type of sequence that would be run.

a. OSE verification test (using subsystem simulator).

b. Ground verification.

c. Initial power application, measure input power, measure temperature rises.

d. TLM/hardwire sensor verification.

e. Command control verification.

f. Nominal parameter test; record the test results, run all modes verifying that all aspects of all commands are executed, verify TLM calibration, verify all interchange of data or control signals, and obtain measurements of all significant parameters, power required as a function of mode, etc.

g. Marginal parameter tests with worst-case input signals and output loads over temperature extremes (starting at nominal). Test sequence as in step f.

h. Repeat of step g for the other environmental extremes discussed in the "OSE Objectives and Design Criteria" (VC260SR101).

Note: Since a subsystem may consist of several bays, steps a through f should be able to be performed on each bay alone (i.e., OSE must supply bay interface simulation). Tests a through h would be done on a complete subsystem.

2.0 SYSTEM TEST COMPLEX

2.1 Studies. During the Phase IA Task A study, a tradeoff study was made concerning the System Test Complex (STC) configuration. The following three configurations were studied:

a. Mariner C approach.

b. Mariner C approach modified such that the Computer Data System (CDS) could provide control functions to the test equipment and essentially control the test from this central source as well as monitor the tests as done during Mariner C.

c. Acceptance Checkout Equipment-Spacecraft (ACE/SC) approach being developed for the Apollo Program.

The conclusion reached during the Phase IA Study was that approach b would be the basic approach; however, many aspects of the other approaches would also be used. A simplified block diagram showing data flow paths for the recommended approach is shown in Figure 2-1. The major blocks and the data paths are discussed below.

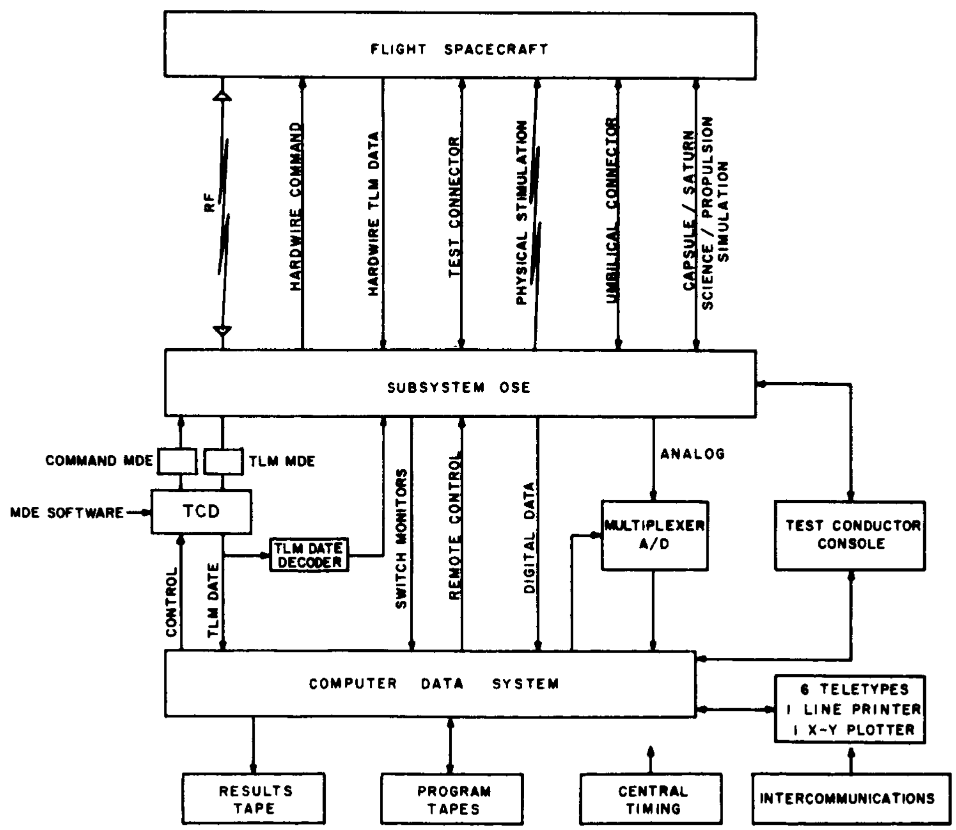


Figure 2-1 First Level STC Functional Schematic

2.2 Computer Data System. System Tests will be accomplished by coordinating the operation of the various subsystem OSE such that precise preplanned sequences of stimulation are executed and the data to evaluate those sequences are collected, displayed, recorded, and analyzed. While all of these functions can be (and the capability must exist such that they can be) performed manually, the approach is that the Computer Data System (CDS) should also provide the central sequencing control, data gathering, analyzing, display generating, and recording functions. The CDS's prime data links (all via the subsystem OSE, where the data is also displayed and recorded) are:

a. TLM data (via hardwire and RF).
b. Analog data from vehicle test points and subsystems OSE generated signals.
c. Discrete data (switch operation from the subsystem OSE).
d. Digital data (vehicle time-station time-digital data, etc.).

The prime control links of the computer (all via the OSE subsystem) are:

a. Vehicle commands via Telemetry and Command Data handling subsystem/radio.
b. Vehicle commands via command subsystem OSE hardwire.
c. Remote control of the subsystem OSE circuitry.

The prime display links of the computer are:

a. Teletypewriters at test stations.
b. Line printer to Test Conductors.
c. Alpha numeric display to the Test Conductor.
d. Status lights to the subsystem OSE consoles.

The implementation of the remote control of the subsystem OSE by the computer will be accomplished in basically two ways, depending on the individual subsystem requirements. These are:

a. Provide means for the computer to operate the subsystem OSE essentially in parallel with the operator's manual controls. This will require the subsystem OSE to provide logic for this remote control and the associated self-check provisions and switch over capability from local to remote control selection.

b. Provide start-stop-type signals to the subsystem OSE to control a manually setup condition or initiate an internally (within the subsystem OSE) generated sequence.

It should be noted that all functions will not be "computerized." A conservative approach is to be taken and no attempt will be made to extend the state of the art for "automatic" control for use in analyzing complex data wave shapes, etc. Where the signals must be controlled and/or analyzed manually, the test sequence will make adequate allowance (i.e., wait for manual go-ahead signals, etc.) to allow the function to be accomplished. It should be noted that the approach is to remotely control a "manual" test sequence. The tests should be so designed such that they can be performed manually and not be dependent upon the Computer Data System.

The STC will also provide other equipment required to support the coordination of the Subsystem OSE into a System Tests Complex. This will include:

a. Primary power (60 Hz) control and distribution.
b. Central Timing.
c. Central recording (for time correlation of analog signals between subsystems). Each subsystem will provide its own recorders to meet its requirements.
d. Intercommunications.
e. Interconnecting cabling.
f. Interface simulators.
g. Vehicle cooling.
h. Test conductor's console.
i. Raw data recording.

2.3 Subsystem Operational Support Equipment. The Subsystem OSE (identical to that discussed in Section 1.0) is used to reperform the subsystem tests as they are introduced into the system. This introduction process will include the following types of tests:

a. Ground loop check.
b. Subsystem test mounted in vehicle.
c. Subsystem test using vehicle power supply.
d. Subsystem test using other vehicle subsystems.

Note that the Subsystem tests are as identical as possible with those performed on the subsystem alone (see Section 1.2).

When the subsystem is connected into the system, its primary interface connections are lost as far as the OSE is concerned. This will reduce the degree of parameter variations that can be introduced directly into the vehicle to the following:

a. Capsule interface.
b. Propulsion subsystem interface.
c. Solar panel power profile (using ground power or solar panel simulators).
d. Battery power profile (when batteries are simulated).
e. Umbilical signals.
f. Saturn interface.
g. Subsystem peculiar stimulation (RF signals - sensor stimulation, etc.).
h. Mechanical loads for drives.
i. Subsystem test points.

Note that items a to d assume that those parts may not be introduced into the system until later and may be removed for shipment to the field.

2.3.1 TLM Data. The subsystem engineers have also lost (in the system configuration) access to their subsystem TLM sensors that had been continuously monitored in the subsystem configuration. In order to compensate for this loss and present this data in a convenient manner, the STC will provide the capability to take the decommutated TLM data words and reconvert them to their original form so as to be able to drive the display on the subsystem OSE as during subsystem tests. By providing this capability, the subsystem engineer will have presented the same type of data in the same form as he had in the subsystem tests. By having this data conveniently presented, the number of hardwire monitors desired should be reduced. The use of TLM monitors in this manner is adequate when:

a. The OSE has an equivalent signal (i.e., accuracy and time availability of the TLM are not critical).

b. Subsequent testing will verfiy the TLM signals as being correct.

c. The TLM signal is monitoring values that can be compared directly to known conditions.

d. Noncritical signals are received.

Note that the TLM subsystem is on continuously whenever power is available to the Spacecraft and will be monitored and decommutated by use of a hardline connection.

It is not proposed that this be done for all signals or that this will allow all test point monitors to be removed, but it can be used to make a significant reduction in the number of monitors.

2.3.2 Command Capability. In the systems configuration, the subsystem OSE engineers have lost the capability to command the subsystems to alter modes except by means of hardwire test points or by the vehicle Command or Controller and Sequencer subsystems. In order to make the vehicle command path available and thereby reduce the number of hardwire control lines required, the computer will generate vehicle commands as requested by the subsystem OSE. These requests will be detected and identified by monitoring auxiliary contacts of the subsystem OSE switches used to generate command inputs during subsystem tests. The computer will output the correct command format via the Command OSE to the vehicle such that the requested command is executed.

2.4 Mission-Dependent Equipment Functions. The STC will be used to demonstrate compatibility between the Mission-Dependent Equipment (MDE) and the vehicle. The MDE consists basically of the TLM Demodulator, the Command Modulator, and the TCD software. Because the latter is required, the TCD hardware is also required. While the TCD normally consists of two redundant SDS 920 computers, it is planned at this time that only one would be used in the STC. Figure 2-2 shows how the MDE is to be connected into the system.

3.0 SUPPORT OF LAUNCH COMPLEX EQUIPMENT (LCE)

3.1 Description. The STC is used to support the LCE by providing the primary means for controlling and monitoring the launch countdown functions. These controls and displays are the same as those used during the systems tests and the intent is that the operators would have the same "feel" whether the vehicle is in the STC or at the Launch Pad. This is accomplished by using the STC as shown in Figure 3-1.

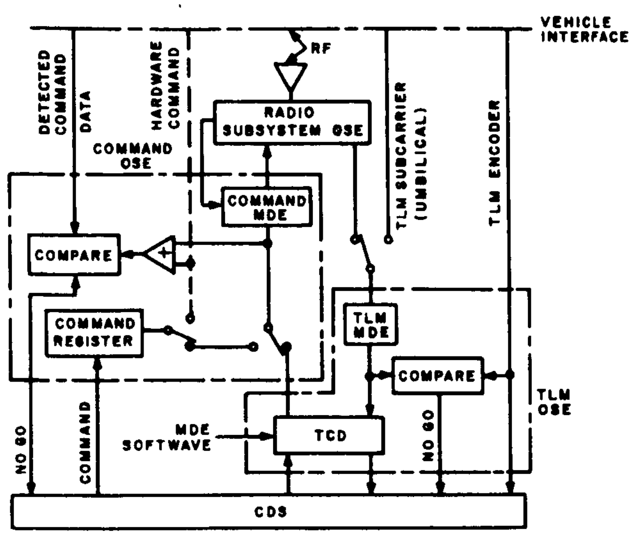


Figure 2-2 Mission Dependent Equipment System Interface Connection

3.1.1 Up-Link (STC to Vehicle or Vicinity Equipment). The STC's computer and TCD complex transmits command data directly to the Spacecraft via the radio subsystem. The CDS also transmits serial digital data to the vicinity equipment, where it is decoded and used to control the vicinity equipment, including transferring data (hardwire commands, etc.) to the vehicle via the umbilical. The data to be transmitted will be obtained by using a combination of three basic schemes. These are:

a. Automatic - The computer will release a pre-programmed sequence of commands based upon countdown time and where required on a basis of the analysis of the data received.

b. Manual - The computer will detect switch operations or changes on the STC Subsystem OSE consoles

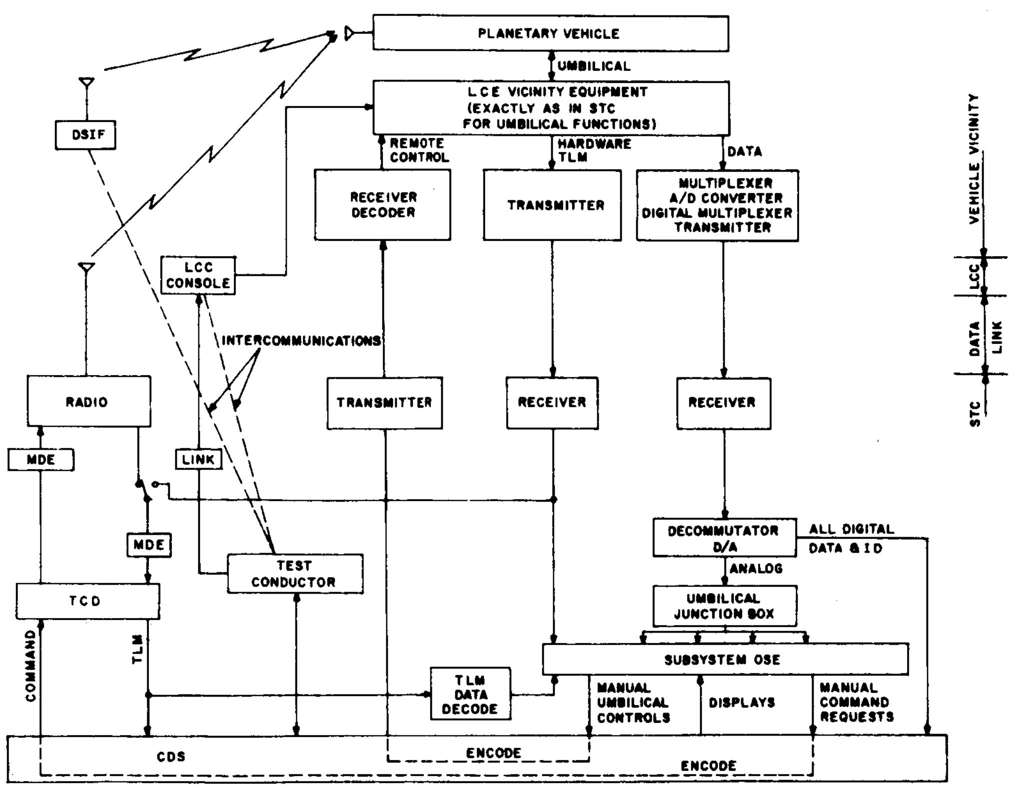


Figure 3-1 STC/LCE Interface (one Planetary Vehicle)

and convert these changes to a particular command or series of commands (either hardwire or RF) to cause the requested change to occur either in the vehicle or in the vicinity equipment.

c. Semi Automatic - The Computer will detect switch operations as in b above but will output the code only if the switch operation occurs in the proper sequence or if it is recognized (as pre-programmed) as being an emergency (i.e., always execute) type of function.

The method or degree of intermix eventually used depends on the computer programming.

3.1.2 Down-Link (Vehicle or Vicinity Equipment to the STC). The STC receives data by: (a) RF vehicle radio/TLM subsystem), (b) TLM subcarrier brought out hardwire via the umbilical, and (c) serial digital hardwire data, consisting of scanned and digitized (as required) umbilical and vicinity generated signals. The TLM data is handled exactly as it is during the STC tests (e.g., decommutated and processed by the computer, as well as reconverted to its original form and distributed to the subsystem OSE for display on meters and/or lights). The received digitized umbilical and vicinity data is fed directly into the computer as digital data and is also reconstructed, via a parallel path, for driving subsystem OSE displays in a manner similar to that of reconstructing the TLM displays. The data available will also include that required to verify the Up-Link data transmission.

3.1.3 Vicinity Equipment. The LCE Vicinity Equipment will be a functional reproduction of the corresponding subsystem OSE used in the STC. This should include controls and monitors for use prior to launch area evacuation and the remote control features required by the LCE usage.

3.1.4 Launch Control Center. The LCE will have a console in the Launch Control Center (LCC) which will contain summary displays and some switch operations. These controls and displays will be in effect tied back to the computer as much as the above is done. In addition to this, the LCC will have hardwire commands directly to the vicinity equipment primarily for emergency types of functions.

3.1.5 Deep Space Instrumentation Facility (DSIF) Station. The DSIF Station at Cape Kennedy will be used during countdown. This will be accomplished by using it to command and monitor the vehicles. The STC will monitor the command loaded via hardline command verification signals. In addition, the STC will also monitor the transmitted TLM data, as well as the hardline umbilical data.

3.2 Equipment. In order to perform the above functions, the STC will have to provide some additional capability above what is required for the basic STC. This equipment will consist of:

a. Data link drivers and receivers.

b. Digital decommutator and reconversion equipment

c. Switch detection circuitry.

d. LCE facilities control and monitor console.

Since it is required to have two vehicles processed essentially side by side through system test at the hangar and since one STC cannot support two simultaneous independent system tests, two STCs will be required in the field. Since two are available, it is planned to use them each to support the LCE functions associated with one planetary vehicle. By providing a few switching paths in the data links and between the computers within an STC and computers between STCs, a large degree of redundancy can be achieved. Since the functions required during the critical phases of the countdown are relatively few (i.e., power transfer, monitor TLM and umbilical signals, loading a few commands), one STC with only one of its computers operating could adequately handle the tasks required. The same can be said if only one set of vicinity LCE was operating in addition to two Ground Power Supplies in operation (note: three being planned). This last condition would impose requirements that hardwire control (i.e., loading commands via hardwire) would have to be done serially between vehicles, but this should not cause undue hardships. With the redundancy planned, it is expected that the launch hold limitation will not be the hardware but will be that the software to handle all of the possible alternatives was not adequately planned.

4.0 SUPPORT OF SPECIAL TEST STATIONS. During the factory acceptance test flow, the majority of the test sequences are performed at a Central Test Station where the spacecraft is in the immediate vicinity of the STC. However, the spacecraft is moved to special test areas for vibration tests, thermal vacuum tests, magnetic mapping, etc. The tests at these locations are more specialized and are not in as much depth throughout the system as required during the earlier and subsequent systems tests. These tests could be supported by moving a reduced version of the STC into the test area or by using the LCE philosophy using augmented LCE to support these tests. This latter choice was selected, and therefore a fixed STC will support these special tests using the standard LCE as vicinity equipment augmented to provide signal sources, stimulation, and monitoring that is not normally a part of the LCE but required to support more extensive testing than that capable via the umbilical. This additional capability is required primarily for the Guidance and Control stimulation and monitoring thrust vector and gas jet feedback signals, as well as a few additional test connector signals. This additional capability will be accomplished as done for the LCE by providing additional data link equipment and additional vicinity equipment.

5.0 DESIGN RESTRAINTS. The following sections summarize the restraints imposed on the subsystem OSE and the STC peculiar equipment as a whole. It is to be expected that others will be required for specific subsystems.

5.1 Vehicle Safety. The vehicle must be protected from damage resulting from equipment, personnel, sequence failures and similar failures. This will be accomplished at least to the following degree:

a. Circuits delivering power to the vehicle will have overload protection (current, voltage, frequency, etc.) devices operating continuously. These devices (if nondestructive) should be verified during self-test operations.

b. The vehicle circuitry will provide limiting circuitry in test points such that any test point can be shorted to any other or to ground and not cause the vehicle function to be degraded.

c. Connection keying should make it physically impossible to connect cabling together improperly.

d. Where a sequence of signals, power, etc., is critical to proper operation or safety, interlocks must be provided to prevent improper operation.

e. Electrical interfaces will be as specified in Section 10.1.3 of the JPL General Specifications.

f. Environmental conditions will be continuously monitored and alarmed before potentially damaging levels are reached.

g. Backup power supplies must be provided to protect against power failures.

5.2 Personnel Safety. Operating personnel will be protected by employing normal safety engineering practices to include at least:

a. High-voltage barriers and warnings.

b. High-pressure gas test cells and warnings.

c. Alarms when potential safety hazards exist, i.e., gas to be vented, movements to be expected, etc.

d. Toxic alarms.

5.3 OSE Reliability. Since it is uneconomical to make the OSE more reliable than the vehicle, great care must be taken to prevent damage and/or verify that the OSE is or is not operating properly. This latter function should be done on-line to a maximum degree and must definitely be done off-line as discussed below:

5.3.1 Off-Line Self-Check. A method will be provided to check the capability of the OSE to perform its functions before it is connected to the system or subsystem. This will be accomplished by providing a simulator with each piece of OSE and for the STC that provides the circuitry required to check all aspects of that OSE (including cabling) in modes or sequences best suited to detect hardware malfunctions. This should include verification of:

a. Proper connector interface.

b. Proper grounding.

c. Characteristics of OSE-generated signals, including power.

d. Indicators and controls.

e. Overload protection circuits.

f. Sequence interlocks.

g. Alarm circuits.

5.3.2 On-Line Self Check. When the OSE is connected to the vehicle and tests are being performed, the proper generation of signals within the OSE should be detected and analyzed and the capability to sense signals properly should be verified. This is done by:

a. Monitoring and recording all prime analog signals on local meters or recorders.

b. Monitoring switch operation (preferably on the signal going to the vehicle); where this is not possible monitoring an auxiliary contact.

c. Providing "calibration" signals that can periodically be switched to sensing devices.

d. Providing capability for the CDS to monitor the output signals of the OSE.

e. Detecting within the OSE the digital data streams sent to the vehicle and displaying the results locally and at the Computer Data System.

f. To ensure that the remote control signals from the CDS are received and executed properly, the subsystem OSE consoles, remote control circuitry will have a high degree of self-check capability designed in.

g. Recording critical signals.

h. Providing independent alarming for critical input signals.

i. Providing capability to switch vehicle input signals to OSE loads or signal sensors such that OSE outputs can be verified prior to switching to the vehicle.

5.3.3 Redundancy. The OSE will be designed by using high-grade commercially available equipment. Conservative application of this equipment as to loading, temperature extremes, voltage variations, etc., will be used. The STC is inherently redundant in many areas (vehicle commanded via the Radio or the Command or the Controller and Sequencer subsystems either from the Computer, the TCD, the Command Subsystem OSE, or the Controller and Sequencer OSE (two power sources, the solar panel and the battery simulators, one backed by power failure immune source, etc.). The OSE should therefore make no special provisions for redundancy unless it does not exist within the system, and damage can be done if the OSE fails.

5.3.4 Availability. No availability time criteria are now established; therefore, the following additional ground rules are required to establish a low mean time to repair.

a. Equipment is to be interchangeable.

b. Plug-in equipment is to be used wherever available.

c. Test points will be made readily available for use by a knowledgeable operator for fault isolation.

d. Accessibility is required for all equipment, and the equipment should be capable of operating extended from consoles as required for access to wiring during troubleshooting.

e. Self-check routines will isolate faults to problem areas within three replaceable modules.

5.4 Construction. The following additional restraints will be imposed on materials, and design to be used by all STC/OSE.

a. Cabinets will be as specified.

b. All meters, lamps, switches, general purpose instruments, power supplies, logic, etc., will be identical and as specified.

c. Cabinets will be furnished singly or in pairs.

d. Identical circuitry will be used to perform similar functions.

e. Cabinets will be mounted on wheels to provide convenient means for moving within the building. Means for locking will be provided.

f. Maximum weight of a replaceable module will be established with suitable hand holes, etc., provided.

g. Facility power will be supplied at 120 volts single phase, or 208 volts 3-phase all at 60 Hz. Each console shall provide convenience outlets.

h. Each cabinet will supply its own additional cooling (blower at minimum) considering STC ambient is maintained at 60-90° F.

5.5 Panel Layout. The subsystem consoles control panels should be laid out in a manner such that the operator can ascertain at a glance the status of the subsystem and the functions of his controls should be apparent. This can best be accomplished by laying out the panel controls/displays using good human factor techniques. The method considered best at this time is schematic panel layout similar to those used for process controls. Figure 5-1 shows a panel using this technique for the power subsystem.

5.6 Grounding. The grounding scheme shall be as specified in Section 10.1.3.3 of JPL's General Specification. The intent of this is to maintain isolation between subsystems. The OSE must not do anything to negate this requirement. The signals between OSE and spacecraft and between OSE and other OSE will be done in such a manner as to guarantee

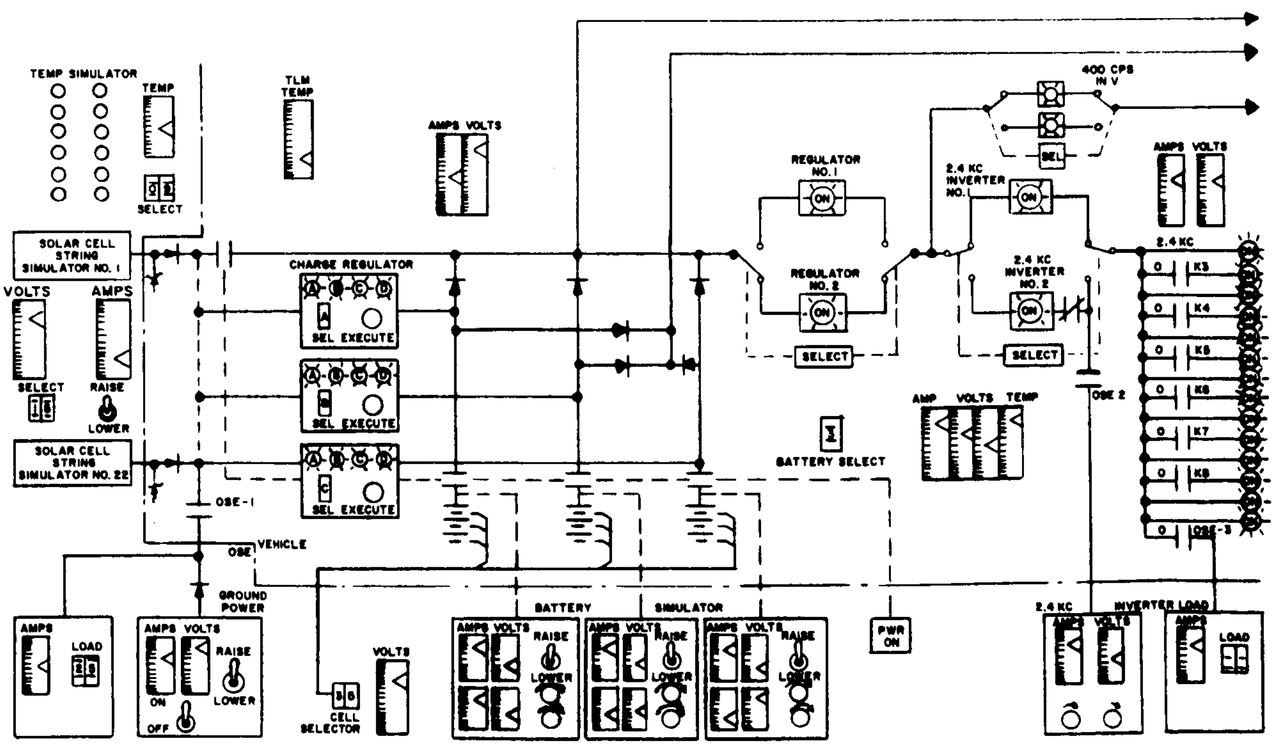


Figure 5-1 Power Subsystem Panel Layout

d. Providing capability for the CDS to monitor the output signals of the OSE.

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g. Facility power will be supplied at 120 volts single phase, or 208 volts 3-phase all at 60 Hz. Each console shall provide convenience outlets.

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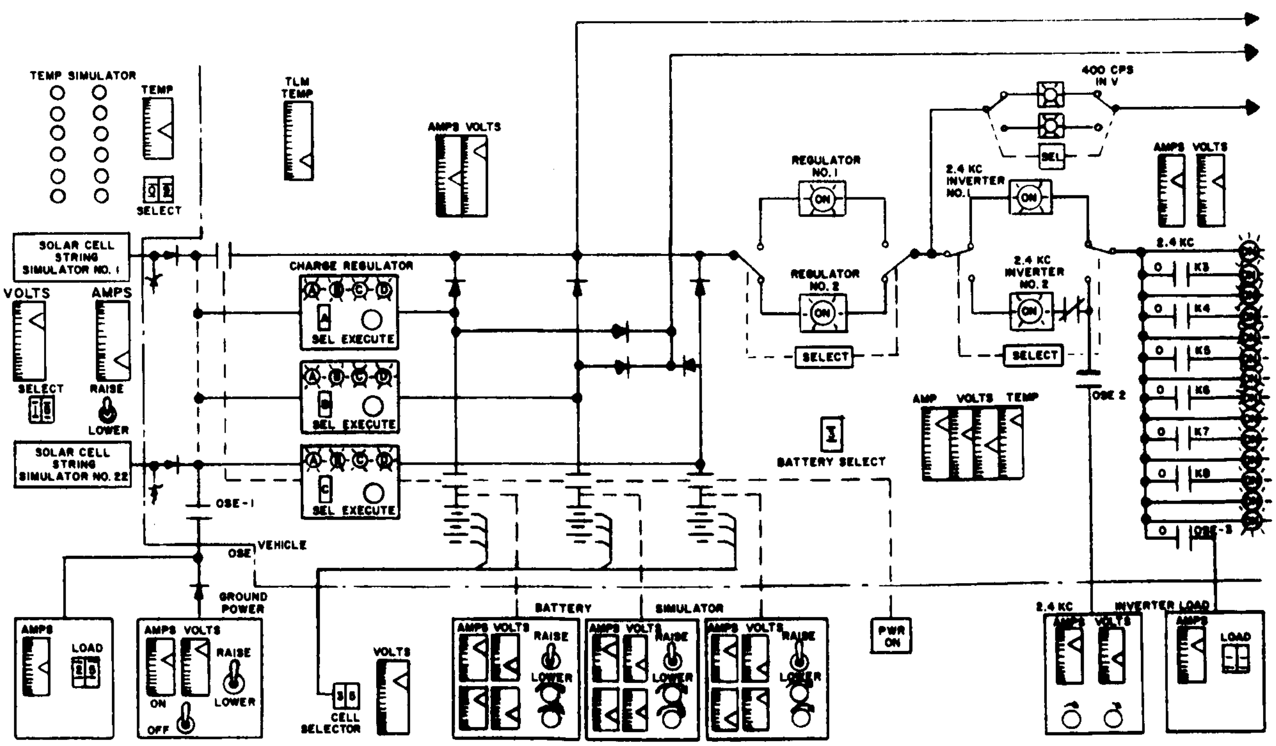

Figure 5-1 Power Subsystem Panel Layout

isolation. This will be accomplished by using transformer-coupled digital interchange, differential amplifiers, discrete data interchange by use of contact closures, etc.

5.7 Degree of Automation. The degree of automation for the subsystem OSE consoles while performing subsystem tests must be decided on an individual basis with the following criteria considered:

a. Automation is justified when:

   1. Accuracy is beyond manual capabilities.
   2. Response is beyond manual capabilities.
   3. Task is repetitive for long periods of time.
   4. Tests are long, and maintaining alertness for that period is not possible.
   5. Large amount of data is to be transmitted/gathered.

b. Any automatic sequence must be easily reprogrammable to accommodate vehicle changes.

c. A conservative OSE approach is to be taken.

d. System tests will be automatic as controlled from the Computer Data System; any subsystem test should be compatible with this approach.

e. The use of computers for the subsystem OSE, as well as the STC, will be as specified in Section 3.10 of the JPL General Specification, with the provision that use of special-purpose types of computing circuits must be separately justified.

5.8 Vehicle Test Points. Special vehicle test points shall be provided only when the command, TLM, umbilical, or OSE-generated or detected signals or procedures are inadequate to perform the required test function. Vehicle fault isolation will not be done on line when additional test points are required solely for that purpose. The tests should indicate that a fault occurred and the signal path that is faulty. Manual troubleshooting using T-connectors to be inserted in flight connectors, test point boxes, and standard test instruments will be used for detail fault isolation.

5.9 Flexibility. Since the OSE is designed concurrently with the vehicle, it must be expected that changes will be required due to vehicle changes, test requirements, test procedures etc. To accommodate these changes requires that the OSE be over-designed so that it provides:

a. Wider and finer signal (stimuli) variation than that envisioned.

b. Extra monitoring and control capabilities than that required.

c. Means for manually controlling sequences for troubleshooting and investigation tests.

d. Modular design such that changes can be incorporated without completely redesigning the equipment.

5.10 Voltage Monitors. Each subsystem OSE will be responsible for monitoring the outputs of its transformer/rectifier. In addition, each subsystem OSE should monitor via hardwire its 2.4-kc input current and provide alarm locally, to the test conductor, and Computer Data System for overcurrent indication.

5.11 STC Displays. The STC (via the computer data system) will display to each subsystem OSE the status of each test sequence. This will be done to: (a) Display Test Sequence Number, (b) Test Time, and (c) Test Running/Hold. In addition to this, each subsystem OSE will contain an intercommunications panel which will connect to the test conductor's panel.

5.12 Cleanliness. The spacecraft will be assembled and tested in 100,000 class clean room. The OSE that must be mounted on the spacecraft or in its immediate vicinity must be compatible with that type of clean room. The two major restraints are that this equipment should not generate dirt (e.g., not flake, have moving elements enclosed, etc.) and should be easy to clean (e.g., have no dirt pockets, have cabling sheathed, have cabling or pneumatic runs separated such that a vacuum cleaner nozzle can fit between, have no sharp internal radii, etc.). Because of the degree of cleanliness, personnel will not have easy access to the spacecraft area, and every effort will be maintained to keep the number of people required small. The OSE vicinity equipment should therefore be able to be operated remotely (from the STC) or by a minimum number of personnel within the test area.

CII VC260FD100

SYSTEM TEST COMPLEX SYSTEM LEVEL FUNCTIONAL DESCRIPTION

SECTION

SYSTEM TEST COMPLEX SYSTEM LEVEL FUNCTIONAL DESCRIPTION

1.0 GENERAL. The System Test Complex (STC) consists of: a) the subsystem Operational Support Equipment (OSE) test sets, each in the same configuration as used during their independent subsystem tests prior to being incorporated into the system; b) the Computer Data System (CDS) which is used to provide centralized data processing and analysis as well as centralized control, consisting of a computer complex and the necessary interface equipment to tie the Computer and the Subsystem OSE together; c) the Test Conductor's Console (TCC) from which the test leader can exert supervisory control over the test operations; and d) ancillary equipment such as power distribution, time distribution, central recording, etc.

2.0 FUNCTIONAL REQUIREMENTS.

2.1 Subsystem OSE. The following summarizes the primary requirements of each of the Subsystem OSE that is required to support the system performance tests and mission profile tests. A more detailed functional description of each appears later.

2.1.1 Power Subsystem OSE. The Power Subsystem OSE provides:

a. Adjustable ground power via the umbilical to the solar array bus.

b. Three battery simulators to simulate variable discharged battery conditions, variable voltage, variable telemetry (TLM), and umbilical sensor simulators associated with flight type batteries.

c. A means for verifying the bus/solar panel interface and variable TLM signals at that interface, as required.

d. A means for varying the amplitude and frequency of the 2.4-kc power sources and for inserting transients.

e. A means for transferring to internal power by gradually reducing the ground power supplier outputs.

f. A means for controlling the Power Subsystem modes.

g. Verification of clock accuracy.

h. A means for monitoring and displaying TLM, hardwire and, OSE data and means for alarming the critical units.

2.1.2 Radio Subsystem OSE. The Radio Subsystem OSE provides:

a. A transmitter that has switch-selected nominal frequencies for each vehicle receiver with manual vernier offset. Power output is to be adjustable.

b. A means for transmitting ranging codes with variable modulation parameters.

c. A means for transmitting command codes with variable parameters, variable PN codes, and data obtained from the Command Mission Dependent Equipment/Telemetry and Data Handling Subsystem (MDE/TCD) or from the Command Subsystem OSE.

d. A means for receiving the vehicle RF transmitted signals, detect range code signals, and the TLM Subsystem OSE.

e. Test equipment, controls, and monitors that are required to measure the significant parameters of the received and transmitted signals, as well as to control the mode of the Radio Subsystem, etc.

2.1.3 Telemetry and Data Storage Subsystem OSE. The TLM and Data Storage Subsystem OSE provides:

a. One TLM MDE demodulator, the input of which can be switched to one of two identical data paths (one from the output of the Radio Subsystem OSE, the other via the umbilical-being the input to the vehicle's Radio Subsystem). The output can be switched into the TCD or into the CDS.

b. During periods that the spacecraft Radio Subsystem is transmitting TLM data, the output of the demodulator is to be compared against the TLM Encoder obtained via hardwire.

c. A means for inserting calibration signals into the TLM Subsystem.

d. A means for controlling the mode, the displays, and the test equipment required to monitor the TLM Subsystem.

2.1.4 Command Subsystem OSE. The Command Subsystem OSE provides:

a. A means for loading commands via the Radio Subsystem OSE. Note that either the Command OSE or the Controller and Sequencer (C&S) OSE will be able to insert commands via umbilical hardwire, by-passing the Radio Subsystem.

b. A means to vary the parameters, format, etc., of the data and of the PN generator.

c. Detection, via umbilical connector, of the command data bits received by the vehicle and compare them to those transmitted. If there is disagreement, it provides an inhibit to prevent the execution of that command and generates an alarm.

d. The necessary controls and displays to monitor the performance of the Command Subsystem.

2.1.5 MDE Functions. The MDE Functions provide:

a. An MDE Command Modulator to drive the Radio Subsystem OSE transmitter with data obtained from the TCD.

b. A MDE TLM Demodulator to operate on the Radio Subsystem OSE output or the umbilical hardwire composite.

c. While the TCD is not MDE, its software is, and therefore it should be used exactly as planned for flight during some portion of the tests with the CDS acting as a Space Flight Operations Facility (SFOF) simulator. In addition, the TCD will be used to aid the CDS in the analysis of the TLM data and in the generation of planned commands. Note that the TCD computer is used as the TLM OSE decommutator during subsystem tests.

2.1.6 C&S Subsystem OSE. The C&S Subsystem OSE provides:

a. A means to load the C&S memory. Data will normally be received from the CDS.

b. A means for reading out the entire memory or selected data words.

c. A means for executing a C&S type command in real time.

d. A means to control the C&S mode and for monitoring and loading the "time-to-go registers".

e. A means to detect a command executed signal.

f. A means for monitoring and controlling other signals associated with the C&S.

g. A means for resetting the subsystem (flip flops, latching relays, etc.) to a known state such that when power is applied to the vehicle, the C&S and others controlled by the C&S are in a known state.

2.1.7 Guidance and Control (G&C) Subsystem OSE. The G&C Subsystem OSE provides:

a. A means to stimulate the Attitude Control Sensors (sun, star, solar aspect, etc.), such that loop verification can be accomplished (i.e., cause position-rate errors to occur).

b. Gyro torquing signals to simulate vehicle motion.

c. That functions (a) and (b) are accomplished in an open or closed loop manner, using jet detectors and thrust vector control feedback data to close the loop.

d. A means for controlling the ΔV accelerometer.

e. A cold gas supply for the G&C Subsystem tank with controlled pressures, etc. Also, measurement of flow rates during solenoid valve operations.

f. Detection of cold gas flow from individual G&C jets.

2.1.8 Propulsion Subsystem OSE. The Propulsion Subsystem OSE provides:

a. Thrust vector controls and monitors (mass injection valve position and/or jet vane loading, force detecting and position indication) that send feedback signals to the G&C Subsystem OSE.

b. Leak test capability.

c. Other monitors and controls as required.

2.1.9 Pyrotechnics Subsystem OSE. The Pyrotechnic Subsystem OSE provides:

a. Resettable squib simulators that exhibit worst case characteristics. These should be able to be mounted in the vehicle and connected to the flight connectors.

b. Squib monitoring displays that are required to determine which squib circuit was actually fired by use of the pyro continuity loop.

c. Monitors for the capacitor banks and for detecting the firing current waveform.

d. Controls and monitors as required.

2.1.10 Thermal Subsystem OSE. The Thermal Subsystem OSE provides:

a. A means of monitoring the thermal characteristics of the vehicle temperatures, louvre positions, etc. The TLM will be a prime source of data, but must be backed up by other special monitors.

b. A means to perform margin reverification tests on the entire vehicle at elevated and low temperatures when used in conjunction with the Environmental Control Equipment described in the Assembly Handling and Shipping Equipment section (CII VC270FD101).

2.1.11 Simulators. The Simulators provide:

a. Verification that the Spacecraft under test can generate the proper signals and can receive data, normally received via its interface connectors, from major elements of the Planetary Vehicle and Launch Vehicle. Simulated components include the Capsule, Saturn booster, Science, and Propulsion.

b. Interface characteristics of the simulators that are adjustable from nominal to worst case.

c. Monitors and controls that will monitor and control the signal interchange.

2.1.12 Articulation Subsystem OSE. The Articulation Subsystem OSE provides: position monitors for monitoring antenna actuating drive positions, load/inertia simulation for simulating the antenna, and monitoring and controls as required.

2.2 Computer Data System. The CDS provides:

a. A means for monitoring and analyzing the data obtained from TLM (in parallel with the TCD), from selected hardwire analog monitors after analog-to-digital conversion, and digital/discrete data.

b. Output data and results of analysis to the test conductor, via his alpha-numerical display and line printer, and to the subsystem engineers, via character printers.

This data output will be as controlled by the program and as requested by the various operators.

c. Record results of tests for further analysis.

d. Control of the test sequence (as governed by the computer program and test conductor), by remotely controlling certain functions of the Subsystem OSE and by originating the vehicle commands to be transmitted (RF or hardwire) to the vehicle.

e. Monitor the Subsystem OSE switch operations in order to verify mode of the OSE as well as the mode of the vehicle.

f. Self-check features, by verifying that vehicle data agrees with what should have been sequenced manually.

g. All of the interface equipment between the Subsystem OSE and the computer.

h. Independent capability to decommutate the received TLM data.

2.3 Test Conductor's Console. The TCC provides the test conductor with displays and controls necessary for him to provide executive control of the STC. This will include:

a. Alphanumeric CRT display of 30 values (selected in groups from 300 values) and their limits expressed in engineering units with English identification obtained from the CDS. Out of limits values identified with asterisks, etc.

b. STC intercom controls.

c. Manual interface with the computer, to control test program selection, mode selection, initiate holds, remove holds, etc.

d. Continuous display of critical signals obtained from the Subsystem OSE.

e. Control/approval of manually selected vehicle commands.

f. High speed line printer.

g. Character printer.

2.4 Ancillary Equipment.

2.4.1 Raw Data Recording. The Raw Data Recording provides: a means of recording unprocessed data to ensure that it is not lost due to equipment failures and a means of reconstructing data leading up to failures or unexplained situations. The data to be recorded is: TLM (output of TLM MDE), converted hardwire analog data, commands sent to and received by vehicle, subsystem OSE switch operation, and control signals between CDS and Subsystem OSE.

2.4.2 Central Recording. The Central Recording provides the capability to record and time correlate signals from several subsystems and provides patch board capability.

2.4.3 Miscellaneous. The Ancillary Equipment provides: a Central Timing Generator similar to that used on Mariner C; a power distribution panel to distribute 110 volts, 1-phase and 208 volts, 3-phase power, 60 Hz; and an umbilical simulator such that the STC interface signals can be verified prior to mating with the vehicle.

2.5 LCE Functions. The LCE within the STC provides the following:

a. A means for reconverting the digitized umbilical signal to its original form.

b. A means for accepting data from the CDS and shifting that data to the Vicinity Equipment. Digital feedback will be provided to the CDS to check proper reception at the Vicinity Equipment.

c. Line drivers, receivers and switching as required.

d. Monitors and controls peculiar to the LCE functions will be provided on a separate STC-LCE console.

e. Alternate paths for all launch control functions including:

   1. Command loading from either STC (RF or hardwire) or Deep Space Instrumentation Facility (DSIF) station

   2. TLM readout from either STC (RF or hardwire) or DSIF station

   3. Umbilical functions from either STC

   4. Power transfer from either STC or from Launch Control Center

3.0 MAJOR STC INTERFACE DESCRIPTIONS

3.1 General. The major interfaces within the STC exist primarily between the CDS and the various subsystems of the OSE. (Note that the subsystem OSE-to-Spacecraft interfaces are discussed in detail in the individual subsystem OSE writeups.) Most of these internal interface signals pass through an STC/computer adapter as shown in Figure 3-1. This adapter is used to buffer the data in and out of the computer, to perform the signal isolation of the computer from the other OSE, to maintain the isolation between subsystems, and to perform by hardware, those functions that could be performed by computer software but may result in overloading the computer. Also shown are the interfaces with the Test Conductor's Console. The following sections discuss and amplify this internal exchange of signals.

3.2 CDS Control of Subsystem OSE. The remote control of the Subsystem OSE by the computer will be accomplished by having the computer control circuits in parallel with the manual controls. This may include setting up a function generator (quantitative command) and then at a later time, give it a start/stop/hold type of command, or it may be implemented by a simple discrete command to initiate a manually set-up sequence or function generator. Generally, when a quantitative command is given, the circuit should be set up to perform a function when later given a discrete command.

Figure 3-2 shows a scheme of how this control data could be read into the Subsystem OSE consoles, as well as reading digital data out of them for use by the computer. The scheme

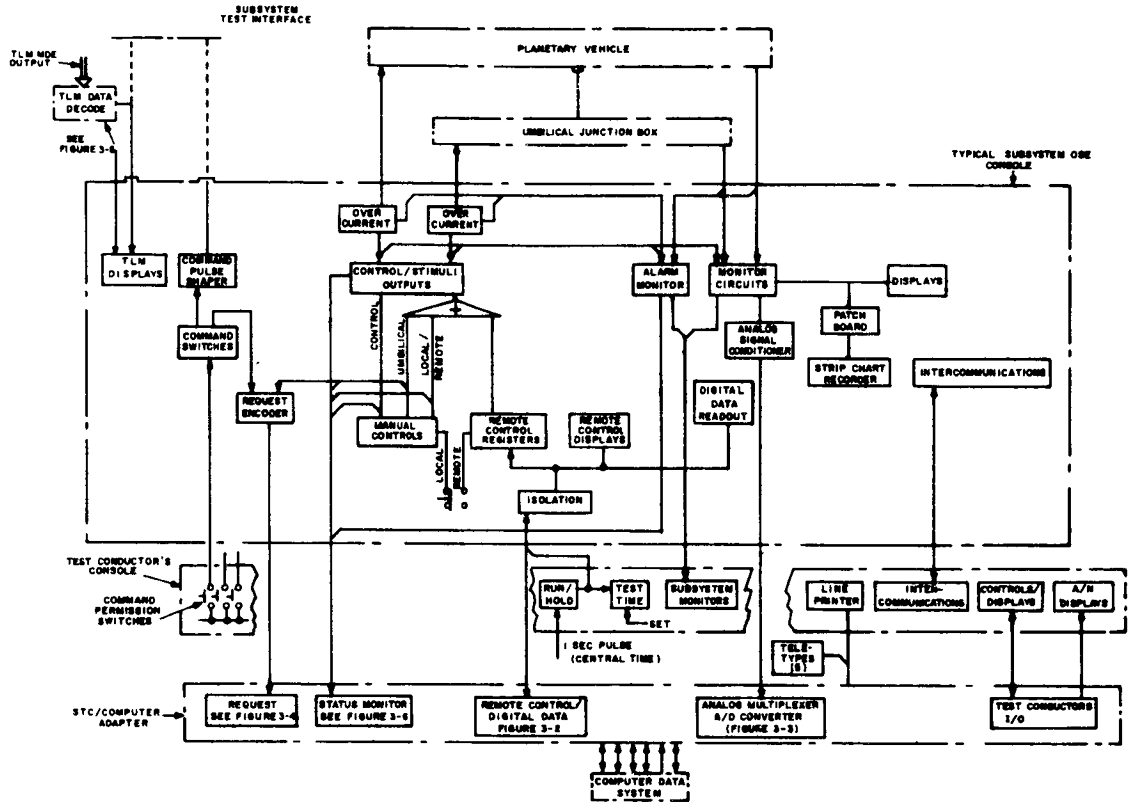

Figure 3-1. Typical Subsystem OSE to STC Interface

presented indicates that a 24-bit buffer in the STC/Computer Adapter is loaded by the computer. The adapter circuitry determines, by looking at bits 1 through 6, whether this is an internal (to the adapter) or an external address and whether that address is an instruction or a data read-in type. When this is established, the applicable sequence, as shown in Figure 3-2, is carried out. The major reason for the complexity and timing of the sequence is to provide a high degree of in-process self-check capability such that it is virtually impossible to load the wrong destination. Several other self-check circuits and routines are also provided to enable verification of the complete integrity of the exchange circuitry.

Not all Subsystem OSE consoles need provide this capability. If only a few simple control functions are required, this will be accomplished by using the adapter to control relays within the adapter by the use of one of its internal addresses which, in turn, will control the OSE by means of contact closures. The types of the functions to be controlled are as follows:

a. Command word to TCD, command OSE and C&S OSE

b. Ground power voltage control

c. Battery simulator control

d. G&C initial position error set (3)

e. G&C initial rate error set (3)

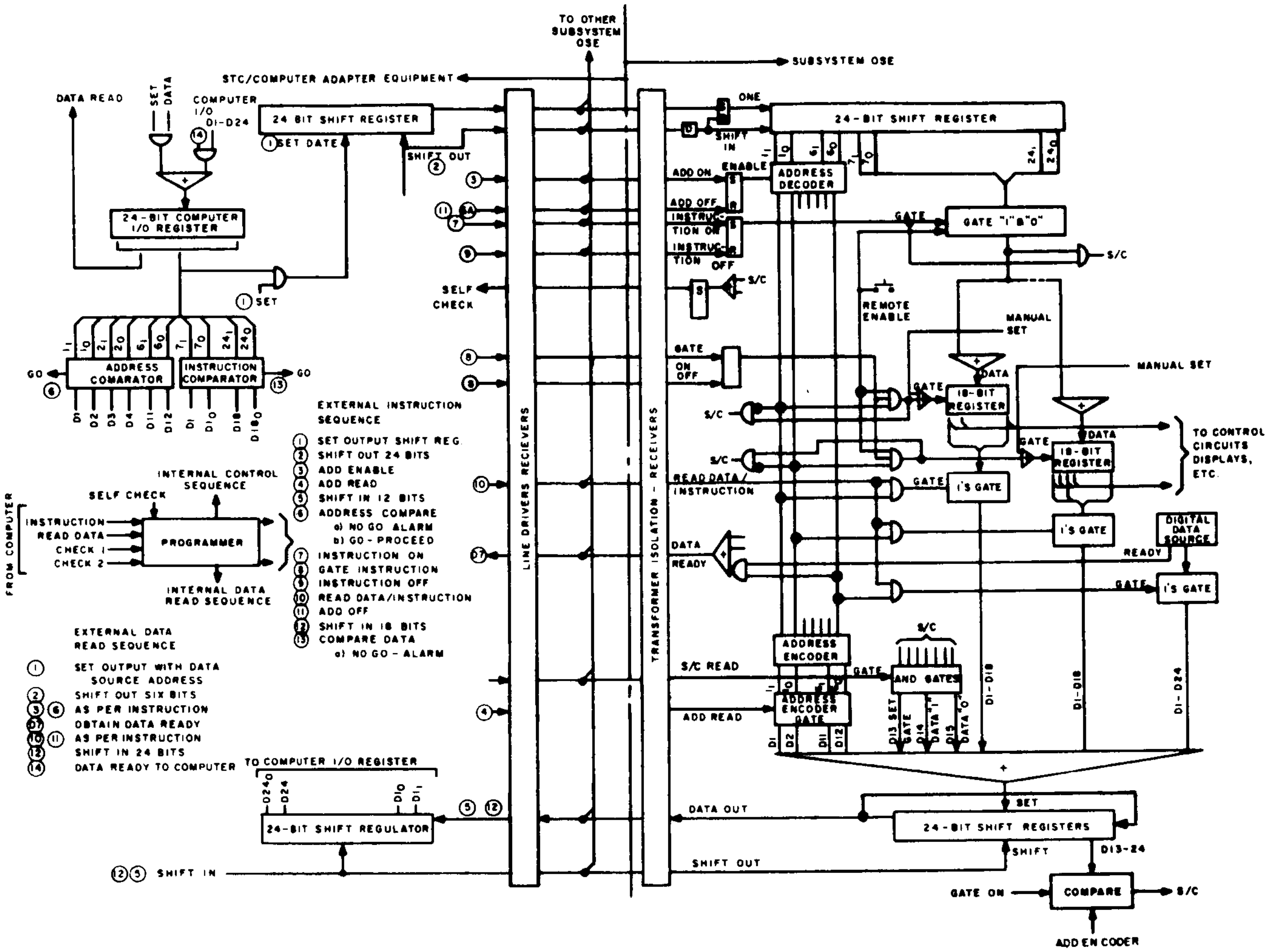

Figure 3-2. Digital Data Interchange, STC to Subsystem OSE

f. Command data format modification

g. Mode controls of all subsystems and OSE

h. TLM homing if provided

i. Test time clocks (set at initial value and updated once per second by central time pulse)

j. Transmitted frequency

k. C&S time to go registers loading

l. Light indications indicating test modes, etc.

m. Read out digital data such as RF frequency, received command, and time to go.

3.3 <u>Subsystem OSE to CDS Data Multiplexer</u>. The computer will require many analog signals in order to evaluate the test. These signals include the umbilical signals, the direct access test connector signals and internally generated Subsystem OSE signals. All of these signals are routed through or from the respective Subsystem OSE to an analog multiplexer controlled by the computer. These signals must be signal conditioned within the Subsystem OSE to one of the following ranges 0 to 3.2 volts or 0 to 100 millivolts. It is desirable that these signals represent the same range as corresponding TLM points, where the data is available on both.

Figure 3-3 shows a proposed analog multiplexer configuration. It consists of the following major elements:

a. Slow speed multiplexer - selects 30 signals (3 wires each) out of 300 (Note that 0- to 100-mv signals are grouped such that 6 of the 30 leads are in this range and that 6 amplifiers are time shared). All inputs are differentially coupled into amplifiers or multiplexers.

b. High speed multiplexer scans (or steps under computer control in any sequence) the 30 selected signals in addition to 20 fixed signals which are considered critical.

c. High speed analog-to-digital conversion, converts the analog from 0 to 3.2 volts to 12 binary bits at a rate of 5000 conversions per second.

d. Six analog voltage comparators that can be switched to any of the 50 inputs to the high speed multiplexer. The reference for these comparators is to detect when the analog signal makes a transition (either high to low or low to high) with the output being used to store a time value (to be read out as in Section 3.2 above) or to interrupt the computer. This allows the system to make precise time measurements, pulse count, etc., without having the computer tied up monitoring one or two signals continuously.

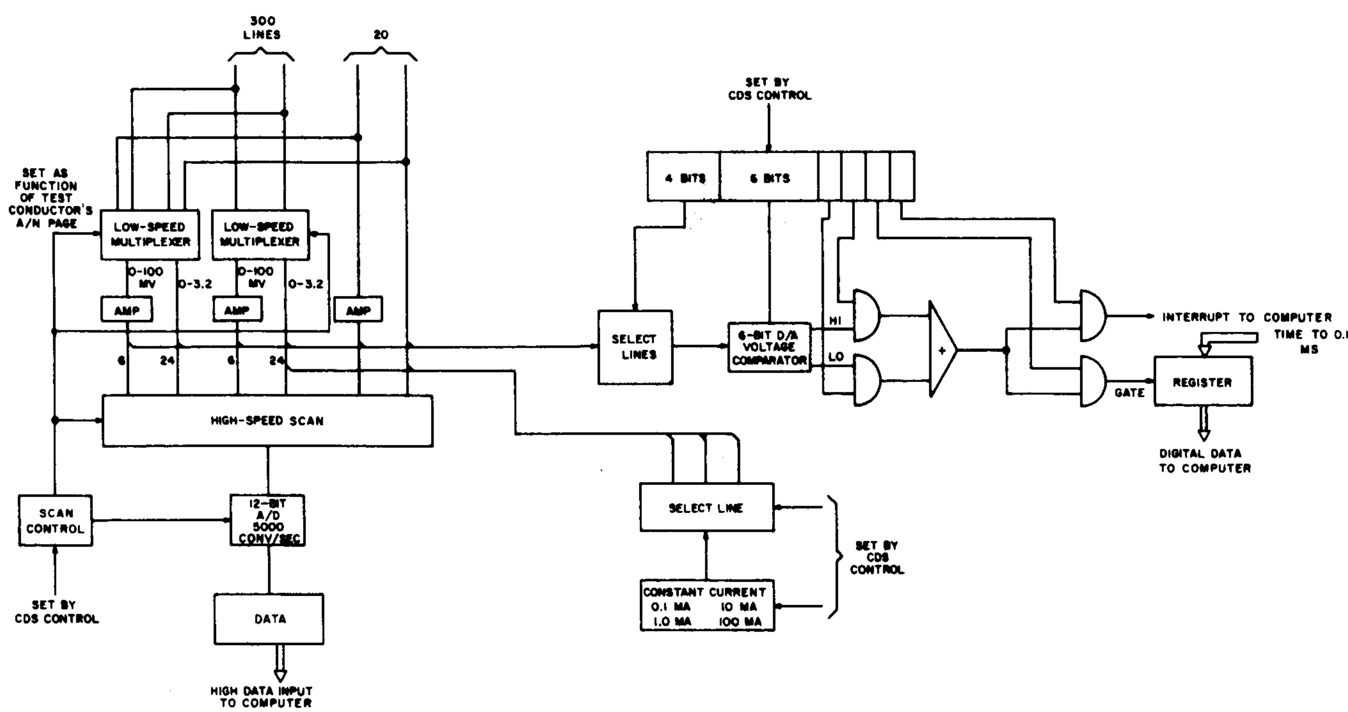

Figure 3-3. Analog Multiplexer

e. A constant current source (adjustable) that can be switched to ____ input lines of the high speed multiplexer. This will be used to measure resistance values.

f. Another slow-speed/high-speed multiplexer combination is provided to obtain blocks of data for the Test Conductor's Alphanumeric display. This multiplexer is kept separate from the other in order to make the gathering of this independently requested data completely independent except for the analog-to-digital converter which is periodically switched over to sample these values under computer command.

The expected type of usage of the data multiplexer would be that the computer would set up the slow speed multiplexer as a function of the test sequence to follow. This operation would make available to the high speed multiplexer all of the analog data required for that sequence (i.e., for the next seconds to perhaps minutes). The high speed multiplexer and analog-to-digital converter would normally sample all selected values at approximately 2000 samples/sec. The computer however would input the data only periodically as the test sequence demanded. All data would be recorded on the raw data recording.

The alphanumeric data slow speed multiplexer would be set up solely as a function of the display requested by the test conductor. Once per second those values would be scanned and inputted to the computer.

3.4 Subsystem OSE to CDS Command Request. In order to increase the flexibility and improve the capability to control his subsystem when connected to the system, the operator has been provided with the ability to cause vehicle commands associated with his subsystem to be executed in near real time. This mode would be used primarily during troubleshooting or investigation type sequences, and possibly during manual system tests.

This capability existed during subsystem tests and it is the intent of this implementation to make use of the same controls (or at least auxiliary contacts). Figure 3-4 shows how this can be implemented. This scheme indicates that the test conductor gives permission to individual Subsystem OSE to enable this capability. This fact is displayed on the subsystem consoles by lights (using CDS power). By operating the command switches used during subsystem tests an 8-bit output-encoded signal is generated redundantly. This code is detected by the STC/computer adapter. The redundant paths are checked and if they compare, an interrupt is sent to the computer. Also a light indication is sent back to the originating Subsystem OSE console, indicating that it has been accepted and inhibiting further command requests. The computer obtains the 8-bit coded word by using its digital data (internal to adapter) read-in circuit (Section 3.2). The computer can then perform any checks desired on that code as provided in the program. The computer will then output, to either the TCD, the Command OSE or the C&S OSE, the command format to be transmitted to the vehicle to cause the requested function to occur. (Note that it is felt that the coded word generated by the switch and its encoding circuit should be in the vehicle command format.) When the signal has been received by the vehicle, the computer lights an acknowledge light on the Subsystem OSE and releases the inhibit of new commands. Quantitative commands are obtained and transmitted in a similar manner.

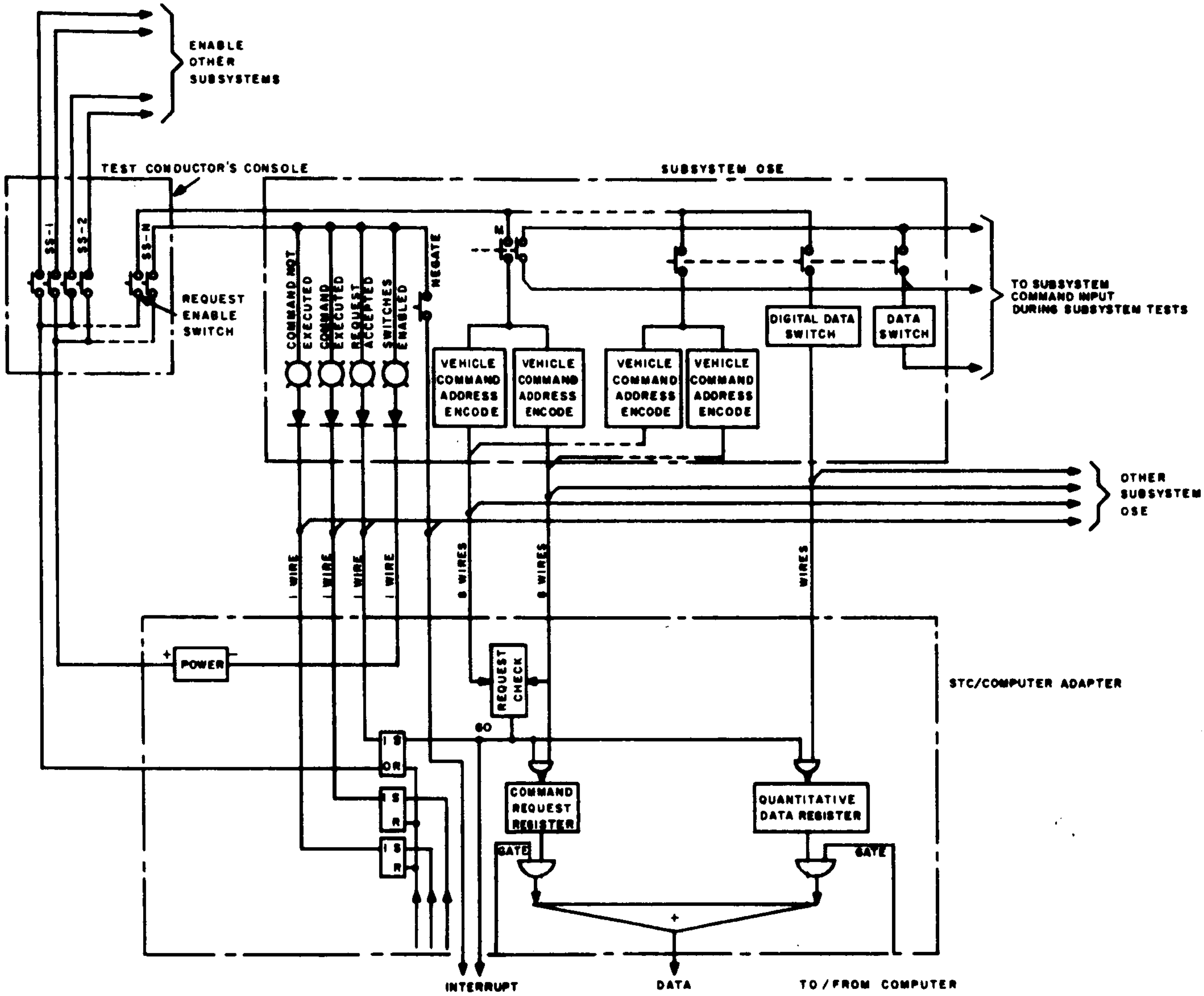

Figure 3-4. Manual Command Request Circuits

3.5 TLM Data Decoder to Subsystem OSE. As an attempt to reduce the number of hardwire monitor leads, it has been proposed to make extensive use of the TLM monitors. This implies that hardwire monitors are not required to calibrate the TLM sensors, signal conditioners, etc. This is not true for all cases but it is true for a significant number; primarily those in which there is already an OSE generated monitor, or for the TLM discretes, or where the calibration is not critical, etc. Therefore, in those cases where the TLM can be effectively used, the STC will provide a means for reconstructing the TLM signal such that it is compatible with the monitors that the Subsystem OSE used during subsystem tests when the TLM sensors were directly available. This use is obviously most useful on the high and medium TLM data rate channels. In order to further improve the usage, consideration can also be given to use of the TLM homing feature which would allow a particular TLM data point to be sampled at a high rate. Figure 3-5 shows an approach to accomplish the TLM data decoding.

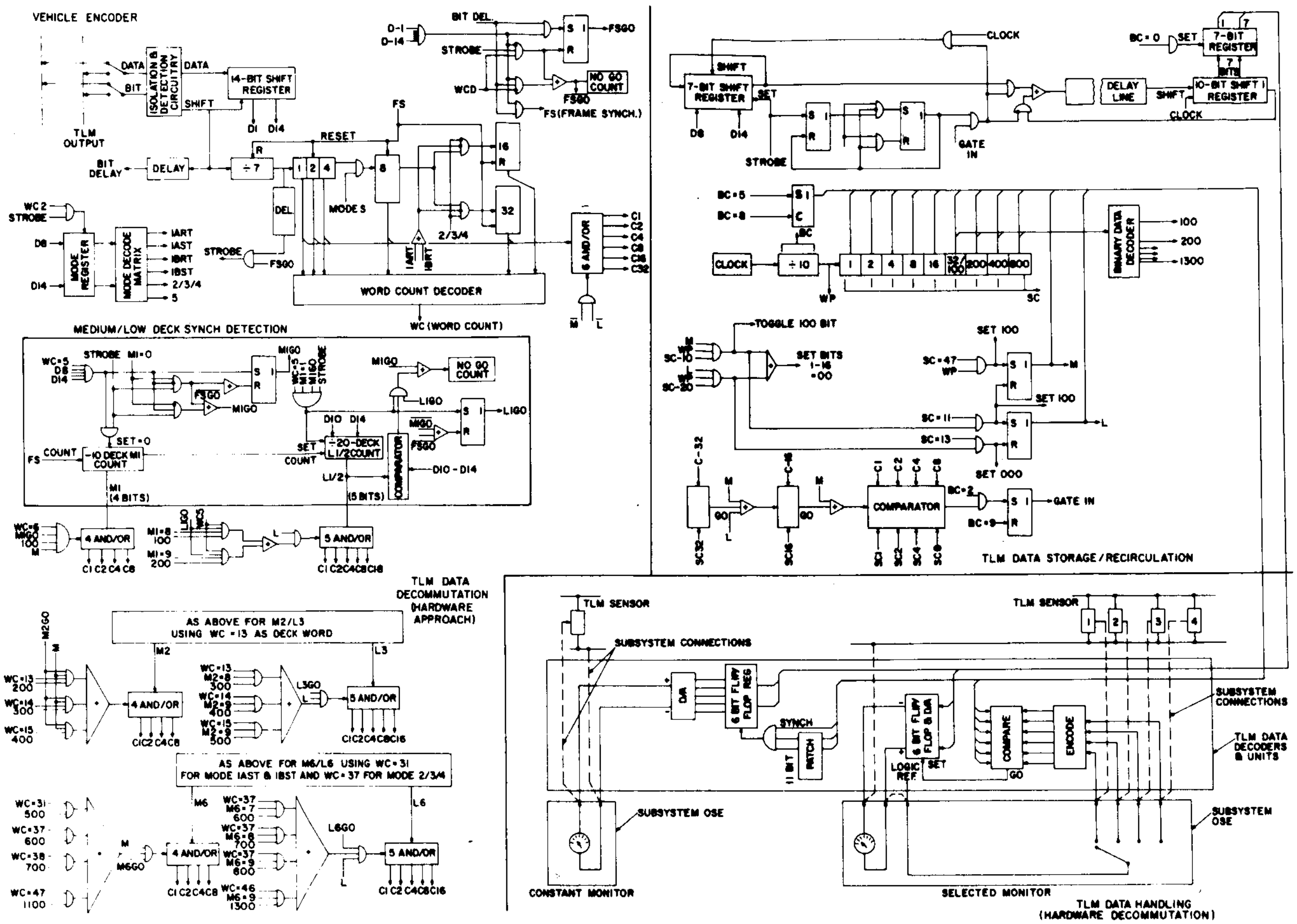

Figure 3-5. TLM Data Decoder

This scheme uses the output of the TLM MDE demodulator or direct access to the TLM encoder, performs a simple decommutation function, and provides a circulating storage for storing the last sample of each data word. This data along with identification bits is outputted continuously. The STC/computer adapter provides the capability to store the data bits of selected words and performs a digital-to-analog conversion for driving subsystem OSE meter displays that had been connected to TLM sensors or lamp driver circuits for driving discrete/digital displays. The words to be displayed can be selected in a number of ways as shown (i.e., continuous monitor or one word selected by prewiring an "and" gate to a particular ID word--select various ID's via switch and prewired ID request encoding circuits--or by simple thumbwheel switches). The circulating nature of the identity and data bits makes the updating of a newly selected value occur within several milliseconds. The data displayed is the last value obtained from the requested TLM sensor. The computer will also be able to select and read out the stored TLM data by an internal data request as described in Section 3.2

The selected data storage, selecting matrix, digital-to-analog converters, etc., will be located throughout the STC so as to reduce the length and number on interconnecting wires.

The scheme shown in Figure 3-5 indicates that the decommutation which is done independently of the TLM OSE (TCD) as specified and the circulating storage of the latest data words (required to allow rapid changing of manually selected displays) are both accomplished by a hardware approach as opposed to using the CDS's computer and appropriate software. There are advantages of doing these two functions (decommutating and outputting data) either way (hardware versus software). This points up one of the major continuing problems involved in applying the computer to the STC (i.e., where is the optimum break point between hardware and software solutions to satisfy the requirements).

3.6 Subsystem OSE Mode Monitoring. The computer is to be able to ascertain whether the Subsystem OSE is in the proper mode to start tests or to be able to detect when the mode, etc., is altered manually. Figure 3-6 shows a scheme of accomplishing this by using Subsystem OSE switch (fixed or momentary) or relay contacts to indicate their mode or to detect when changes occur. The scheme, as shown, scans approximately 500 contacts 2000 times per second in groups of 16 and interrupts the computer only if there is any change from the previous sample. The computer can then read in (using digital data input as described in Section 3.2) which group caused the change or also the group plus a bit for each of the 16 switch positions within that group. (A minor change will also allow the computer to read in a code for the particular bit that caused the change directly.) By reading in each

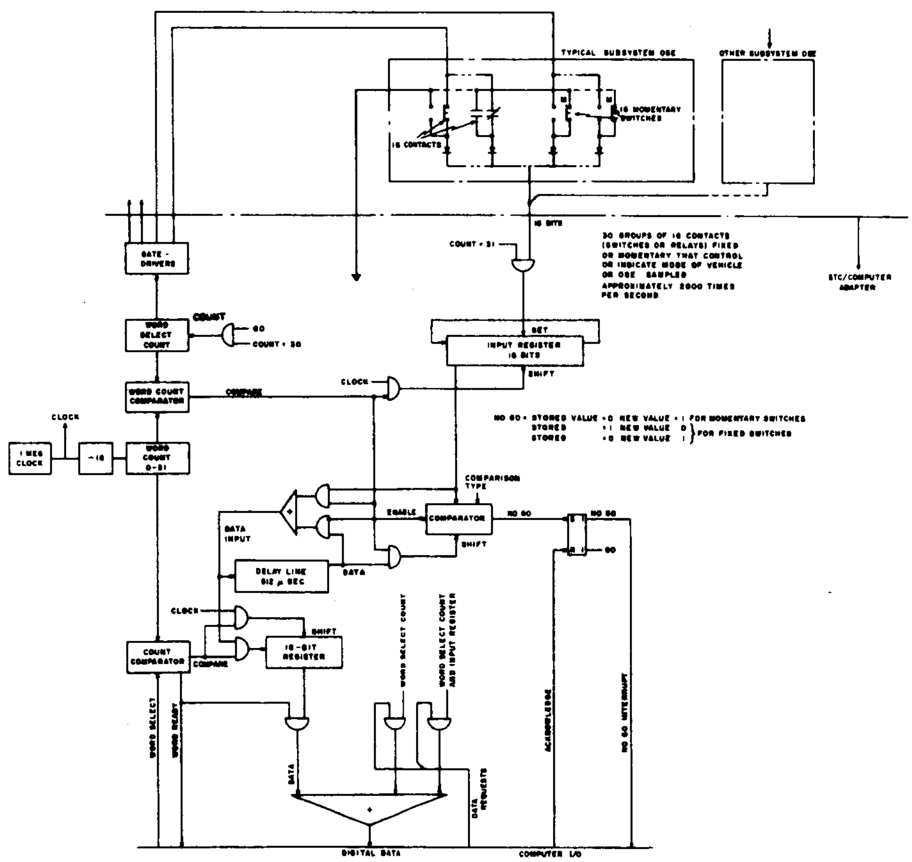

Figure 3-6. Switch/Mode Monitor

of the groups and comparing the 16-bit words to a programmed state, the computer can establish that the mode is either correct or not in order to proceed, and if not correct, why it isn't.

3.7 Teletypes. The STC contains eight typewriters, controlled by the CDS, that are to be used by the various subsystem engineers to obtain a hard copy of the data associated with their subsystem, an analysis of that data, messages, etc. The subsystem engineers can also, by typing in particular request codes, obtain the data at their request and, to some extent, in a variety of forms. This data can be in engineering units, their limits, and whether they are in or out of limits or any other form or content as provided in the computer program. This data can be the TLM data and the hardwire data.

These teletypes are not expected to be used by the subsystem engineers to keep a running tab on their data. That data is available on their meters either from hardwire or by use of the reconstructed TLM data. Since the meters can be calibrated in engineering units and can indicate changing states better than the digital values, the meters are a preferred method. If the subsystem engineers desire a running tab in a printed format, a preferred method would be by use of additional alphanumeric displays similar to that used by the test conductor.

3.8 Test Conductor's Alphanumerical Display. The test conductor's console shall contain a cathode ray tube and its associated controls, on which, groups of data and comments from the CDS will be presented to the test conductor. This display as such has a major interface with the computer. 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 40 groups of 15 lines of data each (i.e., each group consists 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 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: (a) values, identified in English; (b) values, in engineering units or as desired; (c) limits, in engineering units or as desired; and (d) out of limits, indicated by a blinking asterisk preceding the line.

Table 3-1 illustrates a typical display of this type of data.

Each half display will require 6 bits per character, 40 characters per line, and 15 lines per half display, which equals 2600 bits per half display. For 40 half displays this requires a total storage of 40 x 2600 = 104,000 bits of storage. For a 24-bit word, this implies approximately 4400 words of storage to satisfy the total format function. However, at any instant only the format data required by the requested two 1/2 pages need be available (i.e., approximately 220 twenty-four-bit words). By specifying a consistent format and thereby eliminating the requirement to store the blanks between limits and values, etc., the amount of computer storage can be reduced by almost one half again.

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 output need be 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 which two groups are

TABLE 3-1. TYPICAL TCC ALPHA-NUMERIC DISPLAY

Power SS	LL	VAL	HL
Unreg Bus	39.0	40.1	41.0
Reg Bus	27.5	28.3	28.5
*Inv 1	110	109	112
Etc.			

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 an 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. Note, the alphanumeric display contains its own circulating memory to maintain the display.

3.9 Computer Program Criteria. There are basically two approaches to programming the CDS for systems test. One is to program it such that it can establish all pertinent input conditions past and present, and from these, establish how the Spacecraft should be performing at that instant. This approach allows the system to be tested with essentially random inputs, but it is extremely difficult to write such a computer program. The second approach is that the system test is a series of a rigid sequence of events which cannot be performed out of sequence within themselves but the various series can be run in a variety of sequences. This sequence is much easier to program and is the approach recommended.

This sequential approach, whether it is a performance test or a mission profile test, allows the computer to know at any instant what the existing status of the Spacecraft and the OSE should be and what change is to be made next and in many cases exactly when that change is to occur. It is primarily those changes that are to be detected as occurring. Since most of these will be initiated via the spacecraft command functions, only one change can take place per second in that mode. The CDS will not be expected to find transients below levels that cause misoperation (continuous high frequency strip chart recording is the only way of obtaining this on many signals). The CDS program (by use of Subsystem OSE capabilities) should, however, be able to detect transients that cause misoperation such as improper squib operations, transient value operation, mode changes, etc. It should also be programmed to verify that single faults are uncovered (i.e., when a command is executed, other commands with a 1-bit variation from that should be verified as not having been executed).

The computer should record on tape and via the test conductor's line printer, the results of the tests and not just the data from which the results were obtained. Much of the data that the computer receives need only be limit checked (most to fixed limits as a function of existing status, others as a function of some other value) and then can be accumulated such that an average value could be obtained later or some other adequate data compression technique (which may include limit checking and discarding, if good). Note that the raw data recording is gathering all of the available data as backup material.

If the computer detects that the system (Spacecraft under test and OSE) is not in the mode it expects to find it, this should be treated as a failure and the tests held, warnings given, and the sequence reduced to monitoring the critical (safety) parameters until it is instructed to try again. These considerations allow the computer to have in memory, only the test sequences now being executed, the program for the current alphanumeric display to the test conductor, and any emergency or shutdown type of sequences.

The CDS is operating in conjunction with the TCD computer. The TCD will be doing the majority of evaluation of the TLM data, only interrupting the CDS on No-Go status, and giving data to the CDS only on the latter's request. This interface is discussed in greater depth in the "Computer Data System Section", CII VC260FD103.

3.10 Timing. There are essentially two types of timing signals required by the Subsystem OSE; one to indicate the time of day and the other to indicate the time in a particular test sequence. This last time must be displayed and should be recorded on strip chart recordings, etc. In order to prevent each Subsystem OSE from providing a time code translator, the CDS will be used to set the test time (an arbitrary time proceeded by data that indicates which test sequence is being performed) using the digital data input circuit. Then, a 1-second pulse from the central time system will automatically update each console time register. The same time code will be outputted serially by the STC equipment such that recording can also be accomplished.

The CDS will use 1-millisecond pulses from the central timer to measure time increments in conjunction with direct control or from the analog comparators as discussed in Section 3.3. The accumulated count will be read into the computer as digital data.

4.0 SAMPLE TEST SEQUENCE

4.1 General. The following sample test sequence is included to indicate how all of the above is applied to a typical problem (simplified). The problem or test sequence selected is based on the events leading to the attitude control maneuver sequence portion of a mission profile test. The functions chosen were to illustrate the use of the STC and as such are not to be construed as indicating that this is a proposed sequence.

As initial conditions, it is assumed that the vehicle is in a simulated cruise mode consisting of:

a. Radio system in noncoherent mode.

b. Attitude control in cruise mode in simulated closed loop (i.e., jet gas flow detected and assumed as applying constant force/acceleration).

c. Ground power simulation of solar array.

d. Battery simulators used.

e. Test connectors/umbilical, etc., connected.

f. TLM homing feature included to illustrate its usage.

4.2 Sequence. Table 4-1 is a sample test sequence.

4.3 Summary. It should be noted that the variations of parameters in the tests are typical and before the total sequence is complete, all other parameters and combinations would be done at least once if not several times.

TABLE 4-1. SAMPLE TEST SEQUENCE

Step No.	Test Step Description	Subsystem OSE Control	Subsystem OSE Monitor	CDS Control	CDS Monitor
1.0	Establish RF phase lock				
1.1	Measure received noncoherent frequency and power		Radio		Obtain digital value from Radio OSE
1.2	Set ground transmitter frequency to higher value than will be accepted by vehicle	Radio	Radio		Verify as above
1.3	Switch ground transmitter on to vehicle at nominal STC power	Radio			X
1.4	Home TLM to word monitoring AGC of proper receiver		Radio TLM	X	X
1.5	Upon command from test conductor, slowly lower the transmitter frequency; stop when lock is indicated using TLM value read on Radio Subsystem OSE	Radio			
1.6	Measure ground transmitter frequency at lock	Radio			As in Step 1.1
1.7	Measure vehicle transmitted frequency and power	Radio			As in Step 1.1
1.8	Reduce ground frequency to nominal upon instruction from test conductor	Radio	Radio		X
1.9	Check TLM values		X		X
2.0	Obtain PN code lock up				
2.1	Adjust PN code frequency to higher value than can be locked up		Command	Via Command OSE	Obtain digital value via Command OSE
2.2	Switch code on to ground transmitter			Via Command OSE Radio	
2.3	Home TLM to command detector lock word			TLM OSE	X
2.4	Reduce PN code frequency until "lock"			Via Command OSE	
2.5	Measure PN frequency		Command		As in Step 2.1
2.6	Reduce PN code to nominal			Via Command OSE	
2.7	Verify TLM indication		Command		Via TLM OSE
3.0	Test data integrator and dump circuit				
3.1	Send 1-0-1-0 data stream of nominal characteristics--verify proper detection at output of spacecraft detectors	Command Radio	Command	Via Command	Via Command
3.2	Adjust phase relationship between PN code and data stream	Command		Via Command OSE	
3.3	Continue changing phase until data is consistently misdetected (1-bit error/10)		Command		Via Command OSE
3.4	Obtain phase error for Step 3.3		Command		Via Command OSE
3.5	Return to nominal				
4.0	Command decoder tests (verify 111000 preamble detector)				
4.1	With lock established send correct preamble followed by OSE test command code (a command that performs no vehicle function but is brought out via test connector)	Command	Command	Via Command OSE	Only if faulty bits are detected by Command OSE
4.2	Detect command executed in Step 4.1		Command		Via Command OSE
4.3	Send preamble of 111001 followed by OSE test command bits			Via Command OSE	
4.4	Detect no operation from OSE test command		Command		Via Command OSE
4.5	Repeat Steps 4.3 and 4.4 using 111010-111100-110000-101000-011000 preambles.				
5.0	Load command into C&S				
5.1	Load commands into the C&S memory with logically significant time to go labels	TCD Radio		Via TCD	
5.2	Verify reception by reading while loading & reading out after storage		Command		Only if no-go
5.3	Commands to include: Gyro heaters on; Gyro motors on; Switch to inertial mode; Pitch turn start; Pitch turn stop				
6.0	Gyro heater command execution				
6.1	Shortly before gyro heaters are to be turned on, obtain gyro temperature using TLM homing			Via TLM	TLM Decommand
6.2	Home TLM to raw power current monitor measure ΔI when heater power turned on	TLM		Via TLM	TLM
6.3	Monitor command word executed signal, verify time		C&S		Via C&S
6.4	Verify that no other commands were executed varying by one bit from gyro heater power on. Use TLM where possible, operate subsystem in manner to detect mode when TLM does not monitor it adequately.				Via TLM & other Subsystem OSE
6.5	Measure ΔI in ground power at Step 6.2 (TLM and OSE)		Power		Via Power Subsystem & TLM
6.6	Obtain, by TLM homing, gyro temperature every 10 seconds until they stabilize (i.e., reaches a peak)				Via TLM & time
6.7	Obtain temperature cycle period				
7.0	Gyro motors on				
7.1	Home TLM to 400 cps current monitor prior to time for gyro motor on	TLM		Via TLM	

TABLE 4-1. SAMPLE TEST SEQUENCE (Continued)

Step No.	Test Step Description	Subsystem OSE		CDS	
		Control	Monitor	Control	Monitor
7.2	Obtain ΔI in 0.5 second for 60-second increments after command is executed				Via TLM
7.3	Verify no other commands executed as in Step 6.4				
7.4	Measure gyro temperature cycle as in Step 6.7				Via TLM compare with Step 6.7
7.5	Monitor gyro output signals via hardwire		G&C		
8.0	Switch to inertial mode				
8.1	Detect command execution at proper time		C&S		
8.2	Verify sun sensors no longer in control by inserting "solar" errors	G&C	G&C	Via G&C	Via G&C
8.3	Verify star sensor as in Step 8.2	G&C	G&C	Via G&C	Via G&C
8.4	Verify gyros are in control by inserting errors (via test connector) and obtaining solenoid operation	G&C	G&C via hardwire & prop OSE	Via G&C	Via G&C
8.5	Verify no other commands executed as in Step 6.4				
8.6	Verify TLM indicators				Via TLM
9.0	Pitch turn				
9.1	Set G&C into closed loop mode	G&C	Via G&C		Verify
9.2	Verify by inserting disturbing rate and observe nulling out automatically. Set position and rate error signals to 0° and 0°/sec respectively.	G&C	Via G&C	Via G&C	Via G&C
9.3	Verify pitch turn start time		C&S		Via C&S
9.4	Obtain turn polarity and rate by monitoring OSE close loop rate--integrate rate to obtain final position later; also integrate OSE torque current		G&C		Via G&C
9.5	Reduce ground power voltage in straight line manner at start of Step 9.3 at precalculated rate	Power		Via Power	
9.6	Obtain battery cut in voltages (3) by using OSE battery current monitors		Power		Via Power
9.7	Verify no other commands executed as in Step 6.4				
9.8	Verify TLM				Via TLM
9.9	Detect stop pulse command, verify time between Steps 9.3 and 9.9		C&S		Via C&S and time
9.10	Monitor stop jet cycling of G&C		G&C		
9.11	Obtain final position error by integrated rate errors and OSE torque current signals in Step 9.4		G&C		Via G&C
9.12	Verify no other command executed as in Step 6.4				
9.13	Verify TLM				

During the above tests, certain signals are being monitored and checked (by the Subsystem OSE and the CDS) essentially continuously. These would include such things as input currents, voltages, temperatures, mode indications, etc.

The test is preplanned and follows a rigid sequence in which only a few functions are altered at any instant and it is primarily those functions that are to be verified. Any transients must be of a sufficient magnitude to cause a mode change or to cause an output signal to alter. Faster transients or lower level ones must be detected with local recorders.

The subheadings each consist of a test step that provides a convenient place to hold the test sequence (and vehicle timing) if that is desired. Within each test step there are many times in which a hold could not be tolerated. If a hold is required during that period, special provisions in the sequence must be provided.

Certain self-check features would have been incorporated such as the ground transmitter could have been connected to the ground receiver and tested prior to switching to the vehicle, the gyro torque generators could be switched to a ground load and verified, etc. This capability should exist and should be used as governed by an over-all test program.

5.0 <u>STC LAYOUT</u>. Figure 5-1 shows a proposed layout of an STC. The number of racks indicates the current estimate of the various Subsystem OSE consoles required to support all of the tests within the STC, as well as the STC peculiar equipments. A provision for Science and Capsule OSE has also been estimated and incorporated. The total number of racks is approximately 90 and results in a required area of 48 feet by 48 feet.

The CDS is shown as being a part of this layout. While a centralized computer complex of two computers could carry the load of three Flight Spacecraft checkouts, the cost reduction obtained by the elimination of one set does not seem worthwhile at this time because of the following: (a) the inconvenience of having to coordinate in detail the activities of the three STCs (in order to ensure that the computers can handle the peak loads), (b) the fact that the time sharing multicontrol computer program will be more difficult to write and verify, and (c) since it is desirable to treat the computer as a part of the test complex rather than as a data processing center and keep it directly under the cognizance of the affected test conductor.

Due to the large size (internally to the STC, as well as externally), the number of interfaces, etc., it will be difficult to meet the requirement that the entire STC should be able to be moved, set up, and verified in a shorter period of time than the Spacecraft under test can be moved. One alternative is to mount the STC in vans. This is an unsatisfactory solution since the operators will be physically separated and will lose all feel for the test. This also will be a depressing atmosphere in which to work for long periods. Another solution is that the STC peculiar equipment can be treated somewhat like a facility and only move the Subsystem OSE with the vehicle. This "facility" could be verified prior to the move which greatly reduces the interfaces that have to be reverified when the Subsystem OSE arrive. This would increase the number of STC peculiar equipments by approximately two sets, and yet, to a high degree, maintains the STC-Spacecraft interface under test. This aspect of the STC is one of the major problems to be solved early in the Phase II effort.

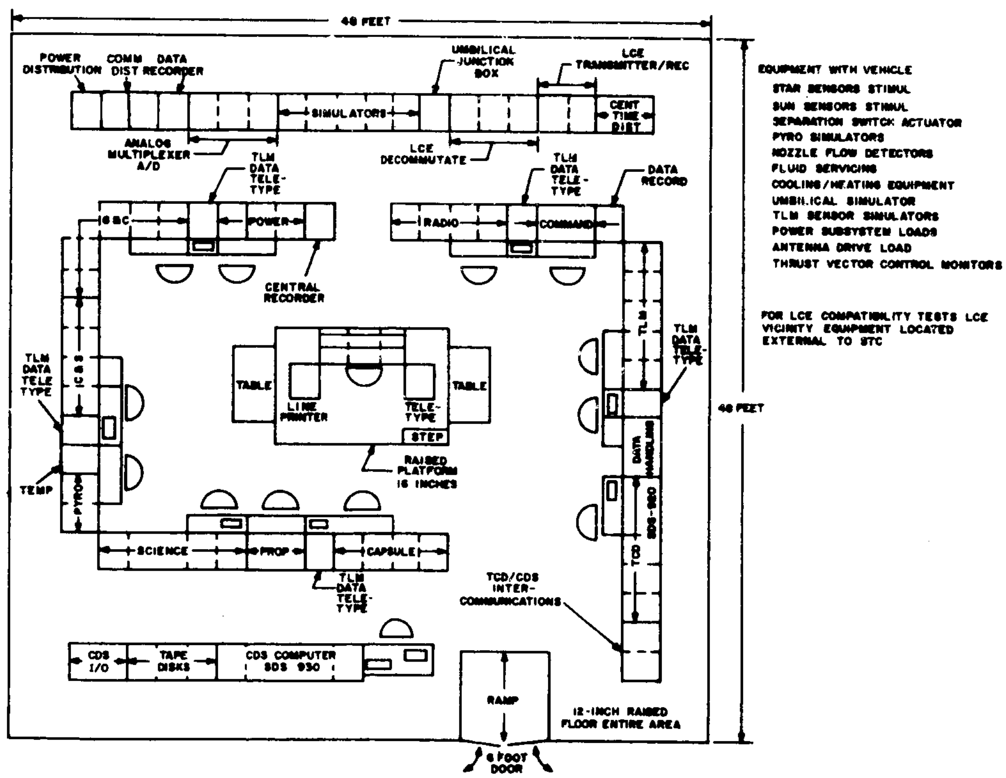

Figure 5-1. Proposed STC Layout

SYSTEM TEST COMPLEX AND SUBSYSTEM OSE EQUIPMENT GROUPS

FUNCTIONAL DESCRIPTIONS

FUNCTIONAL DESCRIPTIONS OF STC AND SUBSYSTEM OSE

The following sections are functional descriptions of the equipment groups within the recommended VOYAGER System Test Complex. Equipment peculiar to the System Test Complex, and Subsystem OSE are included.

VC260FD102

SYSTEM TEST COMPLEX ANCILLARY EQUIPMENT

1.0 SCOPE. For the VOYAGER System Test Complex (STC) to provide the capability of performing system tests, ancillary equipment will have to be provided within the STC. The equipment required is timing, recording, power distribution and voice communications. This document contains functional descriptions of this ancillary equipment.

2.0 TIMING. The STC will have a requirement for two types of timing signals, time code signals for time tagging and correlating information recorded throughout the STC, and synchronizing signals to be used for the processing (multiplexing, gating, etc.) of signals throughout the STC.

2.1 Time Codes. The STC will have a requirement for three different time code signals: two real-time code signals (station time) and the countdown-time code signals. Two separate real-time codes will be required by the STC in order to satisfy the varied station time distribution requirements (magnetic tape recorders, lower bandwidth recorders, Computer Data System (CDS), etc.).

There are a number of time codes available for satisfying STC station time requirements, and it is premature to select specific codes at this time. For purposes of explainlng implementation and distribution requirements, however, IRIG B and AMR B-1 time codes, have been selected as typical examples.

In addition, the STC requires a capability for the playback of magnetic tape data on a subsequent date when recorded time signals will be used for time correlation of events. Recorded time signals, reproduced with the playback of previously recorded data, are defined as playback time.

2.2 Implementation of Timing Functions. The Central Timing System (Figure 2-1) provides timing and synchronization signals for system and environmental tests at the factory and the Eastern Test Range (ETR). As shown in the diagram, a Central Time Generator and a Countdown Time Generator will distribute time signals to the test complexes. The distribution interface at each STC will be a Timing Isolation Unit (TIU). The timing signals are distributed to a TIU at each system test complex so that isolated outputs may be supplied to each OSE subsystem console which uses the timing signals. The TIU's provide either parallel or serial outputs.

To allow for reduction of tape 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.

2.3 Timing Interface Characteristics. The timing interface for each STC will be via the TIU. Interface signals associated with the TIU are defined in the following paragraphs.

2.3.1 High Resolution Time Code. A high resolution (± 1 millisecond) serial time code (IRIG B) will be provided to the TIU from the Central Time Code generator. This code will be amplified as required and otherwise conditioned in a time code distribution unit for parallel and serial distribution throughout the STC. All, or portions, of the format will be distributed depending on requirements of the receiving component.

2.3.2 Time Code for Lower Bandwidth Recorders. The Central Time Code Generator will also provide a lower-resolution time code (one pps AMRB-1) to a TIU Time Code Distribution

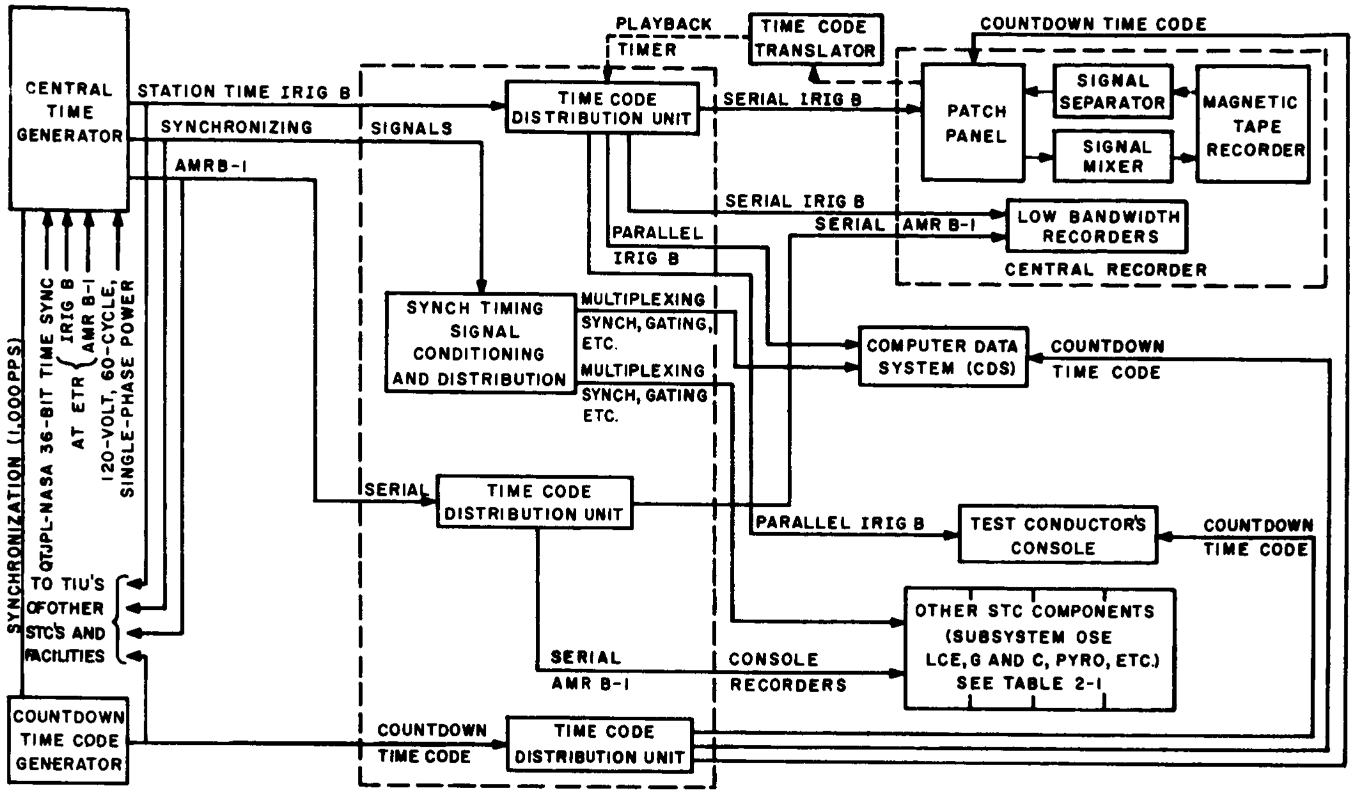

Figure 2-1. Central Timing System

Unit. This unit will perform the same functions performed by the Time Code Distribution Unit associated with the high resolution time code.

2.3.3 Countdown Time. The TIU will receive a countdown time code from a central countdown time generator. This code will also be conditioned and distributed via a Time Code Distribution Unit located in the TIU.

2.3.4 Timing Isolation Unit Interfaces with STC. The TIU will distribute the timing signals discussed in the previous paragraphs to the STC components requiring these signals. Table 2-1 indicates present requirements of the STC.

2.4 Performance Parameters. The TIU's provide an isolated interface between the Central Time Generator 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.

3.0 RECORDING. VOYAGER System Test Complex has a requirement for recording considerable amounts of data during tests. The majority of this data must be time correlated in order to provide a test log, and to verify satisfactory performance by post-test analysis of this data. Present requirements indicate need for two different recording capabilities: first,

TABLE 2-1. TIU/STC INTERFACE CLASSIFICATION

STC Component	High Resolution Time Code	Lower Resolution Time Code	Synchronous Time Signals	Count-down Time
Power Subsystem OSE		X		
Computing and Sequencing Subsystem OSE		X		
Guidance and Control Subsystem OSE		X		
Radio Subsystem OSE (Including Command MDE and TLM MDE)		X		
Telemetry Subsystem OSE (Including TLM Demodulator)		X		
Command Subsystem OSE (Including Command Modulator)		X		
Data Storage Subsystem OSE		X		
Structural and Mechanical Subsystem OSE				
Pyrotechnics Subsystem OSE		X		
Temperature Control Subsystem OSE		X		
Science Subsystem OSE		X		
Propulsion Subsystem OSE		X		
TCD	X			X
Computer Data System	X			X
TLM Data Decoder		X	X	
Multiplexer A/D (Subsystem OSE Analog Signals)				
Test Conductor's Console	X		X	X
LCE Console		X		
STC Recorder, Central Recorder and Data Recorder	X	X		
Simulators (Capsule, Propulsion etc.)		X	X	

a recording capability similar to that used on Mariner C providing a real-time recording/readout of analog signals in the dc-to-1500 cps frequency range; second, a recording capability for higher frequency data (unprocessed telemetry and command pulse trains, voice, etc.) in the dc-to-300 kc frequency range.

3.1 Implementation of Recording Functions. The recording functions described above will be satisfied by providing two types of recording installations as shown on Figure 3-1.

One installation, a Central Recorder, will record the lower-frequency analog signals. The second installation, Data Recorders, will provide magnetic tape recording of the higher-frequency signals and will consist of two separately located, but identical, recording facilities.

3.1.1 Central Recorder. The Central Recorder (CR) should be capable of monitoring approximately 50 analog functions simultaneously. The data will come from the vehicle via the subsystem OSE consoles, and 300 functions will be available for monitoring via a patch panel.

This capability is extremely flexible because all functions are available to the CR and only a repatch is necessary to monitor any data point. As shown in Figure 3-1, variable gain amplifiers and calibration controls will be provided so that proper scale factors, source isolation and impedance matching can be accomplished. The CR is expected to 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 900 cps; the d-c sensitivity will be 10.7

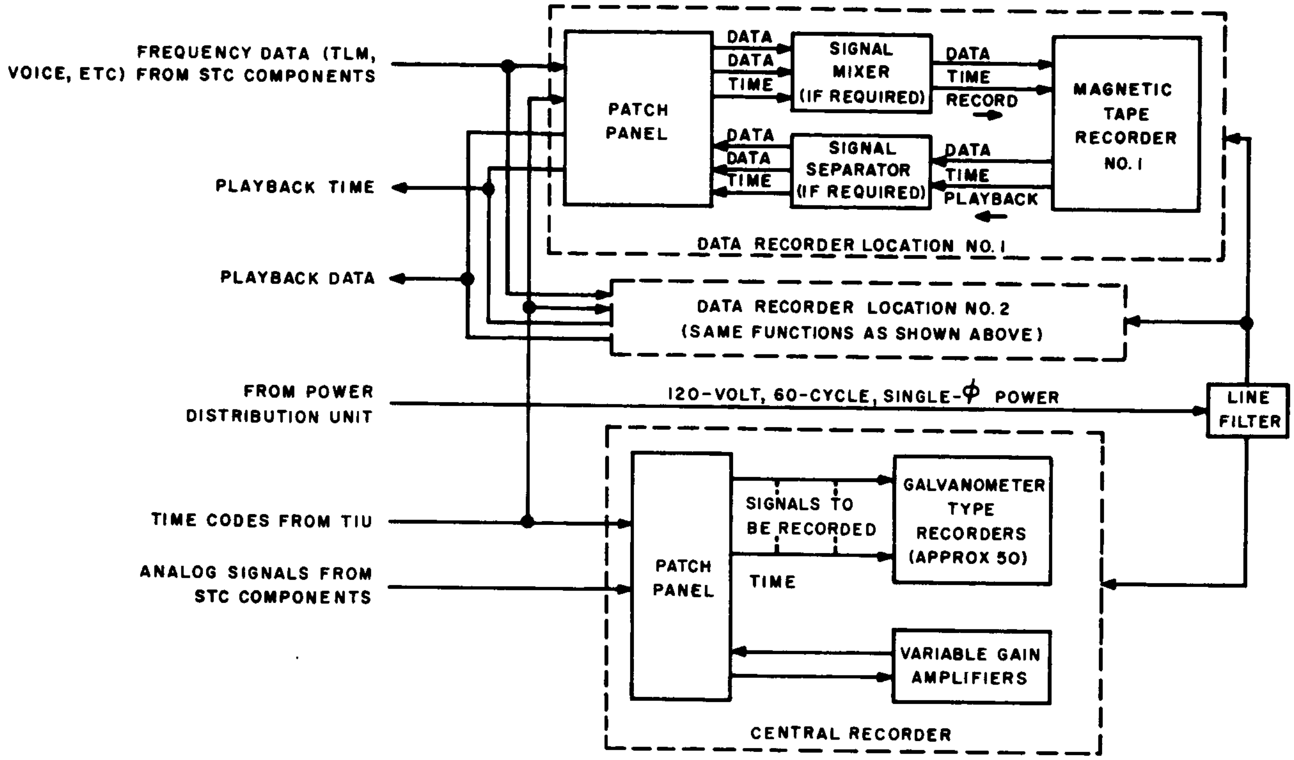

Figure 3-1. STC Recording Functional Diagram

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.

It is anticipated that 48 galvanometer controls will be provided which will handle from 150 volts down to the minimum voltage required for a particular galvanometer deflection. Additional wide-band (0-10 kc) 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.

3.1.2 Data Recorders. The Data Recorders are expected to be two 14-channel magnetic tape recorders with an upper limit frequency response in the neighborhood of 300 K. Recording channel requirements (Direct, FM, PDM) have yet to be determined. Similarly, ancillary equipment requirements, such as the signal mixing and separation equipment shown in Figure 3-1, have not been firmly resolved at this time.

There will be a requirement for a playback capability, as shown in Figure 3-1, in order to evaluate magnetically recorded data using STC subsystem OSE readout instrumentation (monitors, local stripchart recorders, etc.). In this mode, recorded timing signals will be sent to the time code translator associated with the timing isolation unit for distribution over STC timing lines to the OSE subsystem consoles.

4.0 STC POWER DISTRIBUTION AND ELECTRICAL GROUNDING REQUIREMENTS. Sixty-cycle power, three-phase (208 volts) or single-phase (120 volts) will be distributed to System

Test Complex (STC) components by means of a Power Distribution Unit. There will be one Power Distribution Unit for each STC. Any other power requirements within STC components (Subsystem OSE, CDS, Test Conductor's Console, etc.) will have to be provided by power supplies and inverters within the STC component which receive the primary sixty-cycle power as input.

The a-c power distribution and power grounding will meet the requirements of the general specification for performance and design requirements for VOYAGER 1971 Spacecraft System, VC270FD101. Incoming power, obtained from facility power (range facilities in the case of the STC in the Spacecraft Checkout Facility at the Eastern Test Range), will be regulated at the source to the degree negotiated with the power supplier. Should the 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 Unit will incorporate sufficient meters to indicate voltage, current, and power levels. The Power Distribution Unit will provide an emergency main-power distribution to STC components (if any) requiring emergency power for selected spacecraft functions in the event of main power outages. The Test Conductor's Console will have the capability to shut down primary 60-cycle power to all STC components in the event of an emergency.

A separate circuit breaker will be associated with each output power line to individual STC components. The Power Distribution Unit will have metering adequate to monitor frequency, voltage, current, and power.

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 close as possible to the spacecraft 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.

The Power Distribution Unit 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.

It is considered that the breakers and grounding provisions will be adequate protection against surges, overvoltages, and lightening strokes. Interlocks, 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 is the control of electromagnetic radiation, which tends to affect the performance of associated equipment. EMI suppression techniques should be emphasized in the console design.

5.0 STC VOICE COMMUNICATIONS. The System Test Complex will require the following voice communication facilities:

a. Intercommunications system connecting all operating positions in the STC (Subsystem OSE Consoles, CDS Console, Test Conductor's Console, etc.) and remote support areas (Environmental Test areas, Explosive-Safe Areas, etc.).

b. Public address system throughout test facilities.
c. Standard commercial telephone (facility-provided).

The various circuits associated with the inter-communications system will be interconnected via patching facilities located in a Communications Distribution Unit. Each STC will have a Communication Distribution Unit associated with it. Terminal intercom equipment will be located in each control console (Test Conductor's Console, LCC, Subsystem OSE Consoles, CDS console, etc.). The Communications Distribution Unit will also house the amplifiers and power supplies required to support the public address system.

The voice communications equipment (intercommunications and public address system) should also be capable at patching in Missile Operations System (MOPS) nets or their equivalent at the launch facility in order that prelaunch and launch status communications (such as countdown) are available at all STC and LCE associated facilities.

VC260FD103

COMPUTER DATA SYSTEM

1.0 GENERAL. The Computer Data System (CDS) is designed to acquire, process, record and display data from the VOYAGER Spacecraft System Test and to assist in the control of the test. During test, data is acquired from the spacecraft telemetry subsystem and from Spacecraft direct-access and umtilical circuits by way of the subsystem OSE. Additional data is gathered from various OSE such as the Test Conductor's Console, Time Code Generator, etc.

Figure 1-1 is a block diagram of the complete CDS in its preferred configuration. It consists principally of an SDS 930 computer, hereafter called the CDS computer, and its associated peripherals for overall test data processing and control and an SDS 920 computer, hereafter called the TCD computer, and its associated peripherals to perform the telemetry and command data handling task. The SDS 920, which will be used in a manner similar to the Telemetry and Command Data System (TCD) of the DSIF, will physically be the same computer used for telemetry decommutation as part of the Telemetry Subsystem OSE.

The SDS 920 computer is used so as to be able to perform considerable system tests using the MDE software required by the TCD at the DSIF. The Planned Capabilities of the DSM for VOYAGER (EPD-283) specifies the TCD as being two SDS 920, or possibly SDS 930, computers used redundantly. If the SDS 930 computers are used, this function within the STC would also be accomplished with an SDS 930. The exact redundancy is not deemed as being required by the STC and, therefore, only one TCD computer is contemplated.

The SDS 930 CDS computer was chosen largely because of the specified desire to have only one type of computer within the STC, program compatibility, between computers, etc. This was interpreted (considering that a TCD type computer is also required) as limiting the choice to essentially an SDS 920 or 930. Preliminary studies indicate that the SDS 930 capabilities will be required and since this is a competitive machine as to capabilities, reliability and cost, the choice was made.

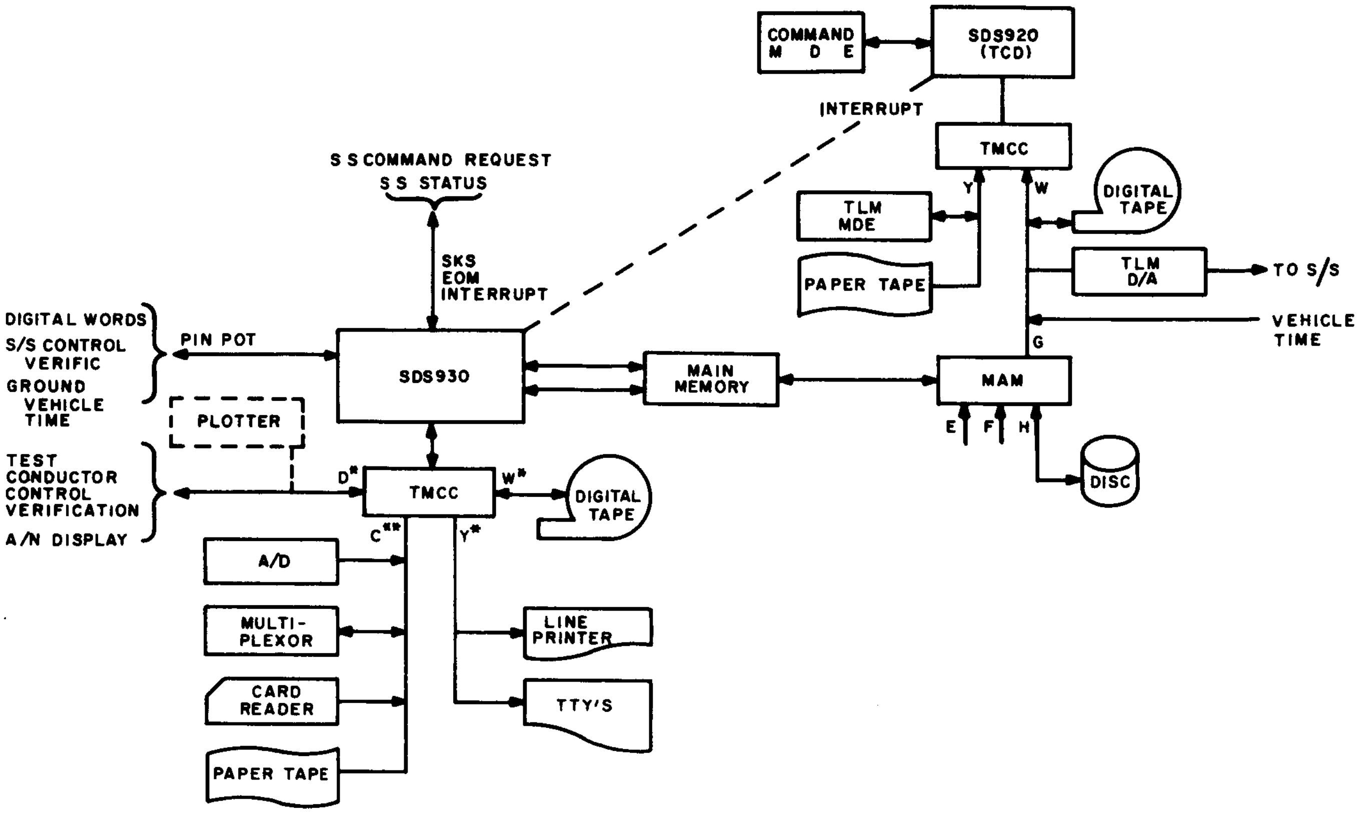

Figure 1-1. CDS Block Diagram

2.0 FUNCTIONS.

2.1 Data Acquisition. The CDS acquires data from various sources as indicated below.

2.1.1 Spacecraft Telemetry. The spacecraft PCM telemetry data will be handled in the STC by two paths. The first will be by the use of the TCD within the TLM Subsystem OSE. The TCD will perform the decommutation function, data analysis, formatting, etc. as normally done at the DSIF. A second path within the STC is specified as being required to decommutate the received TLM with equipment in addition to that furnished by the TLM Subsystem OSE. This decommutation can be done by the SDS 930 computer essentially as done by the TCD SDS 920 or it can be done by relatively simple hardware as described in STC System Level Description (VC260FD100).

The CDS SDS 930 in all cases will have access to the decommutated data and will perform analysis, conversion, and formatting as desired.

2.1.2 Analog Data. Continuous analog data will be received from the spacecraft direct-access and umbilical circuits by way of the subsystem OSE, as well as STC generated analog signals. Selected signals from this group will be sampled and digitized for entry into the CDS. The CDS will receive approximately 300 analog signals. Of these 300 signals, approximately 50 will be selected at any one time and sequentially digitized by an analog-to-digital converter. The scanning and conversion will be asynchronous at a maximum rate of

5000 samples per second and estimated at an average data rate of 2000 samples per second over the length of a test sequence.

2.1.3 Digital Data Signals. Digital input signals in addition to the TLM data include:

a. Command and Control Data from the Test Conductor's Console - These data will normally be entered into the computer as 24-bit parallel data words generated at the Test Conductor's Console. A priority interrupt will notify the computer that input data is available.

b. Control Verification Signals - The CDS generates digital output words for OSE control. The OSE being controlled will return a 24-bit parallel verification word to be examined by the control program.

c. Spacecraft/Ground Time - One or more 24-bit words will be available to the CDS at all times to indicate the current Spacecraft time and ground time.

d. OSE Status Signals - The status of the OSE will be available via approximately thirty 24-bit words generated by the OSE. They will indicate the status of approximately 500 items in the STC. The CDS will be informed, via priority interrupt, whenever these status bits change.

e. OSE Data - Much of the data obtained from the vehicle or internally within the OSE will be digital data. The computer will request this data as governed by the program and will input 24-bit digital words.

2.1.4 Internal Data Sources. The CDS also has the following internal data sources:

a. CDS Operators Control Console and Typewriter - They are normally used only during computer program checkout, program maintenance and troubleshooting.

b. Tab Card Reader - This is used mainly to enter the source program deck when a new program is being assembled. It will also be used during a test to make parameter changes as the test program is moved from bulk storage into core memory.

c. Punched Paper Tape Reader - Used primarily for program assembly time for post-test data processing.

2.2 Data Recording. Figure 1-1 shows the data recording in the system, which is of two types, raw data recording and processed data recording.

2.2.1 Raw Data Recording. Most of the raw data is in analog form. If the computer malfunctions, this recorder serves as a data source after the system is restored. It also serves as a data source for analysis of data not entered during the test. The recorder will be an analog, multichannel unit and will record the following items:

a. PCM Telemetry Bit Stress
b. Converted Analog Data
c. Spacecraft Command Signals
d. OSE Control Signals
e. Station Time
f. Spacecraft Time
g. Voice Annotations
h. Speed Control Signals

2.2.2 Processed Data Recording. Each computer will have one or more IBM compatible magnetic tape recorders. The recorders on the TCD's SDS 920 will be used to record:

a. All of the telemetry data along with an indication of data quality.

b. All of the commands sent to the spacecraft command system. The spacecraft command verification signal will also be recorded.

The recorders on the CDS SDS 930 will be used to record:

a. The results of the test, as a BCD recording of the data printed on the high speed printer at the Test Conductor's Console.

b. The converted analog data (which will be stored in a manner similar to the recording of the telemetry data except that the data will come from the spacecraft direct-access and umbilical circuits). This recording may also contain spacecraft digital data.

c. The test history, including the control request and command request initiated by the Test Conductor and Subsystem operators, and a record of parameter changes entered via the tab card reader, the computer console or the console typewriter.

2.3 Data Display. The CDS will drive five major types of display devices; two types of printers, two visual display devices and a data plotter.

2.3.1 High Speed Line Printer. The Test Conductor will have a high speed line printer adjacent to his control console. This printer will record the results of the test as it progresses. The format will be such as to provide a direct correlation with the test plan and it will indicate the nature of any malfunctions and record any deviations from the preprogrammed plan. During a nominally successful test, this printer will prepare the test data record for the buy-off package.

2.3.2 Character Printers. There will be a number of teletypewriters located about the STC available to the subsystem test operators and the Test Conductor. These printers will be used to display summary data, both telemetry and hardwire, and to send instructions to the various subsystem engineers.

2.3.3 Alphanumeric Display. The Test Conductor will have an alphanumeric device (probably a cathode ray tube) used to display a number of data words. These words will be data and comparison limits in engineering units. The display device will have a memory and regeneration control for maintaining the display which the computer will up-date.

2.3.4 Test Status Display. The Test Conductor will require certain information to constantly be displayed by the CDS at his console. This will include such data as the following:

a. Test Program Name or Number

b. Test Step Number (This is a subdivision of the Test Program)

c. System Status - Ready/Not Ready

d. Test Status - In Progress/Holding/Complete

2.3.5 Data Plotter. The Computer will have available a plotter to provide charts of one variable versus a second variable (and, therefore, one variable versus time). It can be used to plot converted or derived data during the test; however, its use in real time may be limited and it is provided primarily to produce graphs during the post test data processing.

2.3.6 Other Displays. The CDS also provides decommutated telemetry data in digital form to drive analog displays via digital-to-analog converters. This data will consist of seven data bits and identification bits in parallel. The TLM Decoder Equipment will store selected words and perform the digital-to-analog conversion to drive the analog displays on the subsystem consoles or use individual bits to drive status light displays.

2.4 Test Control Data. The CDS exerts control over the test by way of two different paths; the spacecraft command subsystem and the OSE consoles.

2.4.1 Spacecraft Command Link. When the CDS sends a command to the spacecraft, it will be via the Radio OSE or via hardline connection to the spacecraft. Command Verification will be received from the Spacecraft to indicate the command received and to permit comparison. The spacecraft command sequence will be included as part of the system test programming. It will also be possible for the subsystem OSE operators to request the CDS to send commands that are not in the preprogrammed test sequence via the above path.

2.4.2 OSE Control Link. Each OSE console will provide the capability to have the CDS set up its circuitry in the same manner as though it were being set up manually. The CDS will output a 24-bit word with 6 bits of address and 18 bits of data. The data sent to the OSE will be returned to the CDS and verified before the control function is executed. This control loop will enable the CDS to perform many functions automatically which can also be done manually from the OSE console. All functions, however, will not be remotely controlled by the CDS.

2.5 Data Processing. The operations performed on the acquired data can be divided into two major parts; real-time data processing and post-test data processing.

2.5.1 Real-Time Data Processing. Normally the data processing which will be done in real time is that required for the successful completion of the test. Data will be acquired, translated into a form suitable for use by the control program and the display units and recorded on magnetic tape. In general, this means that the data will be acquired, transformed from the input code as required, limit checked, and stored in appropriate buffer areas for use by the various control, display and recording programs. In addition, various derived data will be required (limits for one data point may be a function of a second data point, the rate of change of some variables may be required, electric power and pneumatic budgets may be needed, etc.). These will be calculated and stored in appropriate buffers. Figure 2-1 indicates the type of control program which will be used during the real-time test processing.

2.5.2 Post-Test Data Processing. Post-test data processing can be more varied and complex than real-time processing as it is not constrained by test considerations. It includes that processing required to supply sufficient data for Spacecraft system engineers and subsystem engineers to evaluate the performance of the spacecraft. It should also be recognized that some of the following may be done in real time as machine time allows.

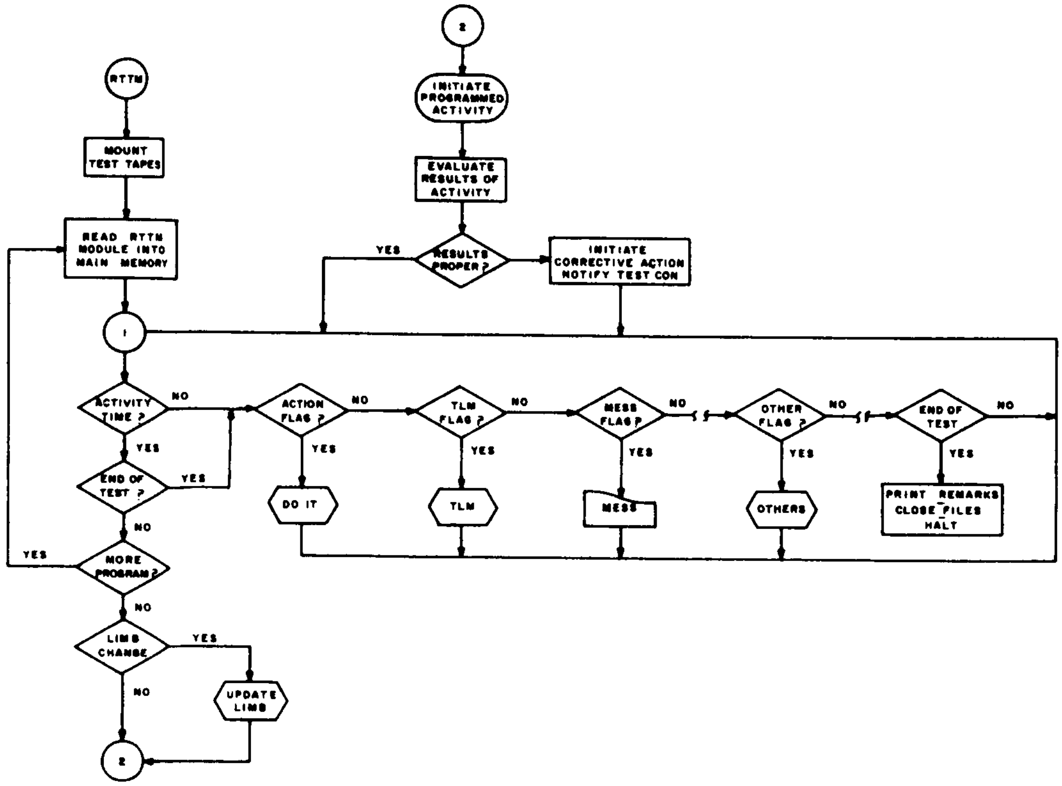

Figure 2-1. Real-Time Test Program (RTTM)

a. Statistical Analysis of the Spacecraft Data - The arithmetic mean and standard deviation of the sensor data can be calculated to assess the trend of data. This can include calculation of dispersion, skewness, etc., of the data to further define its characteristics.

b. Curve Plotting - As indicated earlier, some data must be reduced to graphic form to permit efficient analysis.

c. Calculation of Derived Data - Much of the data necessary for analysis may not be directly available from the raw data. This requires calculation of complicated variables using the raw test data.

d. Fitting Curves to Experimental Data - In many cases the parameters sought from a test are best obtained by fitting some mathematical function to the test data where the coefficients of the fitted curve are then the desired parameters. A typical example of this is the calculation of thermal constants of a piece of equipment by fitting an exponential curve to the temperature versus time data collected during thermal testing of the equipment.

A detailed knowledge of the post-test data processing is not available at present; however, the foregoing examples are typical of the type of processing anticipated.

3.0 DESCRIPTION

3.1 General. The CDS is made of a SDS 930 computer used in conjunction with a TCD SDS 920 computer. These computers are standard computers as covered in SDS literature and their characteristics will not be reiterated here.

3.2 Standard Computer Peripherals. The following standard pieces of computer peripheral equipment will be attached to the SDS 930 and the SDS 920 computers. The exact location of the peripherals with respect to the two computers and their input/output channels is shown in Figure 1-1.

3.2.1 Magnetic Disc File. A disc file mass memory system will be used primarily for test program storage. This will allow the complete test program to reside on the disc while the main core memory will contain only the specific program or step currently in process. The low access time of the disc and the direct memory access feature of the computer will allow for transfer of succeeding program to core without loss of test time. The disc unit is attached to the SDS 930 via Channel H.

3.2.2 Magnetic Tape Units. A magnetic tape unit controller and four magnetic tape recorders will be attached to each computer for recording processed data as described in Section 2.2. The controllers will be attached to the W-channel of each computer.

3.2.3 High-Speed Printer. A completely buffered output printer will be attached to channel Y of the SDS 930 to serve the needs of the Test Conductor. The printer is capable of printing at a speed of 600 lines per minute.

3.2.4 Teletypewriters. Several teletype keyboard/printers will be attached to channel Y of the SDS 930 to provide data to the subsystem engineers. The printers are capable of printing at a speed of 10 characters per second.

3.2.5 Tab Card Reader. A reader capable of reading 400 standard 80-column punched cards per minute will be attached to SDS 930 channel C. This unit will be used while assembling new programs and it will also be used to enter parameter changes during the test.

3.2.6 Paper Tape Punch and Reader. A 300 character-per-second paper tape reader and a 60 character-per-second paper tape punch will be attached to channel C of the SDS 930 and channel Y of the SDS 920.

3.2.7 Graph Plotter. A two-axis digital incremental plotter will be attached to channel D of the SDS 930. It is capable of plotting in increments of 0.01 inch in the X and Y axes at a rate of 18,000 lines segments per minute.

3.2.8 Console Typewriter. A console typewriter will be attached to channel Y of each system. These units will be used by the CDS computer operation to communicate with the system.

3.3 Special Purpose Computer Peripherals. The CDS communicates with the spacecraft and the remainder of the STC by means of a number of special peripheral devices. These devices are used to gather data and transform it into computer words for input and conversely to accept the computer output words and condition them for acceptance by the Spacecraft/OSE system. This equipment is discussed in detail in STC System Level Functional Description (VC260FD100) and will only be summarized below.

3.3.1 Analog Data Group. The analog data group consists of a patch panel programming system, signal conditioning circuits, an analog signal multiplexing system and an analog-to-digital converter.

A slow speed multiplexer is used to select 30 signals out of the 300 available and route the selected signals to the high speed multiplexer. The high speed multiplexer scans these 30 signals plus 20 signals which are permanently connected to it. The normal mode is to scan and record these signals continuously, with the computer taking in only the signals as required. The computer can also assume sequencing control of the multiplexer such that any sequence can be obtained. The analog-to-digital converter and high speed multiplexer can operate at a rate up to five thousand samples per-second.

3.3.2 Digital Data Group. The digital data group is used to gather the following data:

a. Command and Control Data from the Test Conductor's Console
b. Control Verification Signals
c. Spacecraft/Ground Times
d. STC Status Signals
e. Spacecraft and OSE Digital Data Words

3.3.3 Telemetry Decoder Equipment. This unit accepts the telemetry decommutated digital output and generates a group of analog signals representative of the data received from selected pins in the PCM telemetry subsystem. It consists of a number of digital storage registers and digital-to-analog converters.

4.0 SOFTWARE DESCRIPTION

4.1 General. The programs associated with the Computer Data System can be divided into the three groups described below.

4.1.1 Telemetry and Command Data Handling Programs. These reside in the SDS 920 where they operate on the telemetry data to produce the signals required for the SDS 930 and the Telemetry Decoder Equipment. They also receive command data from the SDS 930 for processing and output to the Command MDE Modulator. This program will be similar to those used by the TCD during the mission. Some considerable testing will be done with the identical program.

4.1.2 Real Time Test Program (See Figure 2-1). The real time test program will consist of a single module known as RTTM. RTTM will consist of an extensive executive and a number of interrupt routines. The executive is essentially the test sequence and it will specify a rigid series of events required to control the spacecraft and STC, collect the data, evaluate the data and determine the future course of the test. There will be alternate paths through the executive as well as emergency paths to be followed as a result of the evaluation of the data collected. The subroutines will be those required to collect the input data and generate the output message. Generally speaking the input subroutines will be entered as the result of a priority interrupt while entrance to the output routines will be controlled by the executive.

The anticipated size of the Real Time Test Program indicates that it will not be possible to have the entire program in core memory at any time. To circumvent this problem, it is planned to store the various subroutines in core along with a portion of the executive subroutine. The remainder of the executive will be stored in a disc memory which will have direct access to the main core memory. As the test progresses, additional parts of the executive will be brought into core and written over that part of the executive which has already been completed. It may prove convenient to allow small changes to the program each time the test is executed. These changes will take the form of parameter changes and will be called for from the tab card reader as each section of the executive is transferred from the disc to the main memory.

4.1.3 Post Test Data Processing Programs. The Post Test Data Processing Programs will take a form similar to the Satellite Data Processing System currently in use at GE-MSD. This program is a Master Control Program known as MACOP and it provides the capability of performing multifunctions on satellite data and producing the outputs on a single pass through the input data. Each function is programmed as a module and during execution of the system the control program is used to interconnect the modules and assure a continuous flow of data.

VC260FD104

SYSTEM TEST COMPLEX TEST CONDUCTOR'S CONSOLE

1.0 SCOPE. The Test Conductor's Console is required to provide the controls and displays needed by the Test Conductor to coordinate and control an integrated system test of the spacecraft. This document provides a functional description of those controls and displays.

2.0 FUNCTIONAL DESCRIPTION. The control and display functions required by the Test Conductor are shown in the functional block diagram given in Figure 2-1. There are basically three kinds of functions which have to be implemented: controls, continuous displays, and alphanumeric data displays. The recommended design approach is to implement these functions in three separate bays, on a one-for-one basis.

The alphaneumeric display is recommended as a part of the Test Conductor's Console since it provides the Test Conductor with convenient access to a considerable amount of test data while using a small amount of display area. It is recognized that cost and data processing tradeoffs may eliminate the alphanumeric display in final design configurations. In this case, the ancilliary equipment discussed later (high-speed line-printer and character printer) will have to be used by the Test Conductor as alternate data displays.

2.1 Controls. The Test Conductor's operational controls, and the verification displays intimately associated with them, are to be implemented in the Control Bay. This includes the following.

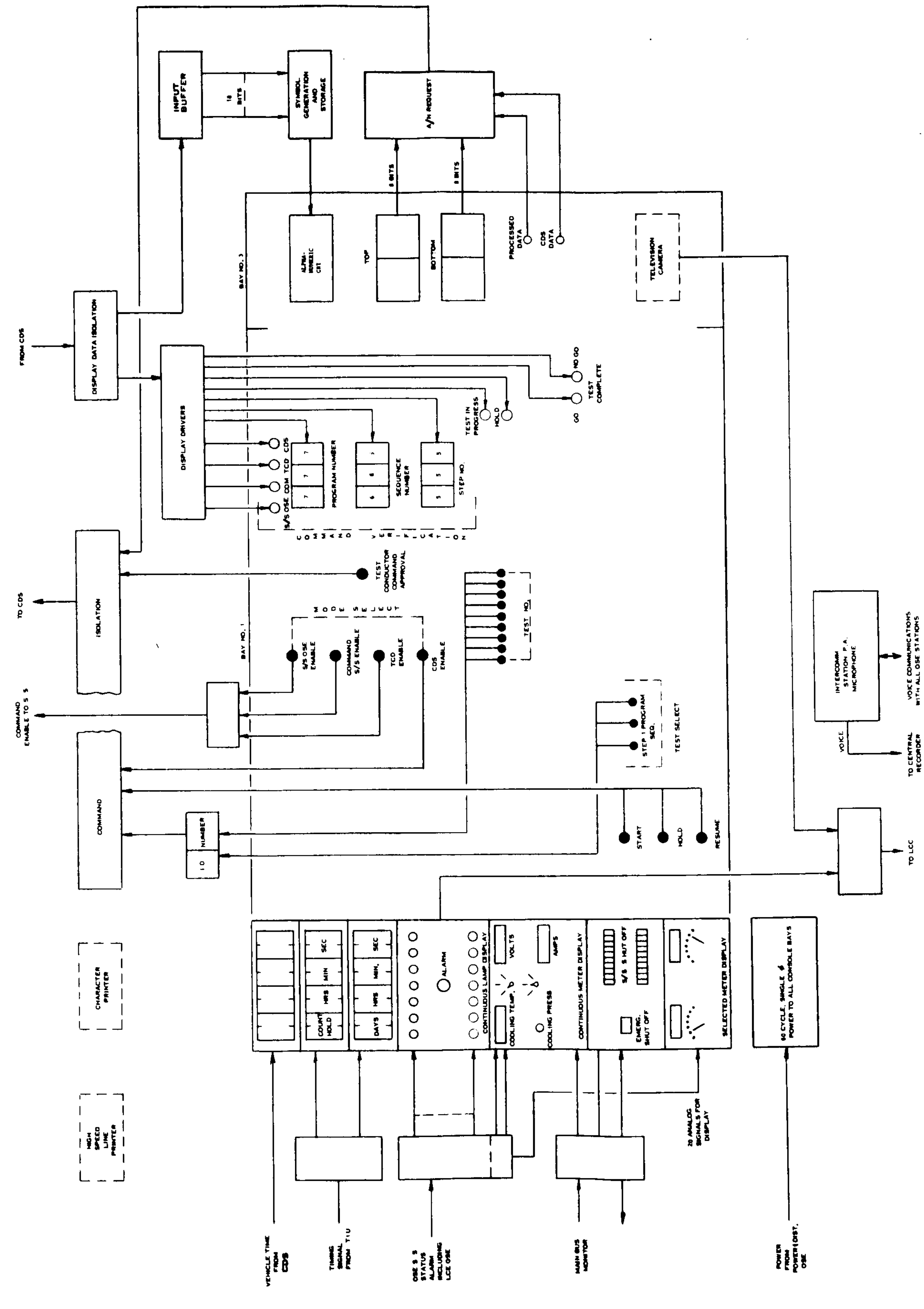

Figure 2-1. Test Conductor's Console (Candidate Layout)

a. CDS Command Controls - These controls will be used to instruct the CDS as to what tests to run, when to stop, when to resume after a program hold, etc.

b. Command Mode Controls - These controls will set up the point from which vehicle commands can be originated. This will include OSE subsystem consoles, CDS (with inputs via a, above), and TCD.

c. Communication - The Test Conductor will have an intercommunication panel which will allow him to talk and listen to any area concerned with the STC.

2.2 Continuous Display. This panel will allow the Test Conductor to obtain, directly, critical readings or those readings that summarize the general state of the system (Planetary Vehicle, STC and LCE). Some, or all, of these signals will also be sent (via the Test Conductor's Console) to the Launch Control Center (LCC). These include:

a. Continuous Meter Displays - Selected analog signals will be displayed continuously.

b. Continuous Lamp Displays - Subsystem status, cooling status, and other critical STC component parameters will be monitored via lamp displays.

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 three time displays, each to seconds. They will be STC station time and countdown time as obtained from the Timing OSE, and vehicle-time (binary code displayed in octal) obtained from the CDS via 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 Shutdown - This panel will contain an Emergency Power Off control. This 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.

2.3 Alphanumeric 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 40 spaces (characters) long, with any one of 60 possible characters 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 consists 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 by the CDS once per second if new data has been received. If an alphanumeric display is incorporated into the Test Conductor's Console, a facility-supplied closed-circuit television could be accommodated, if needed, to allow the transmission of the A/N display to the LCC.

2.4 Supplementary Equipment. In addition to the displays already mentioned, the Test Conductor's area will contain one high-speed line printer and one teletype (character) printer. These printers are controlled by, and considered a part of, the CDS peripheral equipment, but will be physically located in the vicinity of the Test Conductor's Console.

The High-Speed Line Printer will be used to record in near realtime 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.

The Character Printer will print 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.

3.0 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 the receiving end by observing its response to the switch operation). This equipment will, therefore, be verified by co-ordinating the interface equipment using self-test procedures. Calibration and verification are considered sufficient self-test of the displays.

4.0 PHYSICAL CHARACTERISTICS. The Test Conductor's Console will consist of three bays of an operator sit-down type. The displays and controls as discussed in Section 2 and illustrated in Figure 2-1, will be grouped on the sloping panels. A writing surface across the entire width should be provided with drawers for storage of manuals, records, procedures, etc.

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

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.

5.0 SAFETY. One of the Test Conductor's primary tasks is to ensure vehicle 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.

TABLE 6-1. TEST CONDUCTOR'S CONSOLE INTERFACE

Interfacing Equipment	Function	Input/Output	Remarks
Computer Data System	CDS Program Control	Output	
	CDS Program Control Displays	Input	A/N control, address and words
	CDS Command Approval	Output	Switch closure
	Alphanumeric Data Requirement	Output	
All Subsystem OSE	Status and alarm	Input	
	Command Enable	Output	Switch closure
	Communications	Input/Output	
	Selected Analog	Input	
Timing Isolation Unit	Time (station, countdown, digital)	Input	Two, parallel
Ground Power	Power	Input	
	Emergency shutdown	Output	
LCE	LCC Test Coordinator, Status	Output	Repeat of status displays (tone coded)

The Test Conductor also has control by 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 during a particular system test sequence. His other displays and controls are provided as gross vehicle safety type functions (i.e., ground cooling, power monitors, shutdown, status and alarm, etc.).

6.0 INTERFACES. The functional interfaces of the Test Conductor's Consoles with other equipment are summarized in Table 6-1.

VC260FD108

SYSTEM TEST COMPLEX SIMULATORS

1.0 SCOPE. It will be necessary to simulate a number of space vehicle interfaces during STC systems tests. To build up and verify the System Test Complex (STC), an umbilical simulator will be needed. In addition, certain components of the space vehicle will not be available for assembly into a complete system at the time it will be necessary to begin systems testing with the STC. The launch vehicle, propulsion (orbital insertion) subsystem, and capsule subsystem are examples where electrical interfaces will be required for STC tests before these components are mated with the Spacecraft bus. While engineering and proof test models may be provided for some of these components (capsule, propulsion), delivery schedules, availability (models will not be provided on a one-for-one basis with STC's), safety constraints, test/assembly conflicts, etc. will result in STC development and system test delays.

Simulator requirements identified at present are umbilical, propulsion, capsule, science and launch vehicle as shown in Figure 1-1. The simulators are to provide, in effect, a simulated planetary vehicle (or launch vehicle) electrical interface.

The propulsion, capsule, science and launch vehicle simulators are expected to be furnished to the spacecraft contractor. This document identifies the general functional requirements of these simulators. In addition, a functional description of the Spacecraft Umbilical Simulator is given.

2.0 PROPULSION SIMULATOR FUNCTIONAL REQUIREMENTS. The Planetary Vehicle has two propulsion subsystems: (1) Orbital (Mars) Insertion Propulsion, and (2) Midcourse

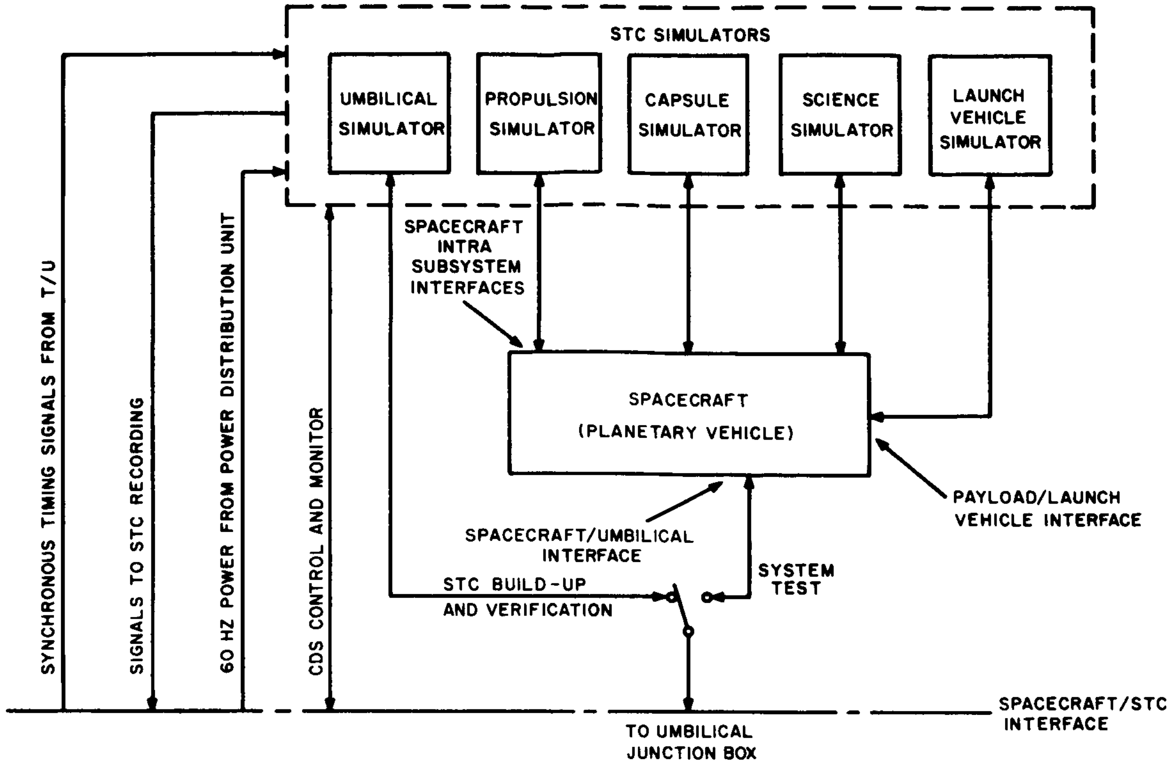

Figure 1-1. STC Simulators, Functional Diagram

Correction Propulsion. It is necessary to have an electrical simulator for the orbital insertion and midcourse correction propulsion assemblies to minimize test delays.

The types of signals and loads requiring simulation are:

a. Orbital Insertion Propulsion (Solid Propellant)

Arming - Motor load, armature "contact complete" discrete.

Ignition - Squib firing load.

Thrust Vector Control - Hydraulic motor load, pintle value feedback, G&C control loads.

Telemetry - Temperature sensors, pressure sensors, indicator and position sensors associated with arming, ignition, and thrust vector positioning.

b. Midcourse Maneuvering Propulsion (Hot Gas)

Actuation and Ignition - Solenoid valve loads.

Thrust Vector (jet vane) Positioning - Motor load (for G&C signals), feedback potentiometer signals.

Telemetry - Temperature sensors, pressure sensors, indicator and position sensors associated with activation, ignition and thrust vector positioning.

3.0 CAPSULE SIMULATOR REQUIREMENTS. The major simulation requirements are:

a. Radio relay (400 mc) signal and associated direct wire link.
b. Data Transmission Link (50 kilobit).
c. Load simulation for spacecraft power subsystem.
d. Discretes associated with separation.
e. TLM monitors.
f. Command data load and detection.

4.0 SCIENCE SIMULATOR REQUIREMENTS. (To be defined later)

5.0 LAUNCH VEHICLE SIMULATOR. Since the launch vehicle-to-payload (two planetary vehicles) interface is to be kept to a minimum it is expected that the Launch Vehicle Simulator will not be very elaborate. There will be a requirement for discrete signal simulation associated with separation and possible telemetry signal loading for signals passed to the launch vehicle for relay to the ground during the launch phase.

6.0 UMBILICAL SIMULATOR FUNCTIONAL DESCRIPTION. The principal function of the Umbilical Simulator is to provide an approximate simulation of the loads, current, voltages, etc., of a nominal spacecraft, to the Operational Support Equipment. This simulation is provided primarily to verify circuit operation and cabling. This is a test or precaution which equally applies 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 planetary vehicle umbilical.

The umbilical connections are simulated although other OSE-to-planetary vehicle connections (the direct-access connections) are not for the following reasons:

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. 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 earlier systems tests due to the increasing time constraint.

The simulation should be to the extent required to verify circuit operation and cabling. 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. The types of electrical signals associated with the umbilical interface (as defined at the present design level) are contained in the umbilical list (Table 2-2) of LCE Functional Description VC280FD100. A summary of this table indicates that the Umbilical Simulator will have to generate, or provide an electrical load for the following types of electrical signals to the extent required to verify cabling associated with the umbilical interface:

a. Radio Frequency (S Band) Modulated Signals - Up and Down Transmission link
b. Load and detect 400 mc signal
c. Generate Subcarrier (270 kc)
d. Generate and Switch Discrete Signals
e. Load and provide indication for discrete signals
f. Generate analog signals (telemetry, monitoring and gyro rate)
g. Provide simulated spacecraft power load

TABLE 6-1. UMBILICAL SIMULATOR INTERFACES

Interface (STC Component)	Function	Input/Output	Remarks
Radio Subsystem OSE	S-Band RF Signals (4) Radio Relay (400 mc)	Input Input	↑
Telemetry Subsystem OSE	Modulated Subcarrier (270 kc)	Output	
Computing and Sequencing Subsystem OSE	High Speed Check, Data, Sync } Digital Pulse Trails	Input	
	Kind of Word, Update Master Sequence, Data Abort, Δ V Abort, Alert TTG's, G&C Commands, PSP Commands, Present TTG } Discretes	Input	
Guidance and Control Subsystem OSE	Gyro Rate Signals (Analog)	Output	
Pyrotechnic Subsystem OSE	Measure Continuity (Analog) Separation Switch Monitor (Discrete) Simulate Separation, Arm Monitor, Arm Enable } Discrete	Output Output Output Output Input	Functions as defined by present design level
Science Subsystem OSE	Undetermined		
Capsule Subsystem OSE	Undetermined		
Propulsion Subsystem OSE	Tank Temperatures (Analog)	Output	
Power Subsystem OSE	Array/Battery Bus Voltage, Battery Voltage Monitors, Battery Temp. Monitors } Analogs	Output	
	Array Enable ON/OFF, Array Enable Monitor } Discretes Power (0-60 V dc, 15 amp)	Input Output Input	
Data Storage Subsystem OSE	Set Recorders to Launch Mode (Discrete)	Input	↓
STC Power Distribution Unit	Main Power for Umbilical Simulator Components	Input	60 Hz, single-phase power

Figure 6-1 is a block diagram of the umbilical simulator which will generate, or provide an electrical load, to verify the capability of transmitting these signals. As shown in the diagram, the simulator will have local controls and indicators associated with discrete switching, detection indicators, etc.

The Umbilical Simulator interfaces with other STC components are shown in Table 6-1.

7.0 PERFORMANCE CHARACTERISTICS.

The basic performance criteria for the simulators discussed is to provide electrical signals and loads that are required to test the spacecraft and need not represent operational interfaces. The simulators must, however, simulate marginal, as well as nominal interface characteristics.

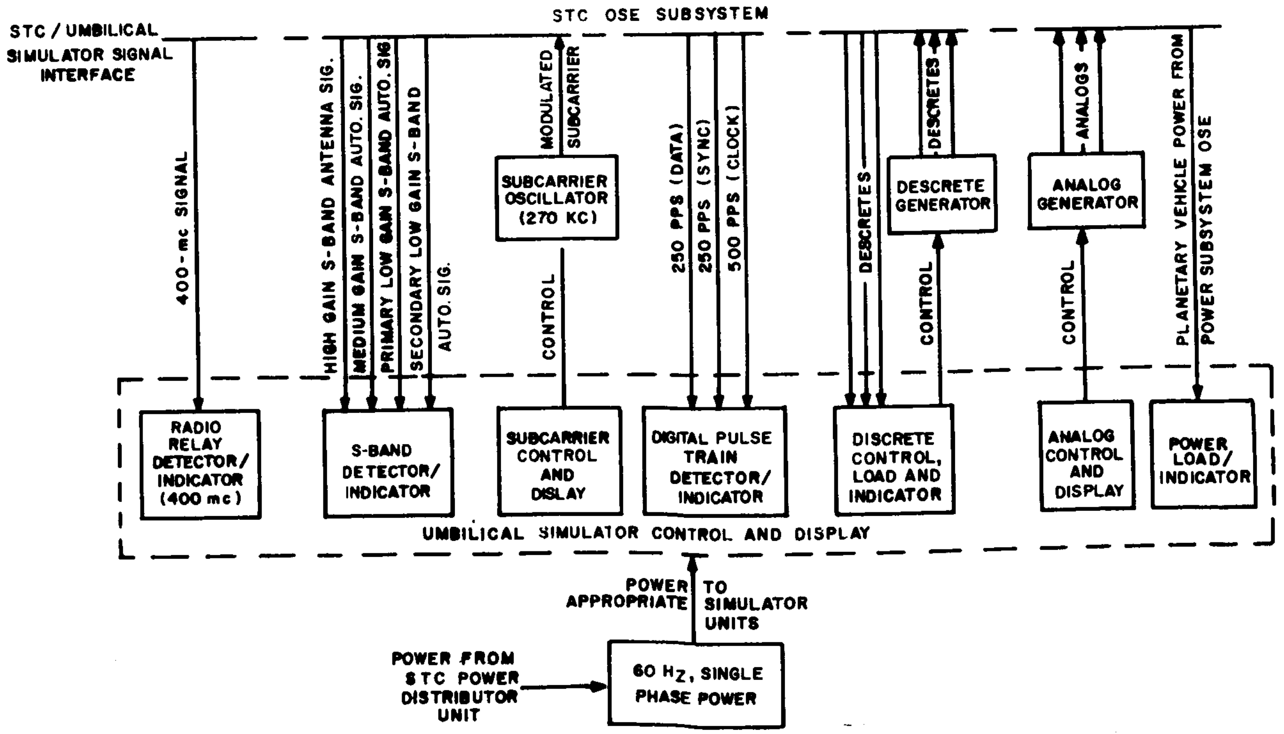

Figure 6-1. Umbilical Simulator, Block Diagram

COMMAND SUBSYSTEM OSE

1.0 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.

2.0 BLOCK DIAGRAM. A block diagram of the VOYAGER Flight Command Subsystem OSE, including interfaces, is shown in Figure 2-1.

3.0 REQUIRED FUNCTIONS. The VOYAGER Flight Command Subsystem OSE will perform the following functions:

a. Generate command words.
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.
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.

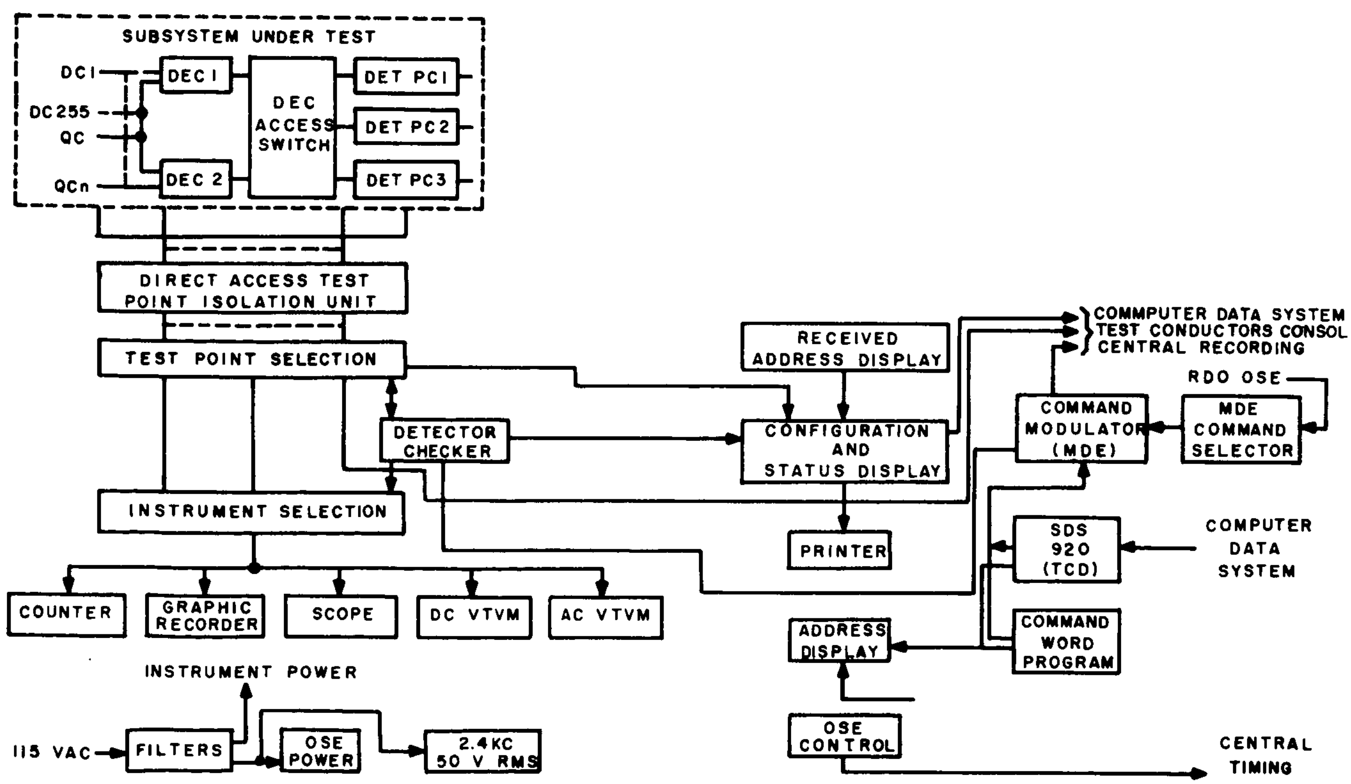

Figure 2-1. Command OSE System Test Configuration

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 Command 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.

k. Provide the capability of self-test.

4.0 SELF-TEST AND CALIBRATION

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 Command Subsystem. Printout 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 correlation with expected error. No anamolous response is shown by the flight subsystem. For example, grounding the isolated direct test point (DTP) corresponding to bit-sync would result in printout 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.

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.

5.0 SUBSYSTEM TESTS

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.

5.2 Detector/Program Control Unit Tests. With a detector/program controller functional element and the OSE in the configuration shown in Figure 5-1, the following tests can be made.

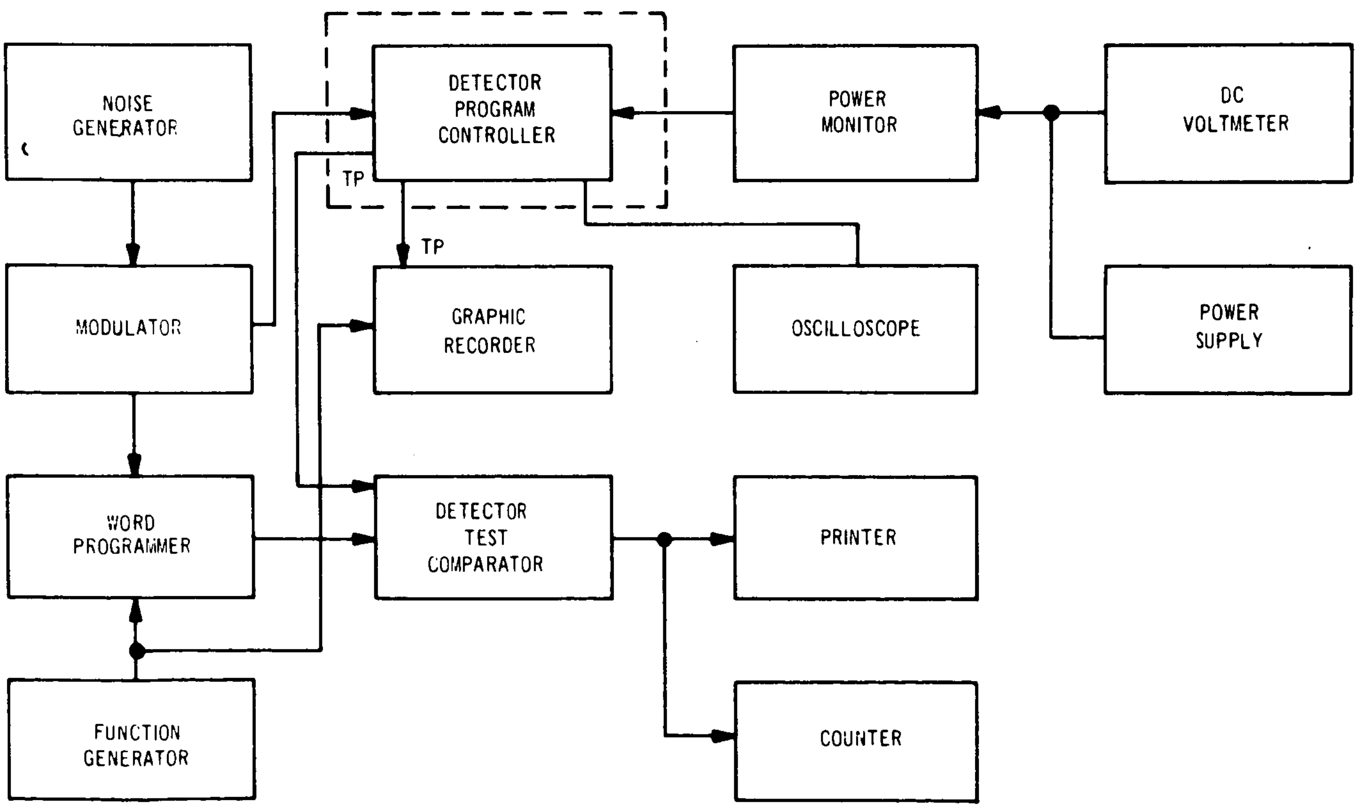

Figure 5-1. Detector/Program Controller, Test Setup

a. Detector Bandwidth Test - 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.

5.3 Decoder Access Unit/Decoder Tests. Since the Decoder Access Unit and Decoders are physically inseparable they are tested simultaneously. However, the address structure and unit switch logic are used to demonstrate the integrity of the redundant decoders. Figure 5-2 shows the active portions of the OSE during these tests.

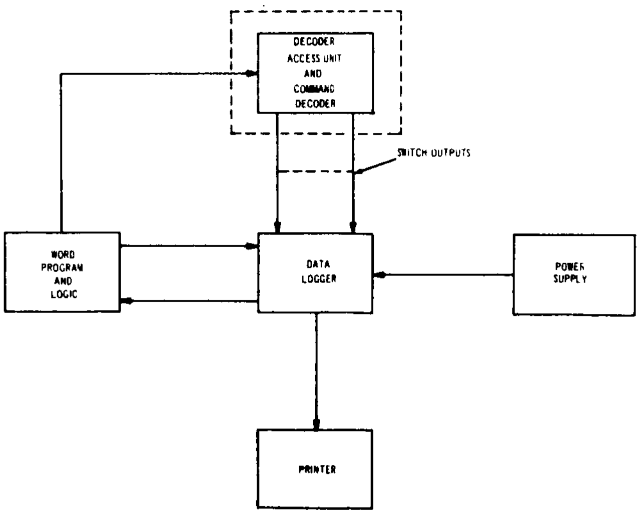

Figure 5-2. Isolated Switch Tests

a. Qualitative Tests - The OSE main consoles are capable of performing fully automatic 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 5-2 is a block diagram of this mode of operation.

c. Power Consumption Test - A test of the decoder similar to that described in Paragraph 5.2c, can be made.

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.

5.5 Subsystem Tests. Tests of the subsystem as a complete unit are the most revealing and comprehensive. 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 2-1.

In the preceding discussion, the subsystem tests have been developed from the theory that the subsystem is the sum of its assemblies and if they are adequately tested separately; and when their performance together is 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.

6.0 SYSTEM TESTS. In tests conducted on the assembled spacecraft the OSE main console only is used. Although the primary mode described in Paragraph 6.1 is normal, sufficient flexibility will be incorporated in the OSE to permit operation in two other modes.

6.1 System Test MDE Generating Commands. The preferred method of generating commands to be issued to the Spacecraft for system tests is to utilize the Telemetry and Command Data System (TCD).

The TCD functions of the DSIF station are duplicated by the STC through use of the SDS 920 computer, and MDE software to cause it to function as a TCD. The SDS 920 in the STC is the same computer described as a telemetry demodulator in the functional description of the telemetry subsystem OSE, VC263FD106.

The TCD will be used to generate command words under the program control of the CDS. These command words are fed to the modulator, which is a functional duplicate of the DSIF MDE. The command words are to be verified in a manner analogous to DSIF usage. The OSE monitors and prints all received addresses and command events. A successful verification inhibits error printouts.

In system tests, the Telemetry and Command Data System (TCD), which is the CDS 920 used as the Telemetry Decommutator, and MDE software to cause it to function as a duplicate of the DSIF TCD, generates the command words under program control of the CDS.

TABLE 7-1. COMMAND SUBSYSTEM FUNCTIONAL INTERFACE

Interfacing Equipment	Function	Input/Output	Remarks
Command Subsystem	Command Words Received Commands Circuit States Power	output input input output	
Computer Data System	Status/Data Command Sequencer Control	output input	
Central Timing	Time and Synch	input	
Ground Power Dist	Facility Power	input	
Radio OSE	Command Message	input	via command detector for verification
Telemetry OSE	Command Words	input	used as TCD
Central Recording	Data	output	
Test Conductor	Status	output	

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.

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 Sections 6.1 or 6.2.

7.0 INTERFACES. The functional interfaces of the Command Subsystem OSE are shown in Table 7-1.

VC263FD104

RADIO SUBSYSTEM OSE

1.0 SCOPE. The VOYAGER Radio Subsystem Operational Support Equipment is a part of the System Test Complex and is used to test and monitor the S-Band Radio Subsystem from initial subsystem tests through STC tests and also supports testing at AFETR.

2.0 FUNCTIONAL DESCRIPTION. A block diagram of the Radio Subsystem OSE is shown in Figure 2-1.

The VOYAGER Radio OSE shown in the block diagram is required to perform the following functions:

a. Provide stable transmitter signals for checking the spacecraft receiver performances.

b. Provide a phase coherent receiver for checking characteristics of the spacecraft transmitters.

c. Provide for measuring, monitoring, and recording of spacecraft powers and frequencies, spacecraft receiver and transmitter functions, and spacecraft transformer rectifier (TR), and dc-dc converter voltages and currents; monitor the Spacecraft Radio Subsystem operating mode and reaction to spacecraft and OSE commands.

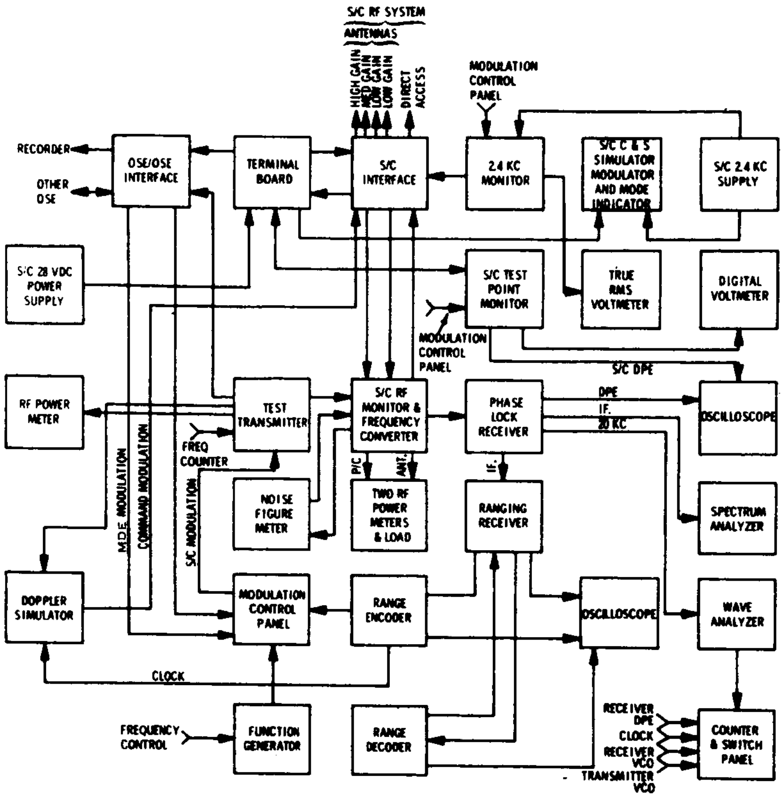

Figure 2-1. Radio Subsystem Functional Block Diagram

d. Provide the spacecraft with a pseudo-random ranging code via the RF link.

e. Demodulate the ranging code from the spacecraft 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 spacecraft RF package when spacecraft power is not available, and simultaneously provide loads for the spacecraft power.

g. Monitor and record operating times of the Spacecraft Radio Subsystem.

h. Provide power and signal isolation.

i. Provide self-test capability.

j. Receive the spacecraft RF telemetry signal and provide a demodulated telemetry signal to the Telemetry OSE.

k. Provide a programmed Doppler sweep and offset to evaluate tracking capability of spacecraft receivers.

3.0 <u>TEST MODES</u>. The Radio OSE has the following modes of operation:

a. <u>Assembly Test Mode</u> - In this mode the individual assemblies (i.e., transponder, power amplifier, etc.) are tested as listed in Table 3-1. The tests are performed using procedures defined by test specifications.

TABLE 3-1. ASSEMBLY TEST MEASUREMENTS

Assembly	Test/Measurements	Assembly	Test/Measurements
Transponder (RCVR)	1. Power Supply Voltages/Currents 2. Command and Verify Response 3. Lock Indications 4. AGC Characteristics 5. Threshold Sensitivity 6. R-F Loop BW 7. AGC Loop BW 8. Phase Jitter 9. Noise Figure 10. Out of Lock Center Frequency 11. Image Rejection 12. Static Phase Error vs. Doppler 13. Doppler Tracking Rate 14. Command Channel, Output Signal Levels and Signal-to-noise ratios 15. Coherent Interference	Transponder (Exciter)	1. Power Supply Voltages/Currents 2. Command and Verify Response 3. Output Power 4. Spurious Output 5. Output Bandwidth 6. Modulation Characteristics 7. Auxiliary Oscillator Frequency and Range 8. Test Transmitter/Radio Receiver Phase Jitter
		Transponder (Ranging Characteristics)	1. Delay (Absolute) 2. Delay Variations 3. Ranging Spectrum
		Power Amplifier	1. Power Supply Voltage/Currents 2. Output Power 3. Bandwidth 4. Spurious Output 5. Delay Characteristics 6. Command and verify response 7. Transfer Characteristics

b. Subsystem Test Mode - This mode includes 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. A functional block diagram showing principle signal flow through sub-assemblies is shown in Figure 3-1.

c. System Test Mode - Here the radio OSE is interfaced with the Command OSE, data storage and display equipments, and recorders in the STC. The requirements are as follows:

1. Permit complete exercising of all spacecraft mechanical and electrical functions.
2. Afford monitoring in realtime of the spacecraft radio subsystem behavior.
3. Afford test repeatibility
4. Have self-test capability
5. Provide real-time test records
6. Provide for detection of failures to a subsystem level.
7. 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 3-2. 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 Computer Data System (CDS) and Command MDE software in the SDS 920, the TCD. The Radio Subsystem OSE will function in this mode.

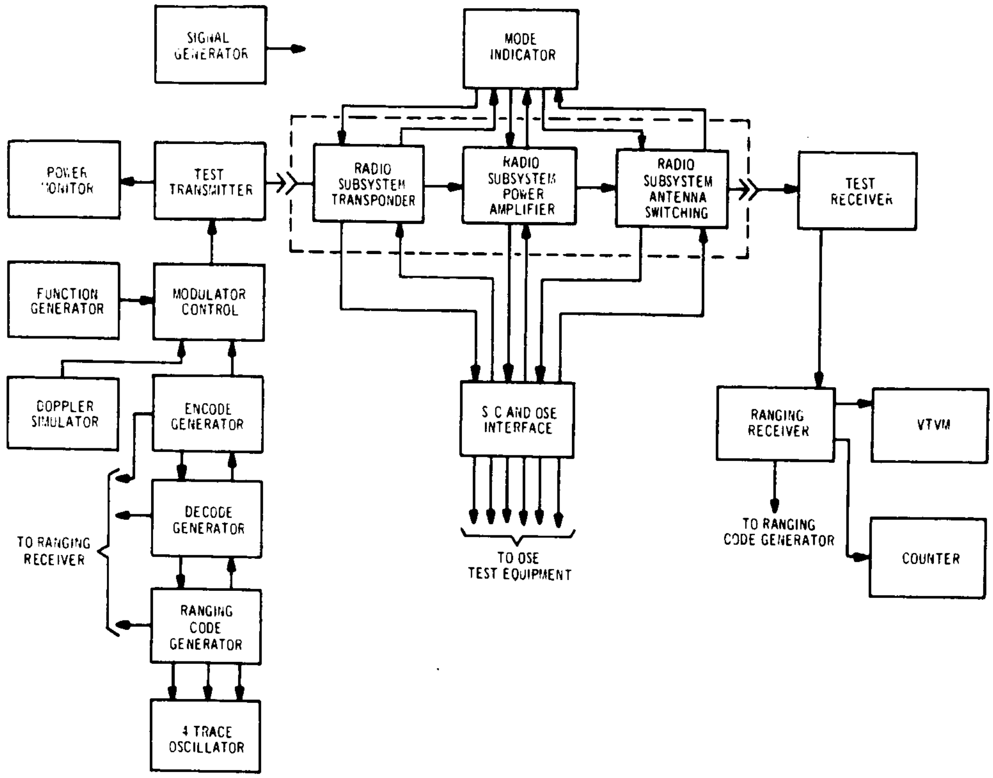

Figure 3-1. Radio Subsystem OSE Subsystem Tests

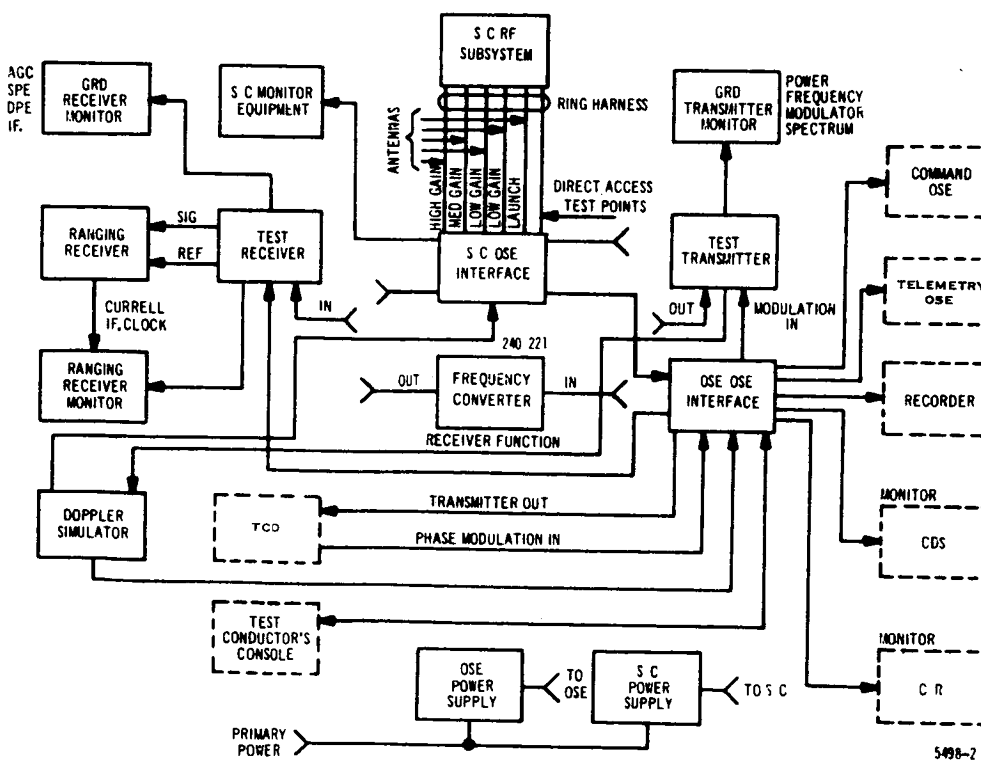

Figure 3-2. System Tests in STC

4.0 INTERFACES. The functional interfaces of the Radio OSE, with the spacecraft and with other OSE, are summarized in Table 4-1.

TABLE 4-1. INTERFACE SUMMARY

Interfacing Equipment	Function	Input/Output	Remarks
Radio Subsystem	RCVR AGC	input	3 CCVRS
	RCVR SPE	input	3 RCVRS
	RCVR DPE	input	3 RCMS
	T/R voltages	input	
	Power amp levels	input	
	Exciter level	input	
	Antenna	in/out	4-RF
	S/C power	output	
Command OSE	Subcarrier	output	
	Synch	output	
	Combined signal	output	

Interfacing Equipment	Function	Input/Output	Remarks
CDS	Direct measurement from RDO	output	
	Gnd RCVR SPE	output	
	Test Xmitter freq.	input/output	
	Doppler start	input	
Central Recorder	Direct measurement from RDO	output	
	Gnd RCVR SPE	output	
Test Conductor's Console	Status/Data	output	
Telemetry Subsystem OSE	Phase Mod'n	input	(SDS 920 acting on TCD)
	Transmitted freq.	output	

VC263FD105

DATA STORAGE SUBSYSTEM OSE

1.0 SCOPE. This section contains a functional description of the VOYAGER Data Storage Subsystems Operational Support Equipment (OSE). The Data Storage Subsystem OSE is for use in subsystem tests and for system testing in the System Test Complex (STC).

2.0 FUNCTIONAL DESCRIPTION.

2.1 General. The purpose of the Data Storage Subsystem OSE is to monitor and control the Spacecraft Data Storage Subsystem during subsystem and system tests as required. The Spacecraft Data Storage Subsystem consists of six magnetic tape recorders (MTR) and their associated power supplies, status monitors and control circuitry. The Data Storage Subsystem OSE (Figure 2-1) performs its tasks by having the capability to simulate all inputs (data and controls) and output loads while monitoring and displaying the results to verify subsystem operation.

2.2 Data Simulator. The data simulator simulates all data input interfaces to the Spacecraft Data Storage Subsystem. The simulator generates a known data stream of variable length to be employed in measuring bit error rate, skew or wow, and flutter. The data will be generated and stored in a manner to allow use of majority logic in determining bit errors (or dropouts).

2.3 Monitor and Control Unit. The Monitor and Control Unit (MCU) simulates the direct-access and umbilical-access interfaces not included by the data simulator or spacecraft power simulator. The interfaces simulated are: telemetry, command, control and sequencer, and data automation equipment data sources. This simulation is required to conduct a subsystem test on the Data Storage Subsystem.

The MCU controls the Data Storage Subsystem OSE and flight equipment in both subsystem and system test modes. The MCU provides data conditioning to the monitor points to provide

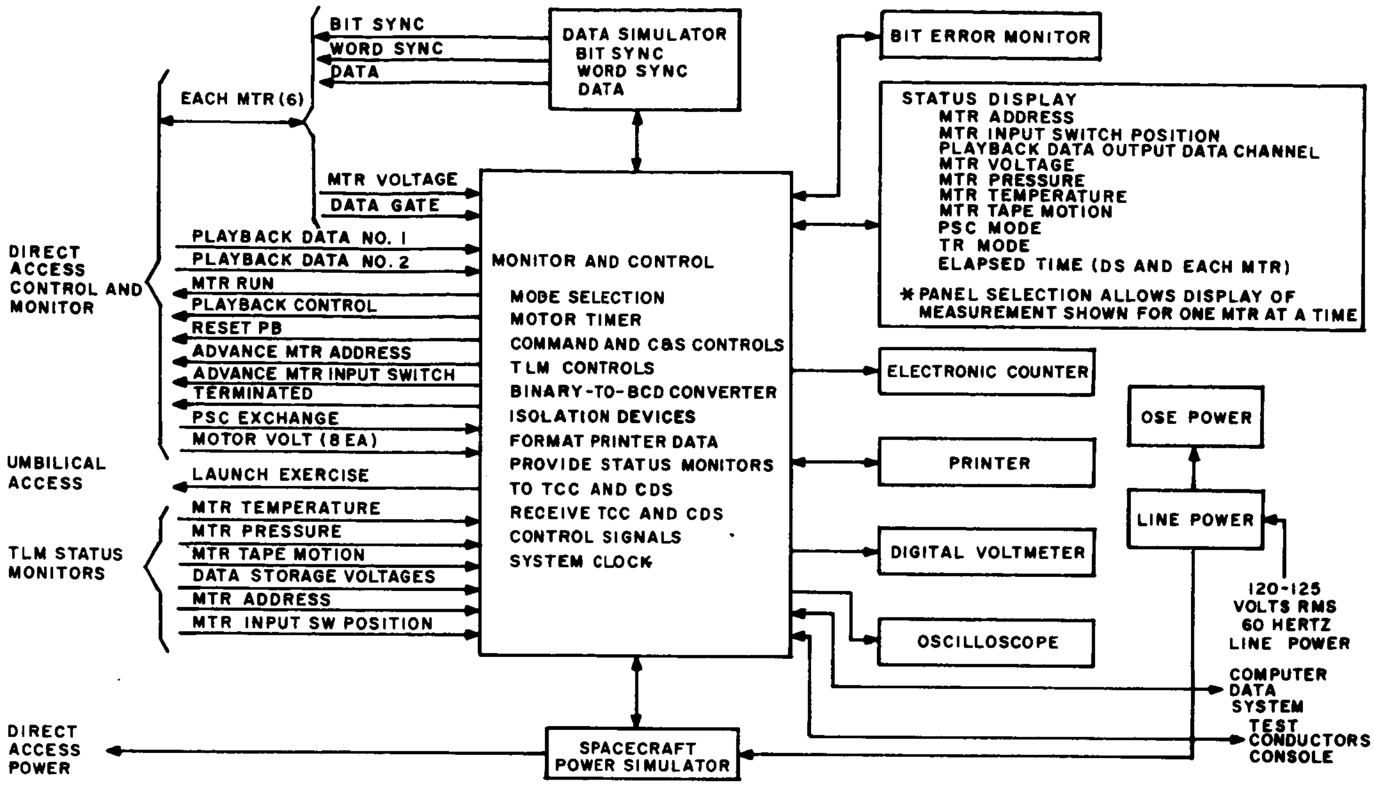

Figure 2-1. Data Storage Subsystem

a more versitile display system in subsystems testing as well as allowing data to be presented in the same manner from the Computer Data System (CDS) while in a system test configuration in the STC. The data storage OSE will accommodate automatic testing by the CDS. The Data Storage Subsystem OSE will include self test capabilities.

2.4 Data Display Unit. The data display unit will consist of alphanumeric on-off lights or alarm displays. All inputs to the display unit will be controlled by the MCU. The extent of display capability is shown in Figure 2-1.

2.5 Test Equipment. The Data Storage Subsystem OSE employs the following items of test equipment: bit error monitor, Spacecraft Power Simulator, electronic counter, printer, digital voltmeter and oscilloscope. The test equipment will be used in subsystem testing and in self-testing of the OSE.

3.0 INTERFACE DEFINITION. The Data Storage Subsystem OSE interfaces with the Spacecraft Data Storage Subsystem, the Test Conductor's Console, and the CDS. The OSE interfaces are listed in Table 3-1.

4.0 PERFORMANCE PARAMETERS. The data storage OSE provides the following measurement/control capability.

a. Measure d-c voltages to ±1 microvolt on the most sensitive scale.
b. Monitor and record elapsed time of the Data Storage Subsystem and each of its MTR's.

TABLE 3-1. DATA STORAGE OSE INTERFACES

Interface	Function	Input/ Output	Remarks
Data Storage Subsystem (Flight connectors)	Playback Data	Output	Digital NRZ Data
	Playback Control	Input	Control Discrete
	Playback Reset	Input	Control Discrete
	Advance MTR Address	Input	Control Discrete
	Advance MTR Input SW	Input	Control Discrete
	PB Terminated	Output	Monitor Discrete
	PSC Exchange	Input	Control Discrete
	Bit Sync	Input	Digital Wavetrain
	Word Sync	Input	Digital Wavetrain
	Data	Input	Digital NRZ Data
	MTR Temperature	Output	Analog Voltage
	MTR Pressure	Output	Analog Voltage
	MTR Tape Motion	Output	Analog Voltage
	MTR Address	Output	3 Binary Bits
	MTR Input SW	Output	2 Binary Bits
	Data Gate	Output	Logic Level
Data Storage Subsystem (Umbilical)	Launch Mode Set	Input	Control Discrete
Data Storage (Direct Access)	MTR Motor Voltage	Output	A-C Voltage
TCC	Status	Input	Switch Closure
CDS	Commands	Output	Switch Closure
	Spacecraft Data	Output	Digital/Analog Data

c. Simulated spacecraft power.

d. Measure frequencies to at least ± 0.01%.

e. Simulation of all data and control signals.

f. Measurement of MTR start and stop times.

g. Measurement of bit error rates and bit drop out.

5.0 PHYSICAL REQUIREMENTS AND CONSTRAINTS. The data storage OSE requires 105-to 125-volt, 60-Hz, single-phase power. The maximum current shall not exceed 50 amperes. The data storage OSE shall weigh less than 2000 pounds and occupy two standard 75-inch NASA racks.

VC263FD106

TELEMETRY SUBSYSTEM OSE

1.0 SCOPE. This section describes the Telemetry (TLM) Subsystem Operational Support Equipment (OSE). The TLM OSE is for use in subsystem and systems testing in the STC.

2.0 FUNCTIONAL DESCRIPTION

2.1 General. The TLM OSE is required to support system and subsystem tests by stimulating, controlling, and monitoring the spacecraft TLM subsystem. The TLM OSE, through umbilical and/or direct access will control TLM modes, decommutate the data, and monitor and display the results. During subsystem and inter-system tests, the TLM OSE will provide stimulation signals such that each channel can be calibrated. Stringent isolation requirements of the STC dicate use of isolation amplifiers and isolation switches between the TLM OSE, the spacecraft, and other ground equipment.

The TLM OSE, shown in Figure 2-1, consists of the following major functional blocks: Input Switching Unit, Telemetry Demodulators, Telemetry Simulator, Magnetic Tape Recorders, Monitor and Control Unit, TLM Stimulator/Calibrator, Decommutator, Test Equipment, and Power Supplies. The bit rates, modes and description of data are defined in the spacecraft TLM section of the report.

2.1.1 Input Switching Unit. The Input Switching Unit receives the composite bi-phase modulated telemetry signal from the Radio Subsystem OSE, tape recorders, TLM simulator or spacecraft TLM. The switching unit routes the incoming signal through its required

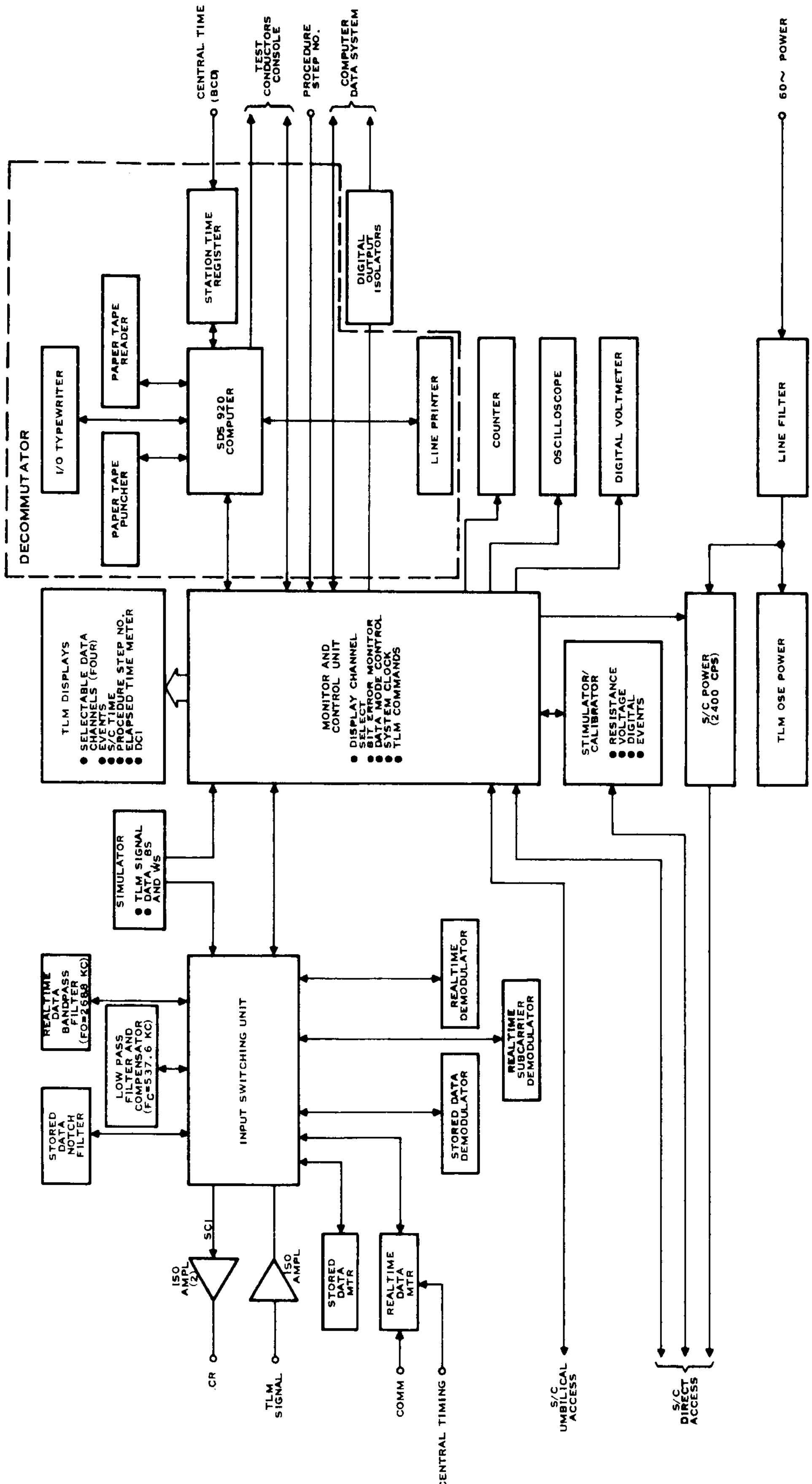

Figure 2-1. Telemetry Subsystem OSE, Block Diagram

signal paths, filters, demodulators, etc., to the appropriate output such as magnetic tape recorders (MTR), and monitor and control unit.

2.1.2 Telemetry Demodulator. The TLM Demodulator has the task of detecting and decoding a noise-corrupted, bi-phase modulated telemetry signal. When frequency multiplexing is employed the signals will be separated before demodulating. This is more fully described in VC280FD105, TLM MDE Demodulator.

2.1.3 TLM Simulator. The TLM simulator functionally duplicates the spacecraft TLM in every respect in order to test and verify the TLM OSE operation without the use of a spacecraft TLM. The simulator is discussed in further detail in VC280FD105.

2.1.4 Telemetry Magnetic Tape Recorders. Two MTR's will be employed for storing Real-Time (RT) TLM and stored data. Stored data will be recorded on one MTR with the other one recording RT TLM data. The RT TLM recorder will record data on seven tracks as follows:

a. Ground Instrumentation
b. Data
c. Composite telemetry signal
d. Bit sync
e. Time and wow/flutter compensation
f. Word sync
g. Voice lable and intercom

Monitor oscilloscopes are included to verify performance of the tape machines during all recording operations.

2.1.5 Monitor and Control Unit. The Monitor and Control Unit (MCU) will provide the manual control of the spacecraft TLM required to test, calibrate, and verify its operational readiness in the STC. The MCU design will allow central control of the TLM OSE by the CDS. The MCU provides the following functions:

a. Rate Commands
b. TLM Mode Commands, etc.
c. Channel Display (Four Selectable Channels)
d. Data Conversion
e. Bit Error Monitoring
f. Basic Clock Generation
g. Stimulator/Calibrator Auto/Manual Control

While receiving or distributing:

a. Data
b. Data Mode
c. Commutator Deck Position
d. Spacecraft ID
e. Spacecraft Time
f. Spacecraft Events
g. Procedure Step Number

2.1.6 TLM Stimulator/Calibrator. The outputs on the stimulator/calibrator can be connected to the spacecraft TLM case harness for signal simulation for all data channels in the TLM subsystem. The TLM stimulator is mechanized to accommodate automatic calibration. The inputs are divided into four categories.

a. Simulate analog inputs as follows:

Voltage simulators provide adjustable voltages to the TLM subsystem in three voltage ranges: 0 to 100 mv, 0 to ± 1.6 volts, and 0 to 3.2 volts.

b. Simulate digital signals of serial or parallel binary data words.

c. Stimulate transducer channels by simulating a minimum of seven discrete values of resistance between 500 and 600 ohms through automatic means.

d. Simulate event occurances through the generation of Z event pulses with a maximum voltage level of +3.2 volts and a minimum duration of 3 milliseconds.

2.1.7 Telemetry Decommutator. The TLM decommutator consists of a SDS 920 Computer, paper tape punch, paper tape reader, I/0 tapewriter and line printer, and station time register. The decommutator receives the demodulated telemetry data and SCI. The decommutator function and software will be essentially the same as the MDE decommutator discussed in VC280FD105.

2.1.8 OSE Power. The OSE requires 105-125 volt, 60-Hz, single-phase power.

2.1.9 Test Equipment. The TLM Subsystems OSE includes the following test equipment:

a. Bit Error Rate Monitor - A bit error rate monitor unit which can automatically check and monitor the bit error rate of the noise-corrupted data being detected and decoded by the demodulator using the hardline data encoder output as its reference.

b. Spacecraft Power Supply - A 2400-Hz spare wave power supply is provided. The power supply provides primary power to the spacecraft TLM subsystem during subsystem testing.

c. Counter

d. Digital Voltmeter

e. Oscilloscope

2.1.10 Special Equipment. Special equipment such as J-boxes, isolation amplifiers, digital output isolators, and cables are supplied as necessary.

2.2 Self-Test. The TLM OSE is capable of performing self-test during system and subsystem test.

3.0 INTERFACE DEFINITION. The TLM subsystem OSE interfaces are defined in the Table 3-1.

4.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

4.1 General. All incoming and outgoing signals are brought through the TLM OSE junction box. The junction box will provide connector plugs for test cables, umbilical cables, a-c

TABLE 3-1. TELEMETRY SS OSE INTERFACE LIST

Interface	Function	Input/Output	Remarks
Radio Subsystem OSE	Spacecraft Telemetry Signal	Output	0.3-3.0V rms
CDS	Decommutated TLM Data	Output	
	Command Request	Output	Switch Closure
	Telemetry Data	Output	Analog from Data Converter
	Control	Input	
Test Conductor's Console	Status and Mode	Outputs	
Central Recorder	Synch Condition Indicator	Output	Voltage Level
Central Timing	Timing	Input	
Telemetry Subsystem (Test Points)	Mode Control	Output	Discrete (9)
	Rate Control	Output	Discrete (2)
	Power Supply Monitor	Input	Analog (4)
Telemetry Subsystem (Umbilical)	Modulated Subcarrier	Input	270 kc

power, and interfaces with the Central Recorder, Computer Data System, Science, Command Subsystem OSE, Radio Subsystems OSE, and Central Timing and Synchronization System. All incoming and outgoing signals are routed to the proper location in the junction box.

4.2 Size, Weight, and Power. The TLM OSE occupies eight racks plus one lineprinter. The TLM OSE weighs less than 6000 pounds and requires less than 9 kilowatts of 105-125 volt, 60-Hz, single-phase power.

VC264FD101

GUIDANCE AND CONTROL SUBSYSTEM OSE

1.0 SCOPE. This document describes the Operational Support Equipment (OSE) required to support the Guidance and Control (G&C) Subsystem and system tests as described in OSE Objectives and Design Criteria (VC260SR101), STC Design Characteristics and Restraints (VC260SR102) and STC Functional Description (VC260FD100). These documents describe the philosophy and approach to be used, how the subsystem OSE is to be integrated into the System Test Complex (STC), and the interfaces between the subsystem OSE and the STC. This document will, therefore, emphasize the interface between the OSE and the spacecraft Subsystem.

2.0 FUNCTIONAL REQUIREMENTS - SYSTEM TEST

2.1 General. G&C OSE is used during spacecraft system testing to provide stimuli to the G&C Subsystem and to monitor the subsystem response. These functions are controlled locally by the G&C OSE or remotely from the Computer Data System (CDS). Either mode may be chosen as an option of the test being performed. The G&C OSE is used to:

a. Optically stimulate the Sun and Canopus sensors.

b. Electrically simulate spacecraft motion by providing torque signals to the gyros and accelerometers. This circuitry will also have provisions for cancelling the earth's rate effect on each gyro.

c. Monitor and examine the G&C Subsystem outputs with OSE instrumentation.

d. Monitor the gas jet outputs of the cold gas jet subsystem.

e. Monitor feedback signals from the thrust vector control position indicators of the Propulsion OSE.

2.2 Description. The block diagrams of Figures 2-1 and 2-2 show the functional layout of the G&C OSE. Figure 2-1 shows the overall interface layout between G&S OSE and the STC. Figure 2-2 expands this definition to include the functional layout of the G&C OSE. The Electronics Control and Display Equipment (ECDE) contains the main controls and monitors for the G&C Subsystem and forms the principal interface to the STC. The Stimulation Control and Display Equipment (SCDE) is used to provide light stimulation to the Sun, solar aspect, and Canopus sensors. The monitors for the cold gas jet subsystem and the Thrust Vector Control Position Indicators are obtained from Propulsion OSE during system test.

Access to G&C control and monitor points is obtained by means of direct access test connectors which are mounted on the spacecraft. In addition, umbilical data and telemetered data are available from the Umbilical Junction Box and the Telemetry Data Decoder, respectively.

All ECDE input data is distributed to indicators and panel meters in addition to being made available on a patch board for selectable monitor by the test instruments and recorders. Status and alarm indicators are fed to the CDS and the Test Conductor's Console (TCC). Selected analog and digital data is also sent to the CDS.

Automatic control from the CDS is accomplished by decoding the 18 information bits of the 24-bit CDS command word and applying these commands to the same ECDE stimulation controls that are used for G&C subsystem test by itself. Similarly, functions which must be

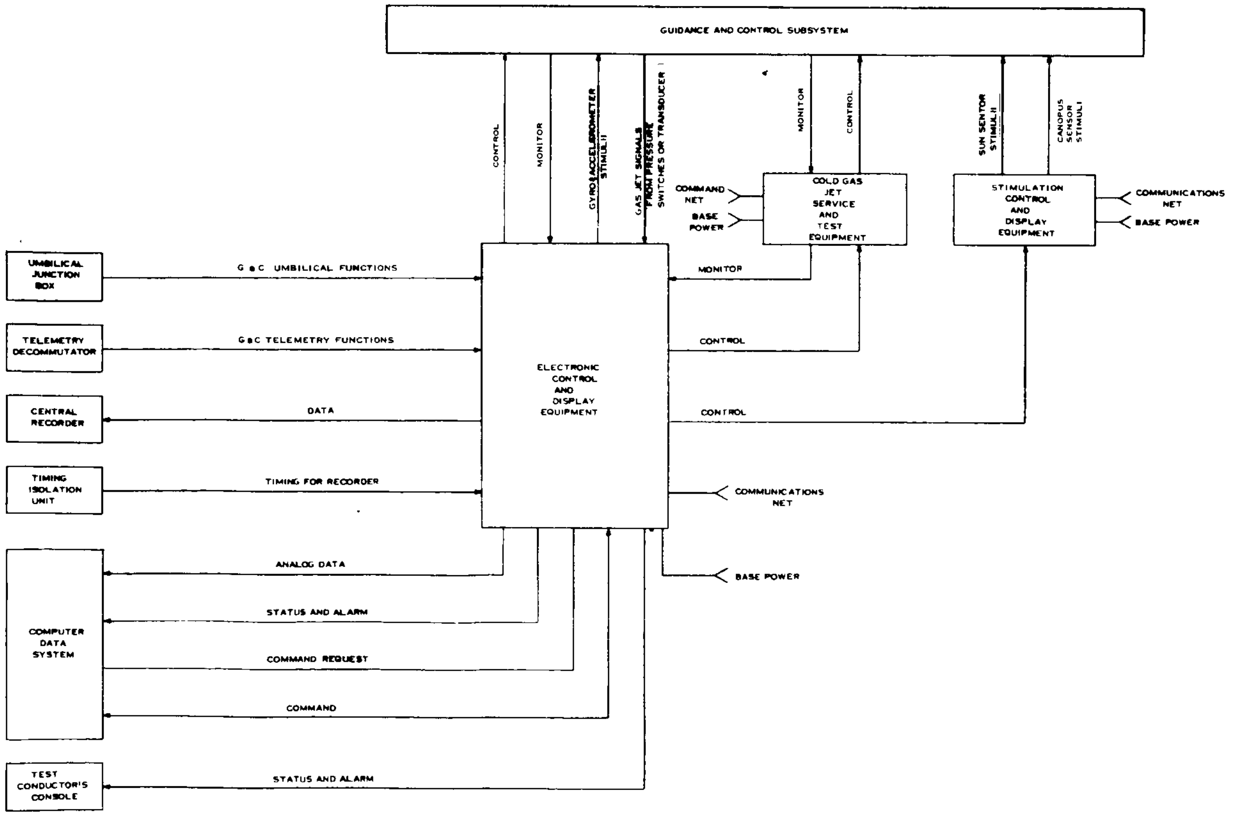

Figure 2-1. Guidance and Control Subsystem OSE, Block Diagram

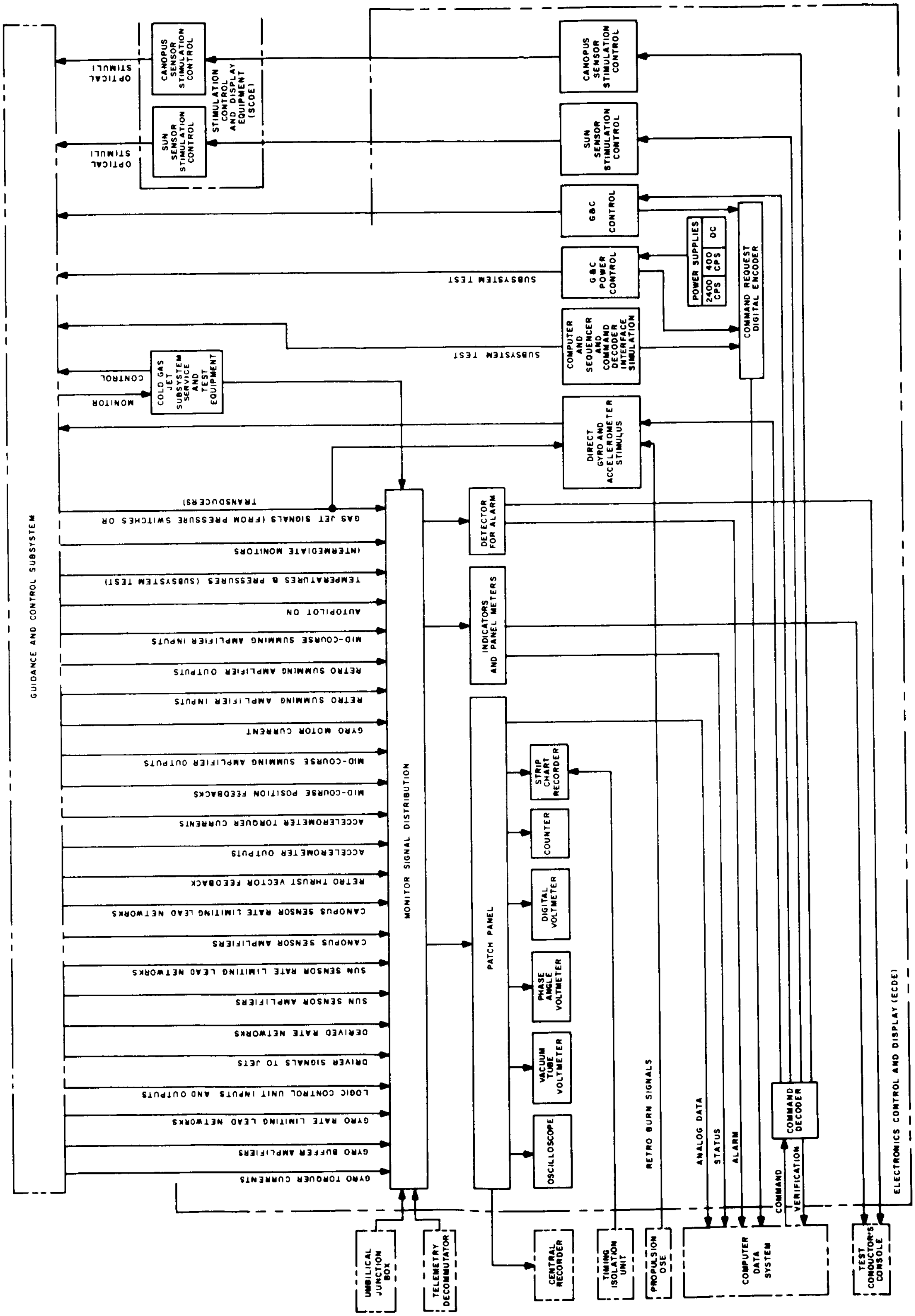

Figure 2-2. System Test Block Diagram

performed on the G&C Subsystem by associated spacecraft subsystems, are requested from the same ECDE controls that would be used for an isolated G&C subsystem test, digitally encoded, and transmitted to the CDS for implementation.

Similarly, if the test plan calls for G&C subsystem test, wherein the G&C subsystem engineer requires commands from the Computer and Sequencer (C&S), then the C&S interface simulation controls on the ECDE will be manipulated. The CDS will interrogate the ECDE, accept the command request, and subsequently command the C&S OSE to initiate the required function through the spacecraft C&S.

2.3 Test Requirements

2.3.1 Monitors and Control Signals. The major electrical functions which are monitored directly from the spacecraft include:

a. Current to gyro torquers to assure nondetrimental values and to monitor commanded error signal inputs.

b. Gyro buffer amplifiers.

c. Outputs of logic control unit to determine the subsystem operating mode.

d. Driver signals to jets to determine that the electronics output is proper for known error signals and to be used to close the loop test when the cold gas supply is not present.

e. Sun sensor amplifier outputs to determine the error signal inputs to the electronics.

f. Canopus sensor amplifier outputs to ascertain the roll error and gating signals to the electronics.

g. Retro thrust vector feedbacks to ascertain the propulsion directional error signals to the autopilot during retro.

h. Accelerometer output to determine the signal to the velocity register.

i. Accelerometer torquer current to assure nondetrimental values.

j. Midcourse loop jet vane position feedback to ascertain the directional error signals to the autopilot during midcourse maneuver.

k. Midcourse loop summing amplifier outputs to determine autopilot midcourse commands for known inputs.

l. Gyro motor currents to ascertain correct operation and to guard against motor damage.

m. Retro loop summing amplifier inputs to establish the reference signals for autopilot output retro commands.

n. Retro loop summing amplifier outputs to determine the autopilot commands for known inputs.

o. Midcourse summing amplifier inputs to establish the reference signals for autopilot midcourse commands.

p. Autopilot on signal to establish when power is applied to the autopilot.

q. Temperatures and pressures to prevent operation when hazardous to equipment or personnel.

Some of the above signals will be duplicated through the umbilical connector or by telemetry.

2.3.2 Sensor Stimulation. For stimulation of the acquisition and cruise sun sensors, lamps in individual light-tight hoods will be used. Since the individual sensors have been thoroughly tested in component test prior to their use in the G&C Subsystem, the OSE excitation of these sensors will be by means of light intensity variation. The GE Quartzline series provides an output in the upper part of the sensor spectrum and may be used for this application. Collimation lenses will be provided as required.

The control for these stimulators will be mounted in the control and display console of the SCDE. This console contains power supplies and control circuits to permit setup of the stimulators without tieing up the ECDE. Control of the individual stimulators to simulate the effect of spacecraft motion will be done by varying the intensity of the lamps in a known fashion. For the gating sun sensors, stimulators similar to those employed for acquisition and cruise sensors will be used.

Stimulation of the Canopus sensor will be accomplished with a collimated source. This source will be variable in intensity between a value below the lower gate threshold to a value above the upper gate threshold. For proper operation, the sensor output should occur only when the stimulator intensity is between the lower and upper gate. Accommodation of pitch deflection steps of the sensor will be included. The light-tight hood will also be used to protect the sensor from handling and bright light. The Sun and Canopus sensor output amplifiers will be monitored to verify the signal to the electronics.

Control of the stimulators will be physically located in the console of the SCDE. This console and the stimulators can be used independently to set up the Sun and Canopus sensor stimulation. This will be a convenient method of operation at initial installation of the G&C Subsystem into the spacecraft or when the ECDE is required for other tasks. When cables are connected between the ECDE and the SCDE, the ECDE will have remote control over the optical stimulation. Since the CDS interfaces to the ECDE, CDS inputs to the stimulators will be implemented through the ECDE.

Cold Gas Jet Service and Test Equipment (GSTE) for the cold gas jet subsystem will be the same as equipment in the Propulsion OSE. Cabling between the GSTE and the ECDE will present pneumatic monitor points to the ECDE and permit a minimum of remote control functions of the pneumatics from the ECDE. Because of the commonality of the GSTE to equipment in the Propulsion OSE, Propulsion OSE will be used as GSTE during spacecraft system testing. A description of the GSTE can thus be obtained as part of the Propulsion OSE section of the report.

2.3.3 System Test Use. The use of these signals is illustrated in Figure 2-3. This figure shows a scheme for controlling and monitoring one channel of G&C. The major aspects of this scheme are as follows.

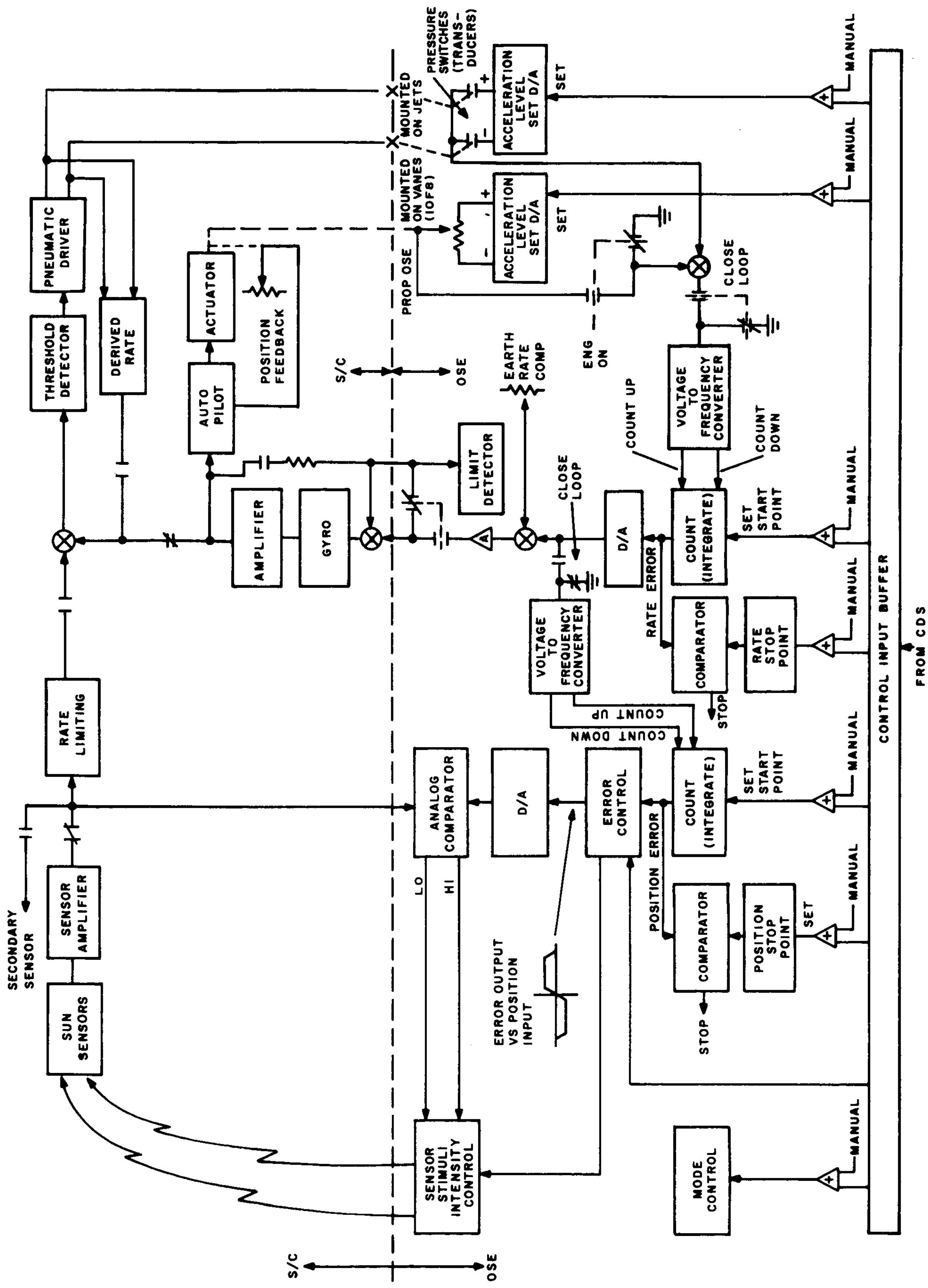

Figure 2-3. Guidance and Control Subsystem OSE, Block Diagram

2.3.3.1 Allows Simulated Close Loop Operation. This is accomplished for the cold gas reaction loop by detecting gas flow (or solenoid operating signal if the gas system is not used) and assumes that this causes a constant (but adjustable) plus or minus acceleration depending on which value is operated. This acceleration is integrated to provide a rate equivalent signal. During midcourse maneuver, the correcting torque or acceleration is derived from an OSE position monitor mounted on the Propulsion Subsystem jet vanes. These signals are summed during engine firing with the solenoid signals as appropriate to establish the net rotational acceleration on the vehicle. This sum is integrated by the same integrator as above to provide a rate equivalent signal. The reference levels for these acceleration signals are adjustable, allowing simulation of various thrust levels, changes in inertias, cg variations, etc., in a static or dynamic manner as desired.

The OSE-derived rate signal is used to provide a signal to the vehicle gyro torque motor which is calculated to simulate the appropriate vehicle rate signal. This gain factor can also be varied as desired to simulate various parameter variations. The OSE-derived rate signal is also integrated in order to establish a position error. The scheme presented shows this OSE-derived position error being converted to an idealized desired vehicle position signal. This desired signal is compared with the actual vehicle position signal (Sun sensor amplifier output in case shown), and the difference between the desired and actual is used to control the sensor stimulation in such a manner that the position, error signal of the vehicle should track the OSE-desired position error.

2.3.3.2 Open Loop Testing. The proposed scheme also allows for open loop testing (i.e., position and rate errors are completely independent and corrective forces are not used to alter the errors). This mode is primarily used to gather data concerning component operation in a simple and direct manner as follows:

a. Sun Sensor Amplifier output as a function of stimulation input.

b. By varying only the sun sensor input, the rate limiting network can be analyzed by using the input to the threshold detector.

c. Switchover from one sensor to another is verified by controlling the individual stimulators in a different manner and observing that the output follows the proper one.

d. The derived rate network can be monitored by fixing the position error when the solenoid fires, and monitor the resulting threshold amplifier input change controlled solely by the derived rate network.

2.3.3.3 Gyro Limiting. The gyro output is monitored in such a manner as to prevent a potentially damaging position/rate signal (one large enough to cause the gyro to be driven into its stop). The OSE will monitor and record the gyro position continuously and will attempt to cage the gyro if the output exceeds a maximum level. Warnings will also be provided such that the operator is aware that the gyro is still coasting (based on maximum stopping time) and, therefore, should not be moved.

2.3.3.4 Remote Control. The largely digital nature of the proposed scheme reflects the requirement that the G&C OSE must be capable of being controlled by digital signals from the STC's CDS. In order to use the same circuitry for subsystem tests and system tests this control is required to be implemented digitally. The functions to be controlled by the CDS are shown as being:

a. Mode - (open/close loop, gas used or not, etc.)
b. Original rate error signal
c. Stopping rate error signal
d. Original position error signal
e. Stopping position error signal
f. Solenoid reference set
g. Thrust vector control reference set
h. Position error form
i. Rate error gain

3.0 FUNCTIONAL REQUIREMENT - SUBSYSTEM TEST. In addition to the requirements for G&C OSE during system test, the following capabilities are needed for G&C subsystem testing:

a. Simulation of all spacecraft interfaces to the G&C Subsystem.

b. Capability to interface with the replaceable G&C Subsystem assemblies to permit progressive assembly buildup and for troubleshooting. This means that adapter cabling for all subsystem internal interface connectors should be available.

c. Provision for representative simulation of selected replaceable assembling interfaces.

d. Provision for monitor capability of all subsystem test connectors and selected representative subsystem internal interface connectors.

e. Verification of gyro and accelerometer package acceptability.

During subsystem testing, the interface to the Power Subsystem, Computer and Sequencer, and Command Decoder are simulated by the G&C OSE. Normal flight connectors to the respective subsystems are used. To simulate the spacecraft electrical subsystem interface, power supplies for 2400 cps, 400 cps, and dc are mounted within the ECDE. These supplies are completely isolated from each other and the console, and are designed to be adjustable from a control panel to permit variation within the design range and at least ± 15% beyond for voltage, frequency, noise, etc. The Controller and Sequencer and Command Decoder interfaces are simulated by pulses generated by logic within the ECDE. For critical drive lengths, driver amplifiers will be employed.

During subsystem test, selective simulation of internal interfaces will be employed for convenience and troubleshooting. For example, when the G&C electronics are separated from the sun sensors at initial spacecraft assembly, or when troubleshooting is required, the electrical output of the sun sensor amplifiers will be simulated by a generator within the OSE. This does not infer that the exact response curve must be generated, but more so that a good test signal, such as a ramp with proper voltage and impedance match should be available in the OSE for test of the interface and the succeeding electronics.

A rate table and a position table are used to verify the acceptability of the gyro and accelerometer packages. The position table is particularly useful for gyro drift measurement using earth rate and for accelerometer test in a one-g field. The rate table provides controlled rates for more detailed gyro performance evaluation, and gives a test of gyro torquer linearity. By taking advantage of the centripetal force of the rotating surface, the rate table may also be used to provide controlled accelerations to the accelerometers. Suitable fixtures for mounting the gyro and accelerometer packages will be provided.

4.0 PHYSICAL. The physical layout of the G&C OSE has been functionally divided in accordance with the previous descriptions and the attached block diagrams. The breakdown with estimated number of electronic racks is listed below:

a. ECDE - 6 racks per ECDE

b. SCDE

  1. Console - 2 racks per SCDE

  2. Sun-stimulators - 14 required per SCDE

  3. Canopus stimulators - 2 required per SCDE

c. Rate table

d. Position table

The G&C OSE is used to test the redundancies of the G&C subsystem as well as the normal operation. Acceptability of the OSE as G&C test equipment is verified by the use of self-check circuitry within the OSE itself.

5.0 INTERFACE (SPACECRAFT G&C-TO-G&C OSE). A tabulation of the functional interfaces to the G&C OSE and the Spacecraft G&C is given in Table 5-1.

TABLE 5-1. SPACECRAFT G&C TO G&C OSE INTERFACES

Interfacing Equipment	Function	Input/Output	Comments
Umbilical Junction Box	Gyro buffer amplifiers output	Input	3 signals per operating package
G&C Subsystem	Gyro torquer current	Input	6 signals
	Gyro torquer amplifiers output	Input	9 signals
	Threshold detector input		3 signals
	Logic control unit outputs	Input	15-20 signals
	Driver signals to jets	Input	16 signals
	Sun sensor amplifier outputs	Input	18 signals
	Canopus buffer amplifier output	Input	3 signals
	Retro thrust vector feedbacks	Input	4 signals
	Accelerometer outputs	Input	2 signals
	Accelerometer torquer currents	Input	2 signals
	Midcourse position feedbacks	Input	Estimate 16 signals
	Midcourse summing amplifier outputs	Input	4 signals
	Gyro motor currents	Input	15 signals
	Retro summing amplifier outputs	Input	4 signals
	Midcourse summing amplifier inputs	Input	15 signals
	Autopilot on signal	Input	1 signal

Interfacing Equipment	Function	Input/Output	Comments
G&C Subsystem	Sun sensor optical stimulation	Output	Variable light intensity
	Canopus sensor optical stimulation	Output	Variable intensity and controlled position
	Gas jet outputs	Input	Via pressure switches or transducers
	Command decoder interface	Output	Subsystem test only
	C&S interface	Output	Subsystem test only
	G&C subsystem power	Output	Subsystem test only
	G&C subsystem internal interfaces	Input	Subsystem test only
	Simulation of selected G&C subsystem internal interfaces	Output	Subsystem test only
	Telemetry sensor monitors	Input	Subsystem test only
Computer Data System	Analog data	Output	
	Commanded operation	Input	
	Verification of command receipt	Output	
	Operation request	Output	
	Status & alarm	Output	
Test conductor's console	Status & alarm	Output	
Base power and distribution	208-v 3-phase 60-Hz	Input	
Communications net	Communications	Input/output	

CONTROLLER AND SEQUENCER SUBSYSTEM OSE

1.0 SCOPE. This document describes the OSE required to support the Controller and Sequencer (C&S) subsystem and system tests as described in OSE Objectives and Design Criteria (VC260SR101), STC Design Characteristics and Restraints (VC260SR102) and STC Functional Description (VC260FD100). These documents describe the philosophy and approach to be used, how the Subsystem OSE is to be integrated into the STC, and the interfaces between the Subsystem OSE and the STC. This document will, therefore, emphasize the interface between the OSE and the Spacecraft Subsystem.

2.0 SYSTEM TEST. During system test the OSE is capable of loading and verifying the memory either under local control or remotely from the CDS, monitoring direct access test points, and monitoring selected telemetry functions.

2.1 Functional Description

2.1.1 Remote Control Operations

2.1.1.1 Memory Loading. The C&S OSE shown in Figure 2-1 will load the C&S Subsystem memory under control of the Computer Data System. The OSE when commanded by the CDS will decode 18 bits of a 24-bit CDS control word and will use the decoded information to prime the control logic such that subsequent control words from the CDS, which contain command data and memory addresses, can be formatted for entry into the memory. In sequence, therefore, the first CDS control word will set up the sync, high or low data read-in rate, the alert, the end of word and the update sequencer discrete. The second and third CDS words will be read into a register such that it contains the format shown below:

1 BIT	8 BITS	8 BITS	1 BIT	9 BITS
FLAG	FUNCTION	TIME TAG	PARITY	ADDRESS

When the register is full, the C&S OSE will shift it serially into the spacecraft memory along with the proper sync and control signals. Data can be read into the spacecraft memory at 15 bps or 250 bps as governed by the high or low data read-in control bits of the initial CDS control word.

2.1.1.2 Memory Verification. Verification is accomplished along the same lines as loading. A CDS control word will command the C&S OSE to read out an address or addresses. The next CDS control word will contain the spacecraft addresses to be read out. This word will be serially shifted out of the C&S OSE register into the spacecraft memory. The spacecraft memory will then interrogate that address and output the contents in parallel through the spacecraft direct-access points or serially through the telemetry interface. Normal readout is one word per second through the direct access or one word per four seconds through the telemetry. By speeding up the 32-pps clock, however, readout can be accomplished at approximately 10 words per second through the direct-access link only. The CDS has the option of selecting the speed to be used. The spacecraft output data is stored in a C&S OSE register and may be compared with data obtained from the CDS or data inserted locally. The

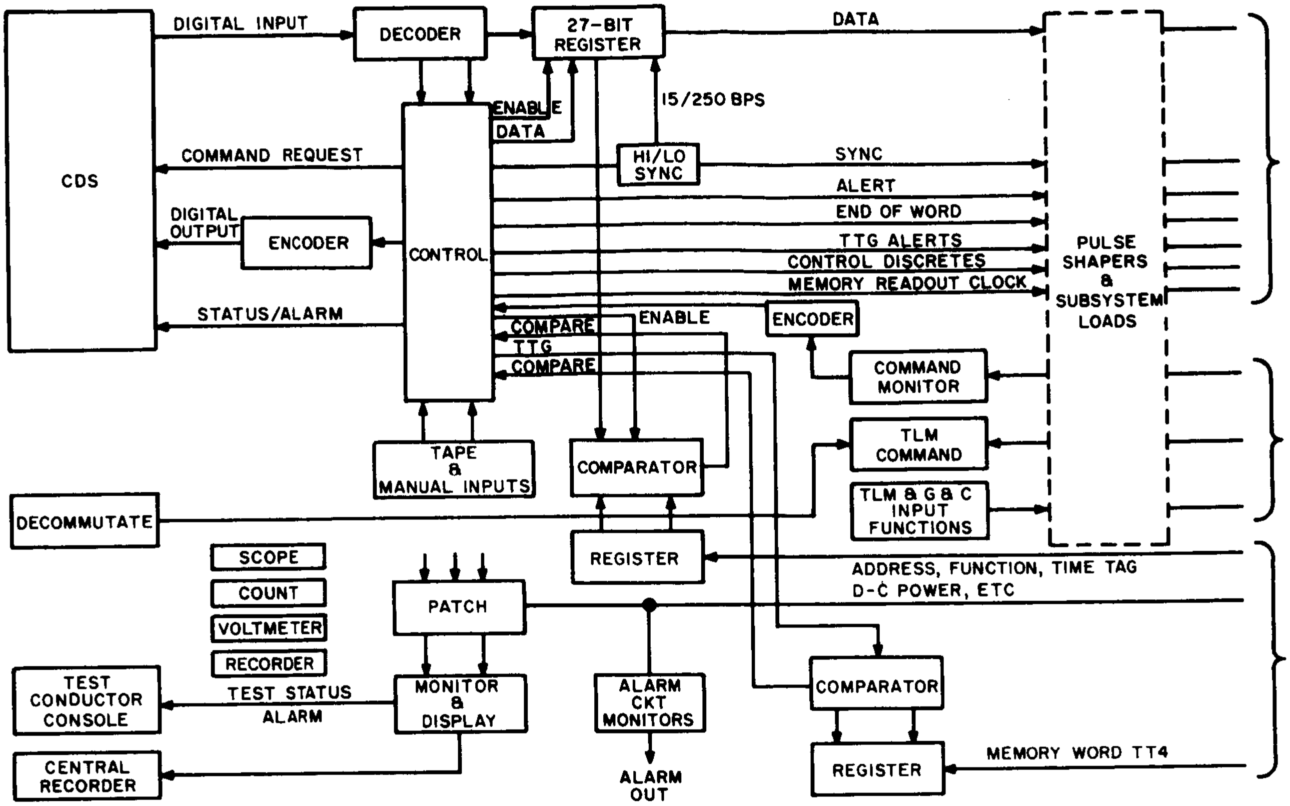

Figure 2-1. Controller and Sequencer Subsystem OSE Block Diagram

CDS can set up the comparison such that it is notified of the results of the comparison at all times, or only if no go, or the word can be read into the computer for comparison there.

2.1.1.3 Time-to-Go Registers. The Spacecraft Time-to-Go (TTG) Registers are loaded in the same manner as the memory with the exception that a TTG alert signal is sent to the register to be loaded, the CDS/C&S OSE can automatically load these registers using the control words described before.

2.1.1.4 Memory Word Time-to-Go. The Spacecraft Memory Word TTG counter is outputted in parallel through direct-access points to a register in the OSE. The TTG time can be compared with a CDS generated value of the expected TTG time. The CDS will input the TTG time using the control word method. This value will be used to verify the proper timing and countdown circuitry and to establish test time to synchronize the STC to the vehicle time.

2.1.2 Local Control

2.1.2.1 Manual Control. In addition to the CDS control described previously, the C&S OSE is capable of manually operating and controlling the C&S Subsystem. This is the normal mode of testing the C&S as a subsystem. Provision is made for manually loading and verifying the memory by utilizing switches that parallel the CDS inputs. In the manual mode the CDS inputs are disabled and the desired memory address and its contents are loaded manually into the OSE data registers. When the register is filled, the control discretes and sync are manually generated and the data is shifted serially into the memory either at the normal clock rate or at a manually generated rate. The TTG registers are loaded in the same manner with the exception that one of the TTG alerts is manually enabled when the OSE data registers are full.

The memory is verified by manually loading the address desired to be interrogated into the OSE data registers and then shifting it serially into the spacecraft memory. The memory location is then interrogated and its contents read out to the OSE where a visual or logical comparison can be made.

2.1.2.2 Tape Reader. A paper tape reader is available for use in the manual mode. Information to be stored in the C&S memory will be pre-programmed on paper tape and read into the memory in blocks of data or stepped by the operator one address at a time. The memory will be verified by having the tape programmed to request the data to be read out and comparing that data with pre-programmed data as required. The paper tape is also used to sequence the C&S subsystem and OSE through semi-automatic test routines.

Double verification of the data in the memory can be accomplished if the memory is first loaded under remote control from the CDS and then verified by using the paper tape to interrogate the memory. This assures that the CDS loaded the correct data by using two independent sources of data (the CDS program deck and the C&S OSE tape). This can also be done the other way around (i.e., the memory loaded from the tape and data verified in the CDS). Capability such as this is desirable to assure absolutely that the pad load is correct.

2.1.3 Displays

2.1.3.1 Status and Alarm. Power supply and clock signals will be continuously monitored for out-of-limit conditions. If an out-of-limit condition is detected, an alarm signal is generated which will both notify the CDS and automatically shut down the OSE in a prescribed sequence. Power-On, Test Mode, and other switches will be monitored to describe the status of the OSE. These switches will also be monitored by the CDS and the Test Conductor.

2.1.3.2 Monitor and Display. All input/output functions are brought into one side of a patchboard. The other side of the patch is connected to test equipment such as an oscilloscope, counter, voltmeter, and recorder. The register contents and telemetry will be continuously monitored and displayed.

2.1.3.3 Fault Isolation. The OSE is capable of isolating a fault to a replaceable assembly of the C&S. Testing for faults can be done manually or with the aid of special programs stored on paper tape.

2.1.3.4 Command Execution. When a command is executed by the C&S, the command decoder input is displayed on the C&S OSE along with a command-executed discrete. This data is to be used to verify what command should have been executed.

3.0 SUBSYSTEM TEST. In addition to the capabilities described in System Test the OSE will provide stimuli and loading that simulate the other subsystems that interface with the C&S.

3.1 Command Subsystem Interface. The OSE is capable of simulating data, sync, alert, and end-of-word signals normally obtained from the Command Subsystem. The same C&S OSE circuitry used to load the memory will be used but entered through the Command Interface.

3.2 C&S Commands. Approximately 200 commands from the spacecraft decoding matrix will be loaded and monitored by the OSE. Since the sequence in which they should be executed is known, one output pulse detector, set to trigger on a good output pulse, will be switched manually or by tape program to the desired point. Upon detection of the pulse the detector will so indicate visually to the operator or logically to the tape reader at which point it will be reset and switched to the next expected point. Time will be used as a basis for whether the output occurred prematurely, or late, or not at all. If this occurs, the vehicle clock will be held, and status made known to the operator.

During the above tests spurious outputs will be detected by using low level pulse detectors connected continuously to all command outputs. When the normal level pulse detector is connected to the output, the low level detector will be disconnected. Therefore, a low level detector firing, indicates improper operation. The vehicle clock will be stopped and by interrogating the low level detectors the one or ones that fired will be obtained, recorded and reset. Proper and improper responses will be recorded manually and/or automatically as selected.

3.3 G&C Interface. The OSE will provide the necessary gimbal and accelerometer signals to simulate inputs from the G&C Subsystem. The OSE will also monitor the quantitative commands that the C&S Subsystem supplies to the G&C spacecraft.

3.4 TLM Interface. The OSE will supply telemetry sync and gating signals to simulate inputs from the Telemetry Subsystem and will load and monitor the C&S Subsystem outputs to telemetry.

TABLE 5-1. CONTROLLER AND SEQUENCER FUNCTIONAL INTERFACES

Interface	Function	Input/ Output	Comments
C&S Subsystem (Umbilical)	Power	Output	
	Command Data	Output	
	Command Discretes	Output	11 Required
	G&C Discretes (load TTG Registers)	Output	6 Required
	PSP Discretes (load TTG Registers)	Output	2 Required
	Present TTG	Output	
C&S Subsystem (Direct Access)	T/M	Input	Subsystem Test Only
	T/M	Output	Subsystem Test Only
	G&C	Input	Subsystem Test Only
	G&C	Output	Subsystem Test Only
	CD	Output	Subsystem Test Only
	Memory Readout	Input	27 Bits - parallel
	TTG Readout	Input	8 Bits - parallel
	Clocks	Input	3 Required
CDS	Load Instructions	Input	24 Bits
	Command Request	Output	
	Output of Memory and Registers	Output	
	Status/Alarm	Output	
	TLM Data	Input	
Test Conductor	Status	Output	
Control Recorder	Status, Analogs	Output	
Ground Power	Power	Input	
Time Distribution	Time, Sync	Input	

3.5 Pulse Shaping and Loading. The OSE is capable of varying amplitude, rate, rise time, fall time, etc., of those signals that simulate other subsystem interface inputs. The OSE is also capable of applying variable loads to all C&S interface outputs.

4.0 SELF TEST. Prior to, and during, any test involving the C&S, the OSE is capable of performing a self-test to ensure that the display, status, alarm, and logic functions are operating properly. Output signals from the OSE will be fed back into the OSE whenever possible. Signals that do not have corresponding input/output will be simulated.

5.0 FUNCTIONAL INTERFACES. The functional interfaces of the computer and sequencer are shown in Table 5-1.

PYROTECHNIC SUBSYSTEM OSE

1.0 SCOPE. This document describes the OSE required to support the pyrotechnic subsystem during all subsystem and system tests described in OSE Objectives and Design Criteria (VC260SR101), STC Design Characteristics and Restraints (VC260SR102) and STC Functional Description (VC260FD100). These documents describe the philosophy and approach to be used, how the subsystem OSE is to be integrated into the STC and the interfaces between the subsystem OSE and the STC. This document will, therefore, emphasize the interface between the OSE and the spacecraft subsystem.

2.0 SYSTEM TEST DESCRIPTION

2.1 Pyrotechnic Subsystem Test Console. Figure 2-1 shows schematically the interface between the OSE and the Pyrotechnic Subsystem (only one half shown). The diagram indicates that the OSE will perform the following functions:

a. Provides means for remotely operating the separation switches during tests or simulating switch operation when the spacecraft is mounted on the booster or booster simulator.

b. Monitors the input 2.4-kc voltage and current and generates an overcurrent alarm.

c. Monitors the input voltage to each capacitor bank.

d. Monitors the total firing current for each capacitor bank.

e. Uses d, above, to generate a squib-fired indication.

f. Provides squib simulators and monitors (see Section 2.2).

g. Provides means for displaying and recording all of the above data such that the signals can be visually analyzed.

h. Transfers all of the above signals to the CDS for its use.

i. Provides means for monitoring the TLM data.

2.2 Squib Simulators. The primary purpose of the squib simulators is to provide a load that will demonstrate that the capacitor bank, power switching, and vehicle cabling can deliver enough energy to fire a worst-case squib. It is required to have an indication of what squib fired; desirable to have the simulator simulate, in a gross manner, the power profile of a squib; and to have the simulator resetable such that it can be used to verify the generation of squib firing commands from more than one source (i.e., C&S or Command) and such that it can be used to verify that no other commands cause it to be fired. This should all be done with a minimum of number wires, etc.

Figure 2-2 shows an approach to accomplishing this using no additonal leads. The squibs are simulated by a resistance and the set coil of a latching relay in parallel. Upon detecting a firing pulse, the relay contact will open the circuit thereby simulating the opening of the squib bridge wire. This opening of the circuit can be prevented by the operation of a manual switch thereby simulating to the capacitor bank, logic, etc., the worst case of a non-clearing squib. The continuity loop is used for two purposes. The relay contact switches-in

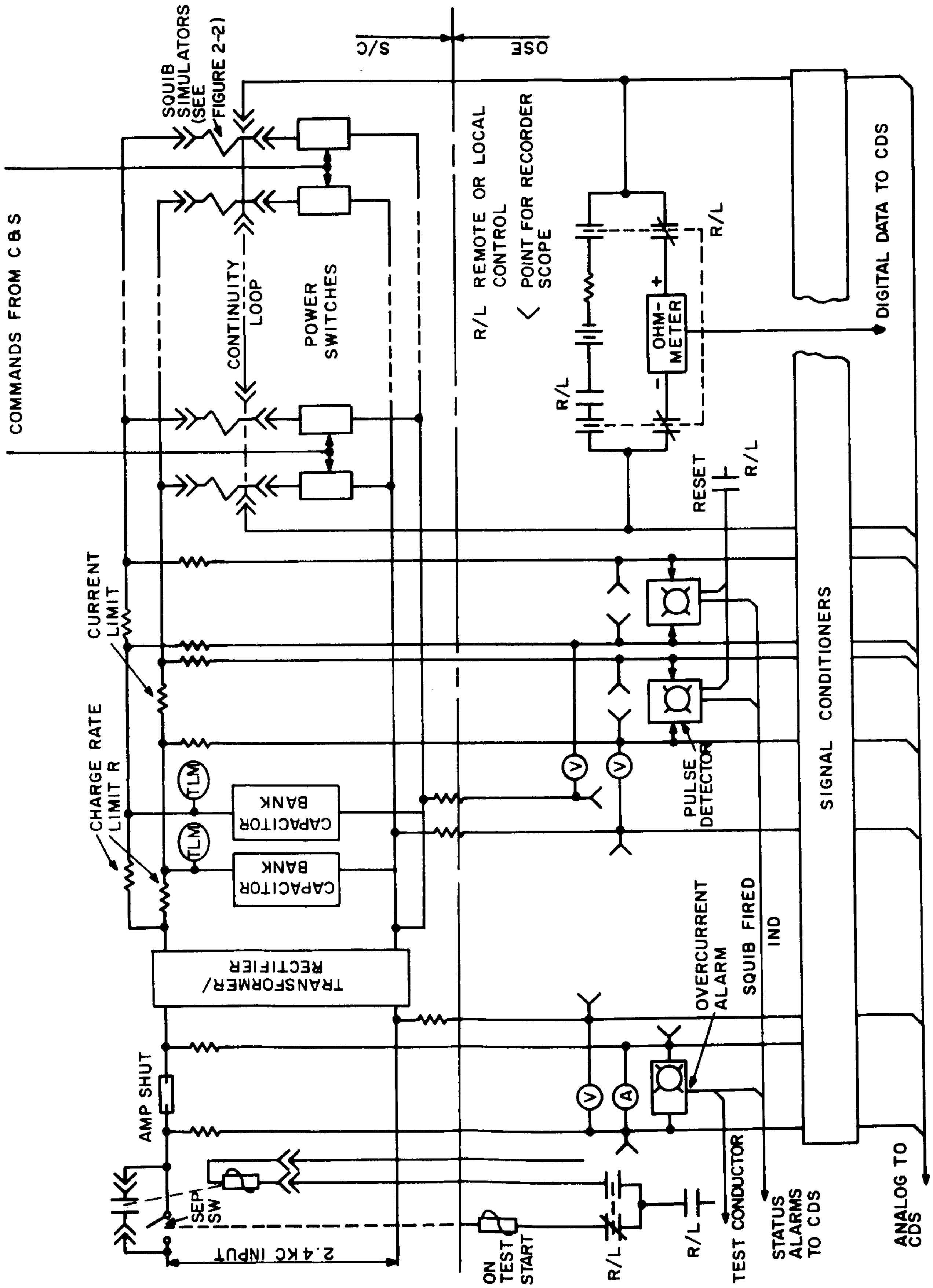

Figure 2-1. Pyrotechnics Subsystem to OSE Interface

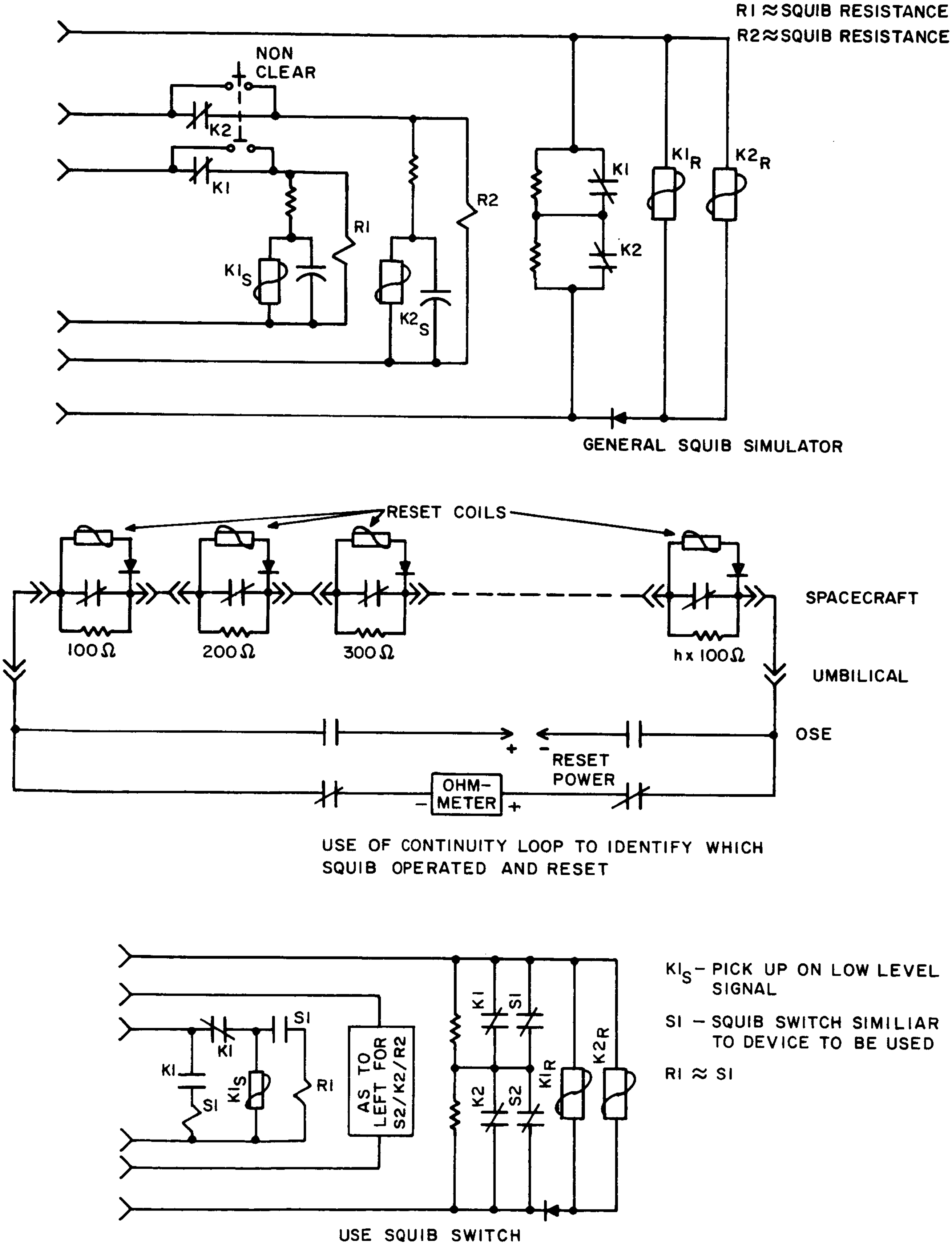

Figure 2-2. Squib Simulator

a value of resistance that is peculiar to that particular simulator and thereby allows the identification of which squib fired by measuring the resistance of the continuity loop. The second use is that the relay reset coil is also in the continuity loop isolated from the resistance by a diode, thus, allowing the resistance measurement to be made unaffected by the reset coils. Resetting of the simulator is accomplished by applying power of the correct polarity.

It should be noted that the limitations of this scheme are the number of squibs to be fired normally before they are reset, or the number that might be fired accidently; the limitations being the resolution to which the resistances have to be measured and the number of reset coils in series that can be reset by the reset signal. Both of these problems can be overcome to a degree by providing additional continuity loops within the spacecraft. It is also expected that normally the squib simulators would be reset after each firing command and verification of the resistance of the continuity loop as being correct.

An alternate squib simulator scheme is shown in which a squib operated switch is used as a simulator for the first firing command which then enables a scheme as above to be used thereon. At the start of each major test, a new squib switch would be installed. This scheme gives a more realistic power profile for the first shot and would allow the relay circuit to be used to simulate a low level squib such that it would fire on noise and yet give reasonable assurance that the squib can be fired from its normal alternate sources as well as maintain the resettable feature.

The squib simulators will be physically the same as the actual devices and will be mounted in the vehicle during tests in the same manner as the real devices. These squib locations must be easily accessible for installing both the simulators and the real squibs.

The squib simulators will be required during vibration and thermal vacuum tests on the system and must be capable of withstanding those environments.

3.0 SUBSYSTEM TESTS. The schematic for the subsystem tests would look very much like that for the systems test with the OSE supplying the 2.4-kc power, command signals required to fire each simulator, meters to monitor the TLM sensors, and additional cabling. The squib simulators should be the same as those used for system test. However, some testing should be done to verify that low level signals are not being generated. This can be done by using low level squib simulators as described above. The limiting input signal characteristics required to fire each power switch will be obtained by using a variable signal generator for command subsystem input simulation.

The effects of 2.4-kc input variations will be obtained by varying all significant parameters one at a time.

4.0 SELF TEST. The Pyrotechnic Subsystem OSE will provide capabilities to completely verify its operation. This will be accomplished by switching-in calibration signals into the sensing circuits, providing convenient means for verifying the squib simulators operation when removed from the spacecraft, providing loads for the reset power supply and 2.4-kc power supply, providing variable signals to test alarm trip points and squib fire points, etc.

TABLE 5-1. PYROTECHNIC SUBSYSTEM INTERFACES

Interface	Function	Input/Output
Umbilical	Continuity Safe/Arm	Input Output
Test Connector	Input Power Firing Signals	Input Input
Squib Simulators	See Section 2.2	
CDS	Analog Signals Digital Data Remote Control Alarm	Outputs Outputs Inputs Outputs
Test Conductor	Alarm Status	Outputs Output
Central Recorder	Analog Signals	Output
Power Distribution	Power	Input

5.0 INTERFACES. The pyrotechnic subsystem interfaces are shown in Table 5-1.

VC265FD102

STRUCTURAL/MECHANICAL SUBSYSTEM OSE

1.0 SCOPE. The Structural and Mechanical Subsystem OSE is required to perform subsystem level tests upon the Structural and Mechanical Subsystem, and to enable this subsystem to be tested during system tests when the OSE is a part of the system test complex.

The Structural and Mechanical Subsystem is defined to be the articulating portions of the Flight Spacecraft, i.e., the Planetary Scan Platform (PSP) and the High-Gain Antenna (HGA). In view of the expectation that the PSP and its OSE are to be furnished as GFE, the Structural and Mechanical OSE is then the OSE required for test of the High-Gain Antenna articulation drive.

2.0 FUNCTIONAL DESCRIPTION. The Structural and Mechanical Subsystem OSE is used to stimulate the High-Gain Antenna drive, and monitor its response during subsystem tests. During spacecraft system tests, it provides the capability of monitoring the response of the drive and gimbals also. Figure 2-1 is a functional block diagram of the OSE.

The OSE is used to perform the following specific functions:

a. Provide controlled power to the HGA gimbal.

b. Monitor HGA gimbal position feedbacks to ascertain gimbal position.

c. Monitor gimbal motor currents to evaluate correct operation of the motors and to guard against damage.

d. Monitor temperatures and pressures to verify operating parameters and prevent conditions which are hazardous to equipment or personnel.

e. Provide input drive signals in digital form to the gimbal drive logic.

f. Monitor the drive inputs to assure that only safe commands are issued to the gimbal.

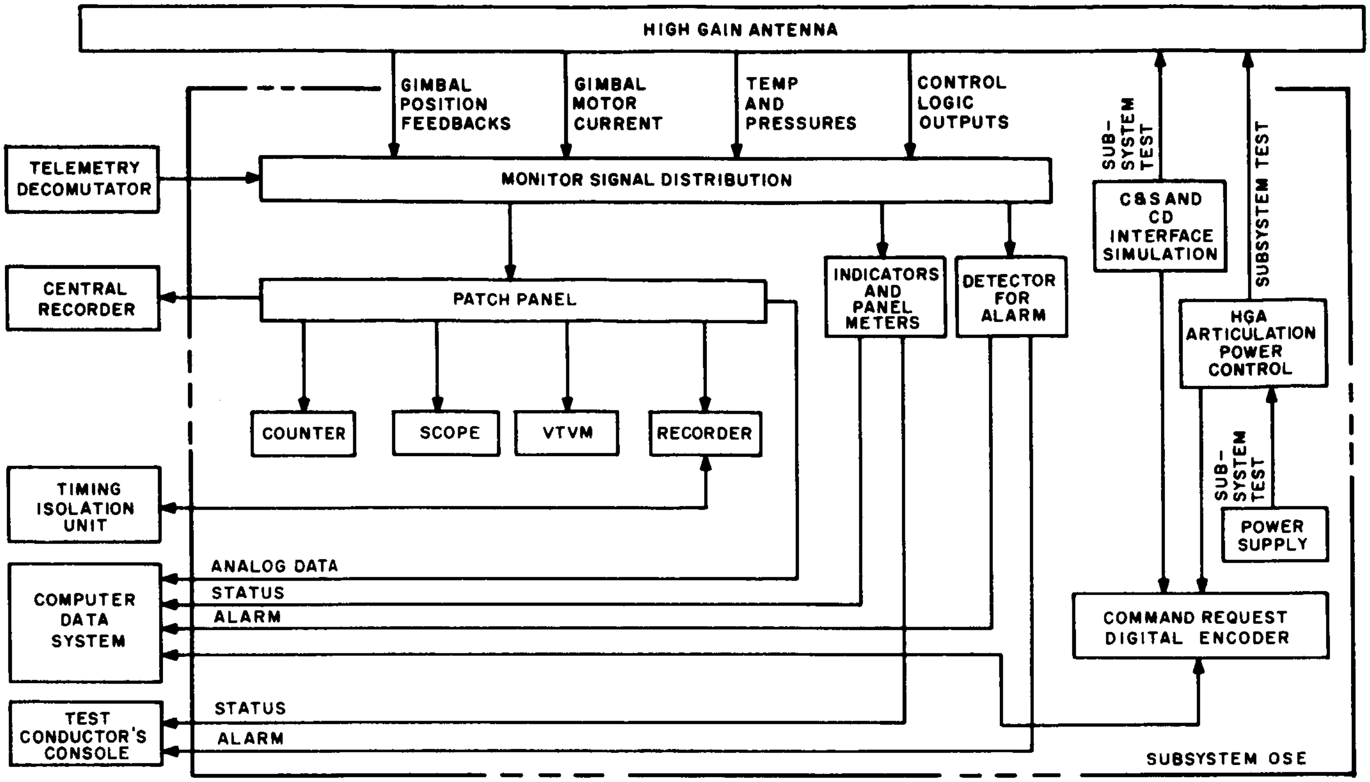

Figure 2-1. Structural and Mechanical Subsystem OSE, Block Diagram

The gimbal position and other gimbal responses and their quantities are monitored with the instrumentation shown in the block diagram. The monitored quantities, from direct access test points and from the flight convectors, is brought to the OSE through a signal distribution system and is displayed on the instruments patched for the measurement. The parameters which are measured by these instruments are also available to the Central Recording System or Computer Data System when a system test is conducted.

Status indicators, such as Power-On etc., and unsafe condition alarm circuits, such as overload detectors, are displayed at the OSE and are also provided to the Test Conductor's Console and to the Computer Data System during system test.

The OSE is required to provide simulated C&S and Command Decoder inputs to the drive during a subsystem test. These inputs are controlled at the OSE where the engineer can cause the OSE logic to generate the proper pulse train. This pulse train can be verified, as a self-test feature, to prevent damage occurring to the flight equipment. These inputs can be controlled remotely by the CDS, if used, in either subsystem tests or system tests.

Power is supplied to the flight equipment under local control from the OSE, or under remote control from the CDS.

Angle displacement indicators, mounted at the gimbal may be used as a direct input of gimbal position to the OSE.

3.0 PHYSICAL CHARACTERISTICS. The Structural and Mechanical Subsystem OSE package concept recommended is to have the OSE self-contained, and mounted in consoles of its own. This requires two racks due to the volume required by the instrumentation recommended. The integral packaging will permit the OSE to be used to support testing of the gimbal package when it is separated from the spacecraft and from adjacent equipment. In the previous Task A report, it was recommended that this OSE be packaged with the G&C OSE in order to avoid duplication of instruments and proliferation of consoles. This recommendation has been reconsidered and it is now felt that the penalties of additional equipment are outweighed by the advantage of being able to conduct tests of the gimbal completely independently of G&C OSE.

4.0 INTERFACES. The functional interfaces of the Structural and Mechanical Subsystem OSE with the spacecraft and with other OSE are shown in Table 4-1.

TABLE 4-1. STRUCTURAL AND MECHANICAL INTERFACE

Interfacing Equipment	Function	Input/Output	Comments	Interfacing Equipment	Function	Input/Output	Comments
Structural and Mechanical Subsystem (High Gain Antenna)	Power	Output		Central Recorder	Analog Data		Selectable from patchboard
	Gimbal 1/4 degree Step Indications	Input					
	Motor Pressures	Input		Timing Isolation Unit	Time Signals	Input	
	Motor Temperatures	Input					
	1/4 degree Step Commands and Slew Commands ± both axis, Stop Command	Output	Subsystem Test Only	Computer Data System	Analog Data	Output	Selectable from patchboard
					Commanded Operation	Input	
Telemetry	Telemetry Sensor Data	Input	Via hardwire from S&M Subsystem during subsystem test		Verification of Command Receipt	Output	
					Status and Alarm	Output	
				Test Conductor's Console	Status and Alarm	Output	
				Base Power and Distribution	208-v, 3-phase, 60-cps Power	Input	
				Communications Net	Communications	Input/Output	

VC265FD103

TEMPERATURE CONTROL SUBSYSTEM OSE

1.0 SCOPE. This document describes the OSE required to support the Temperature Control Subsystem during all system test activities as described in OSE Objectives and Design Criteria (VC260SR101), STC Design Characteristics and Restraints (VC260SR102), and STC Functional Description (VC260FD100). These documents describe the philosophy and approach to be used, how the subsystem OSE is to be integrated into the STC, and the interfaces between the subsystem OSE and the STC. This document will, therefore, emphasize the interface between the OSE and the spacecraft temperature control subsystem.

2.0 FUNCTIONAL DESCRIPTION

2.1 General. The Temperature Control Subsystem OSE provides the capability to verify that the spacecraft components are maintained within the specified temperature limits and to monitor the operation of the temperature control shutter assemblies. The OSE is intended primarily for tests of the assembled Spacecraft Bus, and provides the responsible engineer with a centralized set of displays within the system test complex which enables spacecraft thermal control to be assessed. It can also be used in conjunction with other subsystem OSE to support tests of that subsystem.

2.2 Sensor Output Display. The Temperature Control Subsystem OSE provides the capability to display the telemetry data corresponding to the position of each set of temperature control shutters. This data will be received in analog form from the STC TLM data decoder and will be displayed on the console. The telemetry data corresponding to various component temperatures will also be displayed. Due to the large amount of temperature information available, the data will be displayed in functional groups, selectable by the thermal control engineer. The temperature telemetry data will also be received in analog form from the TLM data decoder.

2.3 Additional Monitors. The Temperature Control OSE will also display monitors associated with the environmental control equipment used during the system parameter variation tests. This equipment is used to allow testing of the spacecraft at reduced and elevated temperatures. While the equipment is operated manually and locally, the Temperature Control OSE will monitor its performance and the additional OSE temperature sensors provided for this test.

During thermal-vacuum tests, this console will be augmented by a special purpose console used to monitor the additional test sensors that will be required to evaluate and monitor the spacecraft during the test.

3.0 FUNCTIONAL INTERFACES

Interface	Function	I/O
TLM Data Decoder	Telemetry	Input
Power and Distribution OSE	Console Power	Input

VC266FD101

POWER SUBSYSTEM OSE

1.0 SCOPE. The Power Subsystem OSE provides all the equipment required to support the Power Subsystem during system and subsystem tests. It provides also for solar panel and battery simulation, and for tests of these components. It forms a part of the STC for System Tests.

2.0 FUNCTIONAL DESCRIPTION. System Tests are provided for as shown in Figure 2-1.

2.1 Simulated Solar Panel Power Source. A variable, current limited power source will supply simulated solar panel power to the power subsystem via the normal solar panel connectors. The power source will be switched to each of the solar panel connections one at a time, at which point the current is measured as the voltage is varied over the range required to verify the proper operation of the zener, isolation diodes, and current carrying capacity of the interface. This will be repeated for various voltage levels of the Ground Power Source.

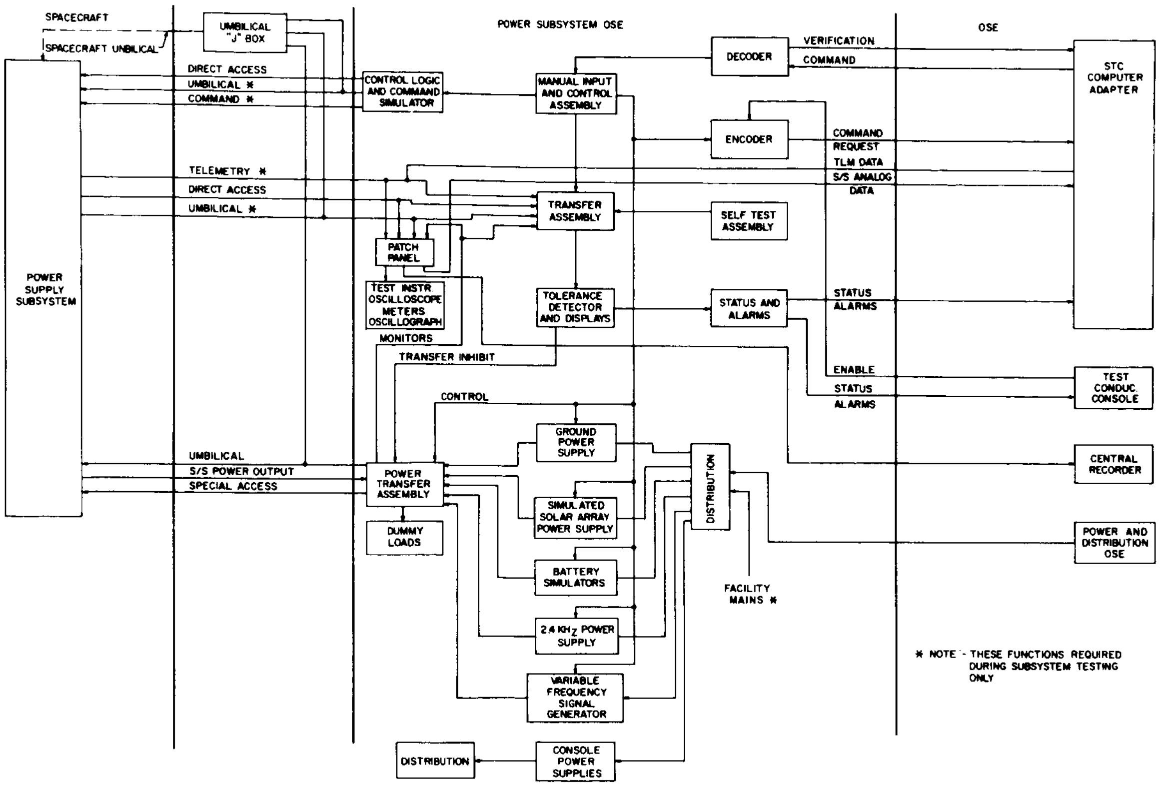

Figure 2-1. System Tests

2.2 Battery Simulators. Three independent, variable, current-limited power supplies will be provided to simulate the spacecraft batteries. The simulated charge/discharge currents will be measured as the voltage outputs of the simulators are varied over the battery charge/discharge limits.

2.3 Ground Power Source. A variable, controlled, current-limited power supply will provide power to the spacecraft via the normal umbilical connection. This Ground Power is connected to the solar array bus and will be used to simulate the solar array. Transfer from external to internal power will be accomplished by decreasing the voltage level of the external power source until the batteries (or battery simulators) assume all of the spacecraft load at which time the external source is switched out. Power transfer from internal to external will be essentially the reverse of the above with the voltage level of the ground power source being gradually increased until it assumes the spacecraft load. The input current will be continuously monitored and recorded.

2.4 Simulated Loads. Controlled dummy loads will be provided to simulate those portions of the spacecraft which normally interface with the power subsystem, but which are temporarily unavailable (e.g., during inter-subsystem testing).

2.5 2.4-kc Power Source. A variable 2.4-kc power source will be provided to replace the spacecraft 2.4-kc power source during parameter variation tests. The significant parameters (amplitude, frequency, transient overshoots) will be variable over the full tolerance limits to verify the capability of other spacecraft subsystems to operate from a marginal power source.

2.6 Master Clock. The OSE will provide the capability to perform a precise measurement of the Master Clock frequency from which an attempt will be made to establish the clock stability. In addition, the OSE will provide the capability to insert a variable frequency (268.8 kHz ±15% in excess of the tolerance limits) signal into the synchronizer chain in order to vary all of the basic frequencies used by the spacecraft simultaneously.

2.7 Measurement and Fault Isolation. All input data from the power subsystem will be brought to a central patchboard. The input data will, therefore, be selectively available for display, recording, and analysis as required for fault isolation using the standard test instruments contained in the OSE console (oscilloscope, frequency meter, ac-dc digital voltmeter, and oscillograph). Panel meters will be provided to continuously display the external power applied to the spacecraft and the power subsystem inverter outputs, regulator outputs, and temperature sensor outputs.

2.8 Tolerance Detection and Alarm. Continuous monitoring of critical parameters (external power sources, regulator outputs, inverter outputs, battery voltages, and subsystem temperature sensor outputs) will be provided to determine if the parameter is in tolerance, out-high, or out-low. Visual and audio alarms will be provided to signify out-of-tolerance conditions. In addition, interlocks will be provided to prevent the application of out-of-limits-high external voltages to the spacecraft.

2.9 Self-Test. Self-test will be performed by simulating the spacecraft power subsystem functions which are measured or controlled by the OSE, and checking the response of all monitors and tolerance detectors to both marginal and nominal conditions.

3.0 FUNCTIONAL DESCRIPTION - SUBSYSTEM TEST

3.1 General. In addition to the capabilities described in system test, the Power Subsystem OSE will provide the simulated spacecraft interfaces that are required to control and monitor the operation of the power subsystem including the solar panels and batteries.

3.2 Power Subsystem Test Console

3.2.1 Spacecraft Command Simulation. The OSE will generate and provide commands directly to the power subsystem. These commands will simulate actual spacecraft commands and will be routed to the subsystem through the in-flight connectors at the power subsystem bays.

3.2.2 Telemetry Data Monitoring. Capability will be provided to supply the required sensor stimulation as well as to measure the outputs of all power subsystem telemetry sensors.

3.3 Solar Panel Test Stimulator. This OSE will provide the capability to stimulate one solar panel at a time, using a tungsten light source. A gross check of the panel output will be made to verify that the panel is functioning properly.

3.4 Battery Charge and Test Set. This test set will provide the capability to test and charge each battery as a component before it is installed in the spacecraft. This set will contain a charge regulator, in series with a d-c power supply to regulate battery charging voltage and current. In addition, it will provide the capability to load the battery for test purposes as well as to measure the battery charge/discharge current, cell voltages, and net ampere-hour discharge.

4.0 PERFORMANCE PARAMETERS. The performance parameters of the Power Subsystem OSE are shown in Table 4-1.

TABLE 4-1. POWER SUBSYSTEM PERFORMANCE PARAMETERS

Parameter	Range	Accuracy	Comments	Parameter	Range	Accuracy	Comments
External power	0-60 V dc 30a			Battery Charging Power	0-50 V dc 0-3a		
Telemetry Sensor Output Measurement	0-3.2 V dc 0-100 mv Digital	0.1% 0.1%	A precision 1-ma supply is required to stimulate the temper-ature sensors	D C Voltage Measurement	0-100v	0.1%	
2.4-KH Power	0-60 V rms 0-6A 20-2.8 KH			D C Current Measurement	0-40a	0.1%	
Simulated Solar Panel Power	0-70 V dc 0-2A			A C Voltage Measurement	0-100v	0.1%	
Simulated Battery Power	0-50 V dc -7 to 25a			A C Current Measurement	0-10a	0.1%	
				Frequency Measurement	1.0H2-500 KHZ	±0.00002%	

5.0 FUNCTIONAL INTERFACES. The functional interfaces of the Power Subsystem OSE are shown in Table 5-1.

TABLE 5-1. POWER PERFORMANCE FUNCTIONAL INTERFACE

Interface	Function	I/O	Remarks	Interface	Function	I/O	Remarks
CDS Adapter	Command	Input	13 required	Power Subsystem (Umbilical) (Cont)	Internal Power Enable	Output	3 required
	Command Verification	Output			Battery Monitors	Input	7 required
	Command Request	Output	23 required		Bus Voltage Monitor	Input	
	Power Subsystem Analog Data	Output			Power Enable Monitor	Input	
	Telemetry Data	Input	3 required		Telemetry	Input	Subsystem tests only
	Status/Alarms	Output			Telemetry (Sensor Stimulation)	Output	Subsystem tests only
Test Conductor's	Command Request Enable	Input			Command	Output	Subsystem tests only
Console	Status/Alarms	Output			Regulator Monitors	Input	4 required
Central Recorder	Selected Data	Output			Inverter Monitors	Input	10 required
STC Power Distribution	Console Power	Input			Battery Monitors	Input	24 required
					Clock Signal Monitor	Input	2 required
					Power Switch Monitors	Input	11 required
Power Subsystem (Umbilical)	Ground Power	Output	3 required		Power Bus Monitors	Input	3 required
					Fault Sensor Stimulation	Output	5 required

VC268FD101

MIDCOURSE PROPULSION SUBSYSTEM OSE

1.0 SCOPE. The Midcourse (MC) Propulsion Subsystem OSE is required to perform subsystem tests on the MC propulsion, when that subsystem is tested by itself, and to enable system tests to be performed on the spacecraft after the MC propulsion has been integrated with the spacecraft. The OSE will become a part of the Spacecraft System Test Complex. The OSE will perform a series of critical component tests in both the subsystem and system test modes.

2.0 FUNCTIONAL DESCRIPTION

2.1 General. The testing of the MC subsystem is constrained to a test philosophy unique to propulsion subsystems. Propulsion subsystems cannot be end-to-end tested, or have their operation simulated, to the same degree that the other spacecraft subsystems can. This limitation, which is principally applicable to flight units is due to the following particulars:

a. Testing of one-time or limited-life-cycle devices (frangible diaphrams, etc.), on a unit ready for flight, destroys the flight ready status of the unit.

b. Simulated subsystem operation of propulsion subsystems tends to produce contamination and wear in reliability critical areas.

This philosophy, therefore, dictates the following approach to OSE for testing flight units of propulsion.

a. Development testing will have established and verified the design and performance capability of the propulsion. (This includes static firing of development and non-flight propulsion subsystems.)

b. A statistical test program will have established a data base on characteristics of components and assemblies of the propulsion subsystem.

c. That subsystem and system tests, of propulsion intended for flight, will be limited to tests of critical components and critical interactions, where the testing will not prejudice flight performance.

2.2 Subsystem and System Tests. The following paragraphs identify the MC Propulsion Subsystem tests which will be conducted on flight equipment. These tests are also part of the Spacecraft System Tests in that the test points and access provisions will be made available so that the tests can be conducted on a subsystem installed into the Spacecraft. Figure 2-1 is a functional block diagram of the OSE recommended for this testing. It consists of two consoles, one for pneumatics, which must be in close proximity to the propulsion, and the other for electrical control and display. The control and display console provides the capability for the MC propulsion engineer to remotely control and conduct tests from a position within the System Test Complex, or from any other location at which the control and display console may be located.

The components which are to be tested by the OSE are:

a. Electrically energized valves
b. Pressurant gas regulators
c. Relief valves
d. Positive expulsion units
e. Jet vanes
f. Telemetry sensors

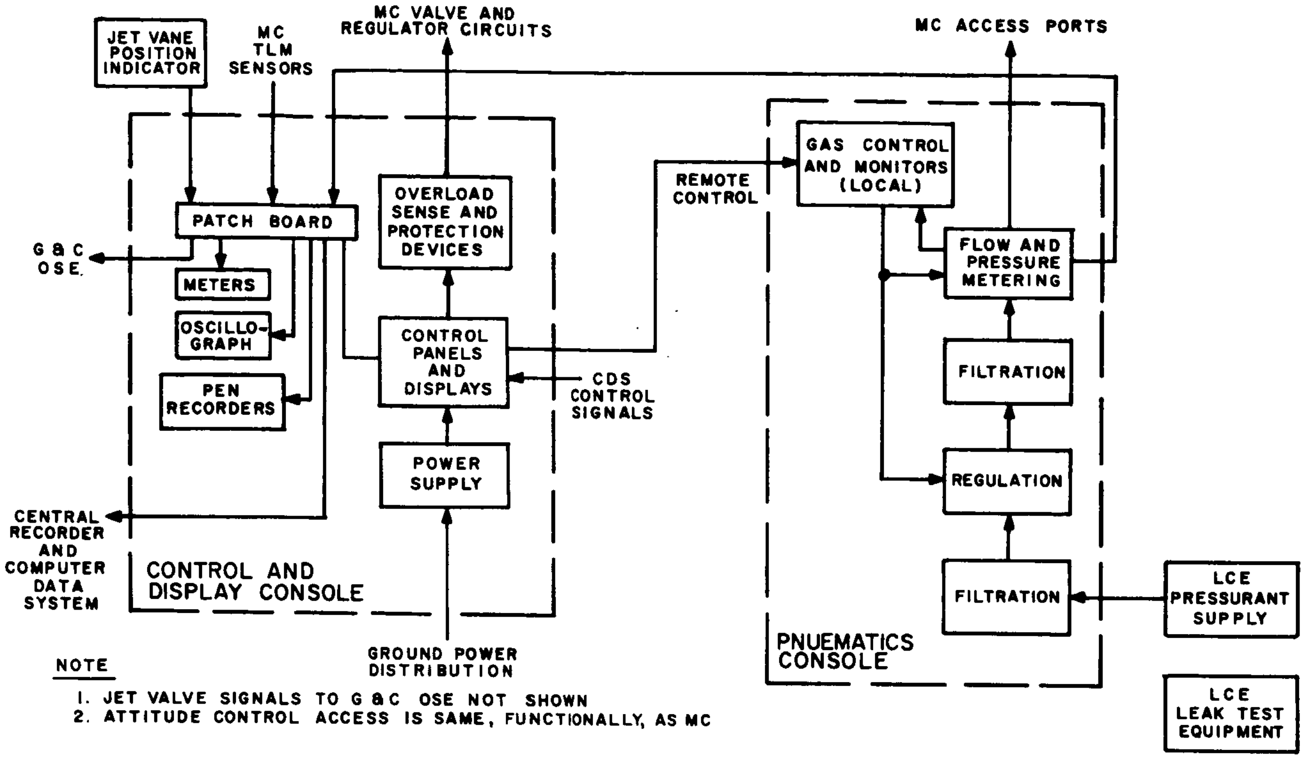

Figure 2-1. Midcourse Propulsion System OSE, Functional Block Diagram

2.2.1 Valve Tests. Satisfactory performance of any individual electrically energized valve in the propulsion subsystem can be demonstrated by evaluating the valve parameters outlined below. It is also necessary to determine the margin that exists between the critical parameter and the nominal system value of the parameter. The console will measure the following parameters on an individual component basis and determine the margin that exists.

a. Minimum actuating voltages and Hi-Pot voltage
b. Energizing current and insulation resistance
c. Valve response time
d. Valve seat leakage
e. Valve flow characteristics and full open determination

These parameter tests are performed by using the test connection port upstream of the quadredundant valve packages.

The OSE will:

a. Supply variable d-c control voltage to the electrically energized valve and record the voltage at which actuation occurs (actuation of valve indicator), at zero line pressure and at partial line pressure. To evaluate the electrical condition of the actuator, a Hi-Pot circuit is used to impress a high potential between the actuator and main valve body.

b. Measure valve current and power drain when the valve is actuated.

c. Measure valve response time with a recording oscillograph. This response time (electrical signature) must be determined for both opening and closing of the valve. Full open position can be verified by this signature.

d. Determine the valve seat leakage by applying normal line pressure with helium gas to the inlet of the valve. A mass spectrometer is then used to measure the leakage rate across the valve seat. (The LCE leak test equipment and the LCE gas supply is used.

e. Sequence check the electrical harness, disconnected from the propulsion system, and connected to a recorder.

2.2.3 Pressurant Gas Regulators. Tests of operating characteristics will consist of: (a) regulation, (b) flow, and (c) lock-up.

A typical test of the gas pressurant regulator unit simulates having the MC regulator subsystem initially actuated. The subsystem characteristics that must be simulated are:

a. Pressurant Tanks - at highest pressure (3600 psi)
b. Downstream Gas Ullage - Minimum (propellant tanks full)
c. Surge flow into regulation system from squib valve
d. Regulation system wide open

The regulator must respond in time to prevent overpressure of the propellant tanks without relief valve actuation. The Remote Pneumatic Unit will be used to apply high pressure gas to the regulator inlet and monitor pressure in the outlet line to the OSE with the outlet shut-off valve closed.

When the propulsion system is cut off and flow from the tank ceases, the regulator must sense this change and shut off gas flow with a minimum over-shoot in tank pressure.

Cold gas is used in these tests in lieu of liquid.

2.2.4 Relief Valve Demonstration. Satisfactory performance of any relief valve is determined by the following type of measurement or test:

a. Cracking pressure
b. Reseating pressure
c. Relief (overpressure) flow characteristics
d. Sealing capability

Operation of the relief valves requires using the test connection upstream of the relief valve and upstream of the burst diaphragm to balance pressure loads on the burst diaphragm. The console contains a filter gas supply which can supply gradually increasing pneumatic pressure in a dead-ended system until the relief valve reaches its cracking pressure and flow starts through the valve. The cracking pressure will be recorded. The pressure can then be increased and the flow characteristic of the relief valve above the cracking pressure recorded in small pressure increments. By gradually decreasing the supply pressure, the point is determined at which the valve reseats itself. The repeatability of the relief valve operation is determined by performing several test cycles on the valve.

2.2.5 Positive Expulsion Test. Positive expulsion propellant supply tanks are used to assure a propellant supply to the engine under any acceleration conditions. The tanks contain an internal bladder connected to the propellant tank outlet. When the bladder is pressurized externally, the propellant is forced to the engine under the proper supply pressure.

To maintain the maximum remaining cycle life on the bladder, the following procedure will be used: the propulsion subcontractor will leave the bladder in an expanded condition with a slight internal helium pressure. The bladder is maintained in this expanded condition up to the time of propellant loading.

The OSE can verify the internal bladder position as follows: from the test connection on the upstream (gas inlet) side of the bladder a line is connected to a gas water trap gage on the test set. A helium pressurization line is connected from the downstream (liquid side) test connector to the console. The helium gas pressure in the bladder is increased to 6 psi. Any gas flow into the water trap indicates an expansion of the bladder and should be measured to indicate the amount of bladder collapse.

The OSE can also be used to verify the condition of the bladder. Helium gas will be stored in the bladder for extended periods. Very low levels of bladder leakage can be detected by using the low pressure helium "sniffing" system and the mass spectrometer on the upstream gas side test connection. Safety features incorporated in the console prevent high vacuum or high pressure from being applied on the upstream or downstream side of the bladder and resulting high pressure across the bladder.

2.2.6 Jet Vane Test. The console has the capability of monitoring the 16 jet vane positions and mechanically loading the vanes and measuring the resultant torque values.

2.2.7 Telemetry Sensor Calibration and Test. The electrical control and display console will be used to control sensor calibration and test. The sensors to be tested are the pressure, temperature and motion sensors. Excitation voltages will be applied to the sensors. The pneumatic supply will be applied at the proper test ports for pressure and movement sensors. Locally applied heat will be used to activate temperature sensors.

4.0 INTERFACES. The functional interfaces of the MC Propulsion OSE are summarized in Table 4-1.

5.0 PHYSICAL CHARACTERISTICS. The MC Propulsion OSE Console for the Systems Test Complex will consist of one control and display console and a pneumatics console. Electrical power supplies, electrical measurements and calibration equipment, selector switches, recording equipment and self-testing equipment will be in the control and display console. During subsystem testing when tests may be performed in the immediate vicinity of the subsystem, the remote pneumatic control unit can be located adjacent to the console.

TABLE 4-1. MIDCOURSE PROPULSION FUNCTIONAL INTERFACE

Interfacing Equipment	Function	Input/Output	Remarks
MC Propulsion	Component Control and Monitoring	-	Pneumatic and Electrical
Test Conductor's Console	Status	Output	Switch closures
Computer Data System	Test Data Control and Enable Signals	Output Input	Analog and Discrete (for coordination of system tests)
Control Recorder System	Analog measurement	Output	
Control Timing	Time signals	Input	
Ground Power Distribution	Ground power	Input	
LCE	Gas supply	Input	Pressurant cart, leak detector
G&C OSE	Cold gas Jet vane tests	Output Output	Gas flow Vane position indications

The Propulsion OSE unit will supply power to the subsystem. The OSE power supply has a bank of overload protection circuits to provide power protection for the specific component or subassembly being tested. Any intermittent or continuous shorting will automatically disconnect the power to the propulsion module and energize an alarm circuit in the OSE.

Pneumatic safety provisions of two types are required within the propulsion OSE; equipment is provided which prevents contamination or overpressuring of the propulsion subsystem in all testing phases. Contamination control is maintained in the helium and nitrogen gas pressurization supply by a dual flow filtration system in the OSE at all propulsion interfaces.

The procedure and requirements discussed in this section assumes that sterilization of the cold gas and MC system is not required. However, if sterilization becomes a requirement, the test procedures will be changed to accommodate them.

The gas supply and leak test equipment required to implement these tests are physically a part of the LCE fluid or propellant service equipment required at the ESF. It is intended that they be used to support these tests, as the two activities are separated into two different portions of the test and prelaunch cycle.

The MC Propulsion OSE, and the gas supply, will also be used to support those Guidance and Control Subsystem tests which involve flowing of cold gas.

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ORBIT INJECTION PROPULSION SUBSYSTEM OSE

1.0 SCOPE. The Orbit Injection (OI) Propulsion Subsystem Operational Support Equipment (OSE) is required to perform subsystem tests on the OI propulsion module when that subsystem is tested alone, and to support system tests to be performed on the VOYAGER Spacecraft after the OI propulsion module has been integrated with the Spacecraft. The OI OSE will become a part of the Spacecraft's System Test Complex. The OSE will also perform a series of critical component tests in both the subsystem and system test modes.

2.0 FUNCTIONAL DESCRIPTION

2.1 Test Philosophy. The testing of the OI subsystem is constrained by a test philosophy unique to propulsion subsystems. Propulsion subsystems cannot be end-to-end tested, or have their operation simulated, to the same degree that the other spacecraft subsystems can. This limitation exists because: (1) testing of "one time" or limited life cycle devices (frangible diaphrams, etc.), on a unit ready for flight, destroys the flight ready status of the unit; and (2) simulated subsystem operation of propulsion subsystems tends to produce contamination and wear in areas of critical reliability.

The OI Propulsion Subsystem consists of a solid propellant engine with a thrust vector control mechanism consisting of pintle valves, Freon injectant, nitrogen pressurant, and associated plumbing and controls. It is the operation of the thrust vector control that is to be tested and verified as well as the safe/arm position of the solid propellant ignitor.

2.2 Subsystem and System Tests. The following paragraphs identify the OI Propulsion Subsystem tests to which flight equipment will be subjected. These tests are also part of the Spacecraft System Tests in that the test points and access provisions will be made available to permit tests to be conducted upon a subsystem installed in the Spacecraft. Figure 2-1 is a functional block diagram of the OSE recommended for this testing. It consists of two consoles: one for pneumatics, which must be in close proximity to the propulsion module, and

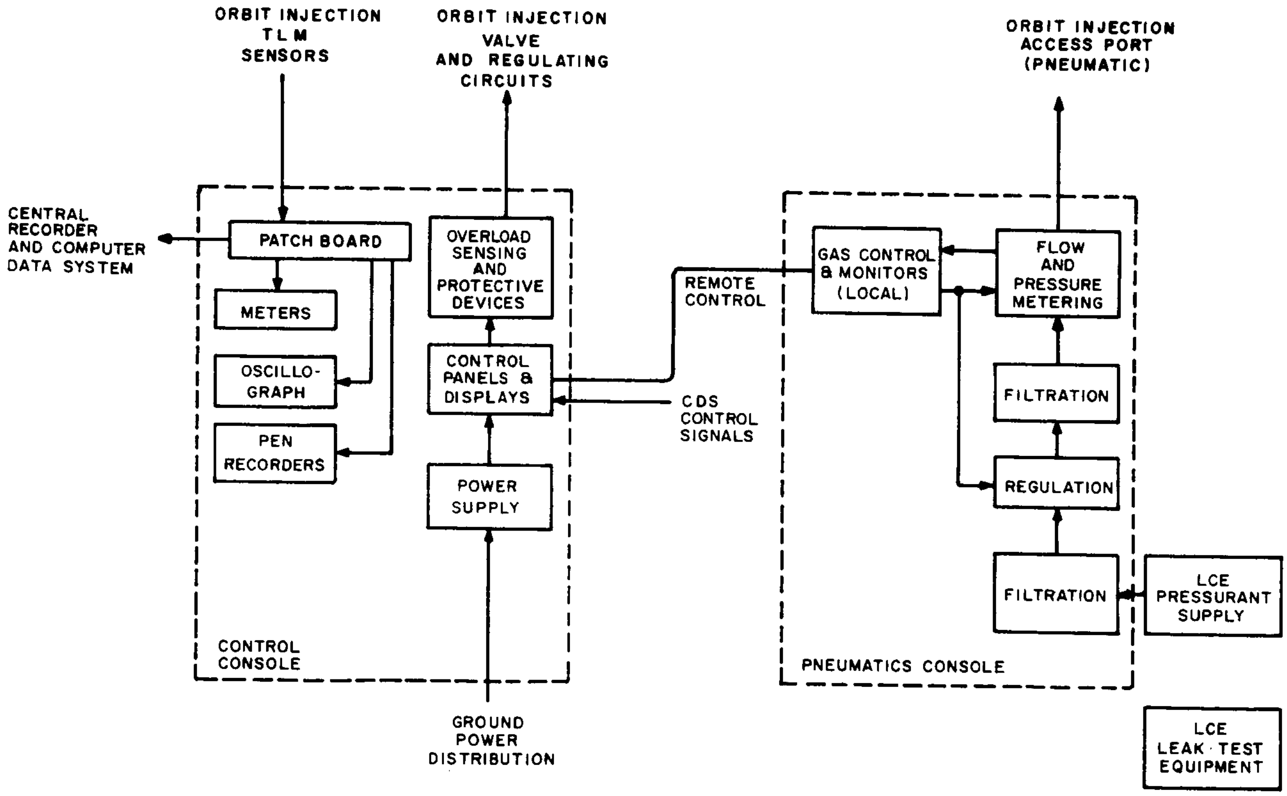

Figure 2-1. Functional Block Diagram, Orbit Injection Propulsion Subsystem

the other for electrical control and display. A control and display console provides the capability for OI propulsion test personnel to remotely control and conduct tests from a position within the System Test Complex, or from any other location at which the control and display console may be located.

The components which are to be tested by the OSE are:

a. Torque motor
b. Pressurant gas regulators
c. Hydraulic motors
d. Positive expulsion units
e. Pintle valves and thrust vector command (TVC) response
f. Telemetry sensors
g. Safe/arm position

2.2.1 Torque Motor Valve Tests. Satisfactory performance of any individual electrically energized torque motor valve in the TVC subsystem can be demonstrated by evaluating the valve parameters outlined below. It is also necessary to determine the margin of safety that exists between the critical parameter and the nominal system value of the parameter. The console will measure the following parameters on an individual component basis and determine the safety margin that exists.

a. Operating voltages and hypot voltage
b. Energizing current and insulation resistance
c. Valve response time
d. Valve seat leakage
e. Valve flow characteristics and full open determination

These parameter tests are performed by flowing hydraulic fluid in a closed cycle through the valve.

The OSE will:

a. Supply a variable control voltage to the electrically energized valve and accurately record the valve position at zero hydraulic pressure. To evaluate the electrical condition of the actuator, a Hi-Pot circuit is provided to impress a high potential between the actuator and main valve body.

b. Measure valve current and power drain when the valve is actuated at zero line pressure and rated hydraulic line pressure. Capability is also provided to measure the resistance from the actuator circuit to the valve body.

c. Measure valve response time and electrical signature by recording control current and voltage signals on an oscillograph. This response time must be determined for both opening and closing of the valve at rated hydraulic pressure. The full open condition of the valve can be verified by this electrical signature.

d. The electrical harness, disconnected from the propulsion system, is connected to a sequence recorder harness, and the sequence recorded on the sequence recorder.

2.2.2 Pressurant Gas Regulators. The OI pressurant gas regulators are tested in a manner almost identical to that described in VC268FD101, Midcourse Propulsion Subsystem OSE. The only significant differences are in pressure (3000 psi vs 3600 psi) gas (N_2 vs He), and slightly different flow rates.

2.2.3 Hydraulic Pump Motor Tests. The OSE unit will have the capability of energizing the electrically driven hydraulic pump in the TVC system. Current, voltage, and power measurements will be made under steady state operation. A simulated command will be given for pintle operation and the transient response of the hydraulic unit and the pintle potentiometers monitored on a high speed oscillograph. The operation of the pintle, through its full excursion, will constitute the full load on the motor. Measurements of motor parameters will require only conventional instruments for observation and recording. The order of magnitude is 870 watts for 120 seconds.

2.2.4 Positive Expulsion Tests. Positive expulsion units are tested in a manner analogous to the test of expulsion units of the MC Propulsion subsystem, as described in MC Propulsion Subsystem OSE, VC268FD101.

2.2.5 Pintle Valve and TVC Response Test. To test the operation of pintle valves and the response of the TVC system, the OSE is required to generate a simulated G&C command signal, which is seen by the TVC as a DC signal proportional to the commanded thrust vector. The torque motor valves, receive this signal and actuate the pintle valves to the proper position. The pintle valve position readout, as identified earlier, is displayed and/or recorded for comparison.

2.2.6 Telemetry Sensor Calibration and Test. This calibration and test will be performed in a manner analogous to that described for the MC Propulsion Subsystem OSE, VC268FD101, except that temperature sensors will not be heated or cooled beyond ambient.

2.2.7 Electrical Test of Safe/Arm and Igniter Assembly. This test is accomplished by removing the assembly from the motor and placing it in a blast chamber. Testing may then be performed using a portable Safe and Arm Device Test Set. Measurements to be made include arming time, disarm time, motor coil resistance, squib resistance, circuit continuity and leakage current.

3.0 INTERFACES. The OI Propulsion OSE interfaces are described in the following tables.

3.1 Propulsion OSE console interfaces. These are presented in Table 3-1.

TABLE 3-1. PROPULSION OSE CONSOLE INTERFACES

Interfacing Equipment	Purpose	Remarks
Facilities Electrical Power	Control power to OSE	120 VAC, 60 hz
Test Conductor's Console	Limited emergency control over OI OSE	Electrical cable
	Test conductors communication network	
	Propulsion OSE status signals	
TLM OSE Interface	DC voltage signals from all OI propulsion subsystem sensors	12 Sensors Electrical cable
Computer Data System (CDS)	Control test functions by CDS	Electrical cable
	Request to perform specific tests	
Remote Control Pneumatic Unit	Instrumentation signals	Electrical cable
	Control signals	
Direct Access Harness	Direct electrical signals from OI propulsion subsystem to OSE unit	Electrical cable
Central Recorder System	Provide additional recording capacity for OI propulsion subsystem test	Electrical cable
Central Timing	Common timing for all tests	Electrical cable

3.2 Pressurant Control Unit Interfaces. Table 3-2 details interfaces for the Pressurant Control (PC) Unit.

TABLE 3-2. PRESSURANT CONTROL UNIT INTERFACES

Interfacing Equipment	Purpose	Remarks
Pressurant Supply Control	N_2 supply for pressurant control unit	Flexible hose
Control Harness	Electrical control to all solenoids in PC unit	Electrical cable
Instrumentation Harness	Excitation and signal readout of transducers in pressurant control unit	Electrical cable
Leak Detection System	Return line to central leak detection unit	Flexible hose
Output Pressure Line	Control pressure line to test connection parts on OI propulsion subsystem	Flexible hose
Return Pressure Line	Return line from OI propulsion module	Flexible hose

4.0 PHYSICAL CHARACTERISTICS AND CONSTRAINTS

4.1 Physical Description. The OI Propulsion OSE Console for the Systems Test Complex will consist of one control and display console and a pneumatics console. Electrical power supplies, electrical measurements and calibration equipment, selector switches, recording equipment and self-testing equipment will be in the control and display console. During subsystem testing when tests may be performed in the immediate vicinity of the subsystem the remote pneumatic control unit can be located adjacent to the console.

The Propulsion OSE unit will supply power to the subsystem. The OSE power supply has a bank of overload protective circuits to provide power protection for the specific component or subassembly being tested. Any intermittent or continuous shorting will automatically disconnect the power to the propulsion module and energize an alarm circuit in the OSE.

4.2 Test and OSE Constraints. Pneumatic safety provisions of two types are required within the Propulsion OSE: (1) equipment is provided which prevents contamination or overpressuring of the propulsion subsystem in all testing phases; and (2) contamination control is maintained in the helium and nitrogen gas pressurization supply by a dual flow filtration system in the OSE at all propulsion interfaces.

The procedure and requirements discussed in this section assume that sterilization of the OI Propulsion Module is not a requirement. If sterilization becomes a requirement the test procedures will be changed to accommodate them.

The gas supply and leak test equipment required to implement these tests are physically identical to parts of the LCE fluid or propellant service equipment required at the ESF.

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LCE DESIGN CHARACTERISTICS AND RESTRAINTS

SECTION

LCE DESIGN CHARACTERISTICS AND RESTRAINTS

1.0 INTRODUCTION.

1.1 Launch Complex Equipment. The Launch Complex Equipment (LCE) encompasses the spacecraft support equipment, over and above the System Test Complex (STC) and Assembly, Handling, and Shipping Equipment (AHSE), required to support a Flight Spacecraft during prelaunch and launch activities. It includes the equipment required at Launch Complex 39 for prelaunch testing and spacecraft control during countdown, and program-peculiar equipment required to maintain the communications links, both voice and signal between the STC, Explosive Safe Area (ESA) and Launch Complex. The LCE also includes the equipment provided at the ESA for propellant loading, pressurant loading, preparation for explosives loading, and system confidence tests.

1.2 Launch Complex 39. Launch Complex 39 is unique in its magnitude and its configuration. The huge size of the facilities limits the usefulness of functional comparisons with other pads. The umbilical tower is as large as most mobile service towers. The mobile service tower is larger than a normal mobile service tower and the Vertical Assembly Building (VAB) with its high level equipment rooms, cannot really be compared with any other facility. The difference between Launch Complex 39 and other pads requires that a unique design approach be taken to the VOYAGER LCE.

The Launch Vehicle flow at Launch Complex 39 is considerably different from the flow at most Launch Complexes. The Saturn V is fully assembled and tested in the vertical position inside the VAB. The Apollo payload is mated to the Saturn V while it is still in the VAB. The entire Launch Vehicle is then moved three miles to the pad in the vertical position, fueled, and launched. An umbilical tower is transported with the booster along with the required support equipment. The service tower can be transported to the pad for service work on the Launch Vehicle or Flight Spacecraft in the pad vicinity. The Flight Spacecraft will be mated to the Launch Vehicle at the pad.

1.3 Pre-Launch Operations. The prelaunch flow of the Flight Spacecraft is shown in Figure 1-1. The Flight Spacecraft will leave the hangar fully tested. It will be transported to the Explosive Safe Facility (ESF) where a "dry run" of ESF activities will take place. Additional pressure and leak tests will be performed, dummy pyrotechnic devices will be installed, and general Spacecraft-ESF compatability verified for later operations. When the Spacecraft leaves the ESF it will be taken to the launch pad and assembled on the Launch Vehicle. A combined systems test is performed proving compatibility with LCE, Launch Vehicle and pad facility. After this test, the Spacecraft is taken back to the Satellite Checkout Facility (SCF) for a final system test and calibration.

The Spacecraft then leaves the hangar for its final trip through the ESF. Pyros will be installed, propellants will be loaded, and the Spacecraft and Capsule prepared for launch. When the Spacecraft again leaves the ESF, it is fully assembled and surrounded by the shroud. Components and subsystems are inaccessible. From the ESF, the Spacecraft is taken to the pad where it will be again installed on the Launch Vehicle. System tests will be performed and the countdown can start.

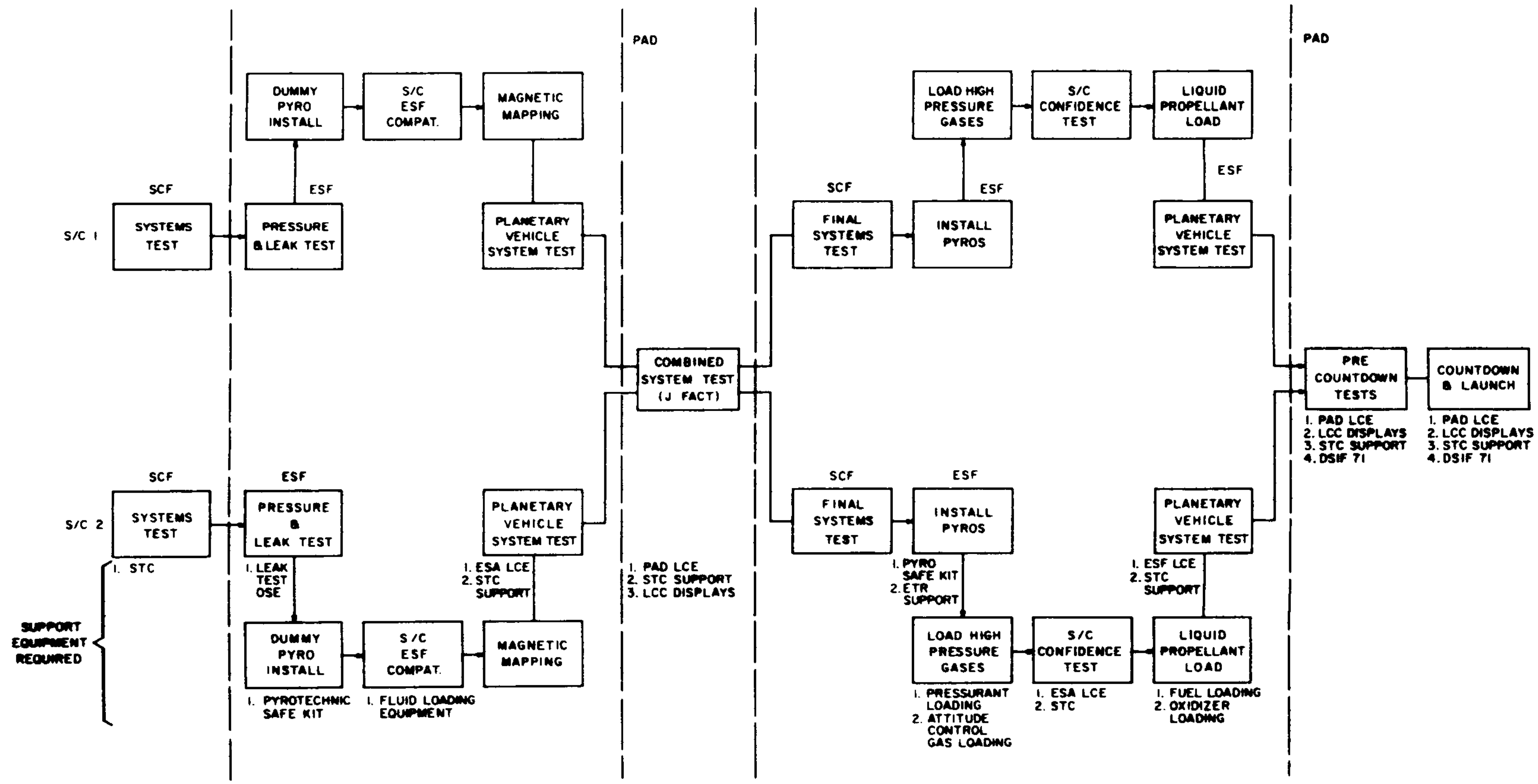

Figure 1-1. Abbreviated Spacecraft Flow ESF-PAD

1.4 Support Equipment Functions. The major assumptions or key points in the flow which relate to equipment location and definition are: (a) Leak Tests will be performed in the ESF, (b) Gases will be loaded in the ESF, (c) Liquid Propellants will be loaded in the ESF, (d) Planetary Vehicle will be checked out in the ESF, and (e) Planetary Vehicles will be placed on the Launch Vehicle at the pad.

1.4.1 Leak Testing. The small leakage requirements on the Spacecraft System will result in extensive leak tests. Most of these tests will be performed in the STC but there will be a need to make leak tests in the ESA, requiring a set of leak test equipment in the ESA. After the ESA, no further leak testing will be performed and no additional leak test equipment will be provided.

1.4.2 Fluid Loading. Loading equipment will be required for liquids and gases for all systems using fluid propellants. The Spacecraft will be turned off during loading but it will be necessary to monitor tank parameters during the loading process. Some equipment must be provided to interface with the umbilical connector and to display tank temperatures.

1.4.3 Pyro Loading. A large number of explosive devices will be used to initiate mechanical operations on the Spacecraft. These devices must be installed in a place where it is safe to do so, where all parts of the Spacecraft are accessible, and where it is possible to check for stray voltages prior to installation. It is assumed that Kennedy Space Center (KSC) will provide means to handle and check the explosive devices before installation. LCE is required to condition the Spacecraft to a safe mode before the explosives are installed and to check the system after installation. There must also be equipment to support a system confidence test after work at the ESA is completed and the spacecraft is ready to be moved to the Launch Complex.

1.4.4 Pad Testing. Pad testing includes all tests performed at the pad. Aside from the countdown, the principal reason for pad tests is to assure that the Spacecraft is operating when placed on the Launch Vehicle and to perform compatibility tests with the Launch Vehicle. The extent of the pad testing capability is limited only by the information available from the Spacecraft. This information will be limited to the umbilical connector and the telemetry link. The principal objective is to provide a reasonable capability at the pad so that a margin of flexibility exists when the test crew defines the test and countdown.

For the Saturn V, the Launch Control Center takes the place of the blockhouse. It is three miles away from the pad. However, it is not a hardened blockhouse, has little unassigned space, and is too distant from the pad to permit an adequate hardwire interface for serving as the Voyager LCE center. Most of the test information travels over data links to and from the pad. Since the information from LCE interfacing with the Spacecraft must be encoded for data link transmission away from the Mobile Launcher, it is equally convenient to send it to the hangar which is only six miles further than the LCE Center. The result is that all information which would normally go to a blockhouse, now goes back to the hangar.

The interpretation taken of Launch Complex 39 constraints is that all personnel are required to be removed from the launch area during the last phases of prelaunch operations.

Therefore, the blockhouse LCE should be remoted (controls and displays) to some acceptable location. The SCF has been recommended, so that STC (Subsystem OSE) controls and displays can be used.

The STC will be used to the maximum extent possible in supporting and controlling pad operations and testing. All data, both RF and umbilical hardwire, will be evaluated by the STC and control of the umbilical lines will be maintained by the STC. The pad LCE will function primarily as a remote control buffer between the STC and the Planetary Vehicle. Sufficient local control will be available for checkout and emergency, but the intent is to control the Spacecraft from the STC. The spacecraft test crew will operate the Spacecraft with the same STC used to check it out, the pad LCE acting as a buffer.

1.4.5 Pad Validation. Before mating a Flight Spacecraft to a Launch Vehicle, it is necessary to completely check the LCE-STC equipment from the umbilical connections back to the STC. Only when the pad LCE is completely operational can a Spacecraft be mated to the Launch Vehicle and the umbilical boom connected. A Spacecraft umbilical simulator is required for this validation.

2.0 DESIGN CHARACTERISTICS.

2.1 General Features. The LCE consists of a set of homogenous equipment for use at the launch pad and a set of functionally identical 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 exercising 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. The STC will support all confidence tests and control of the test will be exercised from the STC. The LCE circuits shall be electrically identical to the circuits used in the STC. Wherever possible, the LCE assemblies should duplicate the assemblies used in the STC.

2.2 Reliability and Maintainability. The LCE and STC are intended to support the launch operation of the VOYAGER. Up to the time of close approach of the launch window, when the prelaunch operation begins, OSE reliability is not critical to launch. That is, assuming that failures are not too frequent, and that restoration in a reasonable time is probable, the mean time between failures (MTBF) and mean time to repair (MTTR) numbers will not influence the probability of a successful launch.

During the prelaunch and launch operations, OSE reliability and maintainability have a more direct effect. The requirements for MTBF and MTTR imposed upon OSE are still hard to assess since launches might well be undertaken with degraded OSE capability. The MTBF and MTTR figures assigned to OSE as design restraints, should come from an apportionment of OSE which considers the probable acceptable degradation modes and the relative criticality of individual OSE groups.

The assignment of MTBF and MTTR figures to OSE is a part of the design. It should be done after the OSE functional design has produced relatively firm Part I specifications, and after those launch criteria which consider the probability of degraded OSE, have been established.

2.3 Launch Area Equipment Characteristics. The launch area equipment must be capable of performing or supporting the performance of specific tasks at the launch pad. These tasks are pad verification, assembly to booster, confidence testing, initializing the Spacecraft, data loading, power control, and 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. Some LCE will be provided for monitoring of the prelaunch status at the Launch Control Center by the test coordinator. This will also include some emergency shut down control capability, for use in the case of normal control loss.

2.4 ESF Equipment Characteristics. The ESF equipment must be capable of performing particular tasks, some of which are unique to the ESA. These tasks are leak testing, liquid propellant loading, high pressure gas loading, pyrotechnic loading, shroud installation, and confidence testing.

The equipment provided for loading fluids 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 spacecraft 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 and provide for propellant sterilization if required.

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 ESF 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-portable, and contain any necessary operating cables.

The LCE provided at the ESF must have the capability of performing confidence tests at various stages of ESF 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.

3.0 DESIGN RESTRAINTS.

a. The LCE, with the support of the STC, will provide for on-pad testing of the Spacecraft. The LCE will interface with the spacecraft umbilical and RF test circuitry to provide STC operation of the Spacecraft.

b. Provide for manual control of the Spacecraft by two modes, one local from the LCE in the Mobile Launcher, and second from the STC through the Subsystem OSE controls.

c. Automatic control of the Spacecraft at selected times during test or countdown by permitting the STC to control the on-stand sequences as during system test in the SCF.

d. Isolate malfunctions or anomalies to either the Spacecraft or OSE.

e. An external dc power supply will be furnished to each vehicle. The power supply will be independent for each Spacecraft with a spare supply which can be substituted by remote control for either power supply.

f. A central facility power control cabinet will distribute facility power to all VOYAGER LCE cabinets.

g. All launch critical parameters will be transmitted to the STC. The STC will provide a permanent digital record of these parameters on magnetic tape.

h. The LCE items, where appropriate, will check their own operation periodically, without disturbing the Spacecraft. This includes the data link as well as operational LCE. In general, the self-checking method will be on line at all times and check performance continuously rather than enter a special self-check mode.

i. Controlled transfer to emergency main power sources will be implemented through controls in the Mobile Launcher when performing pad tests or remotely when the Mobile Launcher is evacuated.

j. A back-up power supply will be available in the Mobile Launcher and in the Launch Control Center for control of the Spacecraft when no facility power is available.

k. Performing tests through hardware data links during periods of RF silence. This will be provided primarily by returning the telemetry composite from both vehicles through two data links to the STC. Stored commands can also be injected into the Control and Sequencer from the STC over the data link.

l. Any spacecraft function relating to Spacecraft or personal safety, will be monitored continuously.

m. Remote capability will be provided to permit relief of dangerous pressures without hazard to personnel.

n. Complete data link capability will be provided to transmit signals to and from the Launch Complex and the STC.

o. RF antennas will be provided to radiate RF signals from the vicinity of the Spacecraft to Deep Space Instrumentation Facility (DSIF-71) and the SCF hangar.

p. As part of the data link, there will be an emergency communication system between the Launch Control Center, the launch pad, and the Planetary Operations Control Center.

q. During automatic testing, the programming safeguards applied in the STC will also protect the Spacecraft from accidental damage at the pad.

r. All LCE subassemblies will be interchangeable with other subassemblies of the same type. Each LCE will be compatible with all Spacecraft.

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LCE SYSTEM LEVEL FUNCTIONAL DESCRIPTION

SECTION

LCE SYSTEM LEVEL FUNCTIONAL DESCRIPTION

1.0 DESCRIPTION. To effectively support prelaunch and launch operations under the indicated restraints, the Launch Complex Equipment must be closely integrated with the facility and the System Test Complex (STC). Most of the capability to perform tests and evaluate tests lies with the STC. The STC has extensive subsystem test equipment, recording capability, computing equipment and, most important, the cognizant personnel familiar with the flight equipment. Placement of an entire STC in the launch complex or at the Explosives Safe Facility (ESF) is precluded by economic and facility constraints. Rather than limit the pad and ESF checkout capability to equipment which can be installed locally, the preferred approach is to provide LCE which acts as a remote extension to the STC, providing personnel using the STC control of the spacecraft at the pad and at the ESF without moving the STC equipment.

A functional block diagram illustrating the concept appears in Figure 1-1. The figure shows two planetary systems on the Launch Vehicle with the major groups of support equipment used. Both Spacecraft communicate directly with the STC and Deep Space Instrumentation Facility (DSIF) 71 by RF transmissions. Both Spacecraft are supported with independent external power supplies. Support equipment which must be close to the Spacecraft is located at the pad, while additional support is provided remotely by the STC. In addition to the RF capability, there is a hardwire link from the Spacecraft to the STC.

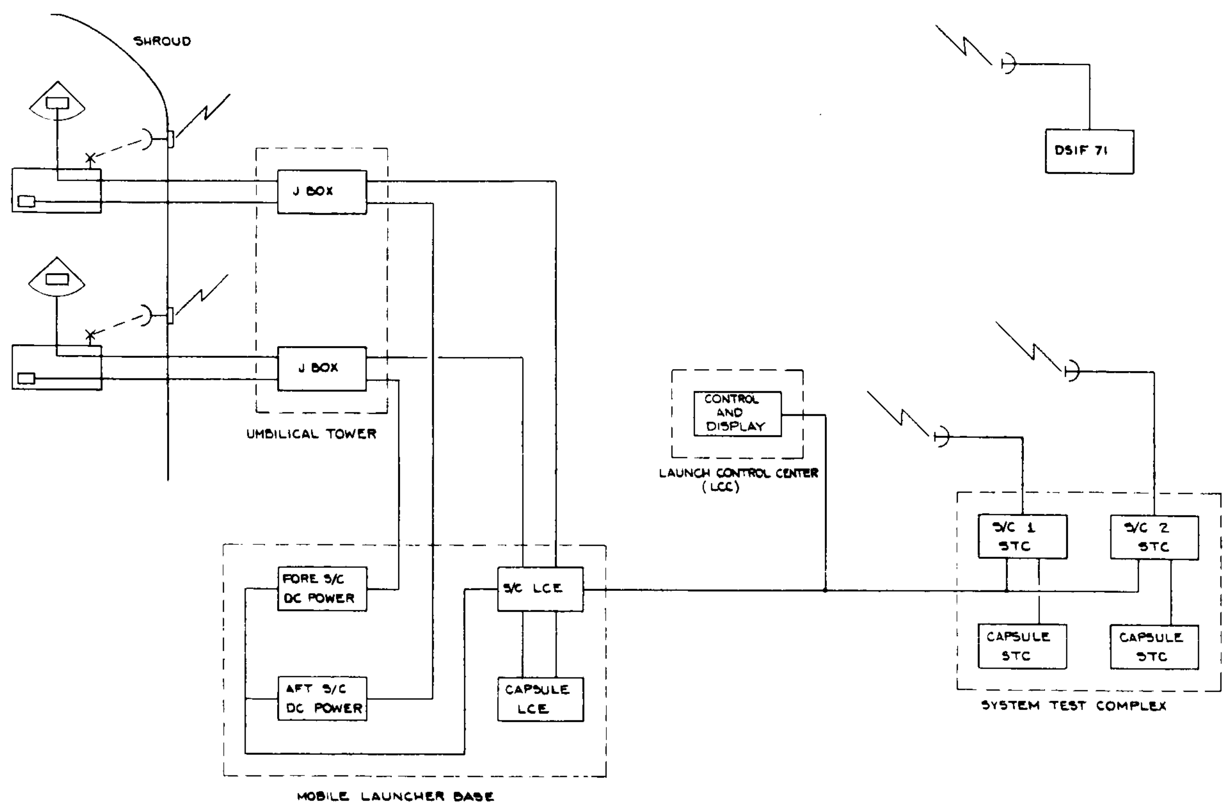


Figure 1-1. Launch Complex LCE Functions

In addition to the equipment provided to extend STC control, additional items of equipment are required to support operations unique to the ESF. Although these items do not operate along with the rest of the STC-LCE system, they are included in the LCE because they are a part of prelaunch operations.

2.0 LAUNCH COMPLEX 39 EQUIPMENT.

2.1 Equipment Location. There are three general equipment locations at Launch Complex 39 pertinent to VOYAGER launch activities. The first is the umbilical tower which is the closest location to the Spacecraft at which equipment can be located. The umbilical tower is unsheltered, and only a limited quantity of equipment can be placed here. The second location is a room within the mobile launcher. This is the closest point to the Spacecraft in which equipment designed for a partially sheltered environment can be placed. There is space available for a limited quantity of equipment in a room reserved for payload equipment. The umbilical tower and mobile launcher base are linked by cables approximately 300 feet long. The third location is the Saturn V Launch Control Center (LCC) where the countdown is conducted. Most communications between the mobile launcher and the LCC are through data links, but there are a limited number of hardwire control lines available. These hardwire lines could be used to control and monitor the Spacecraft from the launch control center during emergency situations. Overall control and monitor information can be transmitted from Launch Complex 39 to the SCF over existing lines. Figure 2-1 shows the

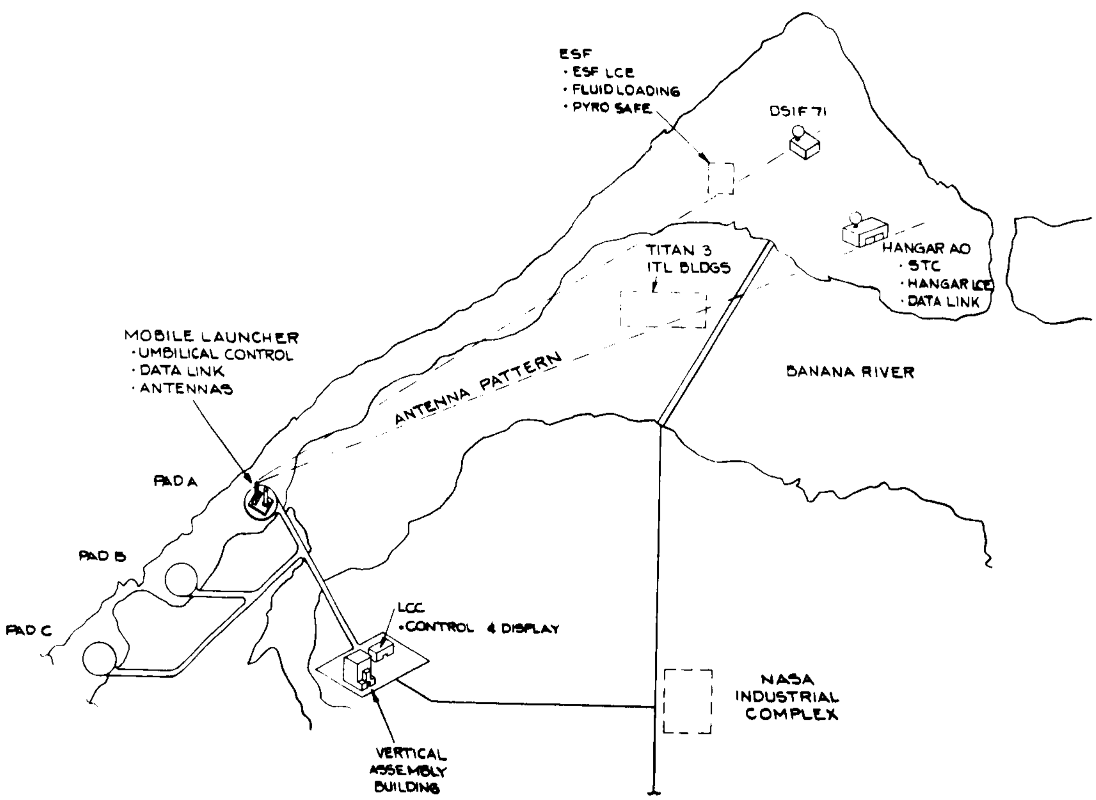


Figure 2-1. VOYAGER Facility Locations

relative location of the buildings used during the prelaunch and launch cycle. The Launch Control Center, adjacent to the Vertical Assembly Building, is 3 miles from the nearest pad and 8 miles from Hangar AO. The hangar is only three times as far from the pad as the LCC.

2.2 Functional Description

2.2.1 Operating Provisions. The launch system should be operated by the same personnel who operate the STC during system tests. They should be supplemented during pad validation tests and J-FACT testing by operators stationed in the Mobile Launcher (ML) and by a test coordinator in the LCC. The entire crew can be integrated through one or more audio communication links under the direction of the test conductor at the SCF.

Operation of a Spacecraft from the STC while it is at the pad can be very similar to operating a Spacecraft located in the hangar. The major difference is the absence of direct access test points, and the objective is the conditioning of the Spacecraft to launch, rather than system testing. The STC would be operated as if the Spacecraft were located in the hangar, the subsystem engineers using the same operating stations as during a system test. In addition, it is necessary to provide operators for LCE control equipment and data link equipment at the hangar. At the pad, in the Mobile Launcher, two or more people are required during pad validation or J-FACT testing. They will monitor equipment performance locally and provide any emergency operation required. During countdown the Mobile Launcher must be evacuated, and LCE operation will be under remote control. At all times during pad activities there must be a test coordinator in the LCC. He will have available sufficient displays of spacecraft and STC status to permit him to coordinate Spacecraft activities with Launch Vehicle System activities. There will also be a need, during countdown, when the ML is evacuated to operate emergency controls and displays controlling the ML equipment and the Spacecraft. Since these controls are backups for controls in the ML, this console need only be operated when the ML is evacuated.

2.2.2 LCE System Operation. On the pad, as shown in Figure 2-2, LCE Functional Block Diagram, the only hardwire interface between the planetary systems and the LCE is through the umbilical connectors. The spacecraft umbilical will be connected to an umbilical in the shroud. The shroud umbilicals will connect through 25 feet of cable to an Umbilical J-Box, located on the umbilical tower. This J-Box, in addition to providing access to umbilical wires in the vicinity of the Spacecraft, will contain any buffer amplifiers or line drivers required to transmit signals to the LCE located in the Mobile Launcher base or to provide isolation for signals entering the Spacecraft. The spacecraft umbilical also carries RF signals to power dividers and antennas which link the Spacecraft with the STC and DSIF 71.

The Umbilical J-Boxes are linked to the LCE control units in the Mobile Launcher base through over 300 feet of cable. The power supplies, command loading equipment, Spacecraft control equipment, and the data link required to link this equipment to the STC are located in the base. The signals coming from the Spacecraft umbilical to the LCE are of four types: telemetry composite, analogs, discretes, and digital words. The telemetry composite is sent directly to the STC through the data link with no intervening processing. Analog and discrete signals from the umbilical are encoded by the data link equipment and transmitted to the STC for evaluation. Appropriate analogs and discretes can be displayed on LCE in the Mobile Launcher to facilitate checkout and troubleshooting. Digital signals coming from the Spacecraft must be processed by an appropriate piece of LCE and transmitted through the data link to the STC.

Signals going into the Spacecraft through the umbilical are handled in an analogous manner. The information is received from the STC over the data link in the form of 24-bit words. A digital demultiplexer separates the information and sends digital words to the Controller and Sequencer (C&S) LCE, the Power LCE, or the LCE Control Unit. The C&S LCE stores the 24-bit words received and clocks commands into the C&S storage at the 250-bps rate required. The Power LCE receives digital voltage adjust and switching commands.

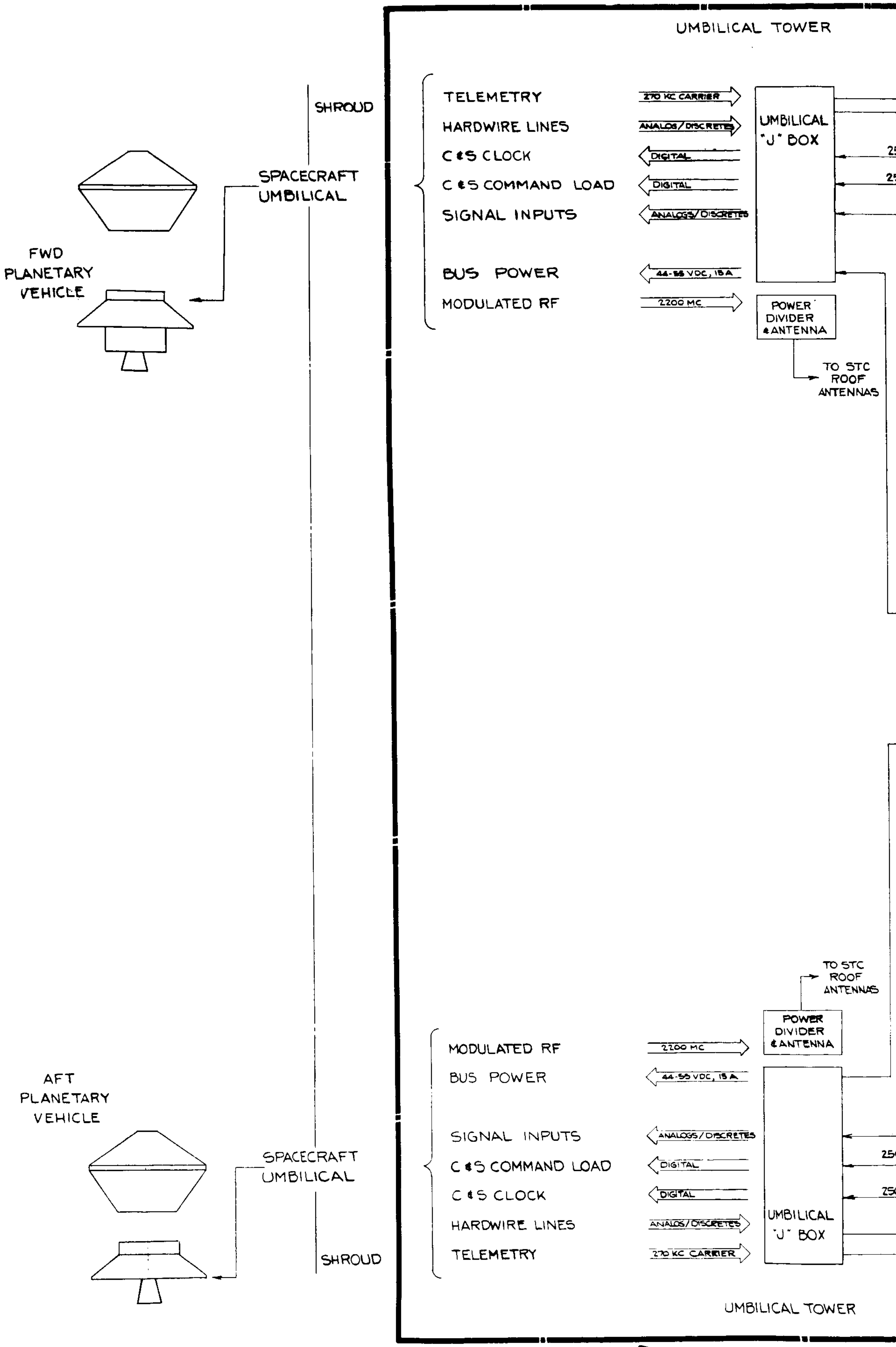
The LCE control unit converts the digital commands to the required discretes, which are relay closures in most cases, for control of Spacecraft, and LCE units. The execution of all signals in the ML is monitored and an indication returned to the STC to confirm the operation.

The Data Link consists of the encoding and decoding equipment, as well as the transmitters and receivers. Cable and terminal equipment is available from Kennedy Space Center for linking buildings together. The existing System permits 4.5-mc bandwidths over controlled video pairs. A bit rate between 50 Kbps and 150 Kbps is suggested because it affords ample capacity and is well within the capabilities of the existing links. The data link itself transmits 24-bit words; each word is transmitted twice and checked before execution. At the STC, or hangar end, the terminal equipment (receivers and transmitters) is the same as the equipment used at the Mobile Launcher. PCM data coming from the ML is decommutated by the data link equipment and sent to subsystem OSE consoles for display and to the CDS for processing.

The STC provides the capability for recording all signals returned from the LCE, both Spacecraft and LCE control signals. In addition to recording all incoming signals, the STC (CDS) provides for encoding into a transmittable format all signals going to the ML. The subsystem OSE consoles will provide the switches for manual control of the Spacecraft. The subsystem engineer will operate the same controls as during Spacecraft testing. Discrete signals are sent to the CDS, where they are converted to the transmittable format. The test conductor can be in complete control of the prelaunch operation at all times. Functions unique to the LCE are controlled from the LCE Console in the hangar.

The Launch Control Center (LCC) equipment provides status displays for the test coordinator and emergency hardwire controls for Mobile Launcher LCE. Status displays are operated from the STC, directly from the test conductors console. The status information is transmitted over conventional telephone lines using a tone code to provide a limited number of discrete status indications. The test coordinator at the LCC will also have closed circuit TV and operational intercom equipment, along with status and temperature signals from the Launch Vehicle System available to him.

In the event that the data link fails, there are hardwire signal lines available between the launch control center and the Launch Pad. A limited number of these lines will be used to


UMBILICAL TOWER
SHROUD
SPACECRAFT UMBILICAL
FWD PLANETARY VEHICLE
TELEMETRY
HARDWIRE LINES
C & S CLOCK
C & S COMMAND LOAD
SIGNAL INPUTS
BUS POWER
MODULATED RF
270 KC CARRIER
ANALOG/DISCRETES
DIGITAL
DIGITAL
ANALOGS/DISCRETES
44-55 VDC, 15A
2200 MC
UMBILICAL "J" BOX
POWER DIVIDER & ANTENNA
TO STC ROOF ANTENNAS
TO STC ROOF ANTENNAS
POWER DIVIDER & ANTENNA
MODULATED RF
BUS POWER
SIGNAL INPUTS
C & S COMMAND LOAD
C & S CLOCK
HARDWIRE LINES
TELEMETRY
2200 MC
44-55 VDC, 15A
ANALOGS/DISCRETES
DIGITAL
DIGITAL
ANALOG/DISCRETES
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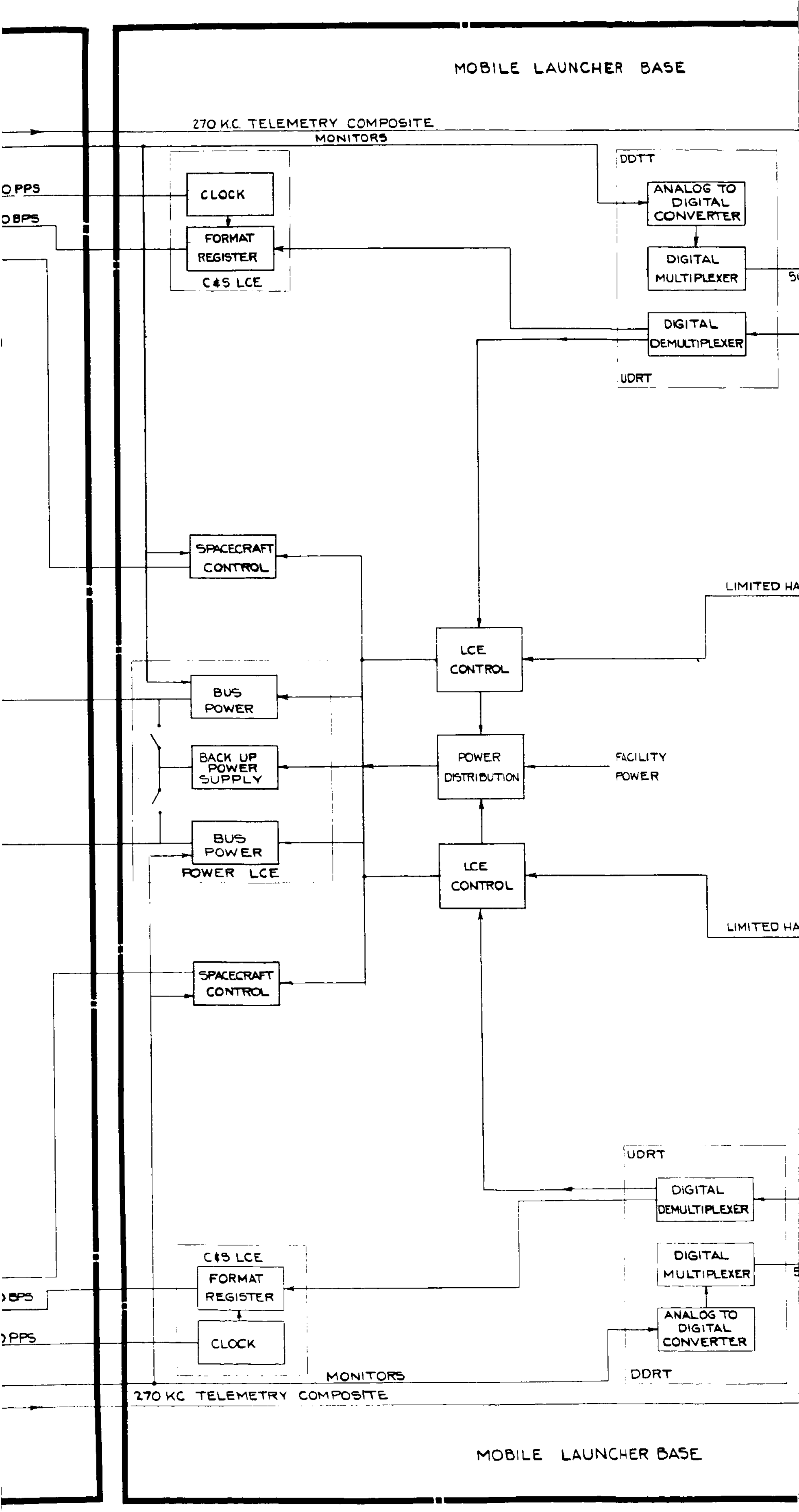
UMBILICAL "J" BOX
AFT PLANETARY VEHICLE
SPACECRAFT UMBILICAL
SHROUD
UMBILICAL TOWER



MOBILE LAUNCHER BASE
270 K.C. TELEMETRY COMPOSITE
MONITORS
0 PPS
0 BPS
CLOCK
FORMAT REGISTER
C&S LCE
DDTT
ANALOG TO DIGITAL CONVERTER
DIGITAL MULTIPLEXER
DIGITAL DEMULTIPLEXER
UDRT
SPACECRAFT CONTROL
LIMITED HA
LCE CONTROL
BUS POWER
BACK UP POWER SUPPLY
POWER DISTRIBUTION
FACILITY POWER
BUS POWER
POWER LCE
LCE CONTROL
LIMITED HA
SPACECRAFT CONTROL
UDRT
DIGITAL DEMULTIPLEXER
DIGITAL MULTIPLEXER
C&S LCE
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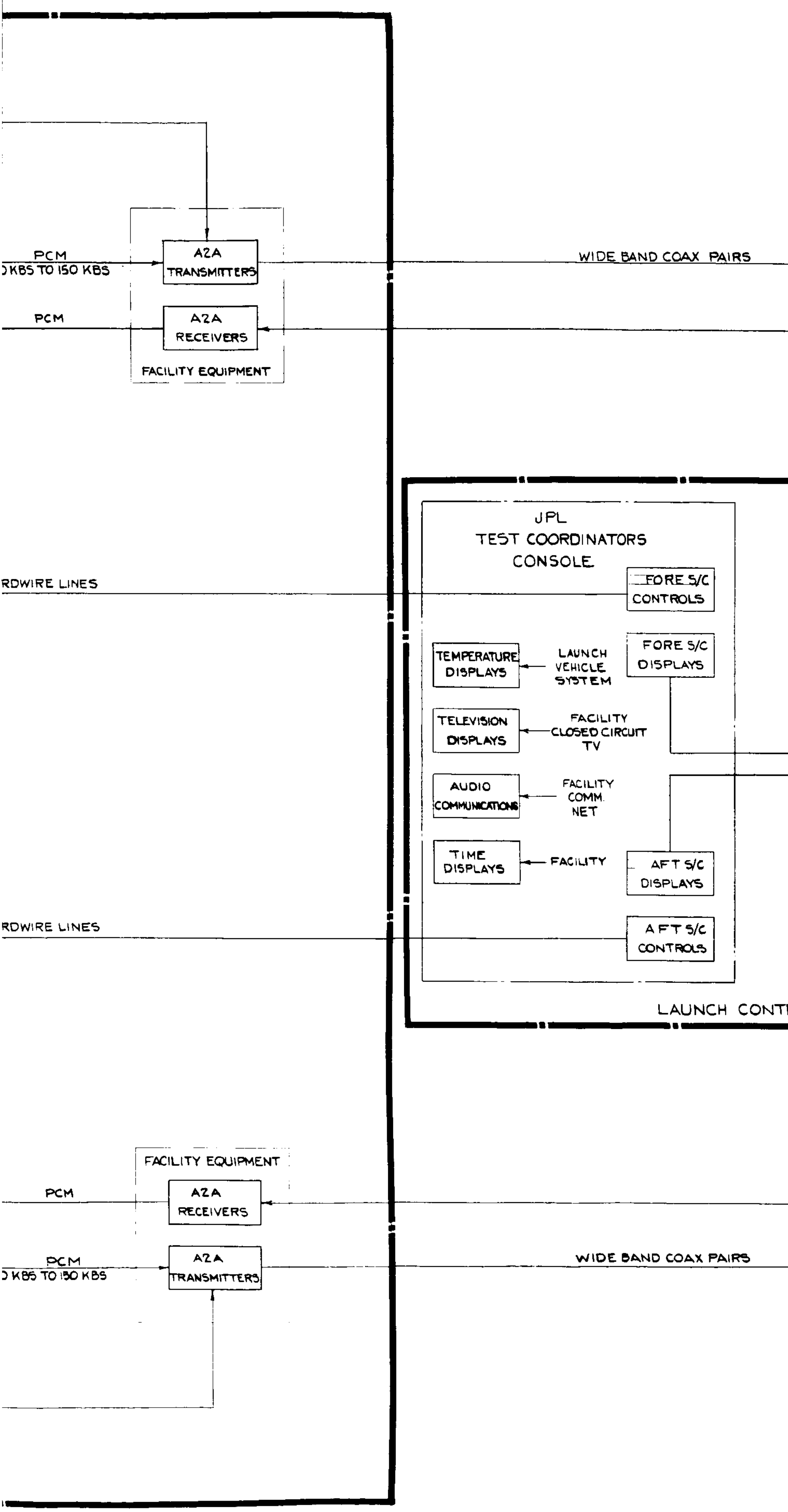
MOBILE LAUNCHER BASE


PCM
) KBS TO 150 KBS
A2A
TRANSMITTERS
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A2A
RECEIVERS
FACILITY EQUIPMENT
WIDE BAND COAX PAIRS
JPL
TEST COORDINATORS
CONSOLE
RDWIRE LINES
FORE S/C
CONTROLS
TEMPERATURE
DISPLAYS
LAUNCH
VEHICLE
SYSTEM
FORE S/C
DISPLAYS
TELEVISION
DISPLAYS
FACILITY
CLOSED CIRCUIT
TV
AUDIO
COMMUNICATIONS
FACILITY
COMM.
NET
TIME
DISPLAYS
FACILITY
AFT S/C
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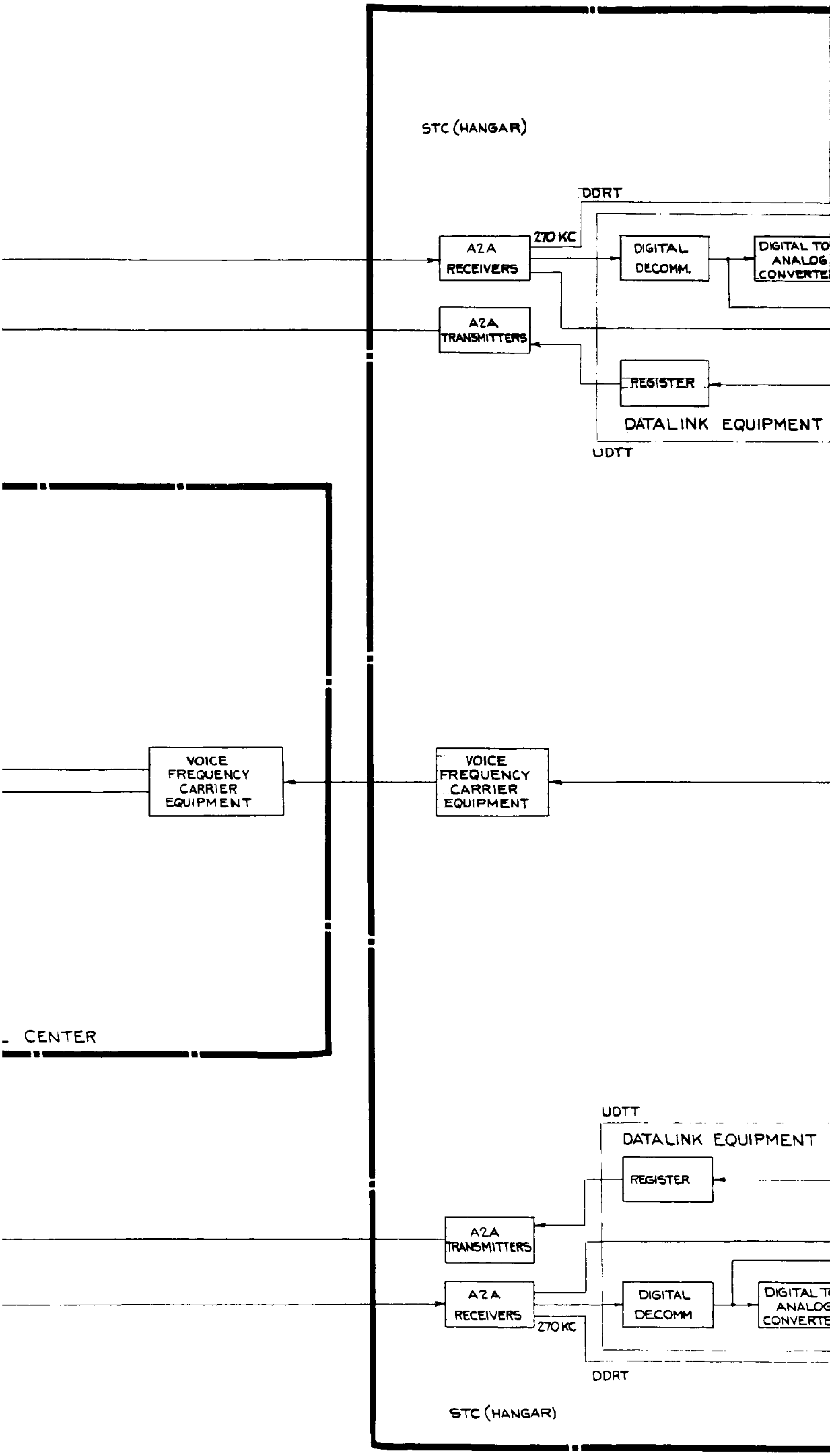
STC (HANGAR)
DDRT
270 KC
A2A RECEIVERS
DIGITAL DECOMM.
A2A TRANSMITTERS
REGISTER
DATALINK EQUIPMENT
UDTT
VOICE FREQUENCY CARRIER EQUIPMENT
VOICE FREQUENCY CARRIER EQUIPMENT
CENTER
UDTT
DATALINK EQUIPMENT
REGISTER
A2A TRANSMITTERS
A2A RECEIVERS
DIGITAL DECOMM
270 KC
DDRT
STC (HANGAR)

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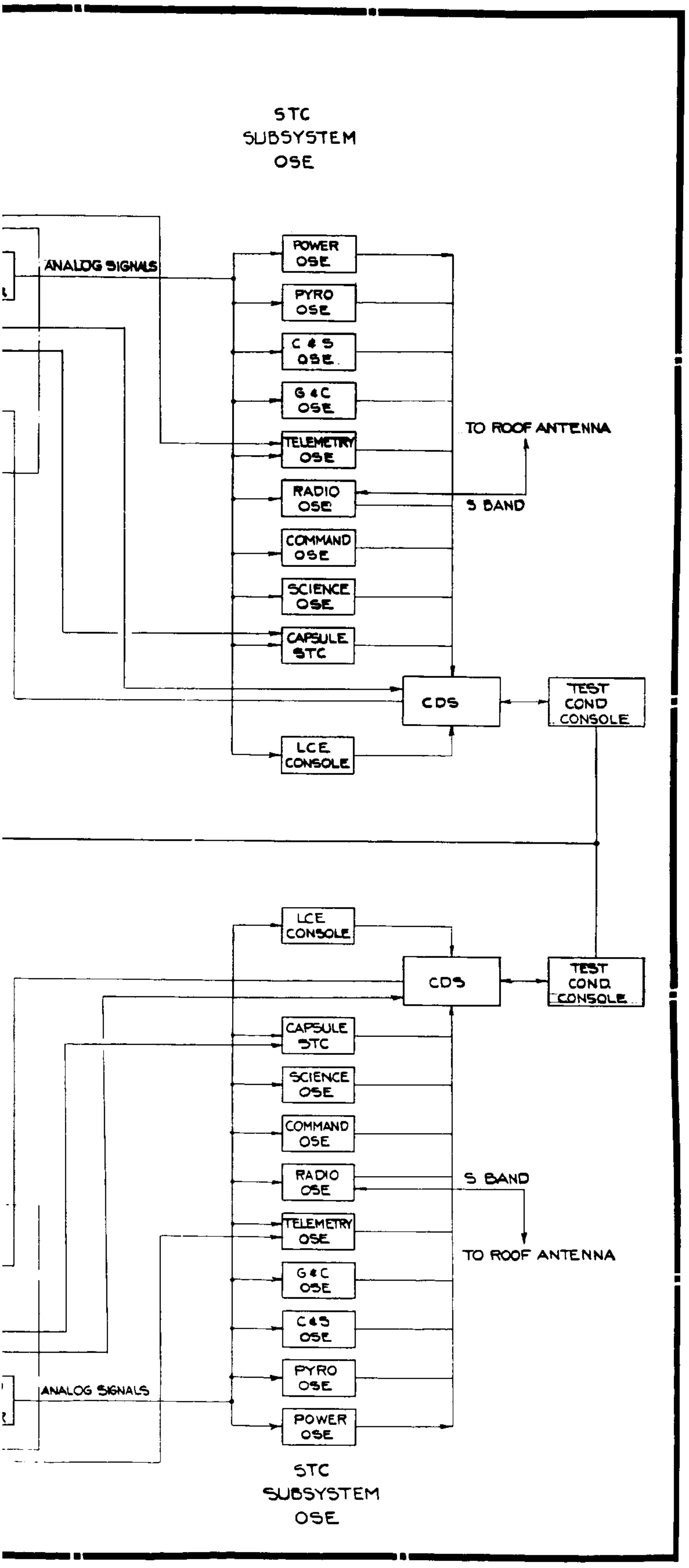


Figure 2-2. LCE Functional Block Diagram Excluding RF Equipment

control the LCE if a countdown emergency occurs. The lines from the ML will terminate in a control panel. Functions to be monitored or controlled include Spacecraft/ground power switchover, arm reset signals, high-pressure gas relief, emergency power switchover, and the monitors needed to implement these commands.

2.2.3 Automatic Operation. The Pad LCE system as presented in the block diagram is arranged so that all control functions are implemented through the CDS and all monitor functions are evaluated in the CDS. This enables the CDS to be programmed to effect complete control of the countdown or any Spacecraft test made on the launch pad. There will be ample computer capacity available because direct access points are not utilized while the Spacecraft is at the pad. If the computer is not operational, the subsystem OSE displays provide backup indications.

2.2.4 Local Control. During pad validation, J-FACT testing, or troubleshooting, it will be necessary to operate the LCE and limited Spacecraft controls from the Mobile Launcher. The LCE will have the necessary monitors and switches required to control the LCE or Spacecraft when the data link is inoperative or operating in a limited manner. All discretes can be implemented locally, all analog functions can be measured, and limited digital signal generation will be possible at the C&S LCE. The intent is to operate the LCE remotely and automatically but to provide enough flexibility that all operating conditions can be handled with no difficulty.

2.2.5 Self-Checking Features and Redundancy. The equipment will be self-checking during performance. The data link will use redundant transmissions which are then compared; each signal executed will be returned to the STC, and the Analog Channels will be periodically checked by introducing known signals.

2.3 RF Links From Pad To STC. To completely verify the operation of the Spacecraft System and maintain complete control, the STC must be able to communicate with the spacecraft by RF transmissions. Figure 2-3 shows the functional operation of a link which provides this capability.

Each Spacecraft can radiate and receive RF through an antenna in its shroud. In addition, the antennas inside the Spacecraft will be linked through probes to the umbilical connector and thence to antennas for radiating to and receiving from the STC. The umbilical tower is on the north side of the pad, while the hangar and Deep Space Instrumentation Facility (DSIF) 71 are to the south, eliminating any interference problem due to an inopportune location of the tower. The shroud antennas should be on the south side of the launch vehicle, providing line-of-sight RF communication with the STC and DSIF 71. The RF link through the shroud is completely adequate for a command information link with the Spacecraft.

In addition to the shroud antennas, each Spacecraft will have RF probes adjacent to its undeployed antennas. These probes will be connected to the umbilical. Outside the umbilical connector, all of the 2000-mc lines from each Spacecraft will connect to a power divider, which in turn is connected to an S-band antenna. Since each receiver is frequency addressable, there is no reason to switch the S-band antenna between coaxial lines. In a similar manner, the relay radio receiver input antenna will be linked to the umbilical and routed to a UHF antenna also located on the umbilical tower. When the mobile service tower is in

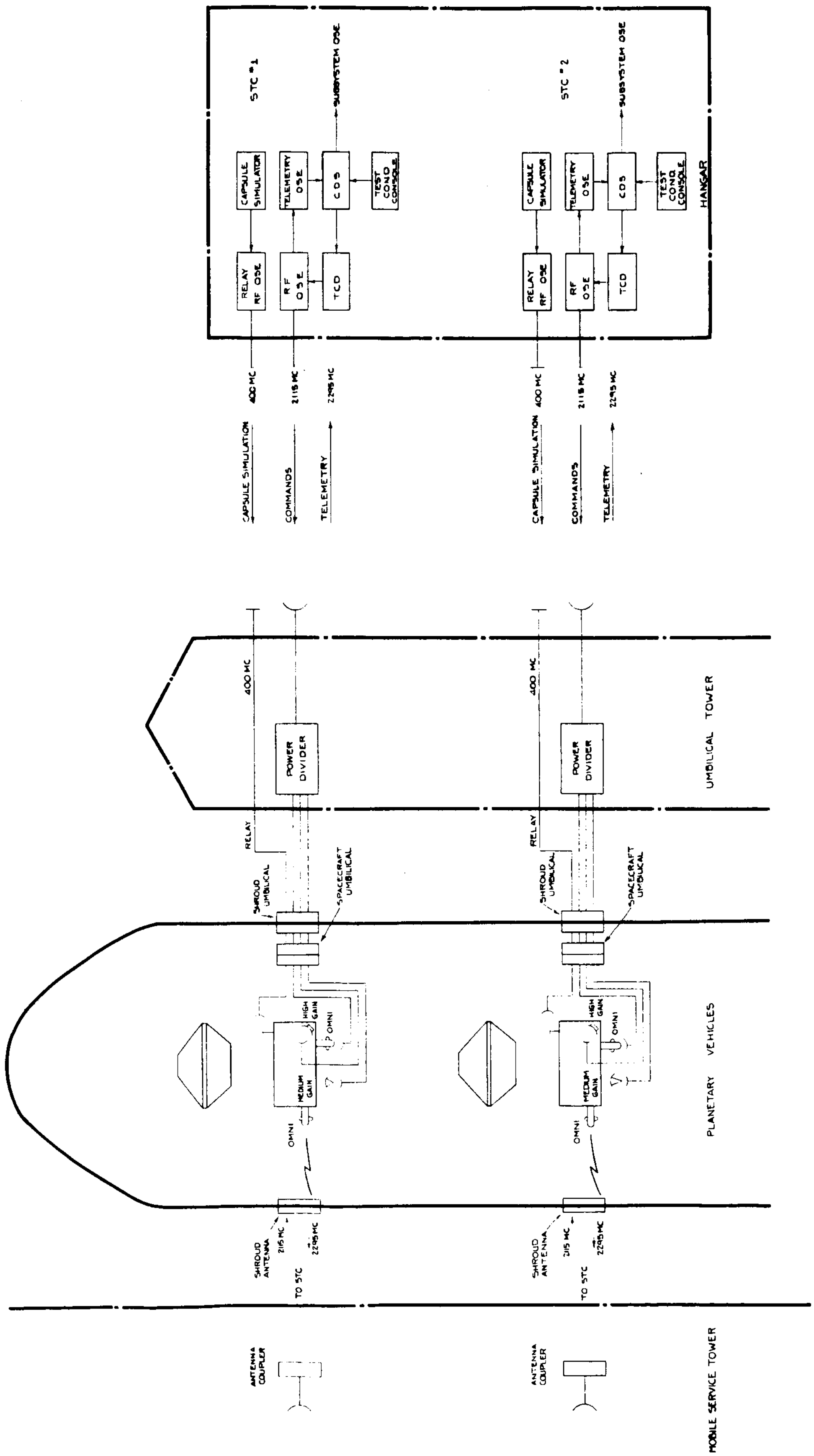


Figure 2-3. Pad RF Link

place, its steel structure will fall between the launch vehicle and the hangar, interfering with RF transmission between the spacecraft and the hangar. Antenna couplers will be placed over the shroud antennas and connected to S-band antennas mounted on the south side of the mobile service tower. If the mobile service tower interferes with transmission from the antennas on the umbilical tower, it will be necessary to provide an additional set of antennas for the mobile launcher. The antennas located at the pad will be pointed at either the STC or the DSIF, depending on the operation in progress. It may be possible to orient the antennas so that both the hangar and DSIF 71 are included in the pattern as shown in Figure 2-1.

At least two S-band antennas and two UHF antennas will be located on the roof of the hangar. One pair of antennas is required for each planetary vehicle. The antennas will connect to the Radio OSE where the signal will be received. Transmissions will be handled in the same manner as during test, except that the Radio OSE will be connected to an antenna on the roof of the hangar and hence to the Spacecraft.

2.4 Data Links.

2.4.1 Use of Data Links. To implement remote monitoring and control of the LCE, a data link is required. The most conservative approach is to use existing lines for data transmission if at all possible. The existing KSC instrumentation communications network was examined. This network will be operationally proven to manned space flight checkout requirements for four years prior to the VOYAGER launch (first Apollo-manned flight in mid-1967). It is assumed that transmission lines, equivalent to those required for an Apollo launch, will be available for VOYAGER utilization. On this basis, the existing planned wide-band transmission capability should adequately handle VOYAGER requirements. A typical A2A (wideband) link consists of a transmitter terminal modem, repeater modems, and a receiver modem all connected in series with 16 PEV-L twin, balanced center conductor video cable.

Figure 2-4 is a line diagram of the transmission network allocated for Apollo. Although the 24 wideband pairs at Hangar AO may be adequate, additional A2A terminations (GFE) would be required.

Table 2-1 lists the major characteristics of the KSC wideband pair (A2A-16PEVL), and audio frequency 19 AWG Telephone pairs. These limitations and characteristics become design constraints for the VOYAGER data link preliminary design. The audio frequency telephone pairs will be used to communicate between STC and LCC.

2.4.2 Wide-Band Data Links - STC/PAD. The overall purpose of the Wide-Band Data Link is to provide an accurate and reliable means of communicating checkout and control data between the LCE at the launch pad and the STC's.

The Wide-Band Data Link consists of a separate Up-Link (STC to pad), and Down-Link (pad to STC) for each of the two independent Planetary Vehicles. Each data link is further subdivided into a Data Transmission Terminal, a Data Reception Terminal, and the portion of the wide-band cable transmission system (Western Electric A2A) that interconnects these terminals.

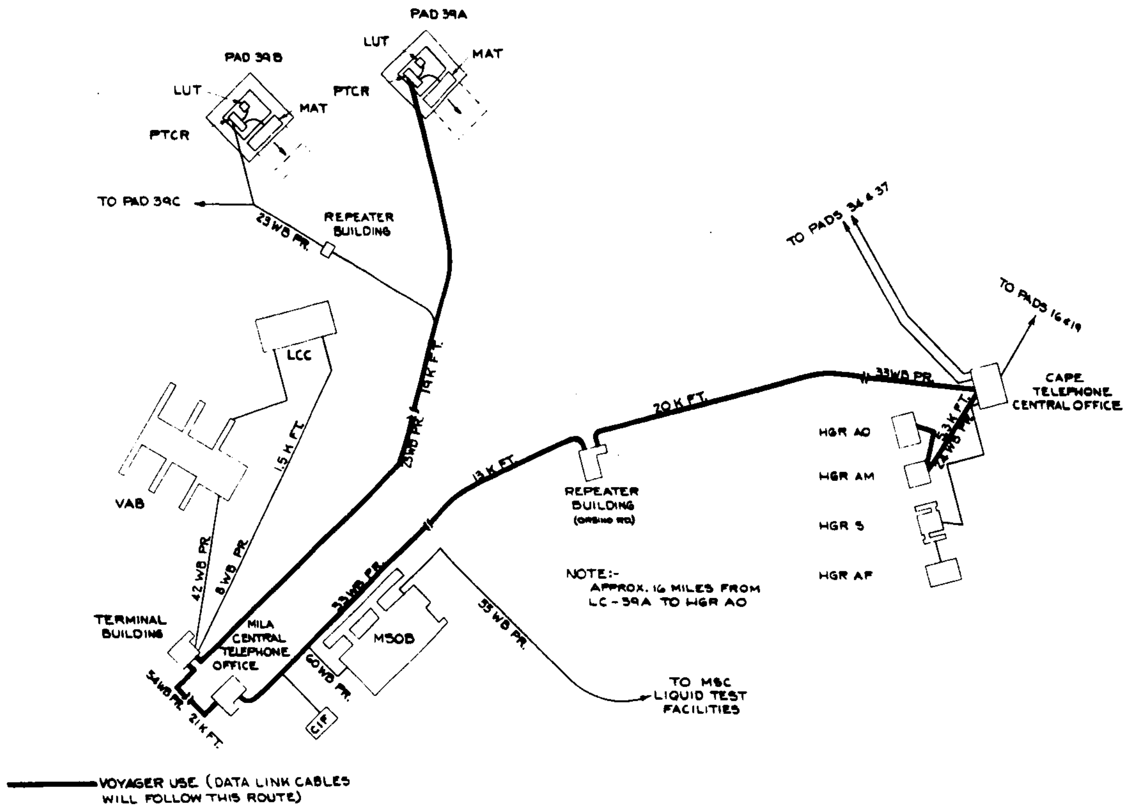


Figure 2-4. Wide-Band Cable Routing at KSC, Simplified Line Diagram

TABLE 2-1. MAJOR CHARACTERISTICS OF KSC WIDE-BAND PAIR AND 19 AWG TELEPHONE PAIRS

Parameter	A2A	Tel. Pairs
Impedance	124 Ω input resistance	600 Ω nominal
Bandwidth	30 cps to 4.5 mc ± 0.4 db	3000 cps nominal to 4300 cps with equalization
Noise Figure	Random noise rejection - 65 db	Noise level - 42 dbm
Cable Type	16 gauge PEV-L co-ax pair	19 gauge wire pairs
Digital Rate Capacity	2.5 mbps with 1 bit error in 10^6	To 1 kbs
Patching Features	At telephone centrals, repeaters, and terminations	At telephone centrals and terminations
Notes	Manufactured by Western Electric Tube type electronics Frequent alignment required Solid-state types being evaluated at KSC and AFETR	Similar to Schedule 4 Commercial telephone lines

The A2A System is common to all of the links and is the standard wide-band cable transmission system used throughout Cape Kennedy and MILA. Figure 2-4 shows the route of the A2A System that would be used to interconnect data terminals at LC39A with the data terminals at Hangar AO.

2.4.2.1 Down-Link Functions. The Down-Link Data Transmission Terminal (DDTT) accepts analog signals, discretes (ON/OFF signals), digital serial signals, digital parallel signals and modulated subcarrier signals. It converts these signals into a form suitable for cable transmission, i.e., a digital serial signal or a modulated subcarrier of the proper level and transmits the converted data to the proper A2A transmitter located in the Pad Terminal Connection Room (PTCR). As a redundant feature, either DDTT can be used with either Planetary Vehicle and its associated LCE. The A2A system transmits the data from the PTCR to the vicinity of the STC, where it is received by the Down-link Data Reception Terminal (DDRT). The DDRT selects the proper A2A receiver and reconverts modulated subcarriers to their original levels and routes them to the STC junction box. It establishes synchronization with the received digital serial signal, converts the digital serial into parallel words for transmission to the CDS, and exchanges control signals with the CDS for computer input operations. Either DDRT may be used with either STC.

Figure 2-5 shows the signal flow in the DDTT from the mobile launcher to the hangar. Although the exact characteristics of the input signals are not defined, it has been determined that the data will be of five major types: analog signals, discretes, digital serial signals, digital parallel signals, and modulated subcarriers. Descriptions of these data follow.

a. Modulated Subcarriers. The signals are routed through signal conditioners to match signal levels to the data link and then through a switching matrix to the available A2A transmitters. This signal is preserved unchanged during transmission.

b. Analog Signals. The analog signals would be scanned by an analog multiplexer which would sample each signal and produce a single time division multiplexed (pulse amplitude modulated) output signal. This signal would be applied to an analog-to-digital (A/D) converter which would convert the amplitude of each pulse into a parallel digital word. The digital words would then be applied to a digital multiplexer to be combined with other digital signals into a single PCM-type of serial digital output signal.

c. Discretes. Discrete signals of various amplitudes would be converted into digital messages by a discrete encoder. These messages would then be sent to the digital multiplexer for combination with the other digital signals.

d. Digital Serial Signals. A digital serial signal such as the assumed 50-kbs Capsule Telemetry Encoder Output would pass through a signal conditioner and switching network to an A2A transmitter in the same manner as the modulated subcarriers.

e. Digital Parallel Signals. Digital parallel messages from the output registers of various pad-vicinity LCE would be accepted directly by the digital multiplexer for multiplexing with other digital signals.

The Downlink Data Reception Terminal (DDRT) would accept three kinds of signals from the A2A receivers as shown in Figure 2-5. These signals are reconverted as follows:

a. Modulated Telemetry Subcarriers. A nominal 1-volt peak-to-peak modulated telemetry subcarrier would be routed through a switching network and a signal conditioner which would restore the signal to its original amplitude. The restored signal would then be routed directly to the Telemetry Subsystem support equipment.

b. Digital Serial Signals. The two types of digital serial signals are the 50-kbps Capsule Telemetry Encoder output signal and the DDTT PCM Multiplexer output. The capsule signal is routed through the switching network and a signal conditioner and then on to the capsule support equipment. The PCM signal is routed through the switching network to a PCM decommutator, which decommutates the PCM signal and distributes digital parallel words to the STC computer complex and to digital-to-analog converters and digital-to-discrete decoders. The analog and discrete signals, reconverted to their original forms, would then be routed to the appropriate subsystem support equipment in the STC.

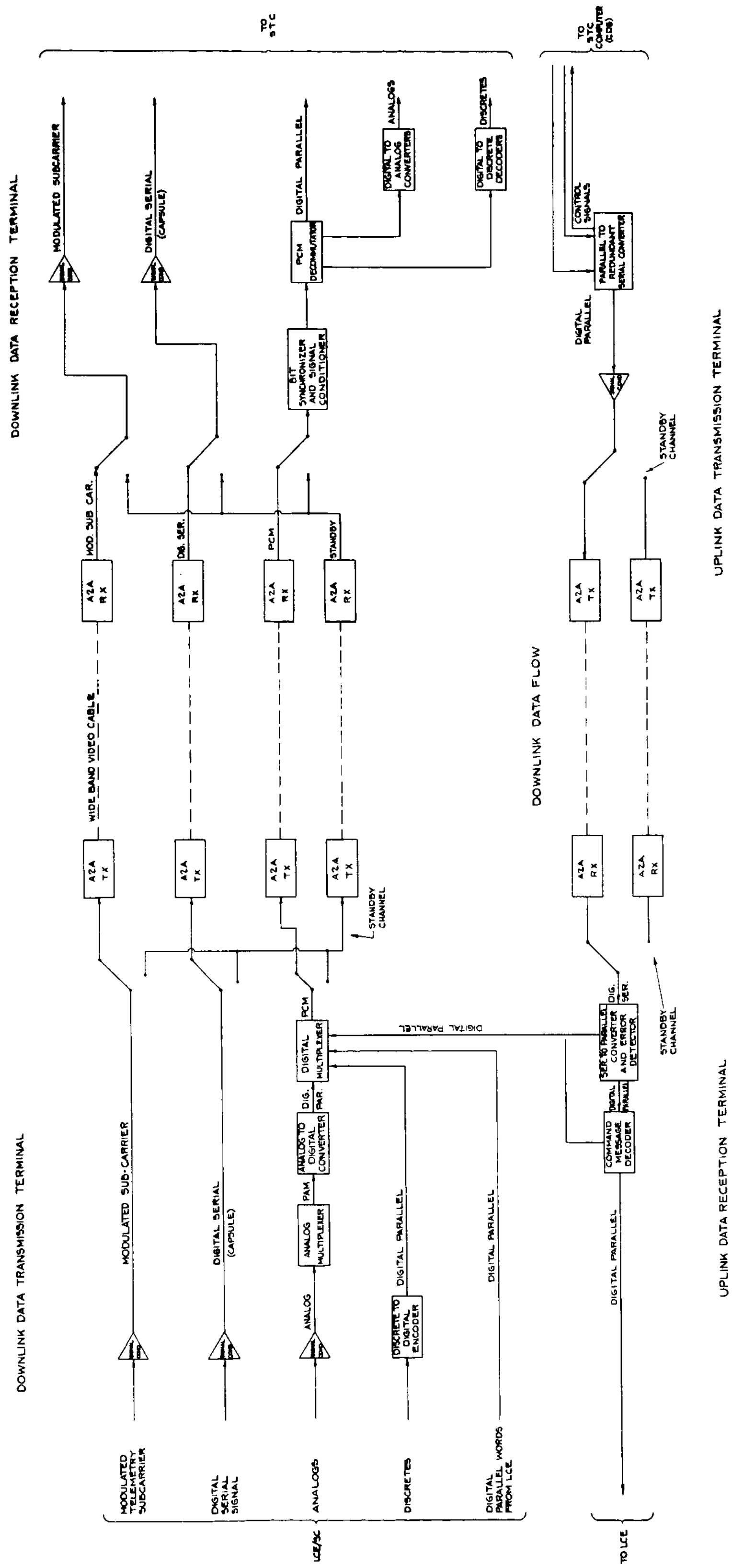


Figure 2-5. Up-Link Data Flow

2.4.2.2 Up-Link (STC-to-pad) Functions. As shown in Figure 2-5, the Up-Link accepts digital parallel command words from the STC computer complex, converts each word into a redundant serial digital word suitable for wide-band cable transmission, transmits each word to the Mobile Launcher, checks each word for transmission errors, verifies the accuracy of received commands, and decodes and/or distributes commands to the proper LCE in the vicinity of the Mobile Launcher.

a. The Up-Link Data Transmission Terminal (UDTT) as shown in Figure 2-5 exchanges control signals with the CDS and receives digital parallel words from the computer. Each word is transmitted in series twice, in such a manner that a single double-length digital series word is transmitted for each parallel word received. Each series word would then be routed through the switching network to an A2A transmitter.

b. The Up-Link Data Reception Terminal (UDRT) as shown in Figure 2-5 uses an error detector to check each received word for transmission errors and places the reconverted command message into a Command Message Decoder Unit. The decoder unit distributes the command messages to the C&S LCE, the power LCE, or LCE control. The message decoder also converts data-link commands to discretes for data-link control.

2.4.3 LCC-STC Data Link. In order to have STC control of firing room status displays it will be necessary to transmit discretes from the test conductor's consoles in the STC's to the VOYAGER consoles in the LCC Firing Room. The system suggested here can be used to transmit as many as 24 independent discrete levels over a single pair of 19 gauge telephone wires. Figure 2-6 shows the route which would be taken by the STC-LCC data link.

At each STC test conductor's console, provisions will be made for the generation of ON/OFF discrete states corresponding to console panel displays. These discretes would be used to key Voice Frequency Carrier (VFC) transmitters, each of which operate at a different voice frequency. An OFF state would be transmitted as the nominal center frequency of a given transmitter. An ON state would be transmitted as a frequency which would be shifted 30 Hz from the center frequency.

The outputs of up to 24 transmitters would be applied through a line termination module to a single pair of telephone lines for transmission directly to the LCC Firing Room approximately 11 telephone miles away. At this distance no repeaters would be required.

The VFC receivers would be located within the LCC Firing Room. Each VFC receiver is tuned to a separate transmitter and will provide an output corresponding either to the received center frequency or to the received shifted frequency. The VFC receiver output states would then be used to actuate relay lamp drivers in the VOYAGER status display consoles.

2.5 LCC Control and Display. In the Launch Control Center the personnel-to-facility interface is extremely important. The LCE and its operation must be integrated into the firing room equipment and procedures. It is recommended that the VOYAGER LCC equipment group be installed in the racks allocated to Apollo Launch Equipment. This should present no difficulties because the firing room designated for VOYAGER will require no other payload equipment. Space in this room is at a premium, and there will be no other convenient place to put it.

The VOYAGER test coordinator and any supporting personnel will be in the firing room during prelaunch testing and countdown providing a face-to-face interface between Voyager and Saturn programs during pad activities.

Figure 2-7 shows a typical LCC firing room layout with the preferred VOYAGER LCE location identified. A minimum of five of the designated twin consoles will be needed, primarily to insure adequate sit-down space for the personnel.

The test coordinator has access to current Spacecraft status at all times. He receives status displays operated by the STC, has access to closed circuit TV monitors, has operational intercoms available, has access to direct communications with STC and launch conductor, and has access to countdown clock displays. In addition to the displays for use in coordinating tests there will be limited controls and displays for emergency use. These controls will operate critical functions in the Mobile Launcher.

The equipment provided in the LCC must be capable of displaying selected, launch critical parameters supplied by the STC; of commanding predetermined hardwire lines for Planetary

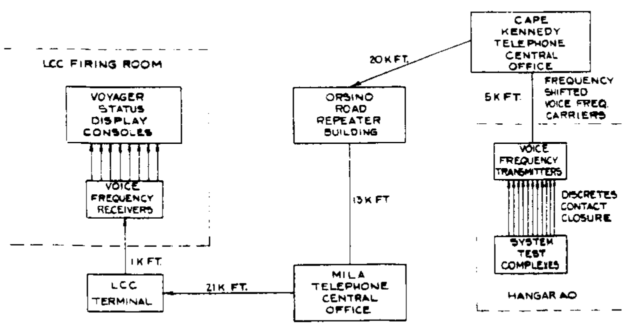


Figure 2-6. Voice Frequency Carrier System, LCC to STC

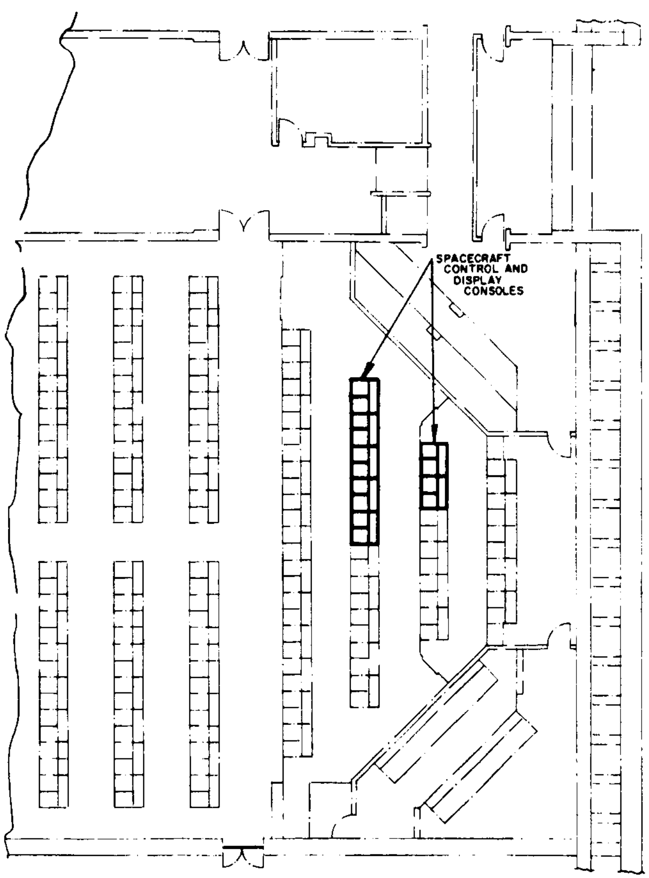


Figure 2-7. LCC Firing Room

vehicle control backup purposes; providing for exchange of countdown status information between the Planetary Vehicle, Launch Vehicle, and Launch Complex; and of utilizing facility standard supporting services and equipments, such as operational intercom (OIC), closed circuit television (CCTV), weather data, range safety status, and power.

2.5.1 Equipment Description. The LCC Controls and Displays are shown in equipment block diagram Figure 2-8.

The Planetary Vehicle Status Panel receives summary status information from the Spacecraft and Capsule test conductors in the STC and displays these data by illuminating status advisories. Discrete signals are received through the low-speed data link between the LCC and STC. Displayed information should include overall L/V Status, Fore Spacecraft Status, Fore Capsule Status, Aft Spacecraft Status, and Aft Capsule Status. To complete this panel's utility as a top-level quick-look status advisor, the SFOF, DSN, and DSIF status should be included.

The Subsystem Status Panels contain displays that are similar to, and supplement the Planetary Vehicle status panels by providing more detailed status on those Planetary Vehicle subsystems most critical to the launch process. These are tentatively identified for each flight spacecraft as Power Subsystem, Pyro Subsystem, Thermal Subsystem, and Science Subsystem.

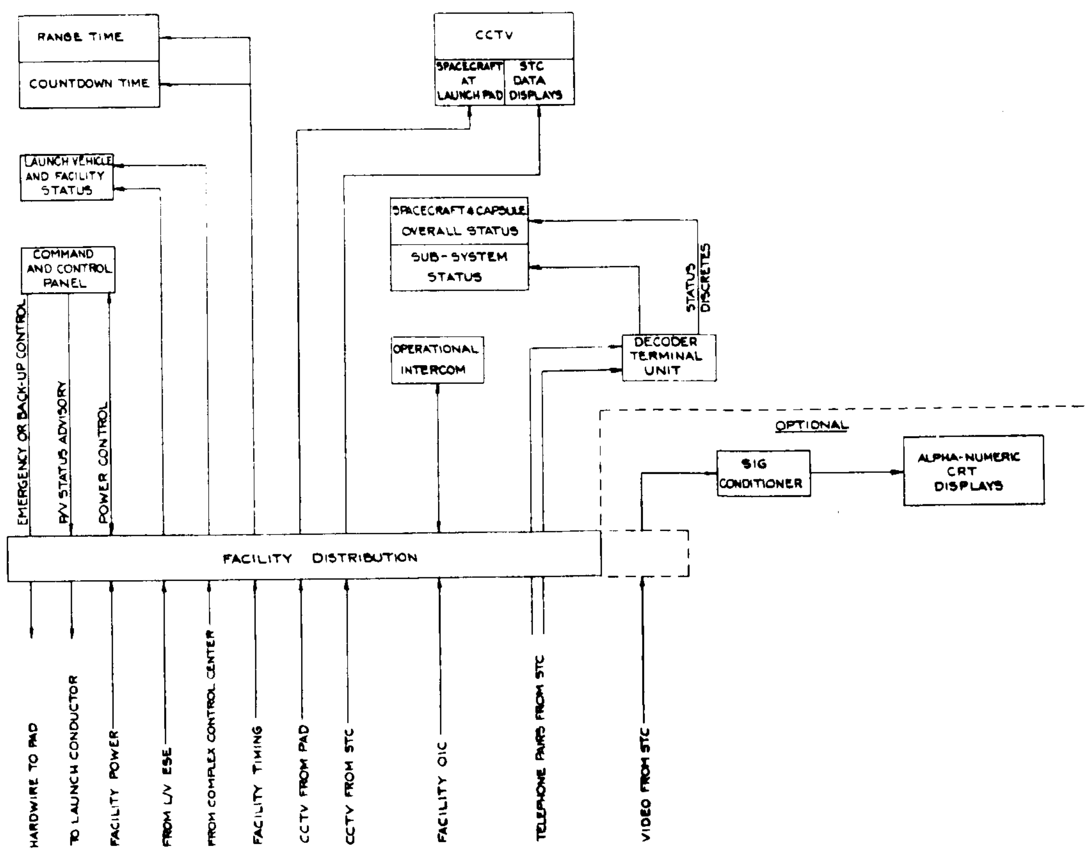


Figure 2-8. LCC Control and Display Group, Functional Block Diagram

For closed circuit television, a requirement would be placed on KSC Facilities to furnish at least two CCTV displays at the Test Coordinators console. One would view the Planetary Vehicle and the umbilical connections during pull-away. The other would provide a view of test activities at the STC. Selectable cameras at the STC will be focussed on test conductors consoles, CRT displays, or teletype printouts to furnish personnel in the LCC with quantitative test data, plus a visual "feel" for STC test activities.

The Operational Intercom (OIC) is a standard LCC Intercom Panel (GFE) that is integrated into each control and display console. Approximately six channels are required, one to the pad, two to the STC, two to DSIF 71 and MOS, and one to launch control.

The Time Displays are standard numeric displays (GFE) of range time, local time, and countdown time that are integrated with the Test Coordinator's displays. Manual controls are provided for operating the mobile launcher LCE through direct connections from LCC to the mobile launcher. These controls assure positive control of safety critical power, pyrotechnics, and facility elements at all times. The direct connections to the ML also provide means to monitor critical temperatures, switch positions, and Spacecraft status. These controls can also be used to implement control functions, such as facility power control and data link switchover, to a backup mode. These controls, however, are not intended to be used except during emergencies or other unusual conditions when normal control from the STC cannot be effected.

An additional possibility for presenting information to the test coordinator is the use of an alphanumeric CRT display. This would be possible only if AN/CRT displays are used in the STC, with a duplicate display in the LCC. This display would provide additional, detailed information to the test coordinator. Since the display would require a wide-band data link between STC and LCC and there is no certainty that the additional information is useful, the AN/CRT display is not included as a recommended item, but only as a possibility.

2.6 Command Insertion

2.6.1 RF Commands. Commands can be inserted into the Spacecraft from the STC through the RF link. The Radio OSE in the hangar transmits the commands which are received by the Spacecraft and processed in the same manner as any other transmitted command. Real-time commands are executed, and stored commands are routed to the computing and sequencing system.

2.6.2 Computing and Sequencing System Launch Complex Equipment. The C&S LCE provides the necessary interface and controls such that command data and control signals can be entered into the C&S subsystem on the pad. The C&S LCE is similar to the C&S OSE in the STC. The LCE circuits are identical to the output circuits in the C&S OSE.

The CDS will generate 24-bit control words. The first six bits are the address of the C&S LCE. The remaining 18 bits are digitally encoded control functions or spacecraft memory contents. The words are transmitted to the C&S LCE through the wide-band data link. The C&S LCE requires longer words; two or possibly three control words will be required to make up one C&S memory word. The spacecraft memory word is assembled by storing the control words in a format register, then reading them out when the proper configuration is reached. To keep the bit rate at 250 bps, two identical sets of format registers can be used, loading one while the other is read into the C&S subsystem. Data from the C&S subsystem is handled in the same manner. The data is read into registers and then formatted into 24-bit words for transmission to the CDS through the data link.

Discrete C&S control signals are also implemented through the C&S LCE, which decodes 24-bit control words as above but uses them to operate the appropriate discrete signal. Each discrete is verified to the CDS through the data link with words generated by the C&S LCE.

The C&S LCE also has controls so that memory data and control signals can be manually entered into the C&S subsystem.

2.7 Ground Power Source. A ground power source will supply power to the Spacecraft power subsystem via the normal umbilical connection. The power supply will be located in the mobile launcher equipment room for on-pad checkout and countdown. The power source will normally be remotely controlled by the STC via the LCE data link. However, local power control capability will be provided for use during an emergency or tests requiring local control.

The power source will be voltage and current limited to protect the Spacecraft and cabling in the event of an equipment failure. The voltage and current output of the power source will normally be monitored remotely by both the CDS and Power Subsystem OSE via the LCE. However, local monitoring and alarm capability will be provided as a backup to the remote capability. In both cases, the input power will be continuously monitored, and adequate alarms (visual, audio, and signals to CDS) will be provided to indicate out of tolerance conditions.

Power transfer from external to internal will be accomplished by decreasing the voltage level of the ground power source until the batteries assume all of the Spacecraft load. The ground power source will then be removed. Power transfer from internal to external will be essentially the reverse of the above, with the voltage level of the ground power source being gradually increased until it assumes the Spacecraft load. External power will be removed if the input power monitors detect an out of tolerance high condition.

There should be a separate power source for each Spacecraft with isolated grounds. Since it appears that these power supplies must operate continuously while the spacecraft is at the pad, a backup power supply should be provided which can be switched to support either Spacecraft in the event of a power supply malfunction.

2.8 Interfaces

2.8.1 LCE-Planetary Vehicle Interface. The LCE-Planetary Vehicle interface consists of the umbilical connector for each spacecraft. A complete list of umbilical functions appears in Table 2-2. This list includes all functions known at this time. Possible functions which may be added because of specific flight hardware selections are not included.

TABLE 2-2. UMBILICAL FUNCTION LIST

Subsystem	Function	Signal Type	Up/Down	Subsystem	Function	Signal Type	Up/Down
Radio	High Gain Antenna Probe (Coax)	RF 2295 MC/2115Mc	D/U	Guidance & Control	Gyro Rate Signals (6)	Analog	D
	Med. Gain Antenna Probe (Coax)	RF 2295 MC/2115 Mc			Initialization	Discrete	U
	Secondary Low Gain Antenna (Coax)	RF 2295 MC/2115Mc		Pyrotechnic	Measure Continuity	Analog	D
	Primary Low Gain Antenna Probe (Coax)	RF 2295 MC/2115Mc	D/U		Separation Switch Monitors	Discrete	D
	Relay Radio Probe	RF 400 MC	U		Simulate Separation	Discrete	U
Telemetry	Modulated Subcarrier	270 KC	D		Arm Monitor	Discrete	U
Controller & Sequencer	Update Master Sequence	Discrete	U		Arm Enable	Discrete	U
	Alert	Discrete	U	Science	Undetermined		U/D
	Sync	Digital 250 PPS	U	Propulsion	Tank Temperatures	Analog	D
	Information	Digital 250 BPS	U	Capsule	Undetermined		
	End of Word	Discrete	U	Power	Array/Battery Bus Voltage	Analog	D
	Δ V Alert	Discrete	U		Array Enable On/Off	Discrete	U
	Alert TTG (7)	Discrete	U		Array Enable Monitor	Discrete	D
	G&C Commands (6)	Discrete	U		Battery Voltage Monitors (3)	Analog	D
	PSP Commands (2)	Discrete	U		Battery Temperature Monitors (3)	Analog	D
	High Speed Clock	Digital 500 PPS	U	Data Storage	Power (0-60 VDC, 15a)	Power	U
	Preset TTG	Discrete	U		Set Recorder to Launch Mode	Discrete	U

2.8.2 LCE-ML Interfaces. Interfaces between the mobile launcher and the LCE are listed in Table 2-3. Location of the equipment is shown in Figure 2-9.

TABLE 2-3. INTERFACES BETWEEN MOBILE LAUNCHER AND LCE

Function	Description
1. Facility Power	208 Vac, 3ø, 21 cycle 4 wire, 20 amperes
2. Emergency Power	208 Vac, 3ø, 21 cycle 4 wire, 20 amperes
3. Data Link Cables	5 A2A links
4. Operational Intercom	6 channels
5. Equipment Space	Mounting space, front and rear access to approximately 10 standard cabinets
6. Umbilical J-Box Mounting Space	Space and mounting brackets for J-Box on tower
7. Antenna Mounts	Space and mounting brackets for 2 S-band antennas, 2 dipole antennas, and 2 power dividers

2.8.3 LCE-LCC Interface. The location of the controls and displays in the LCC is shown in Figure 2-7, LCC Firing Room. The LCE-LCC interfaces are listed in Table 2-4.

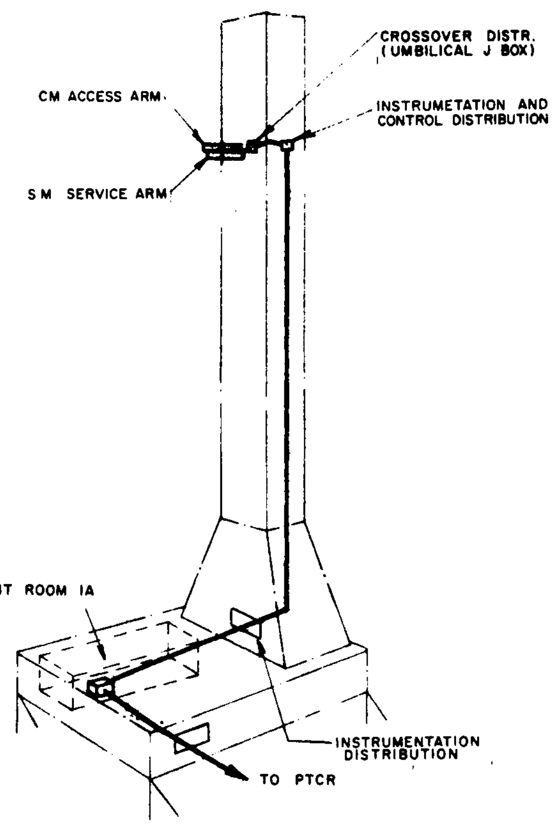


Figure 2-9. Location of LCE in Mobile Launcher

TABLE 2-4. LCE-LCC INTERFACES

Function	Description
1. Facility Power	120 Vac, 30, 60 cycle, thru wire (Approximately 8 circuits)
2. Operational Intercom System	8 Intercom stations, 6 channels
3. Closed Circuit TV	Two CCTV displays convenient to consoles
4. Operator Consoles	Provision of 6 to 12 bays of console space
5. Equipment Racks	Provision of 6 standard racks for mounting of control equipment

2.8.4 LCE-STC Interface. The types of signals transmitted are described in the RF and data link sections. The functions exchanged are listed in Table 2-5.

TABLE 2-5. MOBILE LAUNCHER TO STC AND STC TO MOBILE LAUNCHER

Function	Description
STC to Mobile Launcher	
1. Power Distribution Control	Discretes
2. Power LCE Switching	Discretes
3. Power LCE Voltage Adjust	24-bit words
4. Spacecraft Control	Discretes
5. LCE Control Signals	Discretes
6. Data Link Controls	Discretes
7. C&S LCE Controls	Discretes
8. Command Load	24-bit words
9. Operational Intercom	Voice
10. Command-Radio	S-band RF
11. Relay Radio (Capsule Simulation)	400 mc
Mobile Launcher to STC	
1. Power Distribution Monitor	Analog voltage
2. Power LCE Switch Verify	Discretes
3. Power LCE Voltage	Analog
4. Spacecraft Monitors	Analog/discretes
5. LCE Control Verify	Discretes
6. Data Link Control Verify	Discretes
7. C&S LCE Load Verify	24-bit words
8. C&S LCE Control Verify	Discretes
9. Telemetry Composite	270 KC
10. Operational Intercom	Voice
11. Telemetry Radio	S-band RF

2.8.5 LCC-STC Interface. The functions of the LCC to STC Interface are listed in Table 2-6.

TABLE 2-6. LCC TO STC INTERFACE

Function	Description
1. 20 Discretes	Status advisories operating indicator lights
2. Operational Intercom	Voice

2.8.6 LCC-Mobile Launcher Interface. The functions of the LCC to Mobile Launcher Interface are shown in Table 2-7.

TABLE 2-7. LCC TO MOBILE LAUNCHER INTERFACE

Function	Description
1. Power Distribution Control	Discrete Commands
2. Power Distribution Monitor	Discrete Monitors
3. Emergency Power Control	Discrete Commands
4. Spacecraft Power Switchover	Discrete Control
5. Spacecraft Power Switch-over Control	Discrete Monitor
6. Arm Reset	Discrete Command
7. Arm Monitor	Discrete Monitor
8. Continuity Loop Monitor	Discrete Monitor

2.8.7 LCE-Launch Vehicle ESE. Spacecraft Temperatures Analog Signals.

3.0 EXPLOSIVES SAFE FACILITY EQUIPMENT.

The explosive safe facility should have a set of LCE similar to that at the pad to support spacecraft system tests and operation.

In addition to the test equipment which is a functioning system, additional pieces of equipment are required to support the operations for which the ESF is intended, such as propellant loading, pyro loading, and proof pressure testing.

3.1 LCE (Test Equipment). The system test support equipment located at the ESF is functionally identical to the equipment located at the launch complex. This equipment will include the equipment in the mobile launcher, the equipment on the umbilical tower and the same display panels as those installed in the LCC. The only deviation from the pad configuration will be those deviations required because of differences in the facility. Figure 3-1 shows the equipment to be provided for ESF System Tests.

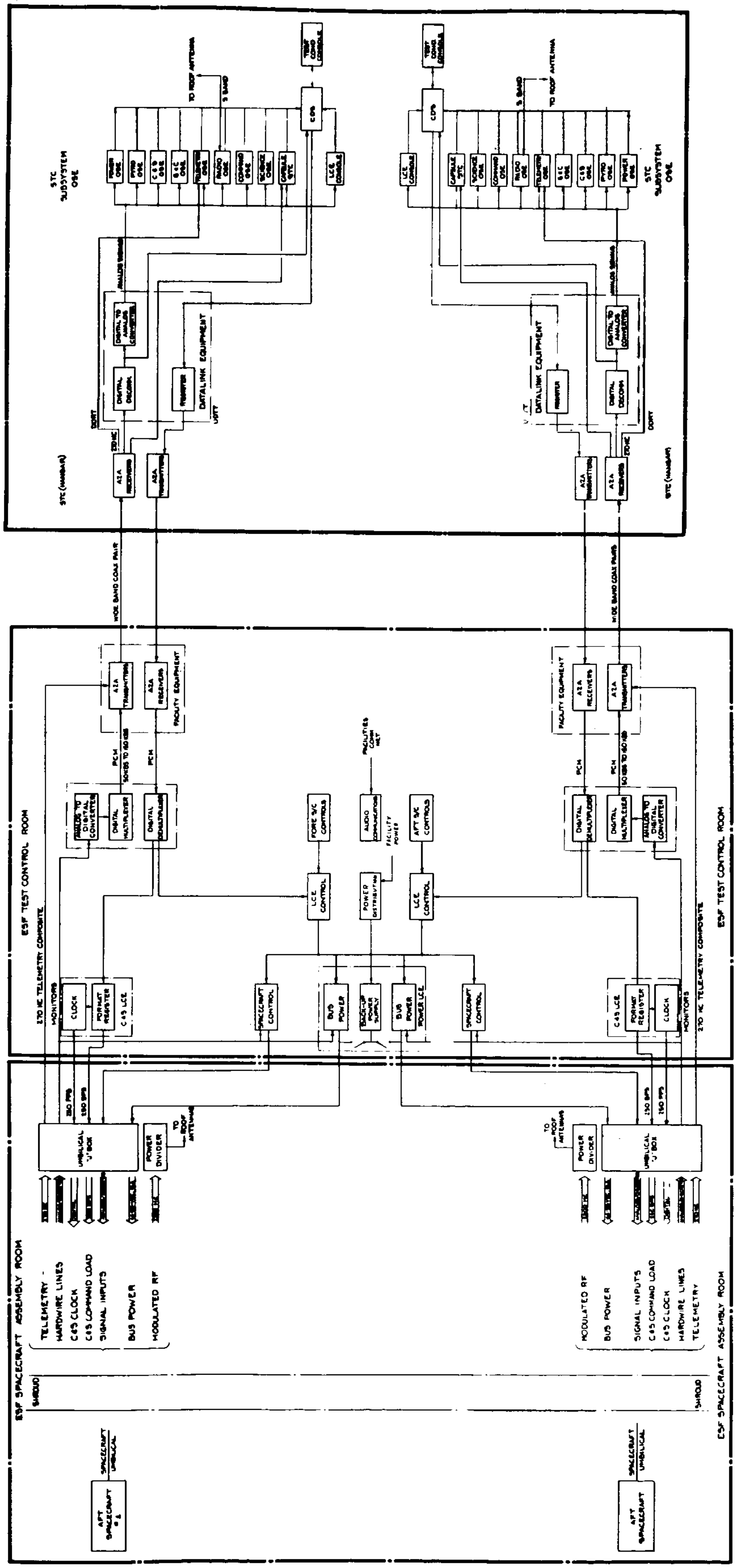


Figure 3-1. LCE Functional Block Diagram, ESF Usage Excluding RF Equipment

The ESF Test LCE need not be dispersed as is the pad LCE. Most of the equipment should be together. The umbilical J-boxes will be portable and located near the two Spacecraft. The rest of the test LCE can be located up to 300 feet away. Elements of the test coordinator's displays should be packaged for specific use in the ESF.

This set of equipment can be operated in the same manner as the LCE at the pad, but it is anticipated that the manual mode of operation and local control would be used much more frequently in the ESF than it is at the pad.

The RF Link between ESF and STC should have the same capability as the Pad-STC link but the line-of-sight capability must be provided in a different manner. The Spacecraft will be at ground level in the ESF as compared to 300 feet in the air at the pad. It is also expected that the RF link will be used with the vehicle in the ESF buildings, outside the buildings, and with or without the surrounding shroud. Figure 3-2 shows the expected configuration of the RF link at the ESF.

Line-of-sight communication for two spacecraft would be provided by mounting antennas on the roof of the ESF buildings. Again, there would be an S-band antenna and a UHF antenna for each Spacecraft, making a total of four antennas on the roof. It is anticipated that line-of-sight transmission is possible by using the Spacecraft shroud antenna when the Spacecraft is outside the ESF buildings.

The data link used between the hangar and the ESF should be identical to the link used between the hangar and the mobile launcher and LCC except for the lack of redundancy since launch pad reliability is not required. The procedures and STC equipment to be used for controlling the LCE can then be realistically checked on a system test before the two spacecraft reach the pad.

3.2 Leak Test Equipment. Prior to propellant and gas loading of the Spacecraft, it is important to reassess the leak profile characteristics and pressure safety of applicable Spacecraft Subsystems. The Leak Test OSE may be used to apply controlled gases to accessible subsystem test points in order to establish proof pressures and differential pressures between subsystem sections and between the subsystems and the ambient atmosphere. The differential pressures cause gas to flow through leak paths, and this gas flow is used to detect and analyze the leaks by measurements of pressure drop, pressure rise, and tracer gas concentration.

The Spacecraft Subsystems should be comprehensively proof-pressured and leak-tested at least three times, the final test to be performed in the ESF. All of the leak testing should be performed without the use of liquids, because gases allow easier, faster, and more accurate analysis of small leaks and because liquids can plug small leaks so that they might not be detectable. Records from this series of tests should be compared to obtain a knowledge of the leak characteristics of the subsystems, such as the incidence and size of new leaks and the trend or rate of increase of existent leaks. Time extrapolation of this data may be made to forecast the integrity of the subsystems during the mission.

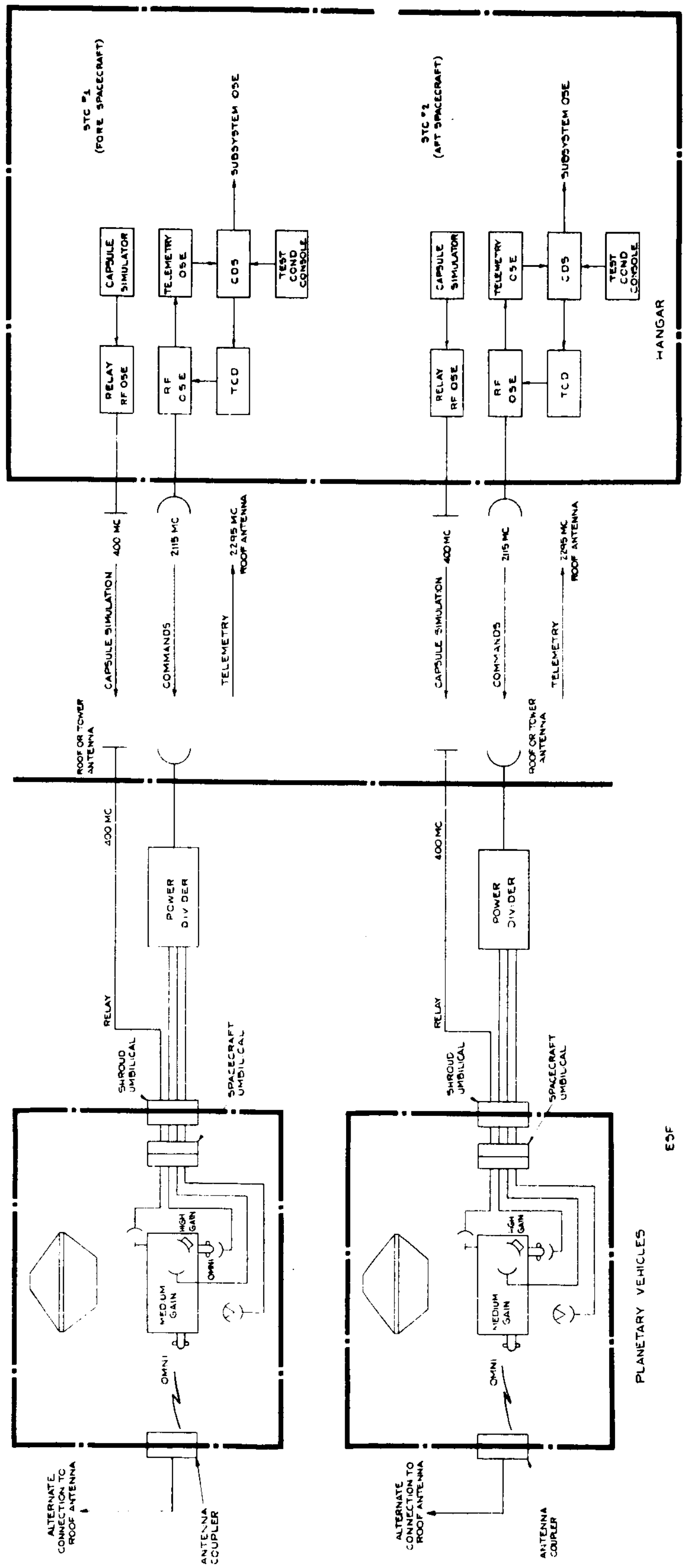


Figure 3-2. ESF to STC RF Link

During and after propellant loading of the propulsion subsystem, gas analyzers of this OSE would be used to "sniff" for traces of propellant issuing from external subsystem leaks.

3.2.1 Description of Leak Testing. Complete test capability of the OSE in detecting, locating, and measuring Spacecraft Subsystem fluid leaks and in application of proof pressures can be realized only when all test access points of the subsystem are accessible for connection to the OSE and when all exterior surfaces of the subsystems are accessible to the near approach or contact by gas analyzer probes. Therefore, complete proof pressure and leak profile testing can only be accomplished prior to Spacecraft assembly, since after this time some test points and exterior surfaces will no longer be accessible to the OSE.

The concept of testing with this proposed OSE would be to perform leak profile tests on subsystems prior to Spacecraft assembly. Thereafter the OSE would be used to continuously monitor for subsystem tank leakage only by detecting pressure drop with portable OSE elements attached to the subsystem service connections until mission gas and liquid propellants are to be loaded. After these fluids are loaded, the pressure drop monitoring elements would be removed and all leak monitoring would be confined to the detection of propellant vapors in the atmospheric environment of the Spacecraft by OSE vapor detectors.

The equipment used for leak testing does not interfere with the rest of the LCE. Since its function is independent of the rest of the LCE, the complete functional description is included in the hardware description in LCE System Hardware Functional Description VC280FD101.

3.3 Propellant Loading Equipment. All fluid propellants will be loaded at the ESF. The propellants to be loaded include attitude control gas, pressurant gases, and liquid propellants for midcourse correction systems. The subsystems supported include the Mid-Course/Orbit Adjust, Attitude Control, and Orbiter Separation Propulsion. Although several different fluids are loaded, there is enough duplication of facilities used and equipment required to justify combining all loading functions in one set of equipment.

Since the loading equipment does not interface with other parts of the LCE system and performs its function independently, the functional description is included in CII VC280FD101, LCE Hardware Functional Descriptions.

3.4 Pyrotechnic Loading. All pyrotechnics should be loaded in the ESF. Loading is done on an electrically dead spacecraft. In order to ensure that the pyrotechnic system is properly prepared for loading (in a safe condition) and to check it after loading, a test kit will be required. Since this test kit operates independently of the other LCE, its functional description is in the LCE System Hardware Functional Description, CII VC280FD101.

CII VC280FD101

LAUNCH COMPLEX EQUIPMENT HARDWARE FUNCTIONAL DESCRIPTION

SECTION

LAUNCH COMPLEX EQUIPMENT HARDWARE FUNCTIONAL DESCRIPTION

1.0 INTRODUCTION. This document contains functional descriptions of the equipment recommended for the implementation of the Launch Complex Equipment (LCE) described in the "LCE System Level Functional Description" CII VC280FD100. It includes the LCE required by each subsystem of the spacecraft. Also described are the equipment required to enable the System Test Complex (STC) to exert remote control over LCE operation and the equipment required to effect an efficient interface with the launch facilities (including established operating procedures for the Saturn V) and LCE for Explosive Safe Facility Operations.

2.0 UMBILICAL PROCESSING AND CONTROL EQUIPMENT. This section provides a functional description of the umbilical processing and control equipment. The umbilical processing and control equipment includes the electronic assemblies installed in the mobile launcher. These assemblies are needed to control the Spacecraft on the launch pad.

2.1 LCE Power Distribution Equipment

2.1.1 Purpose. The LCE power distribution equipment will control, distribute and monitor facility power used by the LCE in the mobile launcher.

2.1.2 Functional Description. The LCE power distribution equipment should contain circuit breakers for each output line, ammeters and voltmeters, and provisons for remote monitoring of distribution voltages. Contacts in each circuit would be operated locally or remotely to turn ac power to LCE on and off. Provision should be made for installation of line filters. A functional block diagram appears in Figure 2-1.

2.1.3 Interfaces

a. Mobile Launcher: LCE receives 208 vac, 3 phase, 60 cycle, 4 wire.
b. LCE
    1. 120 vac, 60 cycle, 1 phase to all mobile launcher LCE.
    2. 208 vac, 60 cycle, 3 phase to power LCE.
    3. Monitor signals to data link equipment.
    4. Control signals from LCE control.

2.1.4 Physical Description. This equipment can be housed in one standard rack of electrical equipment.

2.2 Power Subsystem LCE

2.2.1 Purpose. The power subsystem LCE provides power to all spacecraft vicinity equipment required to supply, control, and monitor the ground power input to the spacecraft. The power subsystem LCE consists of three ground power sources, one for each spacecraft and one as a backup source for emergency use.

2.2.2 Functional Description of Ground Power Source (See Figure 2-2). The ground power source will be located in reasonably close proximity to the Spacecraft at both the Explosive

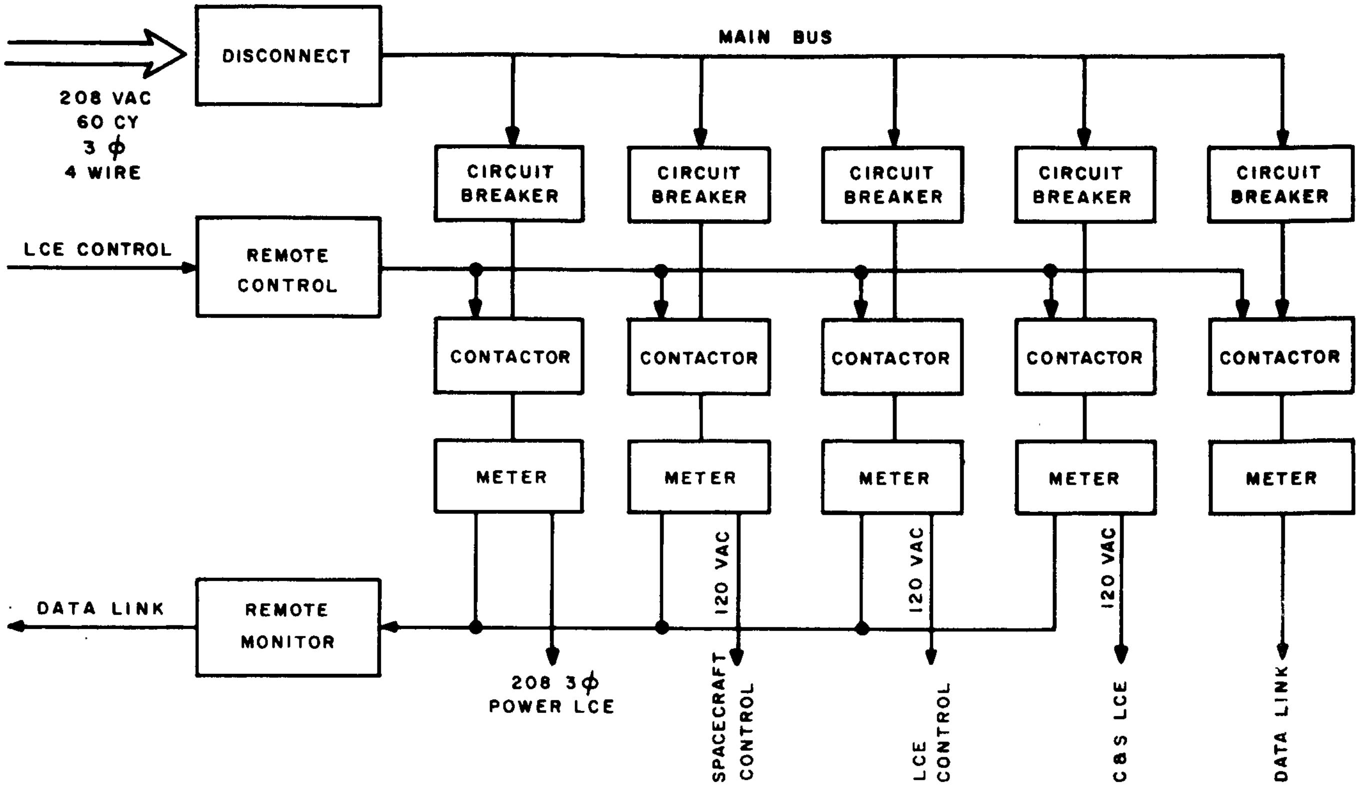


Figure 2-1. Power Distribution Equipment Functional Diagram

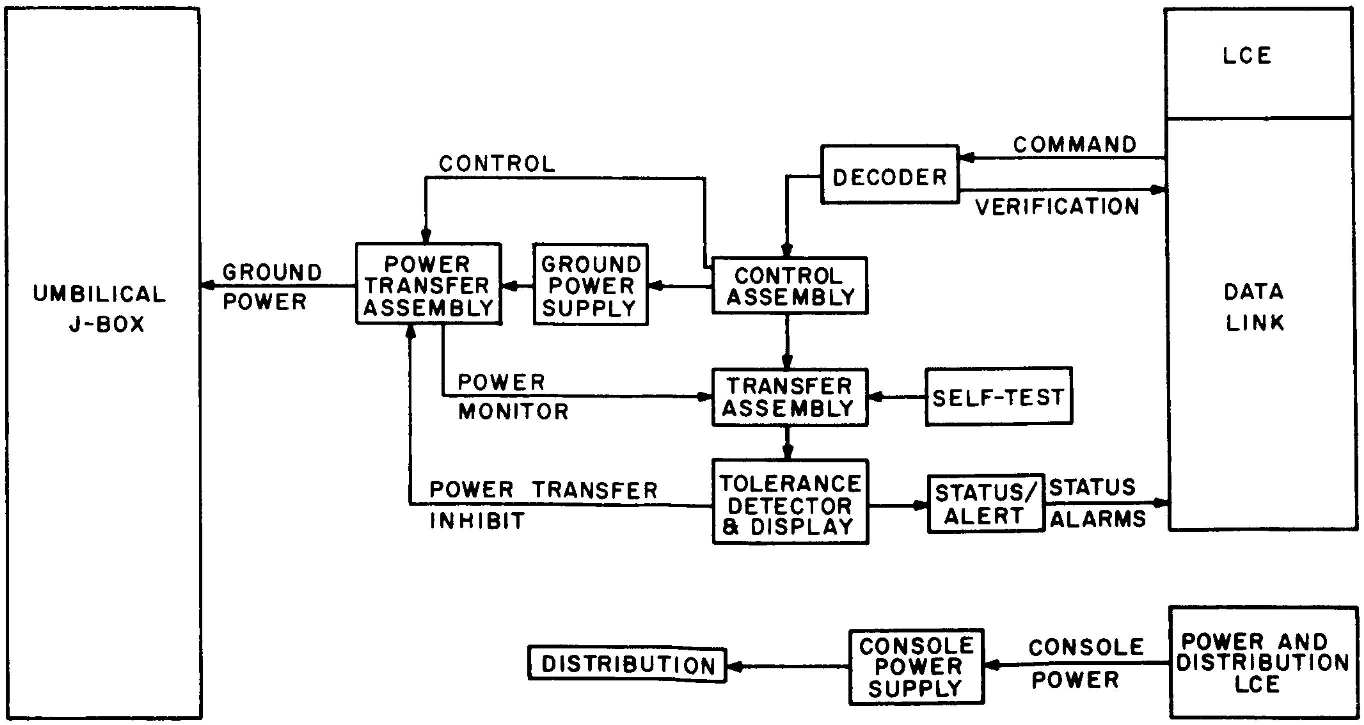


Figure 2-2. Power System LCE Functional Diagram

Safe Area (ESA) for support of confidence tests and in the mobile launcher equipment room for on-pad checkout and countdown.

2.2.2.1 Ground Power Supply. A variable controlled current-limited power supply will provide power to the Spacecraft via the umbilical connection. Remote sensing of the voltage at the umbilical connection will be employed to maintain the input voltage at the desired level. Power transfer from external to internal will be accomplished by decreasing the voltage level of the ground power source until the batteries assume all of the spacecraft load. The ground power source will then be removed. Power transfer from internal to external will be essentially the reverse of the above with the voltage level of the ground power source being gradually increased until it assumes the spacecraft load.

2.2.2.2 Control and Display. The power source will normally be remotely controlled and monitored by the STC via the LCE data link. However, local power control and monitor capability will be provided for use during an emergency (e.g., a data link failure prior to pad evacuation).

2.2.2.3 Tolerance Detection and Alarm. The remote ground power source will contain the capability to continuously monitor the ground power input to the Spacecraft. Adequate alarms (local visual and audio, and signals to the STC via the LCE data link) will be provided to indicate out-of-tolerance conditions. Detection of excessive input power will cause automatic removal of ground power from the Spacecraft.

2.2.2.4 Self Test. Upon receipt of the appropriate command, generated by the STC power subsystem Operational Support Equipment (OSE) or locally if necessary, the remote power source will perform a brief self-test. This self-test will consist of checking the response of the tolerance detection and alarm unit to marginal as well as nominal conditions. Once initiated, the self-test can be run automatically without interruption of spacecraft operations.

2.2.3 Performance Parameters. The performance parameters for external power 0 to 60 volts dc at 30 amps ±1% (tolerance detection).

2.2.4 Functional Interfaces. Functional interfaces are given in Table 2-1.

TABLE 2-1. POWER SUBSYSTEM FUNCTIONAL INTERFACES

Interface	Function	Input / Output	Remarks
Data Link	Command Command Verification Status/Alarms	Input Output Output	13 Required
Power and Distribution LCE	Console Power	Input	
Umbilical	Ground Power	Output	4 Required

2.3 Umbilical "Junction" Box

2.3.1 Purpose. The umbilical junction box provides access to umbilical wires close to the spacecraft and provides any required line isolation or line drivers required to transmit signals to the mobile launcher base.

2.3.2 Description. The junction box would contain terminal boards, amplifiers, power supply, and signal converters. It would be a weatherproof box designed for outdoor use in exposed areas, with operating temperature extremes appropriate to the Kennedy Space Center locality. The "junction" box must be securely mounted to the umbilical tower in a position which provides convenient access.

2.4 Spacecraft Control Unit

2.4.1 Purpose. The spacecraft control unit provides the capability for control of spacecraft subsystems through the umbilical for those subsystems which do not require separate LCE units.

2.4.2 Functional Description. The control unit illustrated in Figure 2-3 will include the necessary controls and displays for the pyrotechnic subsystem, propulsion subsystem, data storage subsystem, and guidance and control subsystem.

2.4.2.1 Pyrotechnic Subsystem LCE. The pyrotechnic subsystem Remote Conditioning Unit (RCU) provides all the spacecraft vicinity equipment required to monitor the electroexposive device continuity loop and the state of the safe/arm monitors.

The RCU should provide the capability to continuously monitor the state of the pyrotechnic controller safe/arm devices (separation switches) and the retropropulsion ignition system safe/arm device. The RCU would provide a discrete to the LCE data link in the event of a malfunction in any of the safe/arm devices. Adequate alarms (both visual and audio) would be provided to warn operating personnel of an inadvertent arm condition.

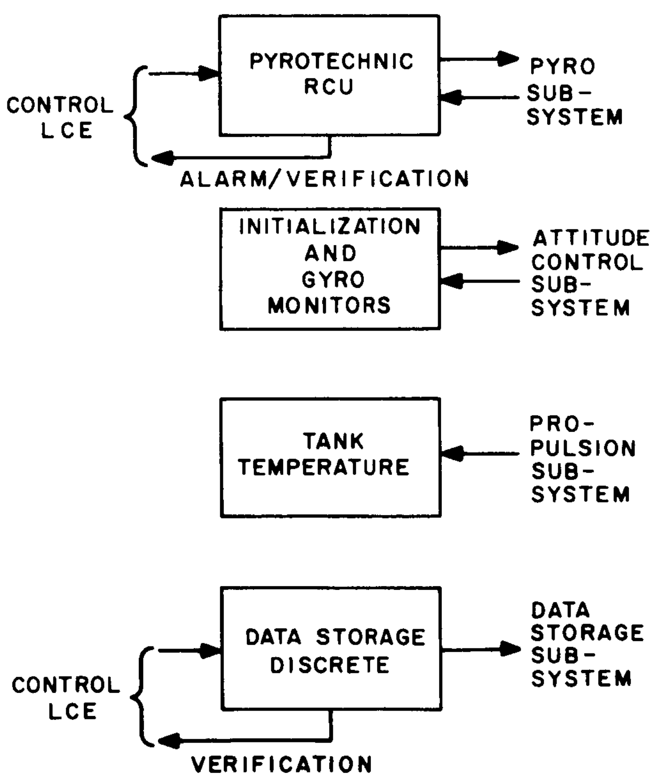


Figure 2-3. Space Control Unit Functional Diagram

The RCU should provide the capability to arm the pyrocontroller during the inert squib firing tests at the launch pad. Upon receiving the appropriate signal via the LCE data link, the RCU would arm the pyrotechnic controller by means of a safe/arm enable unit located in the umbilical tower "Junction" box. The safe/arm enable unit would provide a set of LCE-controlled contacts which parallel the contacts of spacecraft separation switches. This safe/arm capability would be removed before a Spacecraft containing live squibs arrives at the pad, thereby preventing the inadvertent arming of the pyrosubsystem via the OSE.

The RCU would provide the capability to monitor the firing of squib simulators via the total bridgewire continuity loop in the umbilical. This monitoring would be accomplished by measuring the resistance that each triggered squib contributes to the loop. The voltage analog of the resistance will be sent to the STC via the data link.

2.4.2.2 Propulsion Subsystem LCE. Support of the propulsion subsystem consists of providing local displays for monitoring tank pressures and temperatures through the umbilical. These displays would parallel the monitor lines going to the data link and on to the STC.

2.4.2.3 Data Storage Subsystem LCE. This circuit would provide the discrete signal required to switch recorders to a launch mode.

2.4.2.4 Guidance and Control LCE. The spacecraft control unit provides support of the Guidance and Control G&C subsystem by providing local displays of the gyro rate signals in parallel with the lines going to the data link. It also provides initializing signals to the G&C subsystem.

2.4.3 Interfaces. This equipment will interface with the umbilical "junction" box and the LCE control unit. Functional subsystem interfaces are given in Table 2-2.

TABLE 2-2. CONTROL UNIT INTERFACES

Subsystem	Function
Pyrotechnic Subsystem	Squib Monitors Arm Monitors Reset Command Arm Command Continuity Loop Monitors Safe/Arm Enable
Propulsion Subsystem	Tank Temperatures
Guidance and Control	Gyro Rate Signals
LCE Control	Discrete Commands Discrete Verifications Pyro Status/Alarms

2.5 Controller and Sequencer Subystem LCE

2.5.1 Purpose. The function of the Controller and Sequencer (C&S) LCE is to provide the necessary interface and controls to the C&S, such that command data and control functions can be entered into the C&S. The operation and circuitry of the LCE is similar to the STC-OSE.

2.5.2 Functional Description. The C&S LCE functions are shown in Figure 2-4.

2.5.2.1 Memory Loading and Verification. The C&S memory is loaded under control of the Computer Data System (CDS) in the STC. The CDS generates 24-bit control words of which the first six bits are the addresses of registers in the LCE. The remaining 18 bits are read

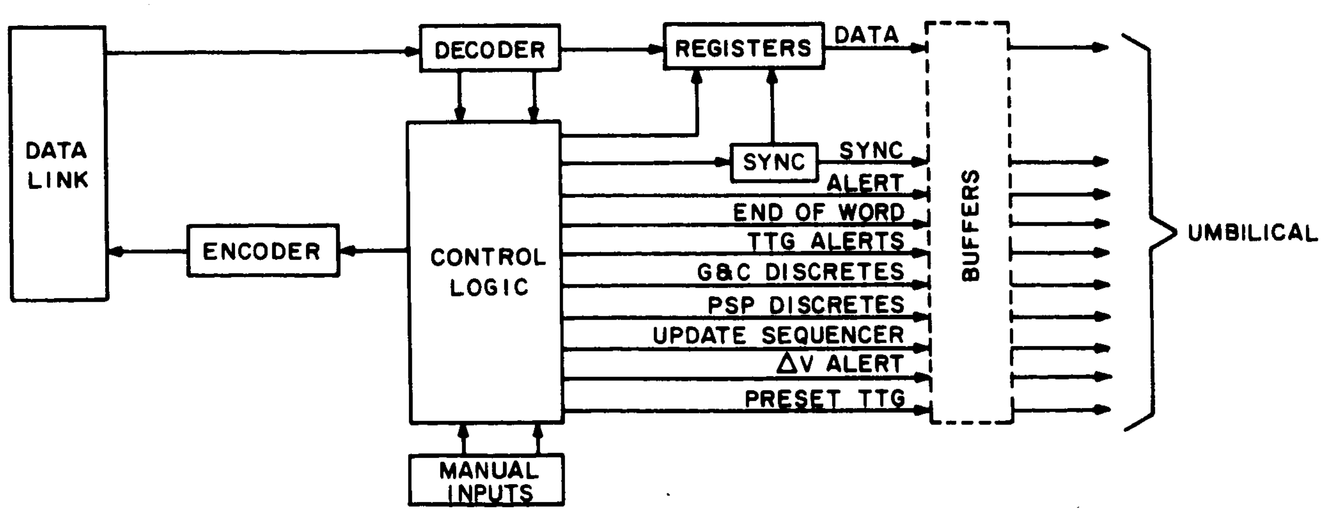


Figure 2-4. Control and Sequencer LCE Functional Diagram

into the registers and can be decoded into control functions or shifted out of the registers as digital data. The C&S requires a 9-bit word to address the memory and an 18-bit word to handle the data. Three LCE registers will be used to format the CDS control words. The first register will store the command to load the memory and the function necessary to accomplish the loading. The second and third registers will contain the address and memory contents, with the second register storing the first 18 bits and the third register storing the remaining nine bits. Readout of the registers will be synchronized by control logic such that the third register is read out immediately after the second, thus forming one continuous memory word. The control logic also contains the necessary counters and delays to implement the function requested by the first CDS control word.

Verification of the memory contents is done in a similar manner to loading, with the exception that only the address of the memory location is generated and the remaining bits only contain logical "0"s. When the memory receives the address it will interrogate that location and shift the contents serially to the telemetry (T/M) subsystem.

2.5.2.2 Functional Units. The decoder-encoder is identical to the decoder-encoder in the STC-OSE (CII VC264FD105). The control logic is similar to the control logic in the STC-OSE. It differs mainly in the fact that the logic for reading the memory and memory-word-time-to-go register is deleted since these functions will be carried out by T/M. The LCE is capable of manually generating the necessary signals to enter and read the C&S memory. Switches are provided for generating the control discretes, the sync, and the digital data. The status of all registers including the encoder-decoder registers will be continuously displayed. The status of the control functions will also be displayed. Outputs from the LCE will be buffered as required to provide the necessary source impedance and isolation to transmit signals through the umbilical.

2.5.3 Interfaces. The C&S LCE interfaces are given in Table 2-3.

TABLE 2-3. C&S LCE INTERFACES

Interface	Function	Input / Output	Comments
Umbilical	Command Data	O	
	Sync	O	
	Alert	O	
	End of Word	O	
	TTG Alerts	O	8 Required
	Guidance and Control Discretes	O	6 Required
	PSP Discretes	O	2 Required
	Preset TTG	O	
	Update Sequencer Discrete	O	
	ΔV Alert	O	
Data Link	Digital Data	I	13 Required
	Digital Data	O	

2.6 LCE Control Unit

2.6.1 Purpose. The LCE control unit receives digital-coded signals from the data link and converts them to discrete output signals for control of subsystem LCE and spacecraft LCE.

2.6.2 Description. Figure 2-5 is a functional block diagram of the LCE Control Unit. The control unit accepts a 24-bit parallel word consisting of a 6-bit address and 18 information bits. The address selects the proper register with the remaining bits paired with each pair controlling a discrete function. In general, a separate bit will be required to initiate a

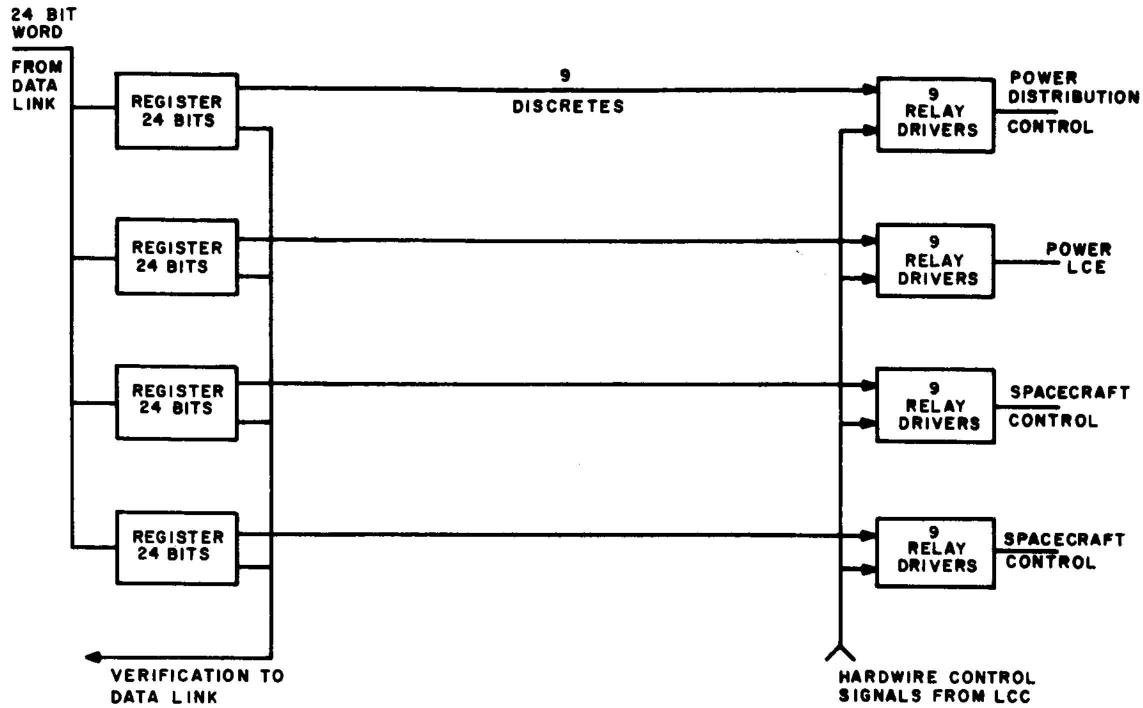


Figure 2-5. LCE Control Unit Functional Diagram

discrete and to turn it off ensuring that a positive signal must be available to initiate each function. Each function initiated provides a return signal to the data link indicating that function has been executed in the support equipment.

The discrete functions will be implemented through form C relay contacts so that the output of the control LCE will normally be a relay contact closure in response to a digital command. In addition to being operated by remote digital commands, each relay circuit may be operated from a hardwire connection which provides parallel external control capability. There will also be provision for local control of all circuits.

2.6.3 Interfaces

Equipment	Function
Power distribution LCE	120 vac, 60 cycle, 3 wire
Spacecraft control	Relay contact closures (control signals)
Power LCE	Discrete control signals (relay contact closures)
LCC control equipment	Discrete outputs (relay or switch closures)
Uplink data transmission terminal	Commands (24-bit words)
Downlink data transmission terminal	Confirmation data

2.7 STC - LCE Control

2.7.1 Purpose. This equipment would be located near the STC. Its function is to provide display and control for the elements of the LCE which are not controlled by the subsystem OSE consoles.

2.7.2 Functional Description. Figure 2-6 is a functional block diagram of the STC - LCE Control. This equipment would provide discrete signals, initiated by switches, which would signal the CDS to generate an appropriate digital word to transmit to the LCE in the mobile launcher. There should be sufficient displays to support the functions controlled. The displays would be activated from discrete signals received from the data link equipment. The discretes will be initiated in the mobile launcher.

The mobile launcher functions that are to be controlled remotely by this LCE, should include power distribution, power supply switchover, back up power supply controls, self-check initiation and any controls required to back up controls on subsystem consoles. The panel should display power distribution monitors, power switchover, back up power supply voltage, Launch Control Center (LCC) status, and countdown status.

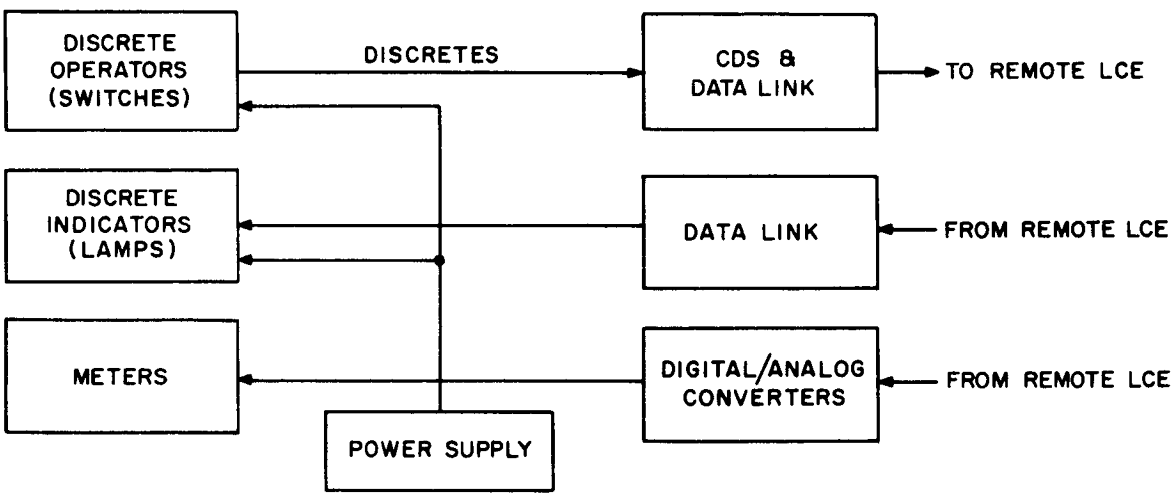


Figure 2-6. STC-LCE Control Functional Diagram

2.7.3 Interfaces. The STC-LCE control interfaces are given in Table 2-4.

TABLE 2-4. STC-LCE CONTROL INTERFACES

Equipment	Function
CDS	Discrete Signals
Data Link	Reconstructed Analog Signals Discrete Signals
STC	120 vac Power

2.7.4 Physical Description. This remote control equipment should consist of two or more panels on which are mounted switches, indicator lights, and meters. The panels may be located in a separate rack or incorporated in a rack with the wide-band data-link equipment.

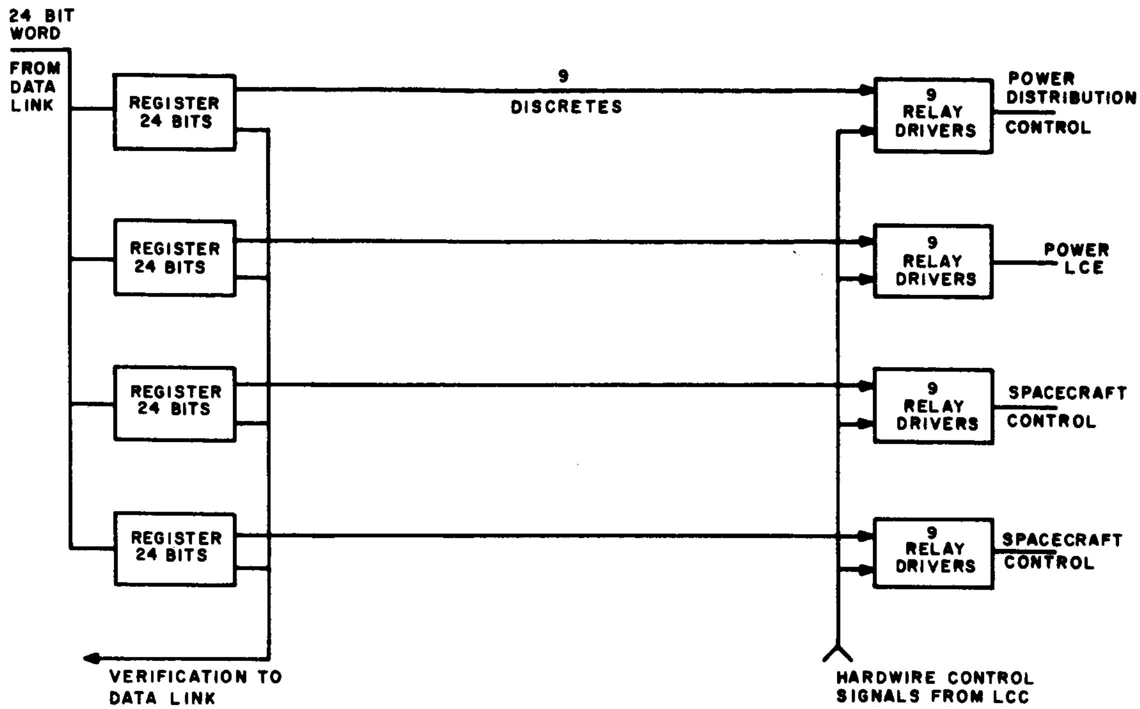


Figure 2-5. LCE Control Unit Functional Diagram

discrete and to turn it off ensuring that a positive signal must be available to initiate each function. Each function initiated provides a return signal to the data link indicating that function has been executed in the support equipment.

The discrete functions will be implemented through form C relay contacts so that the output of the control LCE will normally be a relay contact closure in response to a digital command. In addition to being operated by remote digital commands, each relay circuit may be operated from a hardwire connection which provides parallel external control capability. There will also be provision for local control of all circuits.

2.6.3 Interfaces

Equipment	Function
Power distribution LCE	120 vac, 60 cycle, 3 wire
Spacecraft control	Relay contact closures (control signals)
Power LCE	Discrete control signals (relay contact closures)
LCC control equipment	Discrete outputs (relay or switch closures)
Uplink data transmission terminal	Commands (24-bit words)
Downlink data transmission terminal	Confirmation data

2.7 STC - LCE Control

2.7.1 Purpose. This equipment would be located near the STC. Its function is to provide display and control for the elements of the LCE which are not controlled by the subsystem OSE consoles.

2.7.2 Functional Description. Figure 2-6 is a functional block diagram of the STC - LCE Control. This equipment would provide discrete signals, initiated by switches, which would signal the CDS to generate an appropriate digital word to transmit to the LCE in the mobile launcher. There should be sufficient displays to support the functions controlled. The displays would be activated from discrete signals received from the data link equipment. The discretes will be initiated in the mobile launcher.

The mobile launcher functions that are to be controlled remotely by this LCE, should include power distribution, power supply switchover, back up power supply controls, self-check initiation and any controls required to back up controls on subsystem consoles. The panel should display power distribution monitors, power switchover, back up power supply voltage, Launch Control Center (LCC) status, and countdown status.

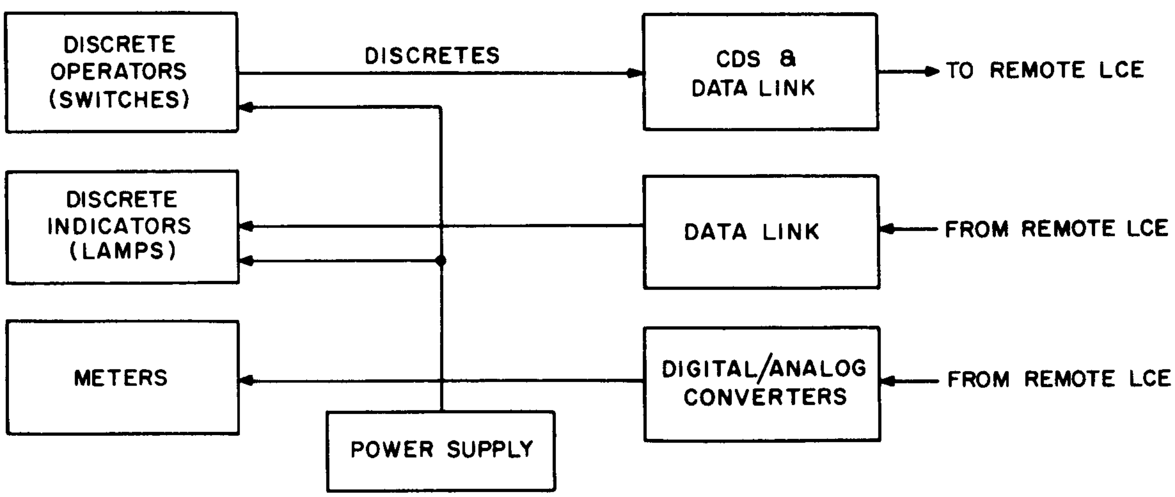


Figure 2-6. STC-LCE Control Functional Diagram

2.7.3 Interfaces. The STC-LCE control interfaces are given in Table 2-4.

TABLE 2-4. STC-LCE CONTROL INTERFACES

Equipment	Function
CDS	Discrete Signals
Data Link	Reconstructed Analog Signals
	Discrete Signals
STC	120 vac Power

2.7.4 Physical Description. This remote control equipment should consist of two or more panels on which are mounted switches, indicator lights, and meters. The panels may be located in a separate rack or incorporated in a rack with the wide-band data-link equipment.

OSE designer, this decoder may be located within the respective consoles or racks to provide special-purpose discrete-actuated functions.

3.1.2.2.5 Switching Network. In the event of a failure, a switchover of STC computers or A2A lines may be accomplished in the DDRT and UDTT by any of the following: manual interconnections, toggle-switch-actuated switching relays, or automatically by computer control. The option depends upon the allowed downtime.

3.1.2.3 Uplink Data Transmission Terminal (UDTT). The UDTT would exchange control signals with the STC computer and would receive either 12- or 24-bit digital parallel command words. Use of the 12-bit option would allow the use of the Control Data 8528 Digital Communications Terminal (proven on the Apollo Program and available at MILA) for the entire UDTT and the majority of the UDRT. The UDTT converts the parallel word into a double-length serial word, the first half of which is identical to the second half. With the 12-bit option the serial word would be 24 bits long; with the 24-bit option, 48 bits long.

A line driver would be required to transmit the serial word on a balanced-center-conductor video cable to the A2A transmitter at a 1-volt peak-to-peak amplitude. A high bit rate is not required, particularly if uplink commands are infrequent. If further analysis indicates that a bit rate of no higher than 40 bps is required, then as an optional transmission link, an inexpensive voice frequency transmission link, similar to that used on the STC-LCC data link, may be used.

3.1.2.4 Uplink Data Reception Terminal (UDRT)

3.1.2.4.1 Receiving Terminal and Error Detector. If the 12-bit option were to be used, this unit would be identical to the CD8528 Data Terminal. Regardless of the options used regarding word length, bit rate, or type of transmission link, this unit would reconstitute the received word, compare the first half of the word bit by bit with the last half of the word to detect errors, generate a GO/NO-GO discrete, and send the reconverted 12-bit (or 24-bit) parallel word to the command message decoder.

3.1.2.4.2 Command Message Decoder (CMD). The CMD would examine each command received from the receiving terminal. If the message is recognized as a data link command, the CMD would decode the message into discretes of the proper levels for operating the data link. If the received message is recognized as an LCE command, it would be routed directly to the proper LCE without decoding. If the 12-bit option were to be used and a 24-bit command message were to be required, the CMD would be designed to recognize two sequential commands as the first half and last half respectively of a double-length command and would store the messages in the proper locations of a double-length register for transmission to the proper LCE.

3.1.2.5 Cable Transmission Systems

3.1.2.5.1 Western Electric A2A Video Transmission System. The A2A system is used at Cape Kennedy and MILA as a universal wide band cable transmission system for video and high-speed digital data transmission. It is used to transmit 1-volt nominal peak-to-peak signals in the frequency range of 30 Hz to 4.5 MHz. An A2A link consists of a transmitter modem, repeater modems, and a receiver modem, interconnected by 16-PEV-L balanced-center-conductor video cable. Each modem contains the proper amount of amplification, attenuation, and equalization to provide a 1-volt output at the receiver for a 1-volt input at the transmitter.

3.1.2.6 Self-Test Equipment. Each of the four wide band data link terminals (DDTT, DDRT, UDTT and UDRT), the A2A system, and the VFC system would have the capability of verifying its own proper operation without the aid of any other elements of the data link. Analog multiplexers, analog-to-digital converters and signal conditioners would require reference voltage supplies and vacuum tube voltmeters or oscilloscopes. Digital devices would require digital message generators and simulators. The A2A system is maintained in accordance with the Bell System practices.

3.1.3 Design Characteristics. As a general ground rule the wide band data link would be designed in such a manner as to eliminate as many constraints as possible upon the design of the OSE for the STC/LCE configuration as compared with the STC system test configuration. Analog voltages generated in the vicinity of the mobile launcher may be received at the STC in either analog or digital form with 3 sigma accuracies of ±1% or less if required. Discretes of any level in the vicinity of the mobile launcher will be delivered at any desired level at any point within the STC. Digital serial signals and telemetry carriers of less than 4.5 MHz may be transmitted with minimal amplitude or phase distortion. Digital parallel words of any length placed in an output register of any LCE in the vicinity of the mobile launcher may be obtained in the STC via the STC computer. LCE functions requiring command discretes or command messages in the vicinity of the mobile launcher may be obtained by providing for the generation of command messages by the STC computer.

3.1.4 Physical Characteristics

a. DDTT - Approximately 3 racks per Planetary Vehicle
b. DDRT - Approximately 2 racks per Planetary Vehicle
c. UDTT - Approximately 1 rack per Planetary Vehicle
d. UDRT - Approximately 2 racks per Planetary Vehicle

3.1.5 Interfaces. The wide band data link interfaces are given in Table 2-5.

TABLE 2-5. WIDE BAND DATA LINK INTERFACES

Equipment	Function	Characteristics
CDS	Data words to computer	24-bit parallel words
	Command words from computer	24-bit parallel words
Subsystem OSE	Data for display	Analogs and discretes
Telemetry OSE	Modulated Subcarrier	270 KC
LCE	Command words to Control, Power and C&S LCE	24-bit parallel words
	Data from spacecraft and LCE	Analogs and discretes
	Verification from LCE	24 bit parallel words

3.2 LCE - STC Data Link

3.2.1 Scope. This is a description of the data link hardware which would be used to transmit discretes from the VOYAGER STC to the VOYAGER consoles in the LCC firing rooms. This hardware is described in more detail in GE/Communication Products Department General Specification GEA-7610, "Type 18 Solid State FS Voice Frequency Carrier System."

3.0 DATA LINKS

3.1 Wide Band Data Link

3.1.1 Purpose. The VOYAGER wide band data link receives data from the umbilical junction boxes and other LCE in the vicinity of the mobile launcher, and transmits this data to the STC. The data link also receives command messages from the STC computer complex and transmits these commands to the pad-vicinity LCE. There is an independent downlink (pad-to-STC) and an independent uplink (STC-to-pad) for each of the two Planetary Vehicles and their associated LCE.

3.1.2 Functional Description. Reference should be made to Figure 2-5, Wide Band Data Link, in CII VC280FD100.

3.1.2.1 Downlink Data Transmission Terminal (DDTT)

3.1.2.1.1 Signal Conditioners. Several types of signal conditioners are required in the DDTT. They may be divided into two major types: those which condition digital or modulated subcarrier signals into the proper form for transmission via the A2A system, and those which condition analog signals into the proper form for the analog multiplexer. The output of the former type of signal conditioner should be a nominal 1-volt peak-to-peak signal on a balanced-conductor video cable. The latter type of conditioner should convert the full range of its input voltage to one of the three output ranges: 0 to ±1 volt, 0 to ±10 volts, or 0 to ±100 volts.

3.1.2.1.2 Analog Multiplexer. The analog multiplexer should be capable of sequentially scanning up to approximately 50 analog signals at the rate of approximately 5000 samples per second and converting the samples into a single time-division multiplexed (PAM) output signal. Each of the analog input signals and its ground should be isolated from all other analog inputs and their grounds. The multiplexer should be able to accept any of the three ranges of analog signal conditioner outputs. The analog multiplexer should also have provisions whereby any analog channel can be selected for continuous scanning.

3.1.2.1.3 Analog-to-Digital Converter. The analog-to-digital converter accepts the PAM output signal from the analog multiplexer and converts each analog sample into a parallel digital word of a fixed length. A word length of approximately 12 bits should be adequate for this application.

3.1.2.1.4 Discrete-to-Digital Encoder. The discrete-to-digital encoder would accept discretes from various LCE and encode the discretes into parallel digital words. A suitable method of encoding would be to assign an address to a portion of each digital word. Each subsystem could be assigned a certain address and all discretes pertaining to that subsystem could be contained in the remainder of that particular digital word. This method would simplify decoding at the Downlink Data Reception Terminal (DDRT).

3.1.2.1.5 Digital Multiplexer. The digital multiplexer accepts digital words from the analog-to-digital and discrete-to-digital converters and from the output registers of various

LCE. It generates frame sync and subframe sync words and multiplexes the various input words into a PCM-Telemetry type of digital serial output signal with a bit rate in the range of 50 kbs to 150 kbs, split phase. It should contain a provision whereby the output format may be changed either by manually operated switches or by receiving a parallel digital command word from the command message decoder in the Uplink Data Reception Terminal (UDRT). The digital multiplexer should also contain a master control and timing unit which provides the necessary control signals required to synchronize operations within the DDTT and LCE registers which interface with the DDTT.

3.1.2.1.6 Switching Network. The switching network provides a means whereby a standby A2A transmission link may be quickly selected in the event of a failure of one of the normal A2A links. The electrically operated switching relays should have the provision of being controlled either by manually operated toggle switches in the vicinity of the switching network or by command discretes from the command message decoder in the UDRT.

Should a future reliability analysis indicate a need for additional redundancy a similar switching network may be used to provide a means of switching the data from one Planetary Vehicle and its associated LCE over to the data link used for the other Planetary Vehicle.

3.1.2.2 Downlink Data Reception Terminal (DDRT)

3.1.2.2.1 Signal Conditioners. Three types of signal conditioners would be required. One type would accept a modulated subcarrier at a 1-volt peak-to-peak level and restore the signal to its original level. A second type would do the same for the 50-kbs type of digital serial signal. The third type would have to be a combination bit synchronizer and signal conditioner. This unit would obtain a phase-lock loop synchronization with the incoming bit rate, reconstitute the serial PCM data and reproduce it at the output at the proper logic levels, and generate synchronized bit rate timing pulses.

3.1.2.2.2 PCM Decommutator. The PCM decommutator performs the following functions: accepts serial PCM data and synchronized bit rate signals from the bit synchronizer; establishes synchronization with the frame rate, subframe rate and word rate of the PCM data; strips selected channels out of the data format for data conversion; and provides digital parallel words for the STC computer along with frame, subframe, and word sync signals. A patchboard-programmable PCM decommutator of simplified design would be adequate for this application.

3.1.2.2.3 Digital-to-Analog Converters. Each digital-to-analog converter would simply receive parallel digital words and word sync signals from the PCM decommutator and generate corresponding analog signals. In some cases, signal conditioners may be necessary to restore analog signals to their original values. One converter channel is required for each analog signal which is to be generated.

3.1.2.2.4 Digital-to-Discrete Decoders. The digital-to-discrete decoders would accept digital parallel words and word sync signals from the PCM decommutator. If the suggested encoding method were to be used, each decoder would have a single address and would generate discretes relating to only one subsystem. At the option of an individual subsystem

3.2.2 Hardware Description. Refer to Figure 2-6, Voice Frequency Carrier System, in CII VC280FD100.

3.2.2.1 Transmitters. Each transmitter would be keyed by operating a relay or switch in the STC test conductor's console. The transmitter oscillator would shift in frequency by 30 Hz, upon closing the relay or switch contacts. The transmitter frequencies would be spaced at 120-Hz intervals ranging from Channel 1 at 420 Hz to Channel 24 at 3180 Hz. Output band-pass filters would provide up to 40 db of isolation between adjacent channels. The maximum keying rate would be 30 changes per second for any given discrete. Each transmitter would contain a self-test option, whereby the transmitter could be quickly test-keyed. A line termination module would be used to match the unbalanced transmitters to a 600-ohm balanced line.

3.2.2.2 Telephone Line. The transmission medium would be a single twisted pair of 600-ohm, 19-gauge telephone line per 24 discrete channels. The lines would be routed through the various repeater building and telephone central offices completely isolated from other instrumentation or telephone lines.

3.2.2.3 Receivers. A line termination module would be required to match the balanced 600-ohm line to the bank of unbalanced receivers. Each receiver would be tuned to a specific transmitter by means of a band-pass filter which would provide up to 40 db of isolation between adjacent channels. The received signal would then be amplified, limited, and applied to a frequency discriminator, to determine the state of the received signal. Each of the two output states would be obtained in the form of the closure of a pair of relay contacts. These contact closures would then be used to actuate the proper relay lamp drivers in the status display consoles.

3.2.2.4 Power Supplies. A power supply module would be required for each set of 8-transmitter or receiver modules required.

3.2.3 Interfaces. All interfaces would be of the nature of electrically isolated relay or switch contacts. The closure of a pair of contacts in the STC would be represented by a closure of relay contacts in the LCC. The signal interfaces are with the STC test conductor's console and the LCC controls and displays.

3.2.4 Physical Characteristics. The VFC mounting frames will accept up to 10 single-width modules. The transmitter, receiver, and line-termination modules are all single-width. The power supply modules are double-width. Three mounting frames, containing 3 power supply modules, 1 line termination module, and 23 transmitter (or receiver) modules, would occupy a total space of 16-1/3 inches high by 12 inches deep in a standard 19-inch relay rack.

4.0 LAUNCH CONTROL CENTER CONTROL AND DISPLAY GROUP. This equipment group includes the monitor, display, and control functions provided in the LCC firing rooms at Launch Complex 39. It must be capable of the following: (1) displaying selected, launch critical parameters supplied by the STC, (2) control and monitoring of hardwire lines for Planetary Vehicle and LCE control purposes, (3) providing for exchange of countdown status information between the Planetary Vehicle, Launch Vehicle and Launch Complex, and (4) using facility supporting services and equipments such as operational intercom (OIC), closed circuit television (CCTV), weather data, standard cabinetry and power.

4.1 Equipment Description. The recommended implementation of LCC controls and displays is shown in the LCC equipment block diagram Figure 2-8, in CII VC280FD100.

4.1.1 Planetary Vehicle Status Panel. The Planetary Vehicle status panel receives summary status information from the Spacecraft and Capsule test conductors in the STC and displays this data by illuminating status advisories with appropriate names. Displayed information should be overall Planetary Vehicle Status, Fore Spacecraft Status, Fore Capsule Status, Aft Spacecraft Status, and Aft Capsule Status. To complete this utility of the panel as a top-level quick look status advisor, the Space Flight Operations Facility (SFOF), Deep Space Network (DSN) and Deep Space Information Facility (DSIF) status should be included.

4.1.2 Subsystem Status Panel. These displays are similar to, and supplement, the Planetary Vehicle status panels by providing more detailed status on those Planetary Vehicle subsystems most critical to the launch process. These are tentatively identified for each Flight Spacecraft as Power, Pyro, Thermal and Science.

4.1.3 Command and Control Panel. Manual operator controls and displays should be provided for the JPL Test Co-ordinator as follows: for control of hardlines between LCC and mobile launcher to assure positive control of critical LCE circuits when the data link is inoperative, for capability to advise the Launch Director of a launch-ready condition, and for a capability to inhibit STC test operations in an emergency situation.

4.1.4 Closed Circuit Television. A requirement would be placed on the Kennedy Space Center facilities to furnish at least two CCTV displays at the JPL Test Coordinator's console. One would view the Planetary Vehicle and the umbilical connections during pull-away, the other would provide a view of test activities at the STC. Selectable cameras at the STC will be focused on test conductors consoles, CRT displays, or teletype print-outs to furnish JPL personnel in the LCC with quantitative test data, plus a visual feel for STC test activities.

4.1.5 Operational Intercom. A standard LCC intercom panel (GFE) should be integrated into each control and display console. Approximately 6 channels are required; one to the pad, two to the STC, two to DSIF No. 71 and MOS, and one to Launch Control.

4.1.6 Time Displays. Standard numeric displays (government furnished equipment), of range time, local time, and countdown time, would be integrated with the JPL Test Coordinators panel.

4.1.7 Alphanumeric CRT Display. An alphanumeric CRT display would be an optional capability in the LCC. The CRT and selection controls subassembly would be identical to that in the STC. Data, as monitored in the STC, would be directly repeated in the LCC. Electronics, ancilliary to the displays, would contain signal conditioning, storage, memory, switching logic, and power sources. Two A2A wide-band video links would be required for transfer of alphanumeric data from STC to LCC.

4.2 Physical Characteristics. VOYAGER control surfaces for use in the LCC must be compatible with Kennedy Space Center Launch Complex 39 physical standards. The Space Center should supply all racks, consoles, cabling, and signal distribution with VOYAGER equipment being designed to fit these pieces of equipment. The display equipment will require between six and twelve display consoles and six upright cabinets.

5.0 FLUID SERVICE EQUIPMENT

5.1 Propellant and Gas Service Eauipment

5.1.1 Scope. This section provides a functional description of the equipment which will satisfy the propellant and pressurant servicing requirements for the spacecraft subsystems. This OSE forms a part of the Launch Complex Equipment, and its use is confined to the Explosive Safe Facility, with the exception of a pressurant supply cart. The pressurant supply cart may also be used for gas control in the flow testing of propulsion subsystem components.

5.1.2 Functional Description

5.1.2.1 Loading Unit. Three subsystems of VOYAGER would be serviced with mission loads of liquid propellants and/or gases by this equipment: the retropropulsion, midcourse and attitude control.

Load transfers of both gases and liquid propellants would be performed through operator control of three physically separate items: a transfer assembly, a control console, and a pressurant supply cart. These three items are linked together for a loading operation by electrical and gas interface connections to form a composite loading unit, as shown in Figure 5-1. One Loading Unit will provide all of the fluid servicing needs of a VOYAGER Flight Spacecraft. The loading unit will enable transferring of liquid propelland and gases from a transfer assembly and pressurant supply cart under operator control of a control console in the vicinity of the Spacecraft. Fluid lines and electrical cables are required to enable a loading unit to transfer the liquid propellant and gas to the appropriate spacecraft tanks. Operational capability of a loading unit is:

a. Storage of sufficient hydrazine, Freon 114B2, and hydraulic oil in self-contained tanks to load one Spacecraft; and transfer of accurate mass quantities of these liquids to the spacecraft tanks.

b. Transfer of pressurant gases to subsystem tanks at controllable pressure and temperature levels.

c. Sampling of the transferred liquids and pressurant gases, for load-assurance analysis

d. Unloading a loaded Spacecraft, and the application of conditioned gas and vacuum to the spacecraft propulsion subsystem by transfer lines for the purposes of purging and drying.

e. Monitoring and control of the equipment in an organized manner such that loading can be directed by one man from a single station.

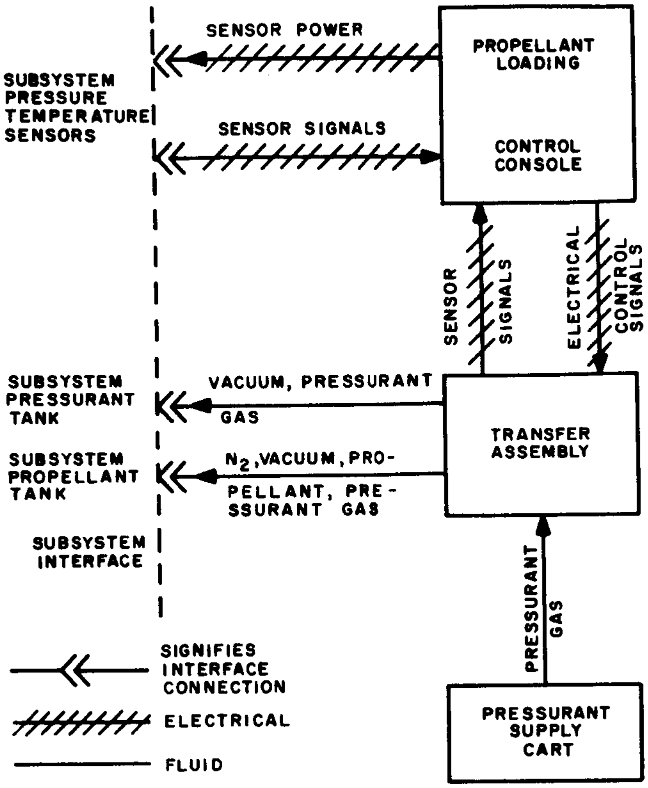


Figure 5-1. Functional Interfaces Between the Three Separable Items of a Propellant Loading Unit and a Propulsion Subsystem

5.1.2.2 Transfers (Refer to Figures 5-2 and 5-3.)

5.1.2.2.1 Propellant Transfer. A high degree of accuracy and control is needed in metering the propellant loaded into the midcourse subsystem tanks, in order to minimize the imbalance in loads among the several tanks of this subsystem. Confidence of high load accuracy also permits the specified loads to closely approach the ideal mission requirements, and may eliminate the need for post-load tests to measure spacecraft flight weight and center-of-gravity. The desired loading accuracy can be obtained by weight measurement of the midcourse propellant and retropropulsion TVC injectant with the use of a variable inductance load cell, its load response readout and recorder, and standardized dead weights for load response calibration. The overall inherent error in measuring weight with this combination is approximately 0.07% of full scale response of the load cell.

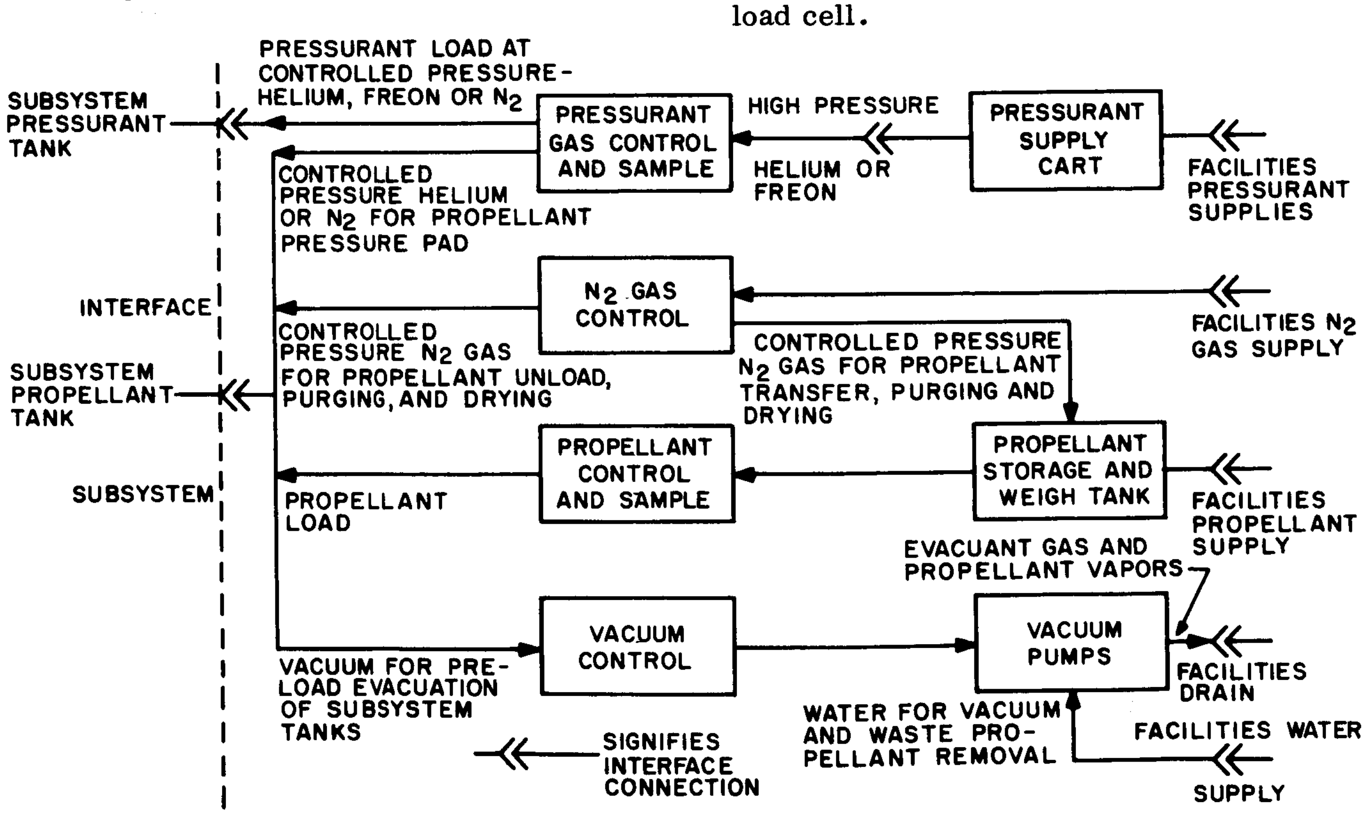


Figure 5-2. Transfer Assembly Functional Diagram

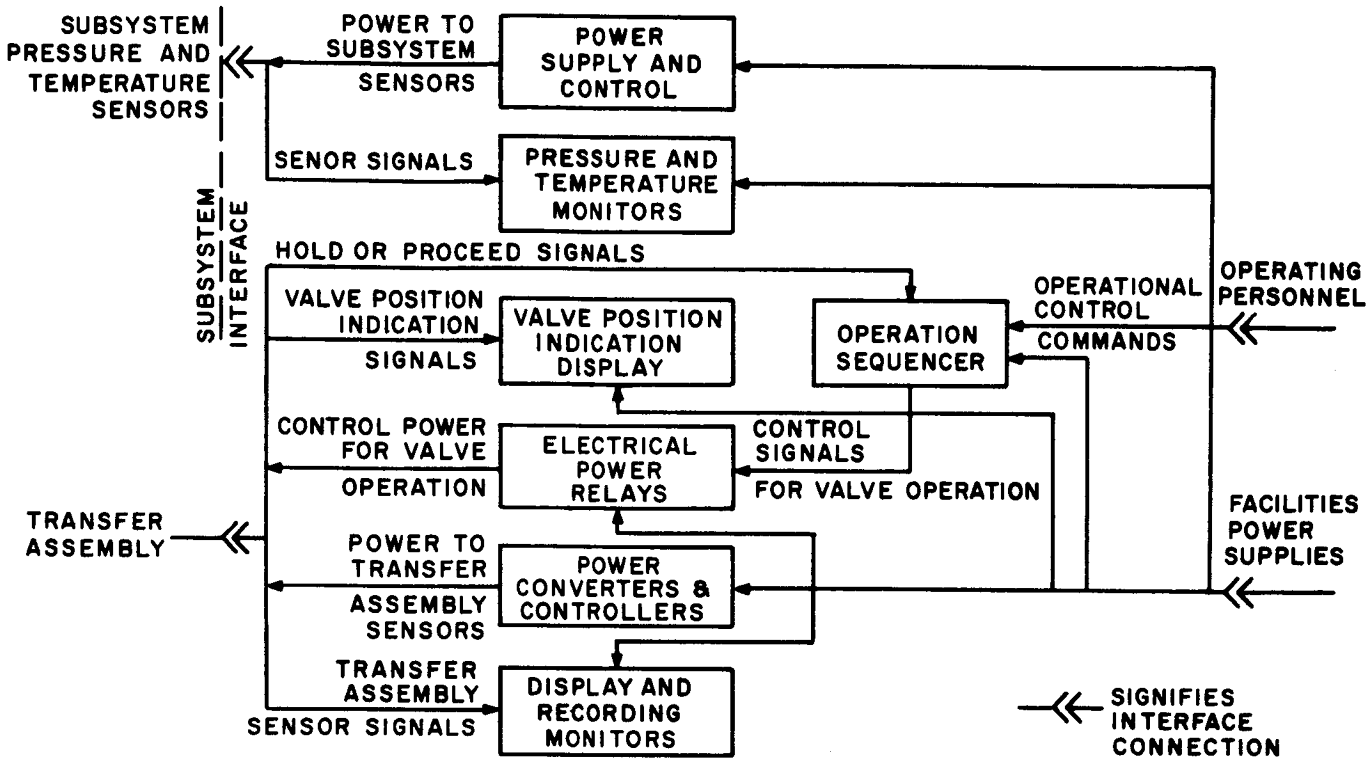


Figure 5-3. Control Console Functional Diagram

A typical propellant transfer would be performed with the hydrazine transfer tank of the transfer assembly suspended from the load cell. After first filling the transfer tank and transfer line with hydrazine at ambient temperature, the tank would be pressurized with N_2 gas as an expulsion medium. The gas supply is then shut off to prevent weight changes except for hydrazine leaving the tank. The load cell output would be adjusted to zero and the required subsystem load transferred, as measured by load cell response. Verification of the quantity transferred would be obtained immediately following the transfer by attaching to the transfer tank dead weights totalling the weight of the hydrazine transferred, and recording the load cell's linear return to zero output as the weights are added. Performance parameters of liquid loading with this equipment are summarized in Table 5-1.

TABLE 5-1. LIQUID LOAD PARAMETERS

Applicable Subsystem	Load Liquid	Maximum Subsystem Load Required (lb)	OSE Load Range Capability (lb)	Allowable Load Tolerance (±%)	Load Verification Sample (lb)	Post-Load Pressurant	Subsystem Post-Load Pressure (PSIA)	OSE Pressurizing Range for Post-Load Pressurant (PSIA)	OSE Post-Load Pressurizing Error (± PSI)	OSE Vacuum Pumping Speed (Ft^3 Displacement)	OSE Vacuum Shutoff Pressure (TORR)	Total Time Required to Load System (hr)
Retro Propulsion (for TVC)	Freon 114 B2	240	To 300	0.5	Approx. 0.4	No Requirement	NA	NA	NA	25	1.0	3
Retro Propulsion (for TVC)	Hydraulic Oil	Less Than 1 lb	To 5 lb	0.5	Approx. 0.4	No Requirement	NA	NA	NA	NA	NA	1.5
Midcourse Propulsion	Hydrazine	1788	To 2000	0.5	0.4	Nitrogen	150	To 300	2	25	1.0	10

5.1.2.2.2 Pressurant Transfer. Gases to be loaded into subsystem tanks must be piped from facilities supplies to the pressurant supply cart for pressure intensifying and temperature conditioning, then to the transfer assembly for pressure regulation and flow control, and then to the subsystem interface. The temperature and pressure of this gas must be continuously monitored during loading, using the output of the subsystem sensors to drive OSE monitoring instruments. This combination will allow gas loads to be made with total mass errors not exceeding 1%. Pressure will also be monitored using OSE sensors, and the OSE will remain attached to the subsystem after loading for relief venting capability until the loaded gas temperature has stabilized to ambient. Performance parameters of this equipment in loading gases are summarized in Table 5-2.

TABLE 5-2. PRESSURANT LOADING PARAMETERS

Applicable Subsystem	Load Pressurant	Subsystem Pressurant Load Pressure (PSIA at 70°F)	OSE Pressurant Load Pressure Range Capability (PSIA)	OSE Pressurant Loading Pressure Error (± PSI)	Verification Sample (Bottle Volume - Ft^3)	OSE Preload Vacuum Shutoff Pressure (TORR)	OSE Temperature Control of Pressurant Being Loaded (Range in °F)	Accuracy of Subsystem Tank Temperature Sensor (±°F)	Total Pressurant Loading Time per Subsystem (hr)
Retro Propulsion (TVC)	Nitrogen	3500	To 4500	10	0.01	1.0	30 to 90	2.5	4
Midcourse Propulsion	Helium	3600	To 4500	10	0.01	1.0	30 to 90	2.5	7.5
Guidance and Control (Attitude Control)	Nitrogen	3250	To 4500	10	0.01	1.0	30 to 90	2.5	8.5

5.1.2.2.3 OSE Vacuum. The vacuum system of a transfer assembly would consist of a water ejector, backing up two stages of rotary dry mechanical pumps, all of materials compatible with the mission propellants. This vacuum system has been operationally proven for use in equipment similar to the transfer assembly described here.

Vacuum is important for purging, creating differential pressures to cause liquid flow, and for drying after a liquid unloading operation. Its primary use, however, is for exhausting the subsystem tanks prior to liquid loading. This makes loading possible without the need for tank vents, avoids high differential pressures on tank expulsion bladders and reduces trapped gas to a minimum in the liquid section of the subsystem tank. The vacuum system described can reduce subsystem tank pressure to one Torr or lower, resulting in a trapped gas bubble of no more than 0.02% of the liquid volume after post-load pressurization. (This is approximately equal to a gas bubble volume of 3 cubic inches trapped in 500 pounds of hydrazine.)

5.1.2.2.4 Liquid and Pressurant Loading Times. Liquid loading times are summarized in Table 5-1 and, for liquid loads, includes consideration of time required for OSE fluid and electrical line connections, OSE preload weighing calibration, vacuum evacuation of the subsystem tanks, transferring the liquid, applying gas pressure to the loaded liquid, OSE post-load weighing calibration, sealing of subsystem lines, and disconnecting the OSE. For pressurant gases, the loading times summarized in Table 5-2 include times required for interface connections, vacuum evacuation of subsystem, pressurant transfer, transferred pressurant temperature stabilization, sealing of the subsystem, and disconnecting the OSE.

5.1.3 Physical Characteristics

5.1.3.1 Transfer Assembly. The transfer assembly is a closed trailer of approximately 3500 pounds empty weight (6000 pounds weight when containing sufficient hydrazine and Freon 114B2 to load midcourse propellant and retropropulsion secondary injectant.) Dimensions could be confined to an envelope 6 feet wide, 7 feet high and 12 feet long. Steerable-wheels, brakes, and jacks allow towing, locating, and level-positioning of the assembly. Both sides open to allow interior access for operational hook-up and maintenance. Fork-lift slots in the base provide for convenience in loading the assembly aboard a truck for transportation.

5.1.3.2 Control Console. The control console weight is 1200 pounds complete and can be confined to a cabinet 3-1/2 feet wide, 6-1/2 feet high and 8 feet long. Casters and fork-lift base slots provide for easy handling and positioning. Service access to three sides would be made possible by removable panels. All operational controls, indicators, and monitoring instruments for propellant and pressurant transfers would be mounted on the panels of one side of the cabinet, located and displayed logically and conveniently for the operator's use.

5.1.3.3 Pressurant Supply Cart (Refer to Figure 5-4.) The operating components would be housed within a 4 feet by 6 feet by 5 feet high closed van trailer, weighing approximately 3000 pounds complete. Flex hoses for connecting to facilities helium and N_2 supplies would be included in this OSE. Either gas would be valved to a pressure intensifier pump within the cart. The pump output pressure would be controllable up to 4500 psi. This pressurized gas would then be temperature adjusted in a coiled tube heat exchanger, after which, it would be filtered, dried, and finally accumulated for operational use in a 1/2-cubic-foot spherical storage tank. A shutoff outlet valve on the tank would be provided with a connector and a transfer line to pipe the gas to the transfer assembly. Since the functional testing OSE for

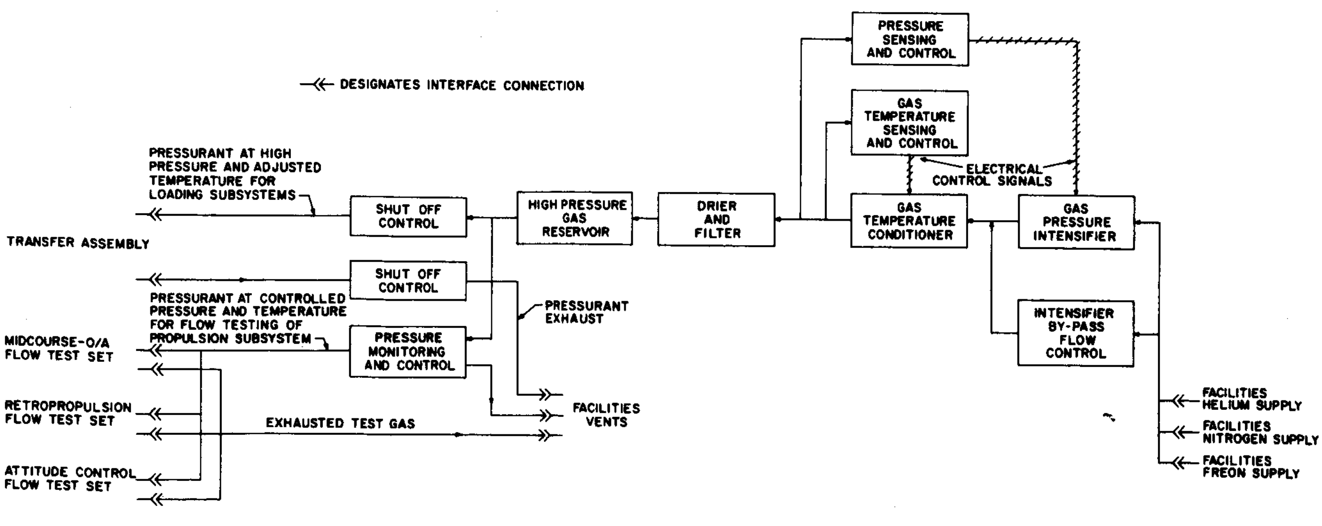


Figure 5-4. Pressurant Supply Cart Pneumatics Functional Diagram

the propulsion subsystems also requires similar conditioned gas supplies, the pressurant supply cart would be given the capability of serving that OSE, as shown in Figure 5-4.

5.1.4 Safety Provisions

5.1.4.1 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. The relief venting valve is adjustable to any desired relief pressure but is nominally set for operation at 110% of normal working pressure of the subsystem tank or section. 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 a transfer assembly 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 cooler pressurant adjusts to ambient temperature.

All pressurized components of the OSE would be designed with a safety factor of at least four, between maximum working pressure and design burst pressure. The OSE would be proof tested to 2.5 times the maximum working pressure.

All mechanical pressure gauges would 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 would 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.

5.1.4.2 Toxicity Safety for Personnel. Operators will require the following facilities equipment during the hydrazine loading operation:

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 and remove corrosive media in case of propellant spill.

5.1.5 Sterilization. If sterilization of pressurized fluid subsystems becomes a requirement, changes would be necessary in the present loading methods and OSE concepts. The following summary outlines the possible minimum effect that a sterilization requirement would have on the loading OSE. The following minimum preparation of a subsystem would be necessary in order that the empty subsystem be in a sterilized condition prior to load operations:

a. Leak test the subsystem.

b. Evacuate all subsystem tanks and isolate the evacuated tank volumes with subsystem valves.

c. Sterilize the subsystem interior with a heat cycle.
d. Assemble the subsystem to the Spacecraft.

With the preparation complete, the subsystem could be loaded at any convenient time. For providing a sterilized load, the presently proposed OSE could be used, except that it would load prescribed quantities of propellant or pressurant gas into special OSE tanks or capsules instead of directly into the subsystem. These capsules would then be separately sealed and exposed to a sterilization heat cycle, and stored until required for loading a subsystem. At the time of subsystem loading, a capsule would be connected to the subsystem fill line. The fill line space between the subsystem shutoff valve and the OSE capsule shutoff valve would then be sterilized by a heat cycle. The heat cycle would be effected by using OSE induction or resistance elements to heat the OSE and subsystem shutoff valves, and the connecting line between the valves. These shutoff valves would then be opened to allow fluid to be transferred to the evacuated subsystem from the capsule, the shutoff valves closed again, and the OSE removed.

Exact pressurant load would be obtained by accurate sizing of capsule volumes and precision control of the pressure and temperature of the pressurant transferred to the capsule. The exact liquid load as preloaded into a capsule would be transferred to the subsystem primarily by using gravity to produce expulsion pressure. Time needed for the actual transfer of propellant or gas to the subsystem may not be increased seriously, if at all, from the loading times without the sterilization requirement, because of the amount of preparation (i.e., capsule preloading) that could be accomplished without the presence of the Spacecraft. Loading time would be primarily dictated by the time length of the heat sterilization cycle of the connecting section of fill line between the OSE and the subsystem. It appears that the temperature of this section could be raised to 500°F or higher for sterilizing, and the time requirement thereby made acceptably low. Subsystem requirements to support sterilization of propellantswould be:

a. The empty subsystem must be capable of sterilization by heat without significant degradation.

b. No subsystem external leaks of a size sufficient that micro-organisms might enter the (vacuum evacuated) subsystem by this route, would be tolerated.

c. The subsystem interior could not be opened to any external exposure after sterilization except under the special conditions of loading, where the connecting volume between subsystem and load capsule has also been heat sterilized.

5.2 Leak Test Equipment

5.2.1 Scope. This section is the functional description of test equipment for leak testing the retropropulsion, midcourse propulsion, and attitude control subsystem assemblies.

5.2.2 Functional Description

5.2.2.1 General Considerations. Prior to propellant and gas loading of the Spacecraft, it is important to reassess the leak profile characteristics and pressure safety of applicable spacecraft subsystems. This OSE may be used to apply controlled gases to accessible subsystem test points in order to establish proof pressures and differential pressures between subsystem sections, and between the subsystems and the ambient atmosphere. The differential pressures cause gas to flow through leak paths. This gas flow is used to detect and analyze the leaks by measurements of pressure drop, pressure rise, and tracer gas concentration.

The spacecraft subsystems should be comprehensively proof-pressure and leak tested at least three times, the final test to be performed in the Explosive Safe Facility. All of the leak testing should be performed without the use of liquids, because gases allow easier, faster, and more accurate analysis of small leaks, and because liquids can plug small leaks so that they might not be detectable. Records from this series of tests would be compared to obtain a knowledge of the leak characteristics of the subsystems, 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 would be used to "sniff" for traces of propellant issuing from external subsystem leaks.

5.2.2.2 Equipment Requirements. The leak test equipment should have the capability to perform these functions:

Leak testing of the MC/OA, Retropropulsion and Attitude Control subassemblies; proof pressure testing of spacecraft subsystem liquid propellant and gas tanks by internal gas pressurization, to satisfy that these tanks will not rupture under normal working pressures; testing of in-line valves, regulators, diaphragms and bladders for through-leaks; testing for external leaks over all of the pressurized portions of the subsystems before propellant and pressurant loading. The OSE should also have limited monitoring capability for sensing external propellant leaks during storage of the spacecraft with propellants and pressurants loaded. There should be no need for interface with other OSE in the performance of leak testing. An accessory feature is desirable which would permit a continuance of pressure-drop leak testing of subsystem tanks while the subsystem or the assembled spacecraft is being transported or is involved in other testing.

5.2.2.3 Procedural Method of Testing

5.2.2.3.1 Functional Diagram. The leak test OSE and its use are illustrated in Figure 5-5. The OSE functions needed for the described methods of testing are labeled in the figure. Also shown are the test lines of the OSE that connect through portable pressure drop sensing elements to the individual test access points of a subsystem. OSE temperature sensors are applied to the exterior surfaces of subsystem tanks. Plastic wrap or tenting may be used to confine the exterior atmosphere of the Spacecraft, a subsystem or parts of a subsystem, and OSE. N_2 gas is used to purge the tent atmosphere of contaminant gases.

5.2.2.3.2 Proof Pressure Testing. After adjusting and testing OSE overpressure relief devices to the desired relief levels for a test, conditioned N_2 gas would be applied through flexible lines to each test point connection of the subsystem. The pressures in these lines

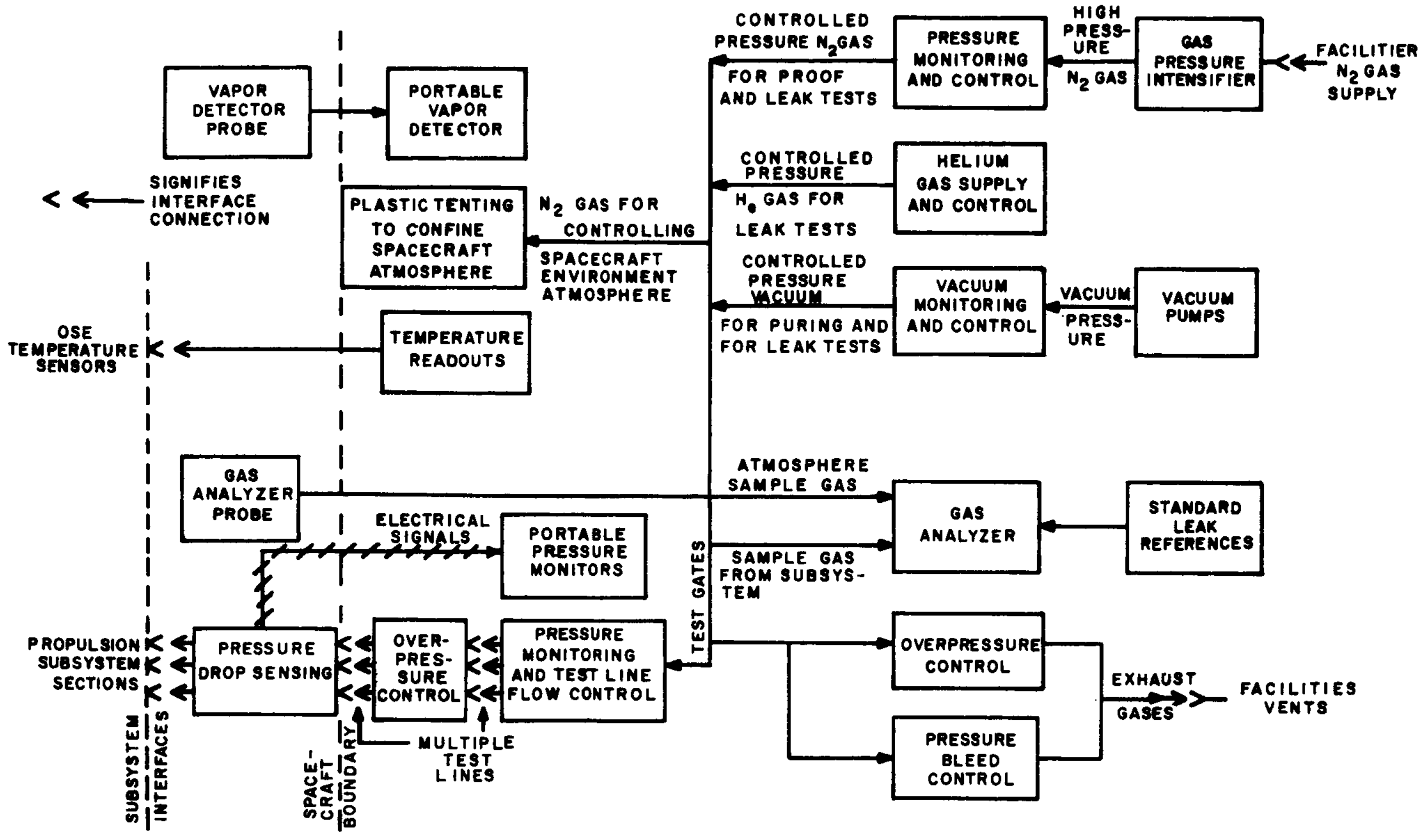


Figure 5-5. Leak Test OSE Functional Program

would be separately preset and automatically regulated to the desired pressure levels. After establishing and maintaining proof pressures for the specified test time period, the OSE pressure bleed control would be used to vent off the test gas in the subsystem.

5.2.2.3.3 Through-Leak Testing. This test capability of the OSE would allow a systematic analysis of subsystem components for through-leaks, and also would serve as a diagnostic aid in identifying the nature of a known leak. Testing would be performed, on one subsystem at a time, by first charging alternate sections with helium enriched N_2 gas to normal working pressures and then isolating these sections by OSE shutoff valves. The remaining sections would next be evacuated with the OSE vacuum pump, and these sections also isolated by OSE shutoff valves. One at a time, the evacuated sections would then be reopened to allow the OSE gas analyzer to detect and measure the concentration of helium tracer leaking from the adjoining pressurized sections. After all evacuated sections have been analyzed, the subsystem sections would be purged free of helium and the pressures reversed for through-leak testing in the opposite directions. Leak rates too high to measure by the gas analyzer may be calculated after measuring the rate of rise of pressure in an evacuated section with the use of an OSE timer and vacuum pressure gauge.

5.2.2.3.4 External Leak Testing. Small external leaks of a tested subsystem of this OSE may be detected and located using gas analyzer probes held against or near the outside surfaces of subsystem tanks and lines, after the subsystem is internal pressurized to normal working pressure with helium enriched nitrogen. The rate of leaks located in this manner may be determined by comparing the gas analyzer response to the leak with the response to a known (calibrated) leak. Calibrated leaks are part of the OSE, as indicated in Figure 5-5. Gas analysis of the air or N_2 gas, within the OSE tent enclosing a subsystem, can also be used as a means of determining overall subsystem external leakage. External leakage from low-volume sections of a subsystem may also be determined from internal pressure decay measured with monitors of the OSE, after pressurizing and isolating the leaking section. However, similar tests for small leaks from large volumes, such as subsystem tankage, require long pressure monitoring periods. To permit an efficient use of time, this proposed OSE includes accessory pressure sensing elements, which may be detached from the primary test equipment but remain connected to the subsystem. This releases both the primary test equipment and the subsystem for other tests or transportation, while pressure drop in the pressurized subsystem can be checked from time to time with hand carried monitors. The sensor elements would be removed from the subsystems when the required monitoring period has elapsed.

5.2.3 Performance Parameters. Performance of the leak test OSE is summarized in Table 5-3. It is not expected that either N_2 or He gas at 4500 psi will be available at the LCE test area, and this OSE is therefore equipped with a pressure intensifier. Overpressure relief adjustment accuracy, monitoring accuracy, pressure control accuracy, and the ranges of operation are based on available components and operationally proven methods and equipment, with certain exceptions:

a. The length of pressure-drop leak testing of tanks will depend upon the accuracy of measurement of the pressurant temperature, or else an extremely temperature stable environmental atmosphere. The likelihood of accomplishment of either of these is presently unknown, so the time for this kind of testing is also relatively uncertain.

b. When the pressure monitoring segments of OSE are eventually removed from the test points of a subsystem, the leak characteristics of the subsequent seal of the test points cannot be ascertained by pressure-drop test methods.

c. Any requirement of sterilization of the applicable subsystems may radically change the OSE and its performance as presented here.

Test times indicated in Table 5-3, for pressure drop tests of tankage, are subject to environmental temperature control or accuracy of measurement of tank temperature. It is assumed that temperature control or tank temperature measurement would be no better than ±1 °F over the test period. A Δt of 1 °F at 530 °R (room temperature) will affect tank pressure by approximately 0.2%. To detect a leak by pressure decay in a tank, about 0.4% of tank pressure would have to decay by leakage in order that the decay would be attributed to leak instead of temperature variation. If the maximum allowable leak were 10% loss of tank pressure per year, the test time would be 0.4/10 year, or about 15 days.

5.2.4 Physical Characteristics. All of the items of this OSE can be confined within a cabinet of about 1200 pounds total weight with dimensions of 4 feet wide, 6 feet foot high, and 8 feet long. Caster mounting the cabinet would allow easy hand towing and positioning, and base slots would make the OSE convenient to handle by fork truck.

TABLE 5-3. LEAK TESTING OSE PARAMETERS

Applicable Subsystem	Nature of Test	OSE Test Medium	Test Pressure Capability (PSI)	Test Pressure Control (%)	Overpressure Relief Set Accuracy	Accuracy Pressure Monitors (% Test Point)	Vacuum Pump Shutoff Pressure (TORR)	Pressure Drop Monitoring Sensitivity	Vacuum Pressure Rise Monitoring Sensitivity	Tracer Gas Concentration Detection Sensitivity (Parts per Million)	Tracer or Contaminant Detected	Approximate Time to Conduct Leak Tests (Except Tank Pressure Drop) (Hr.)	Approximate Time for Tank Pressure Drop Tests (Days)
Midcourse propulsion												10	
Retro propulsion	Proof pressure	Nitrogen gas	100 to 4500	Within 1% of test point	±2% minimum accuracy	+0.5 minimum accuracy		0.1% of test point minimum sensitivity				6	
Attitude control												8	
Midcourse propulsion												20	20
Retro-propulsion	Through leaks	N_2 gas N_2 + He gas	100 to 4500	Within 1% of test point	±2% minimum accuracy	+0.5 minimum accuracy	0.01	0.01	0.2 Torr	20 ppm minimum sensitivity	Helium	6	
Attitude control												12	20
Midcourse propulsion												20	20
Retro-propulsion	External leaks	N_2 gas N_2 + He gas	100 to 4500	Within 1% of test point	±2% minimum accuracy	+0.5 minimum accuracy		0.01		20 ppm minimum sensitivity	Helium or propellant vapors	6	
Attitude control												12	20

Includes making OSE subsystem connections, but no spacecraft transportation or other subsystem readiness time requirements are considered.

Three sides of the cabinet should be quickly detachable panels or doors for interior access for storage, maintenance, and hookup, but one of the 6- by 8-foot sides would be reserved as a console face for mounting operational controls and monitoring instruments. All loose equipment, such as gas analyzer probes, electrical cabling, flexible hose test lines, portable propellant vapor detector, and separable pressure drop monitoring sections, would have storage provisions within the cabinet or in an accessory chest.

5.2.5 Safety Considerations. The proof-pressure testing with gas incurrs 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 inadvertent 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 Spacecraft. After proof-pressure test, subsystem tankage is considered safe for personnel to work around when pressurized to normal working pressures. However, facilities propellant loading, and the air toxicity level must be determined from facilities indicators before approaching the loaded Spacecraft for leak tests at any later time.

6.0 PYROTECHNIC SUBSYSTEM SAFE/TEST KIT

6.1 Purpose. The pyrotechnic subsystem safe/test kit (see Figure 6-1) is a self-contained, portable, battery powered unit intended to be used at the explosive safe area (ESA) to verify the safe condition of the pyrotechnic controller and all squib firing lines before and during squib installation.

6.2 Functional Description

6.2.1 Self-Test. Prior to or during any test operations, the pyrotechnic test kit will be capable of performing self tests to ensure that the measurement and alarm units are functioning properly. This self-test involves simulation of malfunctions and observing the appropriate displays and alarms.

6.2.2 Display and Alarm. Each capacitor bank will be monitored to ensure there are no voltages present. If a voltage appears on any of the capacitor banks, a voltage level detector will trigger an alarm to warn that this dangerous condition exists. In addition, the safe-arm umbilical test points will be monitored continuously. A malfunction in the arm device will cause the OSE monitor to trigger an alarm.

The alarms will provide both visual and audio warnings. The audio alarm will immediately call the attention of the operator to the existance of a malfunction. The visual alarms will provide the operator with the exact nature of the malfunction.

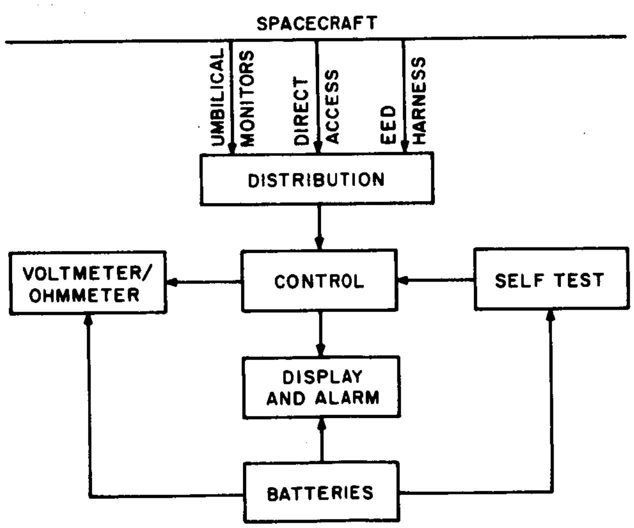


Figure 6-1. Pyrotechnics Test Kit-LCE Functional Diagram ESA

6.2.3 Voltmeter/Ohmmeter. The voltmeter/ohmmeter will provide the capability to measure the voltage present on each of the firing lines before they are connected to the electro-explosive devices (EED). In addition, this instrument will provide the capability to measure the umbilical continuity loops after all EED's have been installed.

6.3 Functional Interfaces. The pyrotechnics subsystem safe/test kit interfaces are given in Table 6-1.

TABLE 6-1. PYROTECHNIC SUBSYSTEM SAFE/TEST KIT INTERFACE

Interface	Function	I/O	Remarks
Spacecraft	Safe/Arm Monitors	Input	3 required
	Squib Line Stray Voltage Monitor	Input	
	Capacitor Bank Voltage Monitors	Input	4 required
	Continuity Loop Monitor	Input	

7.0 RF LINK EQUIPMENT

7.1 Purpose. The RF link equipment includes all of the RF antennas, fixtures, and connections required to support the Planetary System in communicating with DSIF 71 or the STC by the normal spacecraft operating frequencies.

7.2 Interfaces

7.2.1 Facilities

a. Umbilical Tower - Installation of two S-band antennas and two UHF antennas directed toward hanger AO, mounting of two power dividers.

b. Mobile Service Tower - Installation of two S-band antennas, two power dividers, and provision for two antenna couplers for attachment to the shroud.

c. Hangar AO - Installation of two S-band antennas and two UHF antennas on the roof with connections to each STC.

7.2.2 Spacecraft Interface

a. Umbilical - Four S-band signals connected to each power divider. One UHF signal connected to UHF antenna.

b. Shroud - One antenna coupler over each shroud antenna.

c. DSIF 71 - Line of sight S-band transmissions from shroud antenna, umbilical tower S-band antennas, and service tower S-band antennas.

7.3 Performance Parameters. Frequency, S-band, 2000-2300 mc. Frequency, UHF, 400 mc.

7.4 Physical Characteristics. The equipment will consist of: eight S-band antennas, six UHF antennas, four power dividers, connectors, and fittings.

CII VC270SR102

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT DESIGN CHARACTERISTICS AND RESTRAINTS

SECTION

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT DESIGN CHARACTERISTICS AND RESTRAINTS

The Assembly, Handling, and Shipping Equipment (AHSE) will provide the capability for lifting, holding, positioning, transporting, and aligning the Spacecraft and its subsystems during the Spacecraft and Planetary Vehicle ground mission. The ground mission of the Spacecraft will include the general assembly of the vehicle, support of subsystem and system testing, shipment to the field, and the field cycle up to and including mating of the Planetary Vehicle to nose fairing sections at the Explosives Safe Facility (ESF).

Handling, transport, and mate of the nose fairing sections are not presently considered to be within the scope of AHSE. Development testing involving the Structural Test Model (STM), Thermal Control Model (TCM) and the Development Test Model (DTM) Flight Acceptance (FA) testing, and testing of the Proof Test Model (PTM) will be supported by the AHSE.

1.0 <u>DESIGN CHARACTERISTICS</u>. The AHSE shall be designed to adequately satisfy ground mission profile requirements and reflect the following system level characteristics:

1.1 <u>Safety</u>. Design of the AHSE will be based on multiple-use concepts to minimize the number of vehicle moves required during test cycles and, therefore, to minimize opportunity for failure. In addition, all structures will be stress-analyzed for compliance with the following ratio; the more conservative ratio will determine the design.

$$\frac{\text{yield stress}}{\text{limit load stress}} \geq 3 \text{ or } \frac{\text{ultimate stress}}{\text{limit load stress}} \geq 4.5$$

In all cases, limit load will include the weight of the item itself plus any fixturing involved. Each item of equipment will also be proof tested to two times limit load to ensure structural adequacy. During development testing, operating procedures will be checked out using simulated vehicle sections to ensure conservative operation for both vehicle and personnel.

MIL-S-38130 will be used as a guide for monitoring safety in AHSE designs. AHSE interfaces which contact the Spacecraft during electrical testing will be so designed that no electrical continuity will exist between the AHSE and the supported Spacecraft.

1.2 <u>Human Engineering</u>. AHSE design will consider the human factor aspects of accessibility and personnel hazards. MIL-STD-803A will provide the guideline for designs.

1.3 <u>Materials</u>. Selection of materials for AHSE construction will be governed by application requirements; however, wherever practical, off-the-shelf raw materials and components will be specified. In addition, nonmagnetic materials will be emphasized to ensure compliance with AHSE magnetic constraints. Nondusting and nonshedding materials will be used for AHSE, and protective coatings will be rust preventative and nonchipping to promote compliance with cleanliness requirements.

Additional considerations for material selection will include control of outgassing in vacuum environments, hydrocarbons in hypergolic servicing areas, and conductance in plastic covers.

1.4 Transportability. Shipment of the Flight Vehicles to the field will be based upon minimum disassembly for shipment to avoid possible degrading effects on the reliability and integrity of the Spacecraft System as verified by test at the factory. Constraints resulting from compliance with the above are as follows.

a. Flight Spacecraft shipping configuration over-all dimensions shall be kept to a minimum to avoid further complicating the movement of the Flight Spacecraft because of size restrictions on highways and water vessels.

b. Natural and induced transportation environments shall be mitigated to levels which will not functionally degrade the Spacecraft.

c. Explosives and other critical articles to be shipped shall be properly classified, marked, and packaged in accordance with prescribed ICC and Bureau of Explosives regulations.

d. Provisions for mechanized handling shall be provided on all containers over 100 pounds.

e. Shipping equipment shall be integrated with other handling equipment to assure compatibility and full utilization of handling equipment to install, transport, and/or remove items from their shipping containers.

f. Over-all dimensions of containers (other than Flight Spacecraft container) shall be commensurate with size limitations imposed by carrier contemplated for shipment (truck, aircraft, rail car, or water vessel).

1.5 Cleanliness. The following detail design considerations will be included in all AHSE designs to promote ease of cleaning: Pockets in structural members and fittings will be avoided, inaccessible recesses will be avoided, and all tubular members will be capped on both ends; internal radii will be 1/2 inch minimum; external radii will be 1/4 inch minimum. Flat head screws and rivets are preferred, lap joints will be welded or filled, flaking material must be covered or coated, mechanisms should be enclosed in covers, and room must be left between harness and piping runs for vacuuming. In general, all designs provide for practical cleaning access and dirt pocket elimination.

2.0 EQUIPMENT LIFE. End items of equipment and their components will be designed and selected to provide an operational life of ten years when serviced and maintained in accordance with documented recommendations. MIL-M-26512C will be used as a guide to ensure adequate maintainability of the AHSE.

3.0 DESIGN CONSTRAINTS. Equipment will be designed with safety and reliability as prime criteria. Other constraining criteria which will control designs are as follows.

3.1 Environments. Environments to which individual items of AHSE will be subjected as well as the external and internal environmental limitations for the Planetary Vehicle during its ground mission will become specific design requirements for each item. Cleanliness considerations for the 100,000 class environment and 100-micron particle limitation will be considered. Standard environmental considerations, such as humidity, temperature, vibration, shock, contaminants, magnetics, and pressure will be documented in AHSE system level environmental specifications.

3.2 EMI Compatibility. All electrical components included in the AHSE will comply with electromagnetic interference control requirements to ensure electromagnetic compatibility with spacecraft functions and electrical OSE operations.

3.3 Magnetic Compatibility. The use of magnetic materials for vehicle interfacing will be prohibited, and applications of magnetic materials in AHSE designs will be controlled so that at no time will the AHSE used with or near the Spacecraft exert an influence of more than 5 oersteds on the Spacecraft field.

3.4 Facility Interfaces. AHSE design will complement and ensure compatibility with planned facilities by establishing requirements such as required maximum crane hook heights and floor loading and floor areas and by complying with requirements such as supporting cleanliness and interfacing with utilities.

3.5 Spacecraft Structural Parameters. AHSE requirements analysis and detailed concepts presented herein have for their foundation the following Spacecraft structural constraints:

a. Electronic bays will be contained in separable modules weighing 80 pounds or less, removable from the Spacecraft, and removal of any or all modules will not impair the structural integrity of the Spacecraft.

b. The retropropulsion unit will be structurally self-supporting and may be assembled with the Spacecraft by lowering the Spacecraft over the unit.

c. Solar panels will not be an integral part of the Spacecraft structure and will be removable.

d. The Planetary Vehicle-to-nose fairing interface will be arranged to permit applying the fairing over the Planetary Vehicle for assembly.

e. Operation of the Spacecraft in open air at 65° F ambient during system test will not require additional cooling for the electronics.

f. Once proof-tested, the pneumatic systems having tanks designed with a 2.2 safety factor will be permitted a full operational charge without restricting personnel activities in the area.

g. Deployable spacecraft elements may be exercised at 1- g loads without damage.

h. The nose fairing contractor will provide all AHSE required for handling the nose fairing before, during, and after Planetary Vehicle assembly, including on-pad cooling.

i. The Capsule contractor will provide all AHSE required for handling, storing, and shipping the separate Capsule during the ground mission.

CII VC270FD100

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT SYSTEM LEVEL FUNCTIONAL DESCRIPTION

SECTION

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT SYSTEM LEVEL FUNCTIONAL DESCRIPTION

1.0 FUNCTIONAL REQUIREMENTS. The ground mission Assembly, Handling, and Shipping Equipment (AHSE) Requirements at the system level will be defined as functional support of the assembly and test shown on the ground mission diagram of Figure 1-1. It should be noted that ground mission analysis is based on Flight Acceptance (FA) tests; Development and Proof Test Model (PTM) Spacecraft requirements should be treated as complementary increments to be supported by basic FA equipment, supplemented by Special Purpose Test Equipment where required.

2.0 SYSTEM CONCEPT. As a result of analysis based upon system constraints and requirements and the overall size of the vehicle and its relation to size of facilities required, the preferred approach to AHSE is the development of a basic central station which will satisfy the maximum number of functional requirements and thus minimize vehicle handling and redundancy of design features.

The central station concept will permit vehicle assembly, system testing, subsystem testing, leak checking, environmental control, weight and cg location, free-mode testing, interface compatibility checkout, and final system alignments to be implemented at the one station. Close integration with facility design can permit trenching or ducting of System Test Complex (STC) Cabling, as well as built-in features which improve the economics of station design, as well as reduce testing time.

The central station referenced above is designated as the Spacecraft Assembly Fixture.

Test requirements demanding capability beyond that feasible for incorporation in the Spacecraft Assembly Fixture will be satisfied by Special Purpose Test Stations to which the Spacecraft will be removed for the completion of the test.

System rechecks following an environment check will be performed in the Spacecraft Assembly Fixture to which the Spacecraft will be returned if recheck is necessary prior to proceeding to the next Special Purpose Test Station. Whenever practical, special-purpose tests will be run within the same area as system tests to facilitate environmental controls, (cleanliness, temperature, etc.) and minimize vehicle handling.

AHSE will furnish support for the assembled vehicle, regardless of the test phase or the test location, which again introduces economic advantages in design labor, redundance of equipment, and only one excursion of the learning curve.

The general approach to AHSE required for prelaunch operations will follow that outlined above for factory operations. In fact, the central station will duplicate the factory unit.

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

3.0 EQUIPMENT CLASSIFICATIONS. In order to establish convenient reference categories of equipment for work packages and documentation the AHSE has been classified into the following functional performance groups:

a. Assembly Equipment - This equipment includes interface simulators, alignment equipment, and assembly fixtures.

b. Handling Equipment - This equipment includes handling fixtures and magnetic mapping equipment.

c. Shipping Equipment - This equipment includes a Spacecraft Shipping Cover for the transporter and OSE Shipping Containers.

d. Special-Purpose Support Equipment - This equipment includes test fixtures for the following tests: thermal vacuum, vibration, separation, static firing, acoustic, free-mode and environmental control for margin verification testing.

4.0 ASSEMBLY EQUIPMENT. Figure 4-1 is an illustrated flow chart of anticipated assembly and test operations, which show how these types of AHSE are to be utilized. The figure also illustrates some of the major aspects of the AHSE design concept.

4.1 Simulators. Four types of simulators are required during the Test and Verification phases of the assembly of the VOYAGER Spacecraft. The simulators can be supplied by either the spacecraft contractor or the contractor for the interfacing equipment. Table 4-1 and footnotes specify the areas of use for the simulators. See Figure 4-1 for their functions in the context of the assembly and test process.

TABLE 4-1. AREAS FOR USE FOR SIMULATORS

	Development Testing	PTM Testing	Flight Acceptance Testing	Spacecraft Science Subsystem Test Program
Capsule Simulator	x^1	x^2	x^4	
Retropropulsion Subsystem Simulator	x^1	x^2	x^5	
Nose Fairing Simulator		x^3	x^6	
Spacecraft Science Subsystem Test Fixture				x^7

Notes

1. Both the Capsule Retropropulsion Simulators are attached to a dynamically and thermally adequate test model of the Spacecraft while the space simulation (thermal vacuum), vibration, and acoustic tests are being performed. The nose fairing required for these tests will be secured from the nose fairing contractor.

2. Both simulators are used in conjunction with a Spacecraft to support the following tests: (a) weight and center-of-gravity determination, (b) space simulator (thermal vacuum), (c) vibration and acoustic, (d) payload vehicle (PV) - booster and capsule-Spacecraft separation. In addition, the Capsule Simulator will be required to support the static firing test, and the retropropulsion subsystem simulator will be required during the magnetic evaluation.

3. The simulator will demonstrate the mechanical and functional interface compatibility between the PTM Spacecraft and the Nose Fairing.

4. This simulator is used in conjunction with a Spacecraft to support the space simulation (thermal vacuum) test.

5. The free-mode and space simulation (thermal vacuum) tests require this simulator in conjunction with a Spacecraft.

6. The simulator establishes the mechanical and functional interface compatibility between the flight Spacecraft and the Nose Fairing.

7. The fixture provides an integrating structure for the various parts of the Spacecraft Science during the overall JPL Spacecraft Science Subsystem Test Program.

FA OPERATIONS

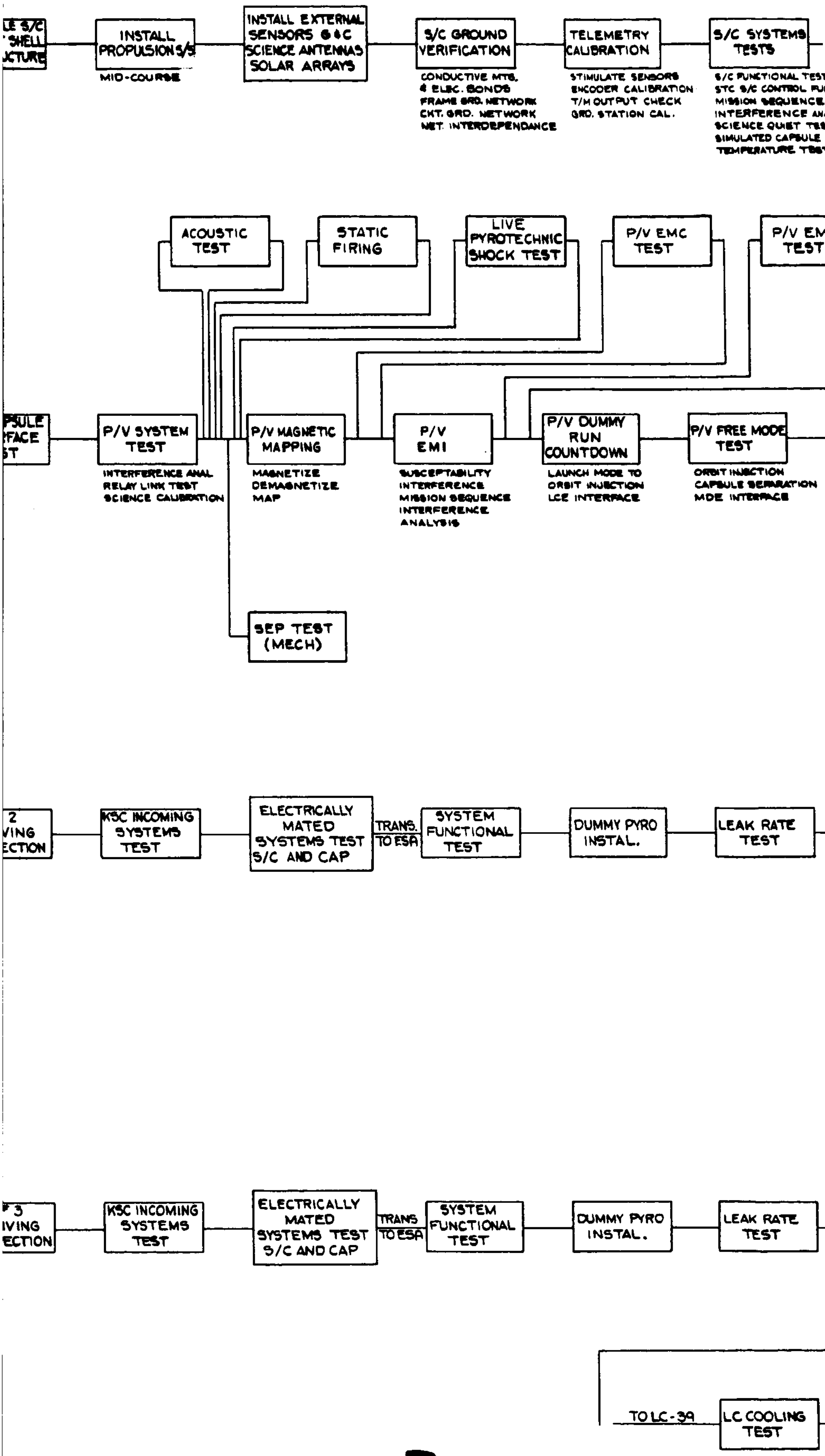
DELTA PTM TESTS

FA OPERATIONS

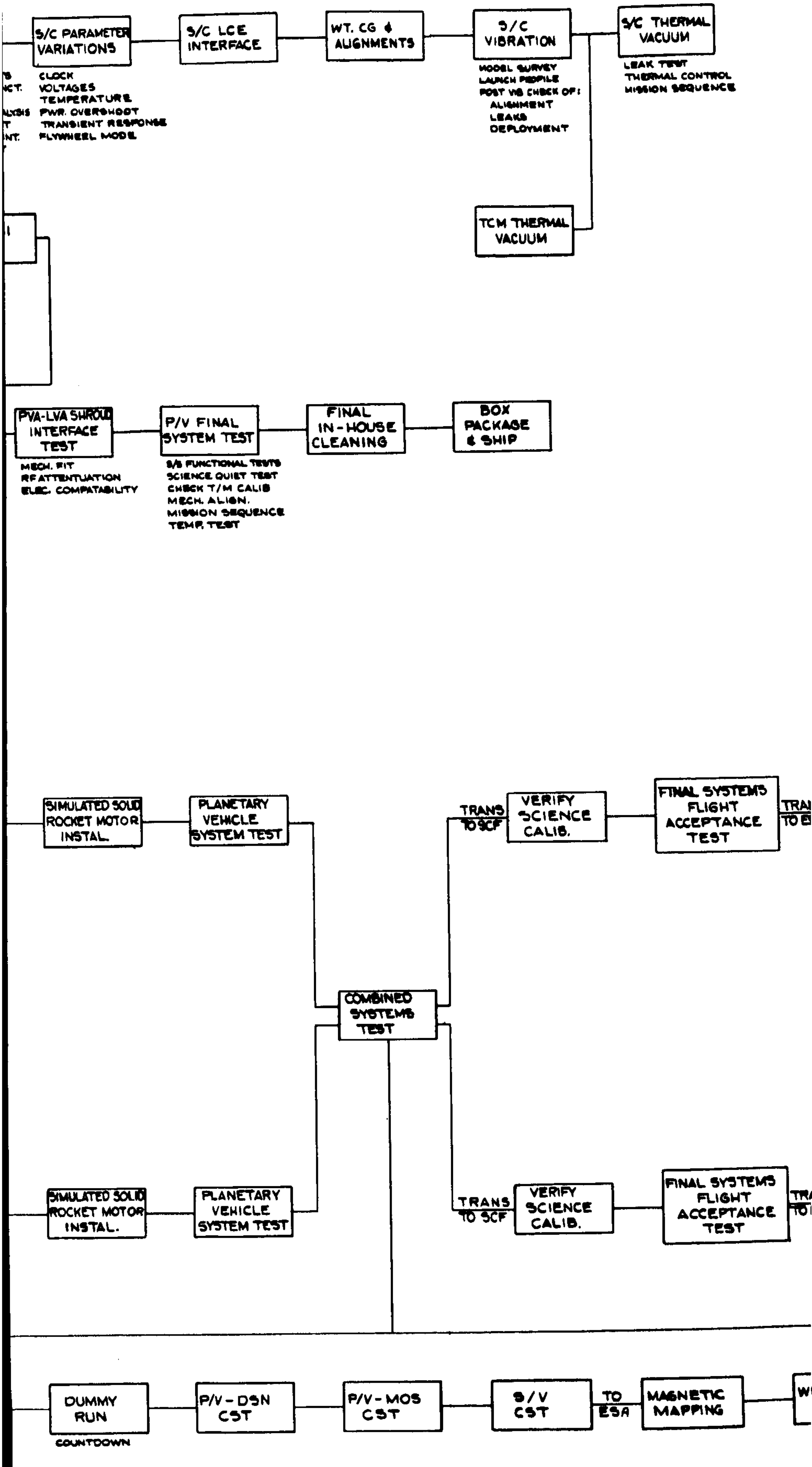
DELTA DEVELOPMENT TESTS

FIELD OPERATIONS

1-3



INSTALL PROPULSION S/S
MID-COURSE
INSTALL EXTERNAL SENSORS S&C SCIENCE ANTENNAS SOLAR ARRAYS
S/C GROUND VERIFICATION
CONDUCTIVE MTG.
& ELEC. BONDS
FRAME GRD. NETWORK
CKT. GRD. NETWORK
NET. INTERDEPENDANCE
TELEMETRY CALIBRATION
STIMULATE SENSORS
ENCODER CALIBRATION
T/M OUTPUT CHECK
GRD. STATION CAL.
S/C SYSTEMS TESTS
ACOUSTIC TEST
STATIC FIRING
LIVE PYROTECHNIC SHOCK TEST
P/V EMC TEST
P/V SYSTEM TEST
INTERFERENCE ANAL
RELAY LINK TEST
SCIENCE CALIBRATION
P/V MAGNETIC MAPPING
MAGNETIZE
DEMAGNETIZE
MAP
P/V EMI
SUSCEPTABILITY
INTERFERENCE
MISSION SEQUENCE
INTERFERENCE
ANALYSIS
P/V DUMMY RUN COUNTDOWN
LAUNCH MODE TO
ORBIT INJECTION
LCE INTERFACE
P/V FREE MODE TEST
ORBIT INJECTION
CAPSULE SEPARATION
MDE INTERFACE
SEP TEST (MECH)
KSC INCOMING SYSTEMS TEST
ELECTRICALLY MATED SYSTEMS TEST S/C AND CAP
TRANS. TO ESA
SYSTEM FUNCTIONAL TEST
DUMMY PYRO INSTAL.
LEAK RATE TEST
KSC INCOMING SYSTEMS TEST
ELECTRICALLY MATED SYSTEMS TEST S/C AND CAP
TRANS TO ESA
SYSTEM FUNCTIONAL TEST
DUMMY PYRO INSTAL.
LEAK RATE TEST
TO LC-39
LC COOLING TEST


S/C PARAMETER VARIATIONS
CLOCK
VOLTAGES
TEMPERATURE
PWR. OVERSHOOT
TRANSIENT RESPONSE
FLYWHEEL MODE
S/C LCE INTERFACE
WT. CG & ALIGNMENTS
S/C VIBRATION
MODEL SURVEY
LAUNCH PROFILE
POST VIB CHECK OF:
ALIGNMENT
LEAKS
DEPLOYMENT
S/C THERMAL VACUUM
LEAK TEST
THERMAL CONTROL
MISSION SEQUENCE
TCM THERMAL VACUUM
PVA-LVA SHROUD INTERFACE TEST
MECH. FIT
RF ATTENTUATION
ELEC. COMPATABILITY
P/V FINAL SYSTEM TEST
S/S FUNCTIONAL TESTS
SCIENCE QUIET TEST
CHECK T/M CALIB
MECH. ALIGN.
MISSION SEQUENCE
TEMP. TEST
FINAL IN-HOUSE CLEANING
BOX PACKAGE & SHIP
SIMULATED SOLID ROCKET MOTOR INSTAL.
PLANETARY VEHICLE SYSTEM TEST
TRANS TO SCF
VERIFY SCIENCE CALIB.
FINAL SYSTEMS FLIGHT ACCEPTANCE TEST
COMBINED SYSTEMS TEST
SIMULATED SOLID ROCKET MOTOR INSTAL.
PLANETARY VEHICLE SYSTEM TEST
TRANS TO SCF
VERIFY SCIENCE CALIB.
FINAL SYSTEMS FLIGHT ACCEPTANCE TEST
DUMMY RUN
COUNTDOWN
P/V-DSN CST
P/V-MOS CST
S/V CST
TO ESA
MAGNETIC MAPPING

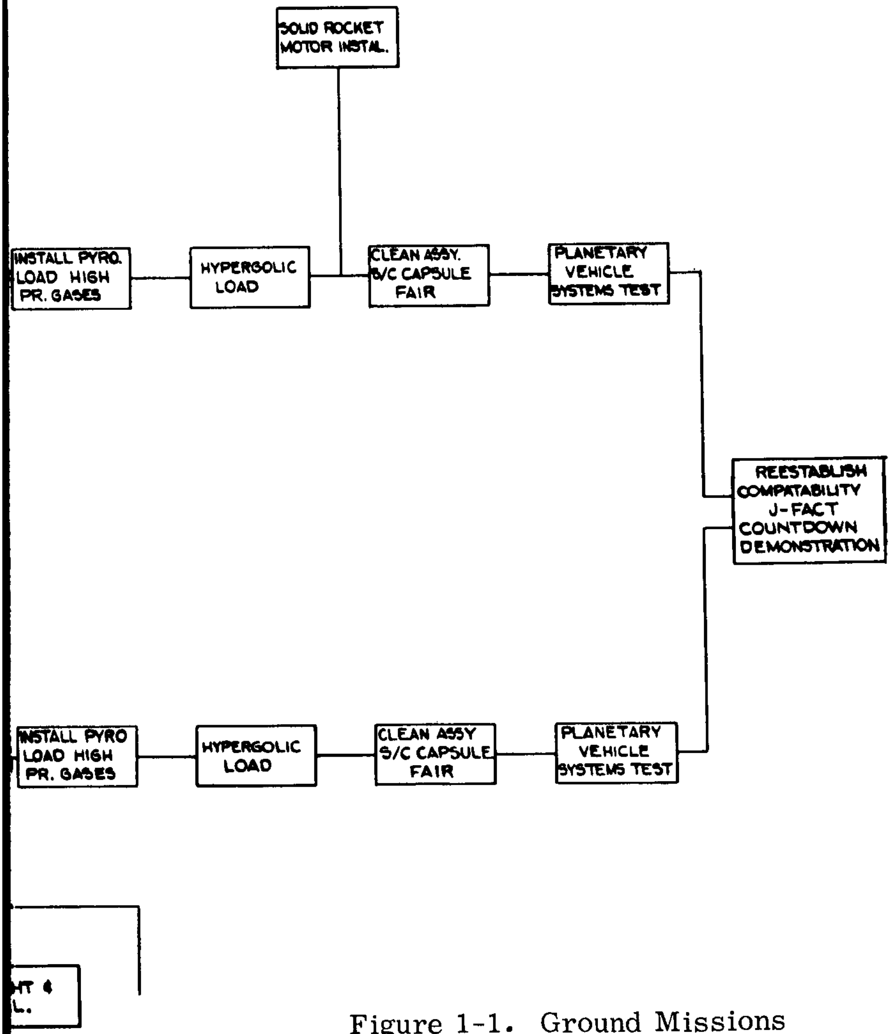
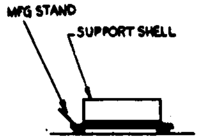


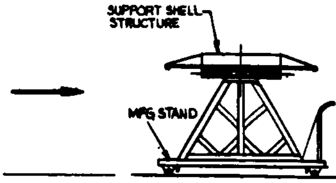
Figure 1-1. Ground Missions (Factory and Field)

3-4

SHOP ASSEMBLY AREA



CLEAN ROOM MECH ASSY AREA

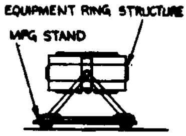


CL

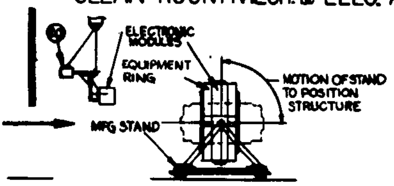
1. FABRICATION OF SUPPORT SHELL STRUCT.
2. INSTALL: SOLAR PANEL STRUCTURE
ANT. & SCANNER ACTUATORS

LATTITUDE CONTROL INSTALLATION OF:
TANKS, NOZZLES VALVES TUBING
2. NOZZLE ALIGNMENT
3. LEAK TESTS
4. PROOF PRESSURE

SHOP ASSEMBLY AREA



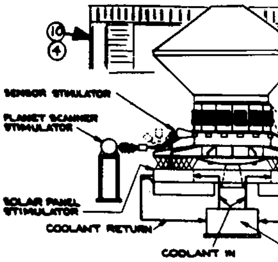
CLEAN ROOM MECH. & ELEC. ASSY AREA



1. FABRICATE EQUIPMENT RING
2. INSTALLATION OF SUPPORTING BRACKETS

1. INSTALLATION OF HARNESS
2. HARNESS TESTS
a. CONTINUITY
b. DELECTRIC STRENGTH
c. INSULATION RESISTANCE
3. INSTALLATION OF ELECTRONIC MODULES
4. INSTALLATION OF SENSORS
5. INITIAL-INTERNAL POWER APPLICATION TEST
6. S/S TESTS ON INTERNAL POWER

7. INTER SUB SYST
a. SYSTEM HAR
b. INITIAL-INTE
c. POWER & SIG
d. COMMAND/
e. STC/SYSTEM
f. PARTIAL SPA
g. INTERFEREN



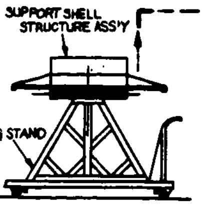
FREE MODE

A→

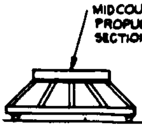
1-4

EAN ROOM ELEC ASSY AREA

CL



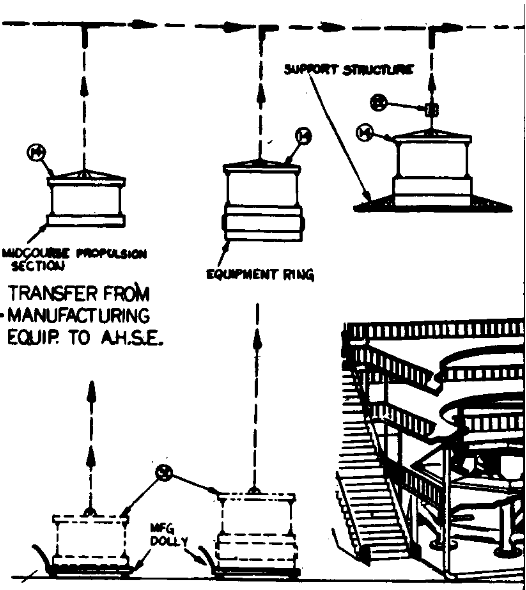
ARNESS & SENSOR INSTALLATION
ARNESS TESTS & GROUND VERIFICATION
. CONTINUITY
. DIELECTRIC STRENGTH
. INSULATION RESISTANCE

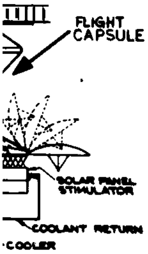


M TESTS
JESS INTERFACE
NAL POWER APPLICATION
AL DISTRIBUTION
CONTROL TELEMETRY DISTRIBUTION
CONTROL
ECRAFT GROUND VERIFICATION
E ANALYSIS

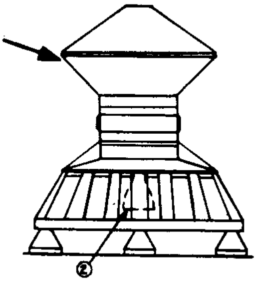
END OF MAJOR MFG CYCLE

TRANSFER FROM MANUFACTURING EQUIP. TO A.H.S.E.



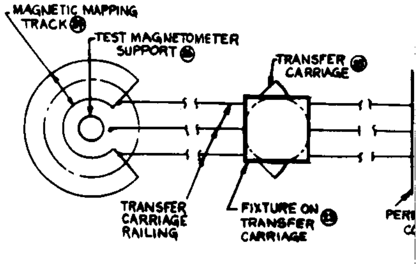


TEST



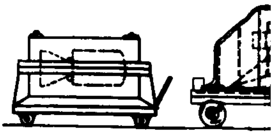
31 VIBRATION TEST

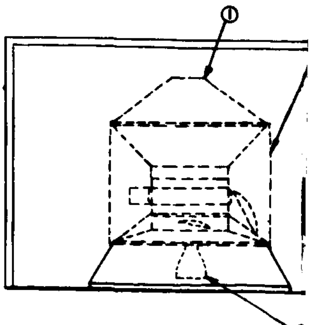
PTM CONFIGURATION SHOWN



MAGNETIC MAPPING

TRANSPORT OUT OF CLEAN ROOM AREA





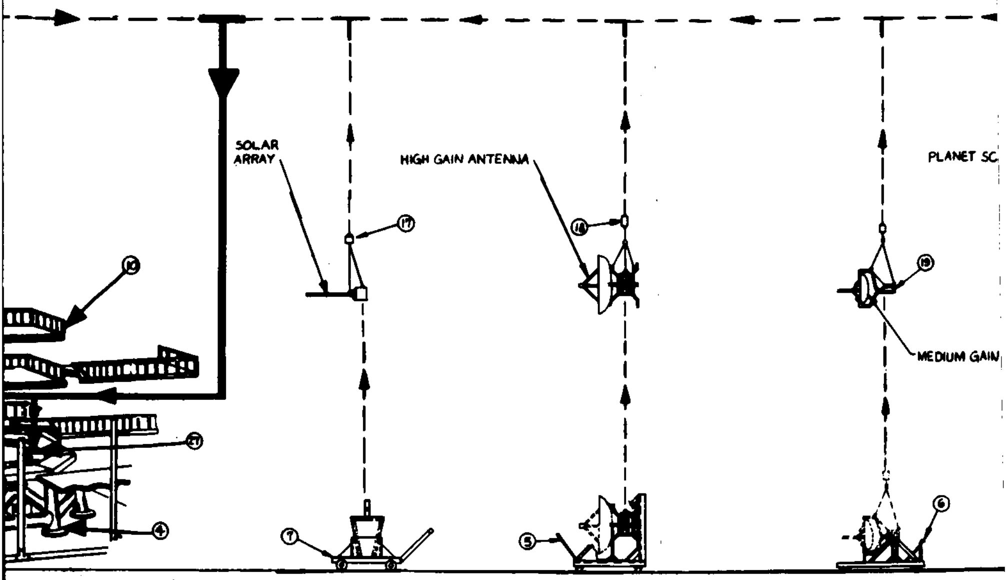
DELTA PTM TESTS

32 ACOUSTIC TEST

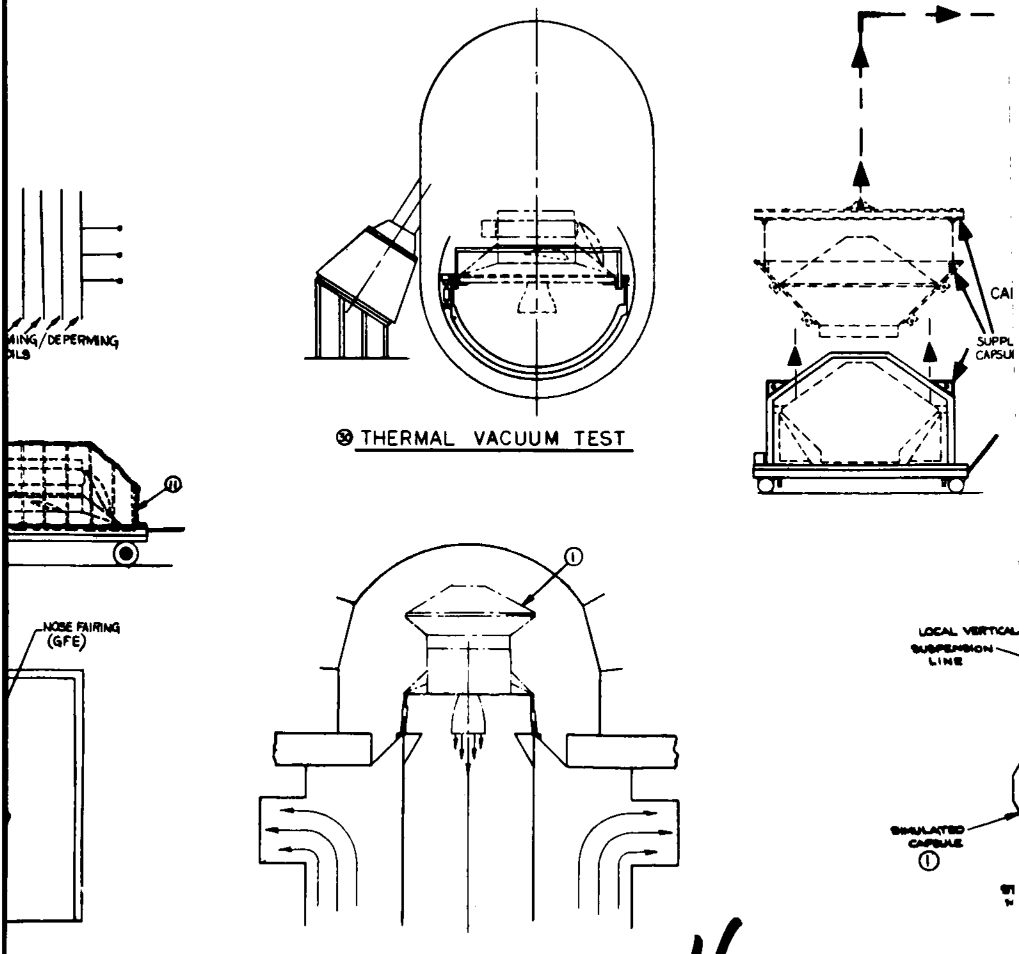
SHIP TO REMOTE AREA

2-4

EAN ROOM ASSEMBLY AREA

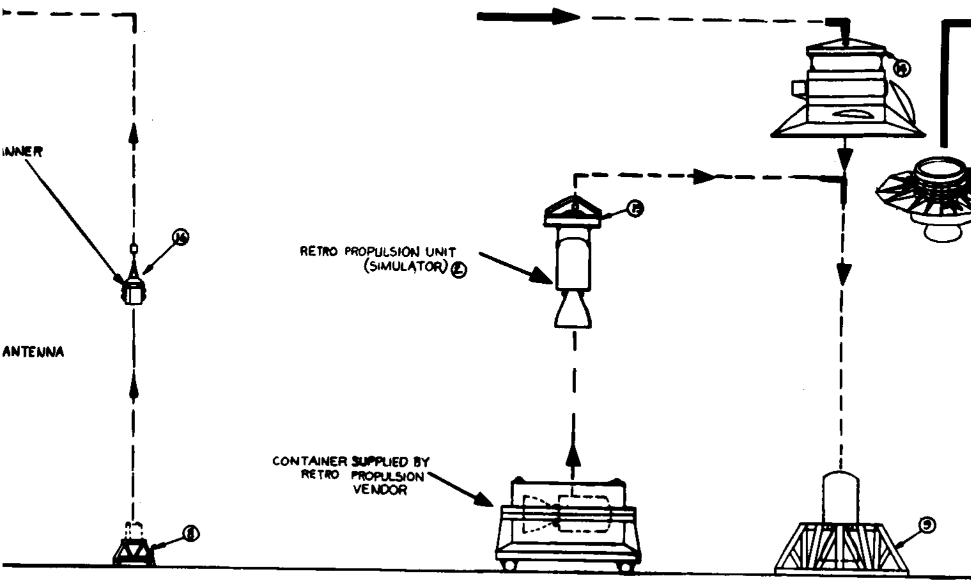


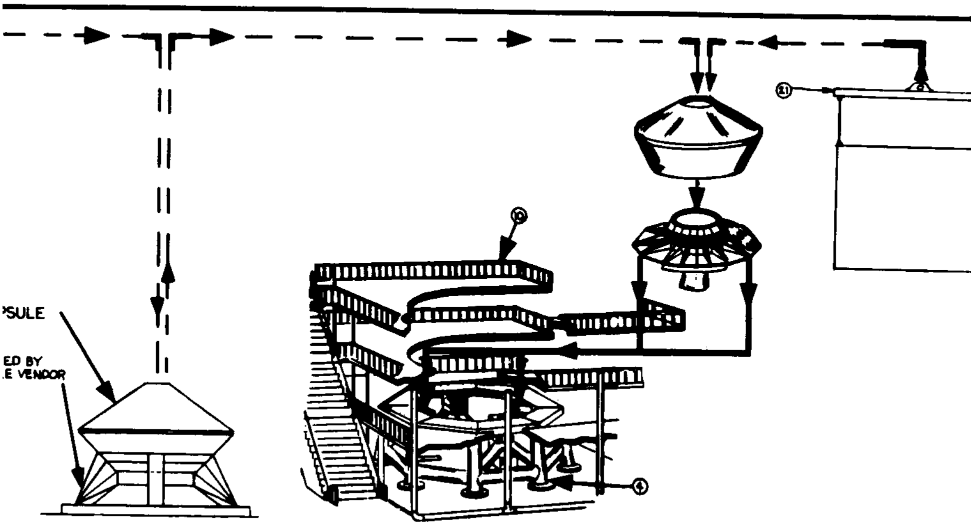
1. INSTALLATION AND ALIGNMENT OF TORUS STRUCTURE
2. INSTALLATION AND ALIGNMENT OF CAPSULE SUPPORT STRUCTURE
3. INSTALLATION OF EXTERNAL SENSORS
   a. SCIENCE
   b. GUIDANCE AND CONTROL
   c. THERMAL
   d. MAGNETOMETER
   e. HIGH GAIN ANTENNA
   f. SOLAR ARRAYS
   g. MEDIUM GAIN ANTENNA



30 THERMAL VACUUM TEST

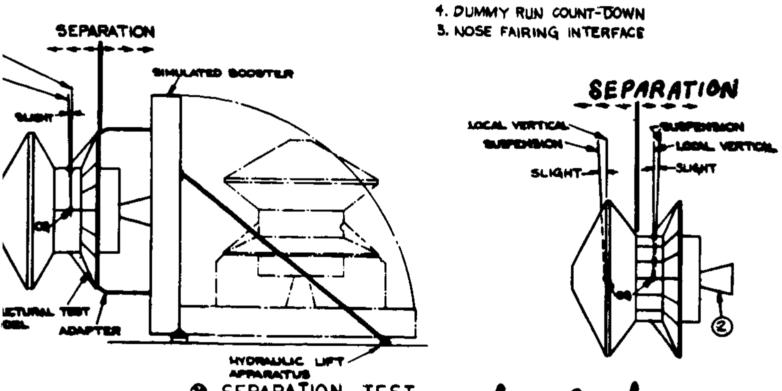
31 STATIC FIRING TEST
SHIP TO REMOTE AREA





1. INSTALLATION/INTERFACE TEST OF CAPSULE
2. PV SYSTEM TEST
3. EMI TEST
4. DUMMY RUN COUNT-DOWN
5. NOSE FAIRING INTERFACE

NOTE
NO A
EQUI
THEY
ASSE



SEPARATION TEST

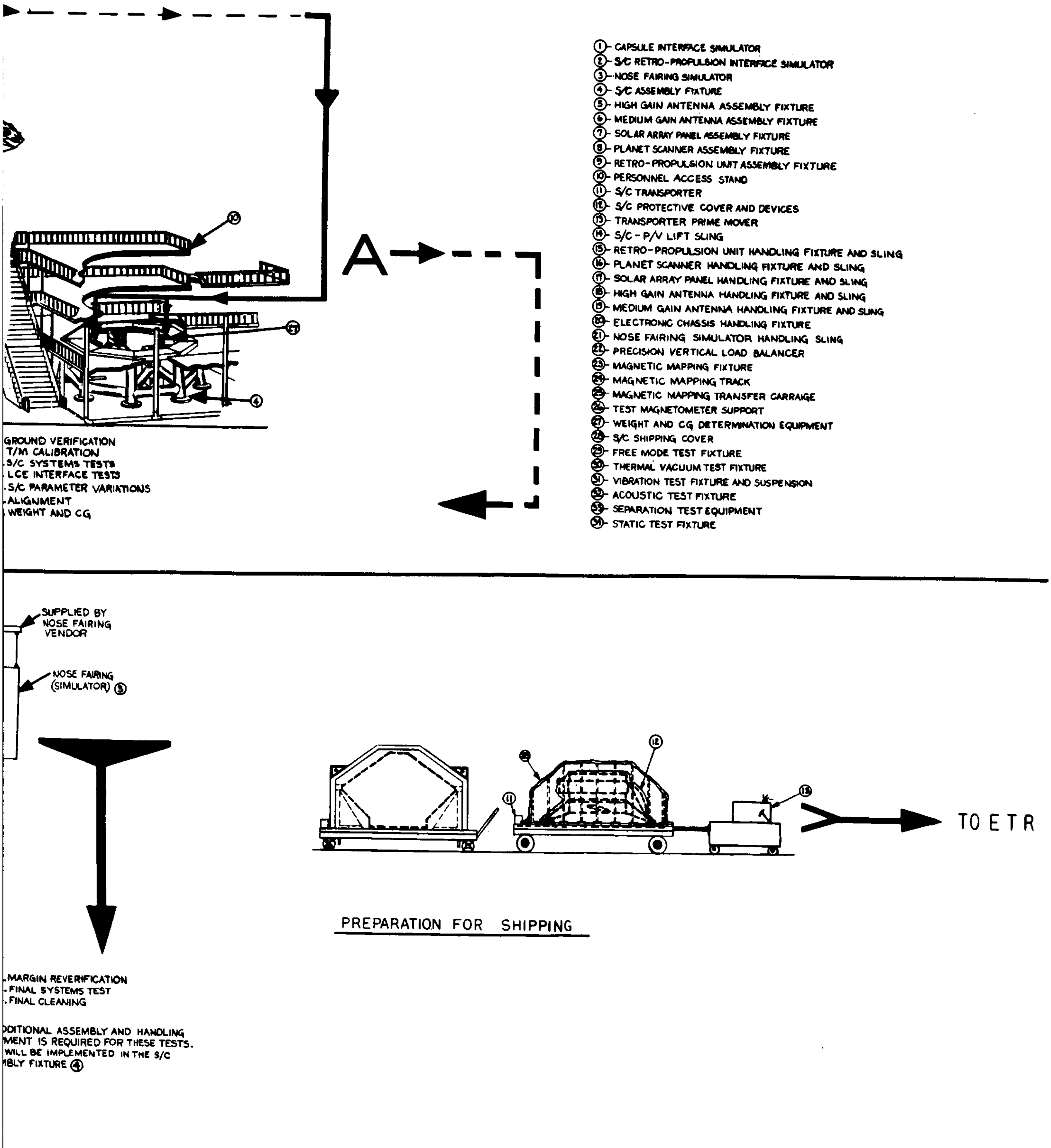


Figure 4-1. Functional Handling Tasks (Factory)

4-5

4.2 Alignment Equipment. To the greatest extent possible, the alignment operations will take place at the final assembly level. The Spacecraft Assembly Fixture will be utilized for alignment. Leveling capability and Spacecraft axes reference will be built into the assembly fixture to enhance the multiple-use concept.

The basic design approach is to use gravity as the common reference between items being aligned and the optical tooling equipment being used. Since the Spacecraft orientation for assembly purposes is with the roll axis vertical, the first step in alignment will be to establish the roll axis vertically. This will be accomplished by leveling the separation plane between the Flight Spacecraft and the nose fairing. For simplification, this plane will hereinafter be called "Reference Plane A". The Spacecraft manufacturing tooling will be used to establish the yaw axis. Through this axis a vertical reference plane (Reference Plane B) will be erected by optical tooling procedures. Individual component mounting surfaces will be checked for parallelism or perpendicularity to Reference Plane A by using clinometers (angle-reading levels) of appropriate sensitivity. For indexing (or pointing) about the roll axis, optical instruments independently supported on stands will be provided. These instruments will be located relative to component and Spacecraft axes by normal optical tooling procedures. Target equipment will be provided. In the case of angular displacements, front surface mirror targets will be used in conjunction with autocollimation. For linear displacements, bifilar targets will be utilized with telescopic instruments having optical micrometers. If the linear displacement tolerance is greater than the range of the optical micrometers (±0.100 inch), optical tooling scales or mechanical staging employing micrometer readouts will be used. The components to be aligned and the alignment tolerances are itemized in Table 4-2.

4.3 Assembly Equipment. The assembly equipment comprises the support equipment upon which a particular assembly is placed for test, servicing, maintenance, or storage during the ground mission cycle. The assembly equipment ranges from the Spacecraft Assembly Fixture, which is the central assembly and test station, through the fixtures to support each removable subassembly.

The Spacecraft Assembly Fixture is by far the most complex item of the group. It is the heart of the AHSE, and the remaining items of assembly equipment support its function.

The spacecraft will be assembled and aligned with the "Z" axis vertical in the assembly fixture. Multistage access platforms will surround this station. Spacecraft movement into and out of the test station will be in the vertical by use of an overhead crane. In order to satisfy the multiple-use concept of the station (such as systems and subsystems tests alignment, temperature tests, etc.), various services will be permanently installed and easily accessible. These services will include but not be limited to:

a. Removable environmental equipment for margin reverification thermal checks on a periodic basis. The environmental equipment will be supplied with necessary devices to obtain temperature variations required at the electronic modules.

TABLE 4-2. ALIGNMENT REQUIREMENTS

Gyro and AccelerometerNozzle Centerline referenceParallel to Reference Planes A and B30 minutes half cone

Gyro and Accelerometer Nozzle Centerline reference Parlalel to Reference Planes A and B 30 minutss half cone

Component or Subsystem	Alignment	Alignment Reference	Alignment Tolerance
Canopus Star Sensor	Sensor mounting surface Sensor indexing reference	Parallel to Reference Plane A	±2.0 minutes ±6.0 minutes
Acquisition Sun Sensors (on Solar Array)	Sensor mounting surface Sensor indexing surface	Parallel to Reference Plane A Parallel or perpendicular to Reference Plane B	±10.0 minutes ±10.0 minutes
Sun Sensor Assembly	Sensor mounting surface Sensor indexing reference	Parallel to Reference Plane A Parallel to Reference Plane B	±2.0 minutes ±6.0 minutes
Gyro and Accelerometer Package	Package mounting surface Package indexing reference	Parallel to Reference Plane A Parallel to Reference Plane B	±10.0 minutes
Cold Gas Nozzles	Nozzle mounting surface Nozzle centerline reference	Perpendicular to Reference Plane A Parallel to Reference Planes A and B	±1.5 degrees
High-Gain Antenna	Reference surface	Reference Planes A and B	±6.0 minutes
Planet Scan Platform	PSP mounting surface	Reference Plane A	
Solar Array Panels	Solar cell mounting surface	Parallel to Reference Plane A	±1.0 degree
Science Payload	Various sensors	Specific alignment requirements will be generated upon definition of the science payload.	
Medium-Gain Antenna	Antenna mounting surface	Fixed angle from Reference Plane A Fixed angle from Reference Plane B	±8.0 minutes
Flight Capsule	Capsule mating surface	Parallel to Reference Plane A	30 minutes half cone
Mid-course Engine	Nozzle centerline	Perpendicular to Reference Plane A	1.0 minute half cone
Retro Engine	Nozzle centerline	Perpendicular to Reference Plane A Radial displacement from Spacecraft Roll Axis	1.0 minute half cone 0.0625 inch

DEFINITION OF REFERENCE PLANES

Reference Plane A: For the purpose of alignment procedures, Reference Plane A is taken to be the separation plane between the Planetary Vehicle and the Nose Fairing.

Reference Plane B: This is a plane perpendicular to Reference Plane A and includes the planetary vehicle Yaw Axis (i.e., the Planetary Vehicle Yaw-Roll Plane)

b. GN_2 and He supply source and leak rate monitoring equipment.

c. Four load cells will be built into the interface portion of the fixture for weight and cg measurement.

d. The station will include design provisions for maximum safety, cleanliness promotion, human standards, and practical conveniences.

e. Adjustable interfaces and accurately oriented alignment references and posts to permit final alignment checks.

The various supplemental assembly fixtures will be specifically designed for their task. In general, assembly fixture interfaces will be a duplicate of that between the subassembly and the Spacecraft (where practical) to ensure compatibility.

Assembly fixture design will be closely integrated with Spacecraft manufacturing and subsystem design to ensure compatibility and economy.

5.0 HANDLING EQUIPMENT

5.1 Handling Equipment - Factory Phase. The functional handling in which the Spacecraft will be involved during the various factory assembly and testing phases is illustrated by Figure 4-1.

A supplemental description of typical handling follows. The following major Spacecraft subassemblies will be transferred from manufacturing equipment in the shop area, to AHSE assembly equipment in the Systems Test station area.

- Support Shell Assembly
- Midcourse Propulsion Section
- Equipment Ring
- Micrometeoroid Barrier
- Support Truss
- Solar Arrays
- Planet Scan Package
- Science Instrumentation
- Medium-Gain Antenna
- High-Gain Antenna

The Support Shell Assembly will be placed on the Spacecraft Assembly Fixture utilizing the Spacecraft Lift Sling. The Assembly will be supported at the inner ring flange by a temporary Fixture Adapter until the Midcourse Propulsion Section and the Support Truss are added. With the structural load path members in place, the partially completed Spacecraft assembly will be supported at the Nose Fairing Interface and the temporary Fixture Adapter will be removed. After Spacecraft assembly has been completed (less Retropropulsion Unit), subsystem tests will be conducted with the Spacecraft supported in the assembly fixture.

Mating of the Spacecraft with the simulated Retropropulsion unit will be accomplished as follows. The simulated Retropropulsion unit is removed from its shipping container utilizing the Propulsion Handling Sling and Fixture and is placed on the Retropropulsion Assembly Fixture. The Spacecraft is then removed from its assembly fixture with the Spacecraft-PV Lift Sling and is positioned over the Simulated Retropropulsion unit for attachment. Safety barriers will be emplaced to protect personnel who will be attaching the Spacecraft to the Simulated Retropropulsion Unit. The Spacecraft with Retropropulsion unit attached is then returned to the Spacecraft Assembly Fixture.

To complete the PV assembly, the Simulated Capsule will be removed from its shipping container with the Spacecraft-PV Lift Sling and placed on its assembly fixture and thence to the Spacecraft. The PV is now ready to be lifted, utilizing the Spacecraft-PV Lift Sling, to the appropriate satellite station for those tests not performed in the Spacecraft Assembly Fixture.

(Note: the Simulated Capsule should have provisions for and be capable of supporting the entire PV load when lifted by the Spacecraft-PV Sling. The flight capsule need not have this capability, as the Flight PV will not be lifted as an assembly by the capsule.)

Those tests not performed in the Spacecraft Assembly Fixture are vibration, magnetic mapping, and thermal vacuum (plus acoustic, static firing, live pyro shock and separation as applied to test models). For transporting to a remote site, the Simulated Capsule is returned to its shipping container, the Simulated Retropropulsion Unit is returned to its shipping container, and the Spacecraft is placed on the Spacecraft Transporter.

At the remote site, the Spacecraft will be removed from its transporter with the Spacecraft-PV Lift Sling and mated to the Simulated Retropropulsion Unit. The completed Spacecraft is now transferred to the special purpose test fixture after which the capsule is applied if a

PV configuration is required. The Spacecraft will be returned to the system test area at the factory by reversing the above procedure.

Before proceeding to the PV-LVA Nose Fairing interface test, the Flight Capsule will be mated with the Spacecraft. The Simulated Nose Fairing will then be lowered over the PV utilizing the Nose Fairing Simulator Handling Sling.

After completion of final factory systems tests, the Flight Capsule and the Simulated Retropropulsion Unit will be returned to their respective shipping containers for shipment to Kennedy Space Center (KSC). The spacecraft will be shipped to KSC on the transporter.

All AHSE, supporting the flight vehicles, will be disassembled and also shipped to KSC.

5.1.1 KSC Handling. The functional handling phase at KSC is illustrated by Figure 5-1. The tests at KSC will essentially be the same types performed in the factory. A supplemental description of the handling anticipated at KSC follows:

The following items will be received in their respective shipping configuration: Retropropulsion Unit, Flight Capsule, and Spacecraft. All of these items will be in shipping containers, except the Spacecraft, which will be on the Spacecraft Transporter.

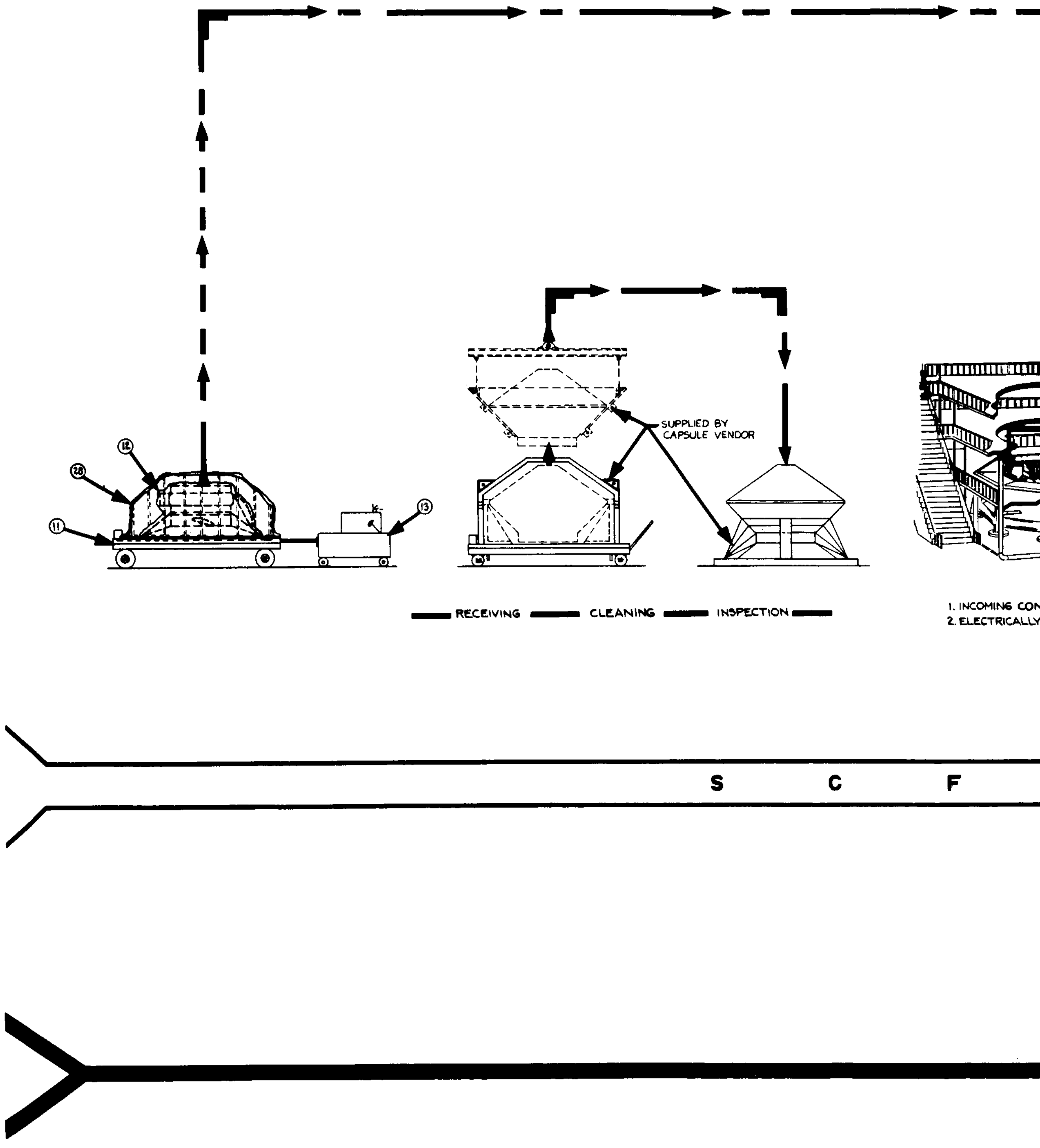
After unpacking, all items will be moved to a receiving-cleaning area, where they will be cleaned in compliance with program requirements and subsequently emplaced in the Systems Test Area of the Spacecraft Checkout Facility.

The Spacecraft will be lifted from transporter by the Spacecraft-PV Lift Sling and placed on the Spacecraft Assembly Fixture.

KSC Flight Spacecraft processing will consist of a confidence test flow of SCF-ESA-Pad, followed by a servicing phase which flows through the SCF-ESA-Pad Route and is terminated by liftoff.

The incoming and systems tests will be performed at the Satellite Checkout Facility (SCF) prior to transfer to the Explosives Safe Area (ESA). All tests in the ESA will be conducted in the systems test stations; the PV in the encapsulated configuration will be taken to the pad for combined system test, after which the PV will be taken to the SCF for further systems tests and then returned to the ESA for installation of pyros, magnetic mapping, weight and balance, fueling, nose fairing, encapsulating mating, and final PV systems tests. All AHSE will be available in the field sequence. For movement between the SCF and ESA, the Capsule should be placed in its shipping container and the Spacecraft on the transporter. It is intended that AHSE will be transferred between the SCF and the ESA to eliminate equipment redundancy. During the transfer, provisions will be made, such as wrapping, transporting in closed vans, etc., to preserve the cleanliness of the equipment.

5.1.2 Magnetic Mapping Equipment. The Magnetic Mapping Equipment, functionally illustrated in Figures 5-2, 5-3, and 5-4, will be utilized during any series of magnetic mapping



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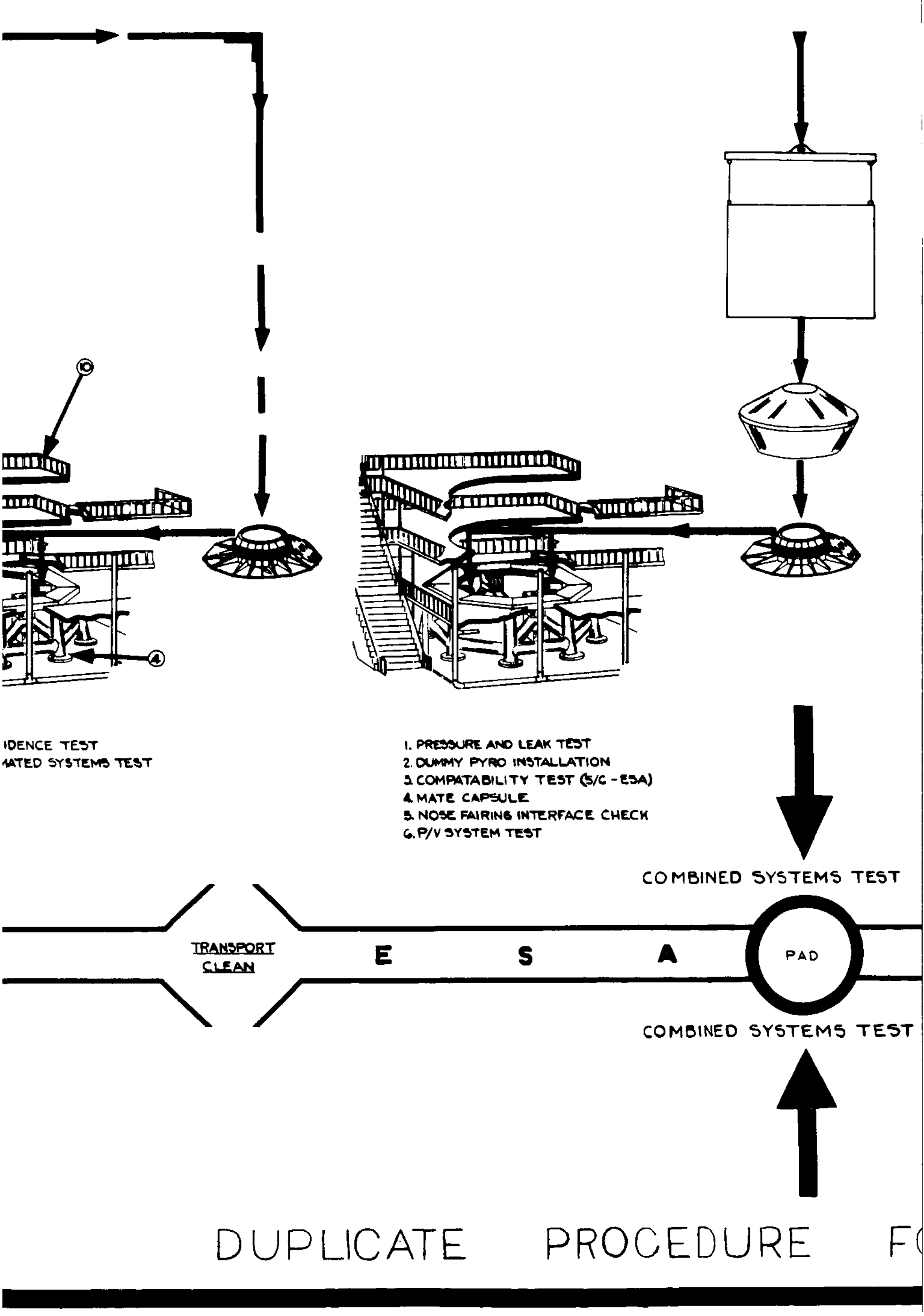
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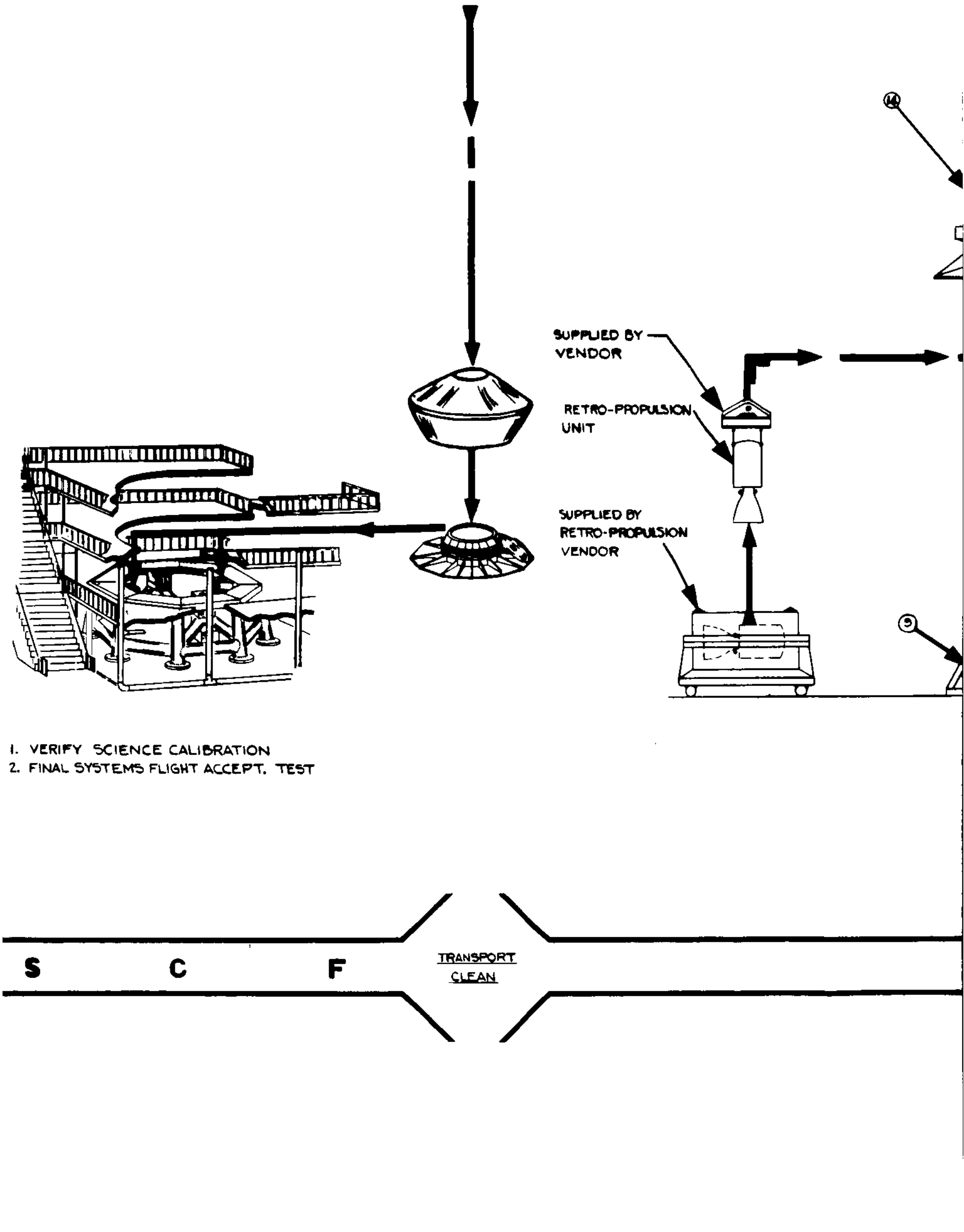
IDENCE TEST
MATED SYSTEMS TEST
1. PRESSURE AND LEAK TEST
2. DUMMY PYRO INSTALLATION
3. COMPATABILITY TEST (S/C - ESA)
4. MATE CAPSULE
5. NOSE FAIRING INTERFACE CHECK
6. P/V SYSTEM TEST
COMBINED SYSTEMS TEST
TRANSPORT
CLEAN
E S A
PAD
COMBINED SYSTEMS TEST
DUPLICATE PROCEDURE F



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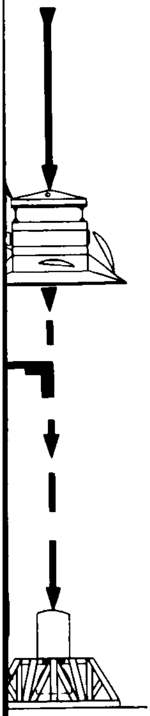

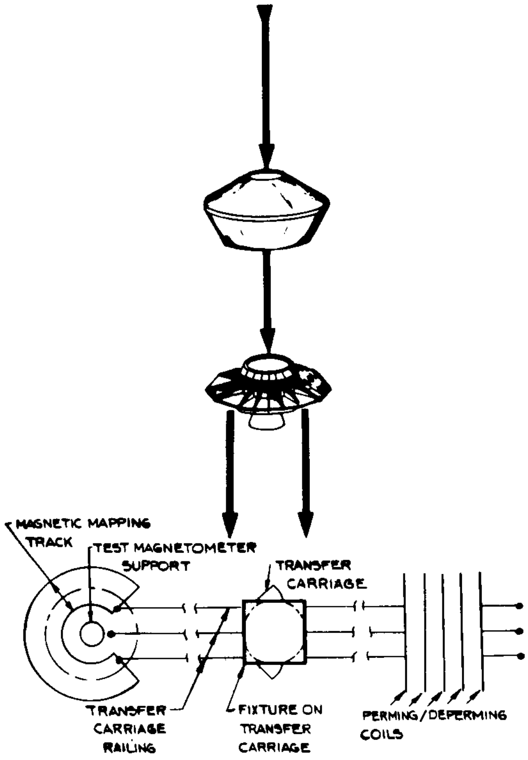

MAGNETIC MAPPING

1. INSTALL PYROTECHNICS A
2. WEIGHT AND BALANCE
3. ALIGNMENT
4. S/C CONFIDENCE TEST
5. MAGNETIC MAPPING
6. HYPERGOLIC LOAD
7. MATE CAPSULE
8. MATE NOSE FAIRING
9. P/V SYSTEM TEST

E	S	A

4-9

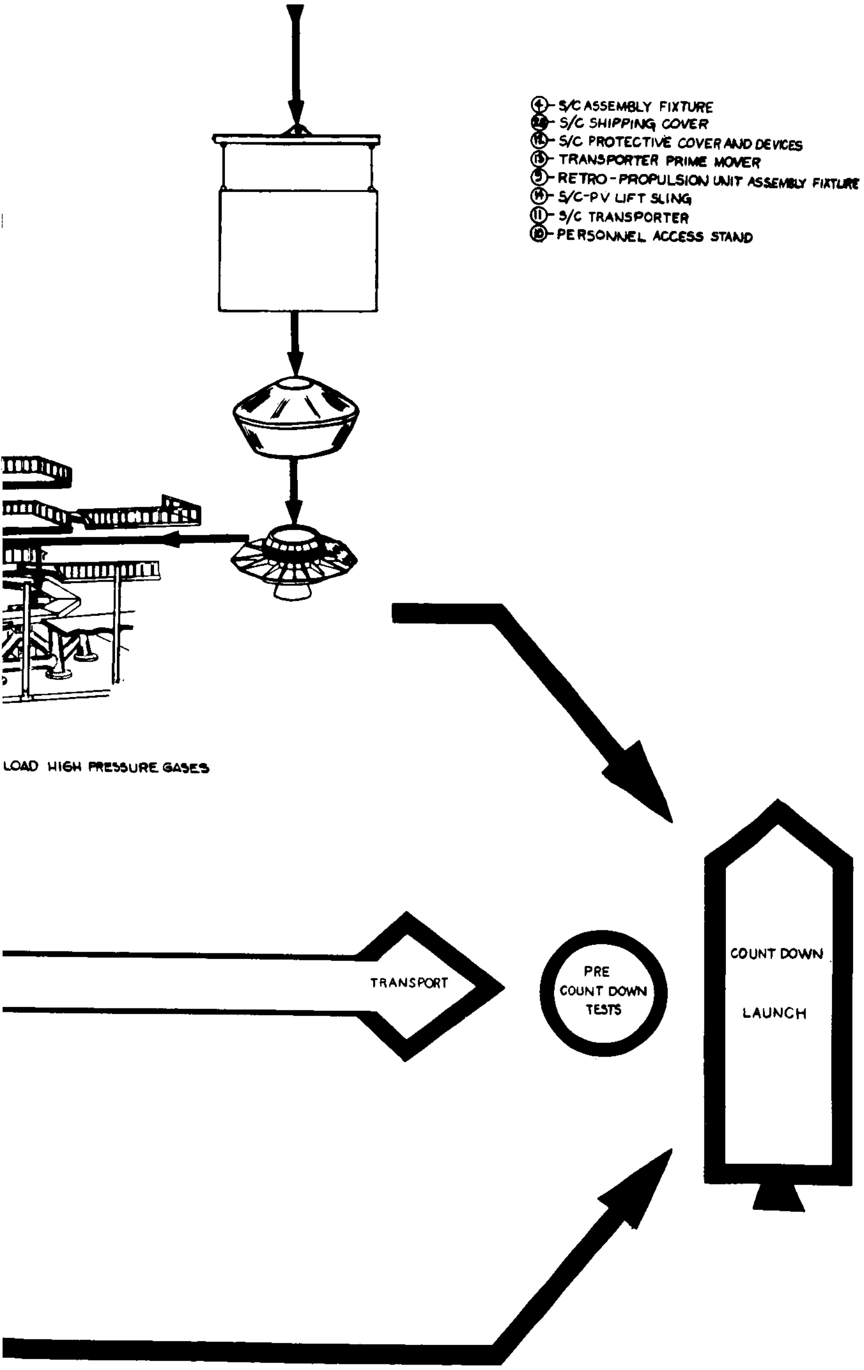

Figure 5-1. Functional Handling Task (Field Flow)

5-9

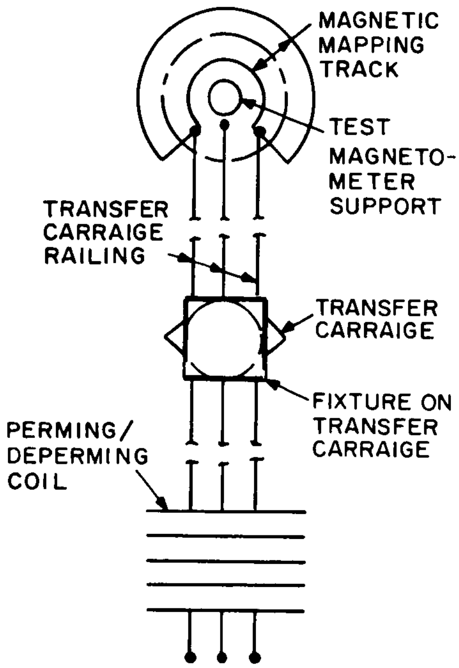

Figure 5-2. Magnetic Mapping Station

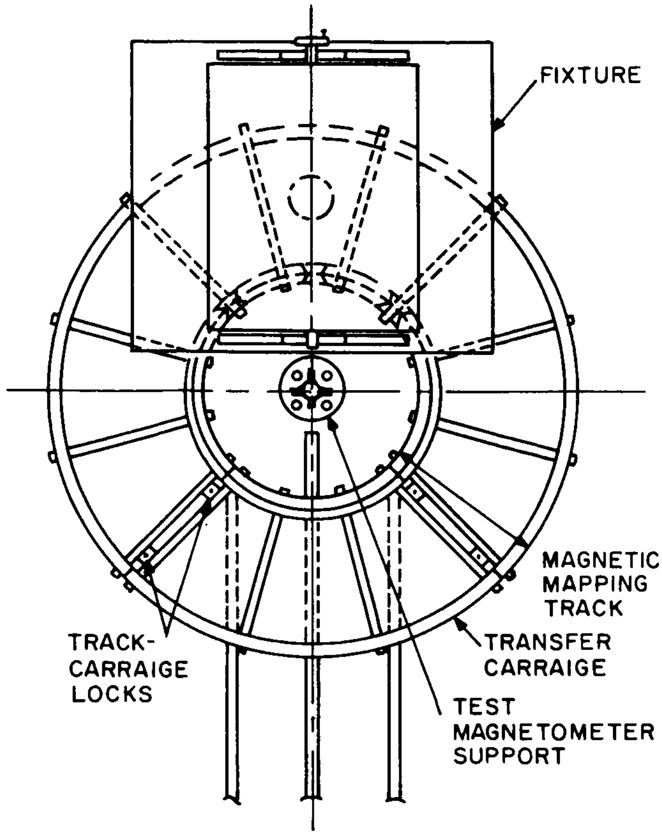

Figure 5-3. Top View of Fixture and Transfer Carriage

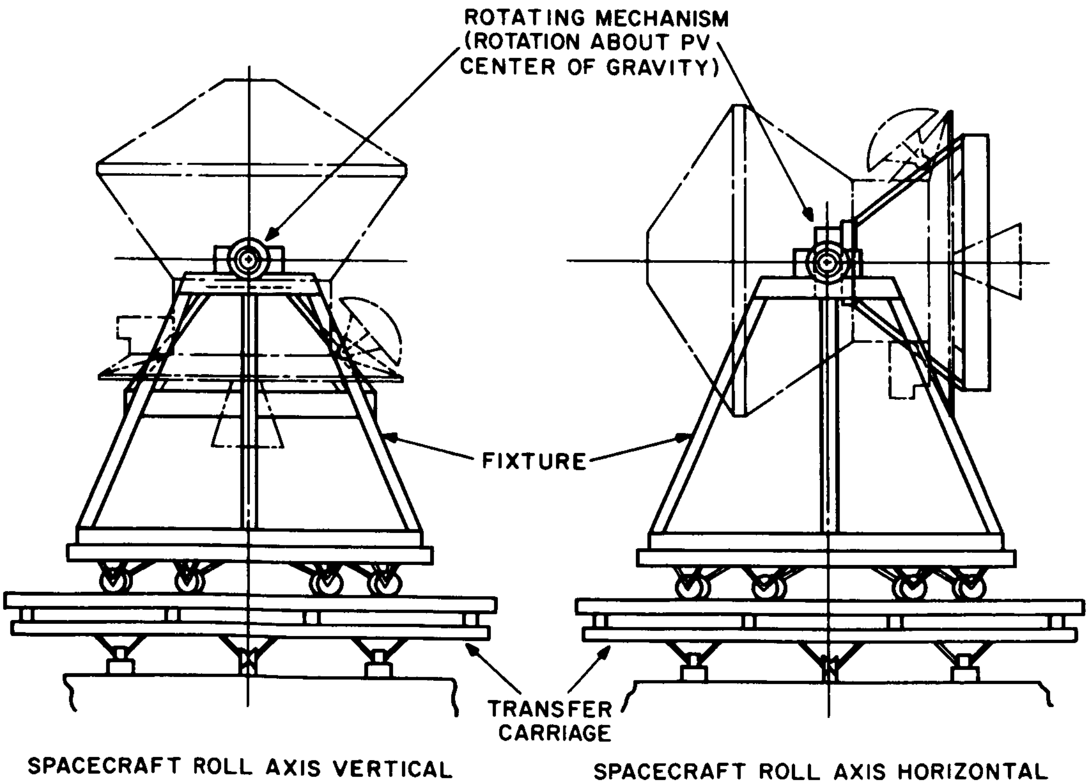

Figure 5-4. Views at Center Line of Railing Looking Toward Magnetic Mapping Track

tests. When perming and/or deperming constitutes an additional requirement in the mapping test sequence, the equipment will be capable of supporting this phase as well.

The Planetary Vehicle is brought to the Magnetic Mapping Station by using a sling and overhead crane facility. The PV is placed on the Magnetic Mapping Fixture, which functions as the means of support during the mapping test sequence.

The Fixture rests on a Transfer Carriage, which operates on rails. The rails may run between a Perming/Deperming Coil, and the Magnetic Mapping Track (as required at the specific facility).

The Magnetic Mapping Track provides a circular path around the Test Magnetometer for the Fixture. The PV (supported by the Fixture) traverses a 360-degree path on the Track for magnetic characteristic determination. The use of a PV lifting device is required during this phase for axis orientation. A Test Magnetometer Support places the test device in the same relative position to the PV as the Flight Magnetometer when deployed.

When perming and/or deperming is required, the Carriage is used to transport the PV (supported by the Fixture) to the Coil where the PV may be permed (or depermed) about three orthogonal axes, each in turn. The use of a PV lifting device is required during this phase for axis orientation.

The perming/deperming Coil and Test Magnetometer are considered as facilities rather than components of the VOYAGER AHSE.

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

6.1 Flight Spacecraft

6.1.1 Shipping Configuration. The Flight Spacecraft shall be shipped fully assembled except for the Retropropulsion Subsystem. This is a prime requirement because of the possible degradation of the reliability of the Spacecraft that might result from disassembly and subsequent reassembly. Disassembly would also tend to negate the effectiveness of the final systems testing at the manufacturing site. Compliance with this "No Disassembly" concept presents a transportability problem because of the large size of the assembled Spacecraft (approximately 22 feet long by 22 feet wide by 100 inches high).

6.1.2 Mode of Transportation. The size of the fully assembled spacecraft eliminates the possibility of shipment by rail. Air shipment is severely limited and can only be accomplished by specially modified and hard-to-get aircraft. Alternate modes of transportation will be over the highway and by water. However, highway transportation shall be kept to an absolute minimum because of the obvious difficulties in moving an item of this size over the road. It is, therefore, planned to move the Spacecraft on the Spacecraft Transporter from King of Prussia by highway to Willow Grove Naval Air Station (or to the Delaware River), whereupon the Transporter and Spacecraft will be loaded onto an aircraft (or barge) for subsequent movement to Kennedy Space Center. The Spacecraft will be protected from the environment by a flexible sealed cover. The atmosphere within the cover will be environmentally controlled by an auxiliary system capable of pumping thermally and humidity controlled gas into the sealed cover. The shipment will be accompanied by qualified technical personnel to assure constant monitoring of the environmental control system. Necessary escort personnel will also be provided for movement over the highway to assure that physical obstructions or other potentially damaging effects are eliminated.

6.2 Development Test Spacecraft

6.2.1 Thermal and Electrical Models. It is not planned to conduct tests on either of these models at remote locations. Shipping equipment in support of these units is therefore not required.

6.2.2 Structural Model. Acoustic testing of this model will be conducted at Huntsville, Alabama. Since it is time consuming to reach Huntsville, Alabama from the Philadelphia vicinity by water, the Structural Model will be shipped by air, with barge transport as a backup. The Structural Test Model (STM) spacecraft air shipment is considered to be less of a problem than flight spacecraft air shipment, since there is no requirement for the maintenance of a flight-ready configuration.

6.3 Proof Test Spacecraft Model. The PTM will be shipped from the assembly point to Huntsville, Alabama for acoustic testing, then to Arnold Engineering Development Center at Tullahoma, Tennessee for static firing. After static firing, the PTM will be transported back to MSD for additional testing. For the reasons pointed out in preceding paragraphs, this unit will also be shipped by air, with barge movement as a backup.

6.4 Shipment of OSE. OSE will be required at each of the test facilities mentioned above, as well as at the launch site. As the logistics plan for OSE is developed, a decision will be made regarding "Exclusive Use" vans for the movement of this equipment when the quantities involved indicate an economic advantage. Individual shipping containers will be considered for each OSE end item as alternates for use in the event that end items must be individually moved from one site to another. In general, the backup containers will be of wooden box or crate construction with appropriate internal shock and vibration mitigation as required. Only limited reusability of OSE end item shipping containers will be required in all cases.

6.5 Spare Parts. Spare parts required to support any phase of testing, and the flight program will be packaged, when possible, by using standard packaging methods already in use by GE-MSD. Packaging of extremely delicate items will be of custom design and will be engineered to meet the protection requirements dictated by the item. Packaging of explosives and other dangerous materials will be of controlled design to assure total compliance with applicable ICC and other regulatory body regulations.

7.0 SPECIAL-PURPOSE TEST EQUIPMENT (See Figure 4-1). The paragraphs describing special-purpose test equipment which is part of AHSE used in connection with tests, vehicles, and locations are shown in Table 7-1.

TABLE 7-1. INFORMATION CONCERNING SPECIAL-PURPOSE TEST EQUIPMENT

Paragraph	Test	Vehicle	Location
7.1	Weight and cg	All	SCF, ESA, and Factory
7.2	Free Mode	Electrical Model, PTM	SCF or Factory
7.3	Thermal Vacuum	Thermal Model, PTM, F/A	SCF or Factory
7.4	Vibration	Structural Test Model, PTM, F/A	SCF or Factory
7.5	Separation	Structural Test Model	Factory
7.6	Environmental Control	PTM, F/A	SCF, ESA, and Factory
7.7	Static Firing	PTM	Remote
7.8	Acoustic	PTM	Remote
7.9	EMI	PTM, F/A	Factory

7.1 Weight and Center-of-Gravity Test Equipment. Weight and center-of-gravity test equipment will be incorporated in the Spacecraft Assembly Fixture. The purpose of this is twofold: to provide a separate weight and center-of-gravity test station, and to provide experimental data continuously so that on completion of the vehicle the preliminary weight report is ready for submittal.

Incorporating center-of-gravity capability in the Spacecraft Assembly Fixture will limit center-of-gravity information to the two components in a plane normal to the roll axis. This, however, is not significant because the spacecraft must be capable of reliable guidance in spite of a large excursion of the center of gravity along the roll axis due to consumption of fuel. Pinpointing the center of gravity along the roll axis thus loses importance to the overall mission. A significant saving of time and cost will result from this approach.

Location of center of gravity along the roll axis can be measured if desired by using appropriate sensing devices in the separation test equipment.

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

Several different approaches to the proper method of conducting the Free-Mode Test were considered. The approach selected (conducting the test indoors by using simulated sunlight) led to the equipment described in Section VC270FD101 and was selected because of the requirement for cleanliness. A secondary consideration was reluctance to allow conduct of the test to depend on the state of the weather, should real, instead of simulated, sunlight have been chosen as the method for delivering power to the solar panels.

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

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

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

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

7.5 Separation Test Equipment. Separation test equipment will be used to test the effectiveness and dynamic response of (a) the PV at separation from the nose fairing interface, and (b) the capsule and the Spacecraft at separation from each other.

The test will be conducted in three phases: (1) separation of the PV from its interface with the nose fairing at the rate of three feet per second, (2) separation of the PV from its interface at the rate of two feet per second, and (3) separation of the capsule from the Spacecraft. The separating sections will be supported by a suspension cable attached at the cg point to permit five degrees of freedom during the separation. The equipment required for this test will provide means to support the vehicles during test, means to measure energy transfer at separation, and means to detect and measure tip-off motion in other directions in addition to the direction of separation. Similar separation test equipment has been used successfully on past programs.

7.6 Environmental Control Equipment. This equipment is used during the margin reverification of the electronic subsystems. Reverification is to be performed at specific intervals after the subsystems are delivered for Spacecraft systems test. Therefore, this equipment will be required at both plant and field facilities. The equipment provides the means for producing and maintaining the required temperature environment while monitoring the performance of all subsystem functions.

7.7 Static Firing Test Equipment. The static firing test equipment will support the vehicle during a static firing test, which simulates Mars orbit injection.

Two approaches are being considered for the implementation of this test: the first approach considers hard-mounting the vehicle on thrust sensors to provide information on vehicle vibration and thrust vector stability; the second approach considers soft-mounting the vehicle on soft spring- or air-suspension members to give the autopilot more realistic input signals.

The static firing test equipment will support the Spacecraft in the Z-axis vertical attitude, as required by the test facility. Suitable sensor and recording equipment will provide information for analysis of the vehicle under a static test firing.

7.8 Acoustic Test Equipment. The acoustic test equipment will be used at the acoustic test facility at a remote location. A nose fairing section, furnished as GFE, will be provided to support the PV in the acoustic chamber. Additional force, equivalent to 2.6 times the actual weight of the capsule, will be required in order to simulate the capsules' inertial force during the boost phase.

The sonic environment will be applied to the exterior of the nose fairing. Suitable sensing and recording apparatus will determine the response of the PV to the applied acoustic forces.

7.9 Electromagnetic Interference Test Equipment. EMI tests will be conducted on the Spacecraft while it is supported in the spacecraft assembly fixture. No additional AHSE will be required to support this test.

CII VC270FD101

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT HARDWARE FUNCTIONAL DESCRIPTIONS

ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT HARDWARE FUNCTIONAL DESCRIPTION

1.0 INTRODUCTION. The Assembly, Handling, and Shipping Equipment (AHSE) required to support the functional objectives of the program are listed below in functional groups. Reference numbers in the Handling and Assembly Sections correspond to the number identification of items of equipment illustrated in the handling sequence diagrams of Figures 4-1 and 5-1 of VC270FD100. Table 1-1 lists the end items.

TABLE 1-1. EQUIPMENT END ITEMS

Equipment	Reference Number Figure 4-1 & 5-1 Of VC270FD100
ASSEMBLY EQUIPMENT	
Simulators	
Capsule Interface Simulator	1
Spacecraft Retropropulsion Interface Simulator	2
Nose Fairing Simulator	3
Spacecraft Science Subsystem Test Figure	-
Alignment	
Guidance and Control Subsystem Alignment Set	-
Solar Panel Alignment Set	-
Science Payload Alignment Set	-
Medium Gain Antenna Alignment Set	-
Capsule Interface Alignment Set	-
Midcourse Engine Alignment Set	-
Retroengine Alignment Set	-
High-Gain Antenna Alignment Set	-
Optical Tooling Kit	-
Assembly Fixtures	
Spacecraft Assembly Fixture	4
High-Gain Antenna Assembly Fixture	5
Medium-Gain Antenna Assembly Fixture	6
Solar Panel Assembly Fixture	7
PSP Assembly Fixture	-
Electronic Chassis Assembly Fixture	-
Retropropulsion Unit Assembly Fixture	9
Spacecraft Storage Stand	-
Personnel Access Stand	10
HANDLING EQUIPMENT	
Handling Fixtures	
Spacecraft Transporter	11
Spacecraft Protective Cover and Devices	12
Transporter Prime Mover	13
Spacecraft - P/V Lift Sling	14
Retropropulsion Unit Handling Fixture & Sling	15
PSP Handling Fixture and Propulsion Sling	16
HANDLING EQUIPMENT	
Handling Fixtures (Continued)	
Solar Panel Handling Fixture and Sling	17
High-Gain Antenna Handling Fixture and Sling	18
Medium-Gain Antenna Handling Fixture and Sling	19
Electronic Chassis Handling Fixture	20
Nose Fairing Simulator Handling Sling	21
Spacecraft Protective Cover Purger	-
Precision Vertical Load Balancer	22
Magnetic Mapping	
Magnetic Mapping Fixture	23
Magnetic Mapping Track	24
Magnetic Mapping Transfer Carriage	25
Transfer Carriage Rails	-
Test Magnetometer Support	26
SHIPPING EQUIPMENT	
Spacecraft Shipping Cover	28
OSE Shipping Containers	-
SPECIAL PURPOSE SUPPORT EQUIPMENT	
Weight and CG Determination Equipment*	27
Free-Mode Test Equipment	29
Thermal Vacuum Test Fixtures	30
Vibration Test Fixtures and Suspension	31
Acoustic Test Fixture	32
Separation Test Fixture and Suspension	33
Static Firing Test Fixture	34
EMI Test Equipment*	-
Environmental Control Equipment	-

*Capability included in Spacecraft Assembly Fixture (Item 4).

A description of functional and design requirements and design concepts for each end item appears in the following text.

2.0 ASSEMBLY EQUIPMENT

2.1 Simulators

2.1.1 Capsule Interface Simulator (Item 1 of Figure 4-1 in VC270FD100). The function of this item is to present all the mechanical, dynamic, and thermal influences expected of a Flight Capsule.

2.1.1.1 Design Requirements. This item will duplicate exactly the Flight Capsule interface with the Spacecraft. Duplication of the Flight Capsule weight and center of gravity is required.

2.1.1.2 Design Concept. The simulator will be a structurally rigid, metallic item duplicating the Flight Capsule except for operative electrical devices. The mechanical interfaces will be duplications of bearing areas, fastening features, and instrumentation connections. The simulated thermal interface will match the radiation and conductive influences of the Flight Capsule and will be removable from the basic simulator for the thermal-vacuum tests which do not require full simulator configuration.

2.1.2 Spacecraft Retropropulsion Subsystem Interface Simulator (Item 2 of Figure 4-1 in VC270FD100). The function of this item is to present all the mechanical, dynamic, and thermal influences expected of a flight subsystem.

2.1.2.1 Design Requirements. The simulator will provide interface compatibility with the Spacecraft and assembly handling devices. Duplication of the subsystem weight and center of gravity are required.

2.1.2.2 Design Concept. The simulator will be a structurally rigid item. It will duplicate the subsystem, with the exception of the engine fuel and operative electrical devices. The mechanical interfaces will be duplications of the bearing areas, fastening features, and instrumentation connections. The thermal interface will match the radiation and conductive influences of the subsystem. Magnetic characteristics of the simulator will be held to that of a subsystem without the exact fuel.

2.1.3 Nose Fairing Simulator (Item 3 of Figure 4-1 in VC270FD100). This item functions as a means of validating the Planetary Vehicle to Nose Fairing Section connections and evaluating the mating procedure.

2.1.3.1 Design Requirements. The simulator will provide interface compatibility with the Planetary Vehicle. Functional compatibility with the Planetary Vehicle on its assembly device must also be considered.

Duplication of the Nose Fairing Section height, inside envelope limits, and field joint are required.

2.1.4 Science Subsystem Test Fixture. The function of this item is to provide an integrating structure for the Science Subsystem during the JPL Science Subsystem test program.

2.1.4.1 Design Requirement. The fixture will have the capability of supporting all parts of the Science Subsystem. The fixture will provide interface compatibility with all the Science Subsystem Devices.

2.1.4.2 Design Concept. The simulator will consist of those portions of the Spacecraft Bus which form a mechanical interface with the Science Subsystem, duplicating bearing areas, mechanical features, and instrumentation connections. Specific areas of attention are the locations of the Spacecraft body-mounted science instruments, the planetary scan platform, the data automation equipment, and the science power switching electronics. Should thermal influence of the Spacecraft be a factor during the Science Subsystem Test Program, the radiation and conduction interfaces with the Spacecraft will be duplicated.

2.2 Alignment Equipment.

2.2.1 Guidance and Control Subsystem Alignment Set. The Guidance and Control (G&C) Subsystem alignment set will be used to check the angular relationships of the mounting surfaces (on the Spacecraft) for the various sensors and components of the G&C Subsystem to Reference Planes A and B. Refer to Table 4-2 of VC270FD100 for alignment tolerances and definition of the reference planes. The alignment set will provide the necessary fixtures and targets which, when used with the general-purpose instruments of the Optical Tooling Kit, will facilitate the alignment operations.

2.2.1.1 Design Requirements. Requirements for the individual alignment fixtures are as follows:

a. Canopus Sensor Fixture - This fixture shall provide a flat test surface which is parallel to the mounting surface within 30 seconds. It shall also have a front surface mirror (of optical tooling quality) mounted at 90 degrees ± 1 arc minute from the indexing surface.

b. Sun Sensor Assembly Fixture - Same as (a).

c. Acquisition Sun Sensor Fixture - The fixture shall provide a flat test surface parallel to the mounting surface within 2 arc minutes. It shall have a mirror mounted as described in a.

d. Gyro and Accelerometer Package Fixture - Same as (c).

e. Attitude Control Nozzle Block Fixture - This fixture shall be a flat plate on which is mounted a front surface mirror (of optical tooling quality). The parallelism between the front surface of the mirror and the rear (or mounting) surface of the fixture shall be ± 30 arc minutes or better.

f. Attitude Control Nozzle Fixture - The nozzle fixture shall be designed such that it will slip over the end of the pitch or yaw nozzles and reference their geometric center lines (i.e. its surface will be perpendicular to the center line within 30 arc minutes). On this surface shall be mounted a circular spirit level of 1-degree sensitivity.

g. Planetary Scan Platform Fixture - While requirements are not defined on this item, it is assumed that the fixture type and accuracy will fall in the same category as the Canopus Sensor Fixture discussed in (a).

2.2.1.2 Design Concept. The design approach to the alignment of the above mentioned items of the G&C Subsystem is based on the assumption that the component mounting surfaces are parallel to Reference Plane A (see Table 4-2 of VC270FD100). An alignment fixture will be placed on the mounting surface and spirit level or clinometer (from the Optical Tooling Kit) will be used to determine the need for shims. The mounting surface will be shimmed and rechecked until it falls within the alignment tolerances established for the task. This is to be done about two axes. The third axis of alignment (or indexing about the Spacecraft Roll Axis) will be checked by optical tooling equipment and procedures. Targets will have been established on the Spacecraft Assembly Fixture to be the reference for the pitch axis and the yaw axis. One theodolite will be positioned to reference the pitch or yaw axis, and a second theodolite will be autocollimated from the mirror on the alignment fixture in question. By collimating the two theodolites and comparing the angles read on their horizontal circles, the pointing accuracy of the indexing stop can be readily determined. The indexing stop must have adjustment capability to correct for any "out of specification" condition.

2.2.2 Solar Panel Alignment Set. The Solar Panel Alignment Set will be used to check the alignment of each solar panel relative to the spacecraft adapter separation plane. If an individual panel is not parallel to the reference plane, which is horizontal, it will be repositioned by shimming. The alignment set will facilitate this procedure by giving an angular displacement from the horizontal which can be converted to actual shim thickness.

2.2.2.1 Design Requirements. Since the alignment tolerance is ± 1 degree, the instrument accuracy should be in the order of ± 6 arc minutes. The instrument should interface with the panel in such a way that the solar cells will not be degraded. Because of the orientation of the Spacecraft, the instrument must be of a type that will function upside down, that is, the readout must be 180 degrees in either direction.

2.2.2.2 Design Concept. The instrument used will be a clinometer, similar to the Hilger & Watts Model C, which has a reading accuracy of 1 minute. For reading convenience, the dial can be locked and the clinometer removed from the surface being measured. A direct reading can be made of any angle in any position (a range of 0 to 360 degrees). Provision will be made for an accessory fixture which, when mounted on the clinometer, will accurately position the clinometer a fixed distance from the panel surface, thus bridging the solar cells. Hand-holding the instrument should be adequate for the tolerance involved in this procedure.

2.2.3 Science Payload Alignment Set. The equipment making up the Science Payload, while not specifically defined, is generally known. Those items requiring alignment would be the primary sensors for planetary/interplanetary environment observations which will be located in various places on the spacecraft. Among this group would be such experiments as magnetometers and energetic radiation detectors. It will be required to align these and/or other sensors to the vehicle axes in much the same manner as the sensors of the Guidance and Control Subsystem.

2.2.3.1 Design Requirements. Assuming alignment tolerances of the same order of magnitude, the equipment provided in this alignment set will be similar to that of the G&C Subsystem Alignment Set.

2.2.3.2 Design Concept. The basic approach is to place a special alignment fixture on the mounting surface in place of the sensor. This fixture is used for all alignment procedures until the shimming is completed and the indexing stop is adjusted. In this approach the handling of the actual flight hardware is minimized.

2.2.4 Medium-Gain Antenna Alignment Fixture. The Medium-Gain Antenna Alignment Fixture will be used to check the angular relationship of the medium-gain antenna mounting surface (on the Spacecraft) to Reference Planes A & B. Refer to Table 4-2 of VC270FD100 for alignment tolerances and reference plane definitions. Items from the Optical Tooling Kit will be utilized.

2.2.4.1 Design Requirements. The angle between the antenna mounting surface and reference Plane A shall be built into the fixture to an accuracy of ± 4 arc minutes. The other angular relationship will be checked with optical tooling instruments and procedures. An optical tooling quality front surface mirror will be mounted on the fixture for autocollimation purposes.

2.2.4.2 Design Concept. The fixture will have the mirror so mounted that its surface is vertical (when the fixture is mounted on the Spacecraft). A theodolite, independently supported, will be autocollimated to the mirror. The angle between this line of sight and the horizontal will be read on the vertical scale of the theodolite. This is the misalignment with Reference Plane A. A second theodolite, similarly supported and located with its optical axis parallel to the Spacecraft yaw axis, will be used. The two instruments will be collimated (their optical axes made parallel) and the angles read on their horizontal scales. The result of this procedure is a triangle in the horizontal plane with two angles known. A simple calculation yields the third angle, which is the angle the antenna axis makes with Reference Plane B. Comparison with the desired angle can now be made and the misalignment found. With the angular errors known, shim thickness can be determined. After the fixture is shimmed, the procedure is repeated to verify alignment.

2.2.5 Capsule Interface Alignment Fixture. The Capsule Interface Alignment Fixture, in conjunction with the instruments from the Optical Tooling Kit, will be used to assure the pointing accuracy of the Flight Capsule. This will be done by checking the parallelism of the capsule mating surface (of the Flight Spacecraft) to Reference Plane A. The tolerance on this parallelism is 30 arc minutes half angle.

2.2.5.1 Design Requirements. The fixture should have a reference surface (for placement of an instrument) that is parallel to its spacecraft interface within 3 arc minutes. Reading accuracy of the instruments should be 1 arc minute (or better).

2.2.5.2 Design Concept. The alignment fixture will contact the Spacecraft at a minimum of four places and will be structurally rigid. A flat surface will be provided parallel (by either machining or shimming) to the Spacecraft interface of the fixture. This surface will be of convenient location and size so that a clinometer can be supported for the alignment procedure.

2.2.6 Midcourse Engine Alignment Set. The Midcourse Engine Alignment Set will be used to check the perpendicularity of the center lines of the four nozzles to Reference Plane A. See Table 4-2 of VC270FD100 for alignment tolerances and definition of the reference planes. The alignment set will provide fixtures and targets to give the capability for optical alignment procedures.

2.2.6.1 Design Requirements. The fixture shall have a front surface mirror (of optical tooling quality) mounted such that the surface is perpendicular to the center line of the nozzle within 15 arc seconds.

2.2.6.2 Design Concept. The design approach is to provide plug targets which will fit inside the nozzles to reference the nozzle center line. In order to achieve the 15-arc-second accuracy the mirror will be mounted in an adjustable mount. A calibration fixture will be provided in the Optical Tooling Kit. Before being used, the mirror on the alignment fixture will be adjusted until it is normal to the fixture center line (and therefore also the nozzle center line). This procedure will entail the use of autocollimation. As shown in Figure 2-1, a theodolite and pentaprism will be used to check the angular alignment of the nozzles to the reference plane. Two alignment checks will be made with the axes of the theodolites 90 degrees apart.

2.2.7 Retroengine Alignment Set. The Retroengine Alignment Set will be used to check the perpendicularity of the retropropulsion engine center line to Reference Plane A. Refer to Table 4-2 of VC270FD100 for alignment tolerances and definition of the reference planes. In addition to the angular alignment function, the Retroengine Alignment Set will provide the capability to check the radial displacement of the nozzle center line from the spacecraft roll axis.

2.2.7.1 Design Requirements. Same as Section 2.2.6.1. In addition, a displacement target shall be provided which must lie on the center line of the alignment fixture.

2.2.7.2 Design Concept. The same design approach will be taken here as described for the Midcourse Engine. The location of the displacement target will be handled in the same manner as the angular adjustment of the mirror. A side viewing plug target will be mounted on the fixture for this task (see Figure 2-2). For both the nozzle alignment operation and the target adjustment, a jig transit with an optical micrometer will be used.

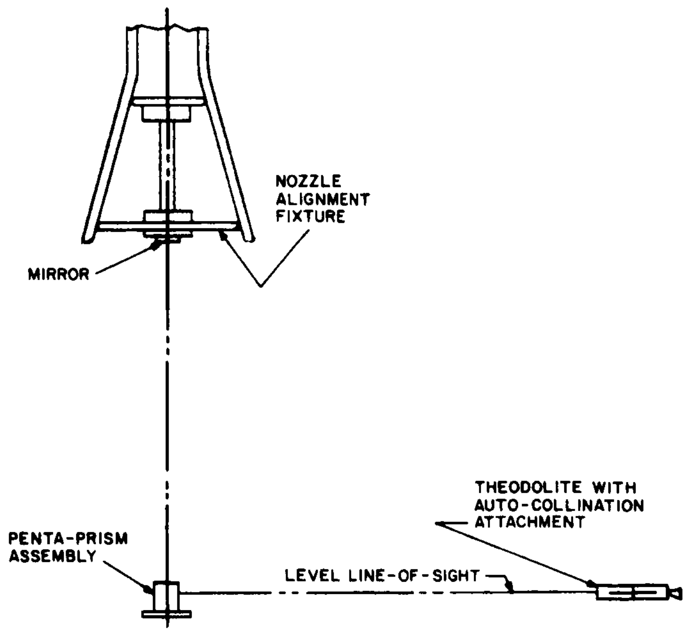

Figure 2-1. Angular Alignment

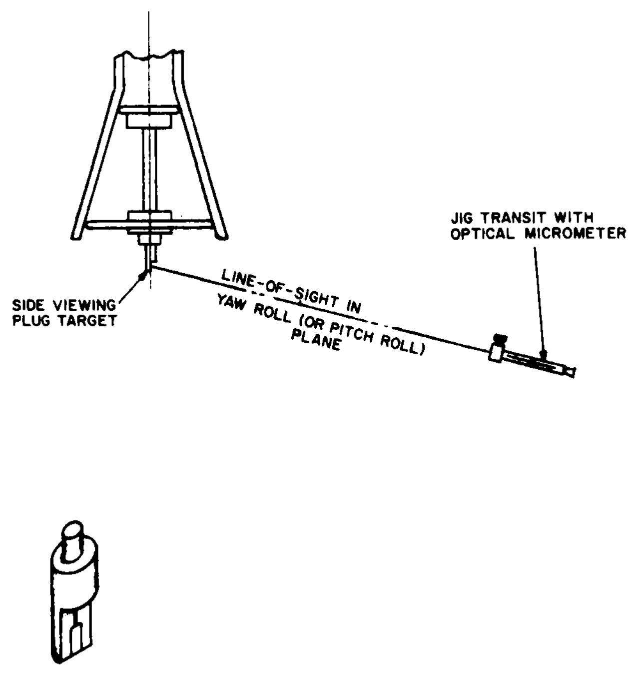

Figure 2-2. Linear Displacement Alignment

2.2.8 Optical Tooling Kit. The function of the tooling kit is to provide instruments and equipment for measuring angles and small linear displacement for leveling and targeting and for calibrating the alignment fixtures.

2.2.8.1 Design Requirements. The equipment selected should be of proper accuracy and sensitivity to perform the alignment tasks defined in Table 4-2 of VC270FD100. Stands should support instruments so that their line of sight will be at the elevation required for each particular task.

2.2.8.2 Design Concept. The Optical Tooling Kit, for the greater part, will be made up of general-purpose alignment equipment of commercially available types. Instruments will be selected so that required functions are either inherent in their design or in readily available standard accessory equipment. Special calibration fixtures and reference targets will also be included in the kit. It will include, but not be limited to, the following equipments:

a. Optical Instruments

   1. Theodolites - Read angles in both vertical and horizontal planes

   2. Jig transits - Similar to above but without angle readout.

b. Accessory Equipment

   1. Angular Alignment Accessories - Autocollimation and autoreflection attachments and pentaprisms.

   2. Linear Measuring Accessories - Optical micrometers (range ± 0.100 inch) and optical tooling scales.

   3. Instrument Stand - With lateral adjusters.

c. Leveling Instruments - Spirit levels, clinometers and remote reading electronic levels (for leveling the Spacecraft Assembly Fixture).

d. Positioning Equipment - Lead screw slide assemblies with micrometers.

e. Target Equipment - Plug targets (both end and side viewing), mirrors*, and porro prisms.

f. Special Equipment - Calibration fixtures for both the midcourse and retronozzle targets.

2.2.9 High-Gain Antenna Alignment Set. The High-Gain Antenna Alignment Set will be used to check the perpendicularity of the antenna axis to Reference Planes A and B (two separate checks). See Table 4-2 of VC270FD100 for alignment tolerances and definition of the reference planes. These alignment checks will be performed with the antenna mounted on the Spacecraft.

*A magnetic mirror will be included in the kit for use in aligning the rate table (part of the G&C Subsystem OSE) to Polaris.

2.2.9.1 Design Requirements. The antenna shall be checked in two positions, null position and rotated 90 degrees. In these positions the antenna axis shall be checked for parallelism to the spacecraft roll axis and pitch axis. An alignment mirror shall be mounted on the antenna fixture such that its front surface is parallel to the mounting surface of the fixture within 1 arc minute.

2.2.9.2 Design Concept. A reference surface will be provided on the antenna for mounting the antenna alignment fixture. With the antenna in the null position, a theodolite will be autocollimated to the mirror, and the deviation from the vertical will be read. This angle is the misalignment from the roll axis. In similar manner, the second position can be checked with a theodolite located so that its optical axis is parallel to the pitch axis (i.e., perpendicular to Reference Plane B) and the misalignment determined.

2.3 Assembly Fixtures.

2.3.1 Spacecraft Assembly Fixture (Item 4 of Figure 4-1 in VC270FD100). The fixture will be used for holding the Spacecraft and Planetary Vehicle during final assembly, inspection, repair, and tests.

2.3.1.1 Design Requirements. The fixture will be primarily of aluminum construction with capability of supporting a load of 30,000 pounds. It will be capable of supporting the Spacecraft with or without the Retropropulsion Unit, as well as the complete Planetary Vehicle with the roll axis vertical.

The assembly fixture will be compatible with Spacecraft-Planetary Vehicle Lift Sling (Item 14 of Figure 4-1 in VC270FD100), System Test Complex (STC) and Launch Control Equipment (LCE) cables, environmental control enclosure, nose fairing simulator and personnel access stands. The assembly fixture will contain leveling features for leveling the Spacecraft and/or Planetary Vehicle to within ± 6 arc seconds. This feature is needed to permit alignment of Spacecraft subsystems. The assembly fixture will contain all necessary reference markings, etc, required for proper optical alignment of subsystems. The assembly fixture will contain weight-indicating devices, such as load cells for weight and cg indication of the Spacecraft. Four load cells at 10,000 pounds each will be used. The fixture will contain a mechanism for blocking out cells when weighing is not required. A built-in load cell calibration unit will be used.

2.3.1.2 Design Concept. The fixture will be a multileg structure, approximately 6 feet high and 20 feet in diameter, and made up of standard structural shapes with the necessary cross-bracing required for rigidity and strength. The structure will be essentially openwork to permit maximum access to the Spacecraft. Floor loading will be maintained at acceptable levels so that a special foundation for the assembly fixture will not be required. The uppermost section of the assembly stand will contain the load cells for determining weight and lateral cg of the Spacecraft and Planetary Vehicle. This capability will permit close weight and cg control during modification and/or substitution of Spacecraft Subassemblies. The necessary readout device will be placed adjacent to the assembly fixture.

The upper section of the assembly fixture will contain a device for leveling the Spacecraft for the purpose of alignment of subsystems. This leveling device will consist of electric motor-driven mechanical jacks with the necessary controls and readouts. Rough leveling will be done automatically; fine leveling will be accomplished by manual operation of controls. The required leveling pads, first-surface mirror mountings and other features required for alignment will be designed into the assembly fixture. Direct line of sight will be maintained at all times. A personnel workstand will surround the assembly fixture. The assembly fixture will be designed such that it is compatible with:

a. Nose Fairing Simulator
b. Alignment Equipment
c. Thermal Control Enclosure
d. Capsule Simulator
e. Flight Capsule
f. Retropropulsion Interface Simulator
g. Flight Retropropulsion Unit
h. Flight Nose Fairing
i. Free Mode Solar Simulators

2.3.2 High-Gain Antenna Assembly Fixture (Item 5 of Figure 4-1 in VC270FD100). The fixture will be used for holding the High-Gain Antenna during checkout whenever disassembled from the Spacecraft.

2.3.2.1 Design Requirements. This fixture will be primarily of aluminum construction with capability of supporting and positioning the 75-pound antenna and handling fixture during testing in which the antenna is not required to be physically assembled to the Spacecraft. The assembly fixture will be compatible with the Antenna Handling Fixture and Sling (Item 18 of Figure 4-1 in VC270FD100) and electrical OSE.

2.3.2.2 Design Concept. The fixture will be a four-leg structure, approximately 6 feet high, 5 feet wide, and 10 feet long, with all-swivel castors. A gimbal device, which interfaces with the handling fixture (Item 18 of Figure 4-1 in VC270FD100), will be provided on the assembly fixture. This device will permit angular positioning of the antenna for checkout and repair. Locking devices will be provided for both the gimbal mechanism and the castors.

2.3.3 Medium-Gain Antenna Assembly Fixture (Item 6 of Figure 4-1 in VC270FD100). The fixture will be used for holding the Medium-Gain Antenna during checkout whenever disassembled from the Spacecraft.

2.3.3.1 Design Requirements. This fixture will be primarily of aluminum construction with capability of supporting and positioning the 7-pound antenna and handling fixture (Item 19 of Figure 4-1 in VC270FD100) during testing that does not require the antenna to be physically assembled to the Spacecraft. The assembly fixture will be compatible with the .antenna Handling Fixture and Sling (Item 19 of Figure 4-1 in VC270FD100) and electrical OSE.

2.3.3.2 Design Concept. The fixture will be a three-leg structure, approximately 3 feet high and 3 feet square, with all-swivel castors. A gimbal device, which interfaces with the handling fixture and sling (Item 19 of Figures 4-1 and 5-1 in VC270FD100), will be provided on the assembly fixture. This device will permit angular positioning of the antenna for checkout and repair. Locking devices will be provided for both for gimbal mechanism and the castors.

2.3.4 Solar Panel Assembly Fixture (Item 7 of Figure 4-1 in VC270FD100). The fixture will be used for holding Solar Panel sections during checkout, prior to installation on the Spacecraft.

2.3.4.1 Design Requirements. This fixture will be constructed primarily of aluminum with capability of supporting and positioning the 10-pound, 4-foot by 5-foot Solar Panels and handling fixture (Item 17 of Figure 4-1 in VC270FD100). The assembly fixture will be compatible with the Solar Panel Handling Fixture and Sling and associated electrical OSE.

2.3.4.2 Design Concept. The panel assembly fixture will be a four-leg structure, approximately 2 feet high and 4 feet square, with all swivel castors. This fixture will support the Solar Panel Handling Fixture and Sling (Item 17 of Figure 4-1 in VC270FD100) for checkout and repair of individual solar panels.

2.3.5 Planet Scan Platform Assembly Fixture (Item 8 of Figure 4-1 in VC270FD100). The Planet Scan Platform (PSP) Assembly Fixture should be used for holding the PSP whenever it is disassembled from the Spacecraft.

2.3.5.1 Design Requirements. The fixture should be constructed primarily of aluminum with capability of supporting the 250-pound platform in correct orientation during tests that do not require the platform to be physically assembled to Spacecraft. The fixture should be compatible with the PSP Handling Fixture and Sling (Item 16 of Figure 4-1 in VC270FD100), the Spacecraft and associated electrical OSE.

2.3.5.2 Design Concept. The PSP Assembly Fixture should be a four-leg structure, approximately 2 feet high and 4 feet square, with all-swivel castors. This fixture should support the PSP Handling Fixture and Sling (Item 16 of Figure 4-1 in VC270FD100) during checkout of the PSP. The assembly fixture should provide angular positioning of the PSP for individual checkout and repair.

2.3.6 Personnel Access Stand (Item10 of Figure 4-1 in VC270FD100). The Personnel Access Stand is required to permit personnel to work around the Spacecraft and Planetary Vehicle during various phases of assembly and test.

2.3.6.1 Design Requirements. The access stand will be of aluminum construction with capability of permitting access to the Spacecraft while supported on the Spacecraft Assembly Fixture (Item 4 of Figure 4-1 in VC270FD100) and when the Spacecraft is supported in the special-purpose test fixtures. Portions of the platform will be movable to permit the spacecraft to be installed or removed without creating the requirement for excessive crane hook height. Castors will be provided for workstand maneuverability where required. The remainder of platforms will be semipermanent; however, the entire platform will have capability of being readily disassembled for shipment. Adequate safety features will be provided to protect the Spacecraft and personnel. The platforms will be compatible with the Spacecraft, electrical OSE and other mechanical OSE.

2.3.6.2 Design Concept. The stand will be a semipermanent multilevel structure, approximately 35 feet square, consisting of standard structural shapes. Both movable and stationary platforms will be provided. Movable platforms will contain vertical as well as radial adjustment. Access to the stand will be by stairways and ladders. Equipment elevators will be included in the workstand design to facilitate the movement of equipment between different platform levels. These elevators will be motorized and will include the required safety interlocks.

The stand will contain adequate areas for toolboxes, special equipment, electrical outlets for drop cords, and power tools. Supports and trays will be included for cables, power tool air lines, GN2 and He, and cooling air lines.

2.3.7 Electronic Chassis Assembly Fixture. The fixture will be used for holding electronic bays or modules during checkout, prior to installation into the Spacecraft.

2.3.7.1 Design Requirements. This fixture will be constructed primarily of aluminum and will be capable of supporting modules and handling fixtures (Item 20 of Figure 4-1 in VC270FD100). The assembly fixture will be compatible with the Electronic Module Handling Fixture and electrical OSE.

2.3.7.2 Design Concept. The Electronic Module Assembly Fixture will be a four-leg structure approximately 3 feet high and 4 feet square with all-swivel castors. The fixture will contain a tilt mechanism so that modules can be positioned for proper checkout and repair. This fixture will support the handling fixture (Item 20 of Figure 4-1 in VC270FD100) through the necessary interfaces.

2.3.8 Retropropulsion System Assembly Fixture (Item 9 of Figure 4-1 in VC270FD100). The Retropropulsion System Assembly Fixture is required to handle the Retropropulsion Subsystem during assembly with, or removal from, the Spacecraft.

2.3.8.1 Design Requirements. The fixture will be primarily of aluminum construction with capability of handling the 15,000-pound Retropropulsion Unit during inspection, checkout and test. Further, it will have capability to permit assembly of the Spacecraft to the Retropropulsion Unit by lowering the Spacecraft over the unit.

The fixture will permit access to all parts of the Retropropulsion Unit. The fixture will be compatible with the Retropropulsion Subsystem, the Spacecraft, the Retropropulsion Subsystem Lifting Slings, and the work platforms.

2.3.8.2 Design Concept. The assembly fixture will be a multileg structure, approximately 7 feet high and 10 feet square, and made up of standard structural shapes with the necessary crossbracing required for strength. The structure will be designed so as to permit the Spacecraft to be lowered over the Retropropulsion Unit for attachment. The assembly fixture will permit complete access to the Retropropulsion Unit for checkout and repair. The interface between the assembly fixture and Retropropulsion Unit will not be the same interface that is required for Spacecraft Retropropulsion Unit Assembly.

2.3.9 Assembly Equipment Spacecraft Storage Stands, Nonflight. The storage stands will be required to support the Thermal Control Model (TCM), Development Model, Structural Test Model (STM), and the Proof Test Model (PTM) vehicles in the storage areas.

2.3.9.1 Design Requirements. The stands will be simplified structures with capability of supporting the various test vehicles when they are not in use. These stands will not necessarily permit access to the various subsystems.

2.3.9.2 Design Concept. The stands will be multileg structures made up of standard, commercially available materials; wood will be utilized wherever practical. Adequate interfaces will be provided so as not to cause damage to the test models.

3.0 HANDLING EQUIPMENT

3.1 Handling Fixtures

3.1.1 Spacecraft Transporter (Item 11 of Figure 4-1 in VC270FD100). The transporter will be used to provide ground mobility between test facilities in-house and between assembly and checkout facilities in the field. Further, the transporter will serve a shipping function between the factory and field.

3.1.1.1 Design Requirements. The transporter will be constructed primarily of aluminum with the capability to transport and protect the Spacecraft from external environments. An airtight cover, together with environmental control, will be part of the transporter used for moving the Spacecraft from clean-room areas. Maximum speed of the transporter will be 15 miles per hour. The transporter will be compatible with the Spacecraft/Planetary Vehicle Lift Sling (Item 14 of Figure 4-1 in VC270FD100) and the Spacecraft. Features will be provided for towing the transporter at either end. The transporter with cover will be approximately 360 inches long, 264 inches wide and 208 inches high. The transporter will contain an air and electrical braking system that is compatible with commercially available prime movers.

3.1.1.2 Design Concept. The transporter will be constructed of standard structural shapes forming a low-bed trailer configuration. Maximum height of bed will be 2 feet; overall transporter dimensions will be 22 feet wide by 30 feet long.

The transporter will have a set of small, steerable pneumatic wheels at each end; these wheels will be capable of being locked in the forward position. The transporter will also have a "gooseneck" fifth-wheel towing attachment that will permit towing by commercially available truck-tractors; air and electric brakes will be provided. Environmental control (self-contained) will be provided as part of the transporter. A Government Furnished Equipment (GFE) vehicle can be modified for this use if such action is found to be practical.

3.1.2 Spacecraft Protective Covers and Devices (Item 12 of Figure 4-1 in VC270FD100). Protective covers and devices are required for securing the Spacecraft and subsystems from contamination and damage when these items are being moved between or within clean areas and to protect the solar panels from physical damage.

3.1.2.1 Design Requirements. Covers will be made of conductive material. They will be contoured when possible and will be marked with clearly designated use areas.

3.1.2.2 Design Concept. The Spacecraft and subsystem covers will be made of acceptable flexible material with the required zippers, grommets, and tie-downs. Provisions will be made for hoisting covers by overhead crane to ease handling. The solar panel protective covers will be made of transparent material contoured to adequately fit each solar panel segment. These covers will contain all necessary fasteners and brackets for securing to the Spacecraft.

3.1.3 Transporter Prime Mover (Item of Figure 4-1 in VC270FD100). The Prime Mover will be used for pulling the Spacecraft Transporter (Item 11 of Figure 4-1 in VC270FD100) between the Satellite Checkout Facility (SCF) and Explosive Safe Area (ESA) at Eastern Test Range (ETR).

3.1.3.1 Design Requirements. The prime mover should be a GFE all-electrical vehicle. This vehicle should be modified to permit effective cleaning in keeping with program cleanliness requirements.

3.1.3.2 Design Concept. The prime mover vehicle should be a commercially available, heavy-duty vehicle containing provisions for either electric or air braking of transporter.

3.1.4 Spacecraft and Planetary Vehicle Lift Sling (Item 14 of Figure 4-1 in VC270FD100). The Spacecraft and Planetary Vehicle Lift Sling will be used to lift the Spacecraft and Planetary Vehicle with simulated Capsule.

3.1.4.1 Design Requirements. The sling will be made of aluminum and stainless steel with capability for lifting the Spacecraft and Planetary Vehicle. The sling will be compatible with the Spacecraft, Simulated Capsule, and personnel workstands. The sling will be capable of supporting 30,000 pounds. The sling will be capable of rotating the Spacecraft and Planetary Vehicle with simulated Capsule from vertical position (roll axis vertical) to the horizontal position (roll axis horizontal) for support of the STM separation tests.

3.1.4.2 Design Concept. The lift sling will be constructed of standard structural shapes with the necessary attachment points for stainless wire rope cables. Quick-release pins will be used to attach the wire rope at the interface points. A special lift ring that will interface with the Spacecraft and Planetary Vehicle will be part of the lift sling assembly; this ring is needed to assure correct loading of the structures.

3.1.5 Retropropulsion Unit Handling Fixture and Sling (Item 15 of Figure 4-1 in VC270FD100). The fixture and sling are required for lifting the Retropropulsion Unit and its simulator.

3.1.5.1 Design Requirements. The fixture and sling will be primarily of aluminum construction with capability positioning the 15,000-pound Retropropulsion Unit as required. This fixture and sling will be compatible with the Propulsion Unit Assembly Fixture (Item 9 of Figure 4-1 in VC270FD100) and shipping container.

3.1.5.2 Design Concept. The sling will be constructed of a standard structural shape with the necessary attachment points for wire rope cables. Quick-release pins will be used for attachment of wire rope at the interface points. A special bracket, that will be part of the lift sling assembly, is needed to assure correct loading of the Retropropulsion Unit.

3.1.6 Planet Scan Platform Handling Fixture and Sling (Item 16 of Figure 4-1 in VC270FD100). The Planet Scan Platform (PSP) Handling Fixture and Sling is required to lift and handle the PSP during test, installation, removal and shipping.

3.1.6.1 Design Requirements. The fixture will be of aluminum construction with adequate features for hoist handling and positioning the 250-pound platform. The fixture will also include features for proper orientation during testing. The fixture will be compatible with PSP Assembly fixture (Item 8 of Figure 4-1 in VC270FD100), PSP shipping containers, hoists, electrical OSE, Spacecraft structure, and other mechanical OSE.

3.1.6.2 Design Concept. The PSP Handling Fixture and Sling will be an open framework made up of standard structural shapes. The sling will have correct interface attachments for proper handling of package. Handles will be provided for safe manual manipulations while hoisting and positioning with overhead crane.

3.1.7 Solar Panel Handling Fixture and Sling (Item 17 of Figure 4-1 in VC270FD100). The Solar Panel Handling Fixture and Sling is required to lift and handle the solar panels during test, installation, removal, and shipping.

3.1.7.1 Design Requirements. The fixture will be of aluminum or plastic construction and have the capability of permitting safe handling of a solar panel. Further, the fixture will have the capability to permit the necessary orientation for individual panel checkout while physically detached from Spacecraft. The fixture will be compatible with shipping containers, Spacecraft, electrical OSE, and other mechanical OSE, as well as the Solar Panel Assembly Fixture (Item 7 of Figure 4-1 in VC270FD100).

3.1.7.2 Design Concept. The sling will be of open framework construction made of standard structural shapes. It will contain the necessary interface points for connection to each panel. Handles will be provided for safe manual manipulations.

3.1.8 High-Gain Antenna Handling Fixture and Sling (Item 18 of Figure 4-1 in VC270FD100). The fixture will be used for assembly of the High-Gain Antenna to the Spacecraft.

3.1.8.1 Design Requirements. The fixture will be of aluminum construction with adequate features for hoist handling and positioning the 75-pound, 10-foot diameter antenna. The fixture will also include features for proper orientation during testing that does not require the antenna to be physically assembled to the Spacecraft. The fixture will be compatible with antenna assembly fixture (Item 5 of Figure 4-1 in VC270FD100), antenna shipping containers, hoists, electrical OSE, Spacecraft structure, and other mechanical OSE.

3.1.8.2 Design Concept. The High-Gain Antenna Fixture and Sling will be an open framework made up of standard structural shapes. The fixture and sling will have the correct interface points for proper handling of antenna. Handles will be provided for safe manual manipulations while hoisting and positioning with overhead crane.

3.1.9 Medium-Gain Antenna Handling Fixture and Sling (Item 19 of Figure 4-1 in VC270FD100). The fixture will be used for assembling of the Medium-Gain Antenna to the Spacecraft.

3.1.9.1 Design Requirements. The fixture will be of aluminum construction with adequate features for hoist handling and positioning the 7-pound antenna. The fixture will also include features for proper orientation during testing. The fixture will be compatible with Medium-Gain Antenna Assembly Fixture (Item 6 of Figure 4-1 in VC270FD100), antenna shipping containers, hoists, electrical OSE, Spacecraft structure, and other mechanical OSE.

3.1.9.2 Design Concept. The Medium-Gain Antenna Fixture and Sling will be an open framework made up of standard structural shapes. The fixture and sling will have the correct interface points for proper handling of antenna. Handles will be provided for safe manual manipulations while hoisting and positioning with overhead crane.

3.1.10 Electronic Chassis Handling Fixture (Item 20 of Figure 4-1 in VC270FD100). The fixture will be used for lifting and handling electronic modules either manually or by sling.

3.1.10.1 Design Requirements. This fixture will be constructed primarily of aluminum with capability of positioning electronic modules in proper positions as required for assembly, testing, and shipping. The handling fixture will be compatible with the Electronic Module Assembly Fixture, Spacecraft, electrical OSE, and other mechanical OSE.

3.1.10.2 Design Concept. The fixture will be an open framework design, fabricated from standard structural shapes. Attachment points will be provided for ease of handling the modules. A universal-type fixture will be used.

3.1.11 Simulated Nose Fairing Handling Sling (Item 21 of Figure 4-1 in VC270FD100). The sling will be used for handling of the Simulated Nose Fairing.

3.1.11.1 Design Requirements. The sling will be of aluminum construction with adequate features for hoisting and positioning the Simulated Nose Fairing for checkout with the Planetary Vehicle. The sling will be compatible with overhead hoists in the checkout areas.

3.1.11.2 Design Concept. The handling sling will be constructed of a standard structural shape with two-point attachment for connecting to the Simulated Nose Fairing. This sling will duplicate, as close as practical, the sling supplied to Kennedy Space Center by the manufacturer.

3.1.12 Spacecraft Protective Cover Purger. The purger will be used to replace the air entrapped by the cover of the Spacecraft transporter with dry nitrogen.

3.1.12.1 Design Requirements. The purger will provide dry nitrogen at 4 psi. Also, the purger will provide vacuum pump capacity for evacuating the transporter cover within 15 minutes.

3.1.12.2 Design Concept. The purger will be a self-contained unit consisting of a vacuum pump, valves, gages, hoses, connections, and other necessary equipment. Commercially available dry nitrogen bottles will be connected to the purger through regulators, and manifolds. Relief valves and other safety devices will be used where necessary. In operation, the vacuum pump will evacuate the cover prior to introducing the dry nitrogen; this procedure will assure complete purging of air.

3.1.13 Vertical Load Balancer (Item 22 of Figure 4-1 in VC270FD100). The balancer will be used to raise and lower Spacecraft and other loads (up to 30,000 pounds) within an increment of 0.003 inches.

3.1.13.1 Design Requirements. The balancer should be compatible with overhead cranes and all lifting and handling devices. It is required to provide precision linear movement for fast, accurate load positioning.

3.1.13.2 Design Concept. The balancer will be a commercially available, hydraulically operated instrument that has the correct interfaces for installation between an overhead crane hook and the lifting devices. Control of the unit can be accomplished through handles located on the balancer or by an auxiliary remote balancer control. One commercially available balancer is known by the trade name of "Hydra-set."

3.2 Magnetic Mapping Equipment

3.2.1 Magnetic Mapping Fixture (see Figure 5-3 of VC270FD100). The functions of this item are to: (a) support the Planetary Vehicle during perming, deperming, and magnetic characteristics determination; (b) provide the means for moving the Planetary Vehicle about the magnetometer during magnetic characteristic determination; and (c) rotate the supported Planetary Vehicle from the roll-axis vertical attitude to roll-axis horizontal attitude.

3.2.1.1 Design Requirements. The fixture provides interface compatibility with the Planetary Vehicle (with the roll-axis vertical or horizontal) and the Magnetic Mapping Track.

Other items used with the fixture, and to be considered for functional compatibility, are the Planetary Vehicle Lifting Device, the Perming/Deperming Coil, and the Test Magnetometer Support. The total capacity of the fixture is 30,000 pounds. The fixture structure is insulated so that large induced current loops do not occur.

3.2.1.2 Design Concept. The fixture consists of a rectangular base mounted on four rail-running wheels. The base contains a device to propel the fixture along its rails and to lock it in any desired position. At two sides, vertical members extend upward from the base to a trunnion bearing. Seated in the trunnion bearing is a table-like structure which duplicates the Spacecraft-Nose Fairing interface. A turnover mechanism provides the means to rotate the interface.

3.2.2 Magnetic Mapping Track (see Figure 5-4 of VC270FD100). The functions of this item are to support a loaded Magnetic Mapping Fixture during magnetic characteristic determination and to provide a path for the Magnetic Mapping Fixture about the Test Magnetometer Support.

3.2.2.1 Design Requirements. The track provides interface compatibility with the Magnetic Mapping Fixture and the Transfer Carriage. The load capacity of the track is a function of the combined Planetary Vehicle and fixture weight.

3.2.2.2 Design Concept. The track consists of two rails laid in a circular path, with an open arc to accept the Transfer Carriage. The fixture railrunning wheels interface with these rails. The innermost rail is the guide rail and the outer rail is a flat raceway. The rails are mounted to crossties, which are mounted to the floor. The diameter of the track center line is 30 feet.

3.2.3 Magnetic Mapping Transfer Carriage (see Figure 5-3 of VC270FD100). The functions of this item are to: (a) support a loaded Magnetic Mapping Fixture, (b) provide a means for the movement of a fixture to and from the Magnetic Mapping Track, and (c) provide a means for the movement of a fixture to and from the Perming/Deperming Coil.

3.2.3.1 Design Requirements. The carriage provides interface compatibility with the fixture and the Magnetic Mapping Track. The Perming/Deperming Coil is used with the carriage and must be considered for functional compatibility. The load capacity of the carriage is a function of the combined Planetary Vehicle and fixture weight.

3.2.3.2 Design Concept. The carriage consists of a pie-slice shaped, tubular base structure fitted with three railrunning wheels. The base contains a device to propel the carriage along its rails. A section of track similar to the Magnetic Mapping Track is mounted on the base. The fixture rail-running wheels interface with this track. Removable limit stops on the track hold the fixture in place during the carriage transfer modes. When the carriage is interfaced with the Magnetic Mapping Track at the open arc, its section of track makes a complete circular path about the Test Magnetometer Support. A locking device holds the carriage in place with the Magnetic Mapping Track.

3.2.4 Transfer Carriage Railing. The functions of this item are to provide: (a) a path for the Transfer Carriage to and from the Magnetic Mapping Track and (b) a path for the Transfer Carriage to and from the Perming/Deperming Coil.

3.2.4.1 Design Requirements. The railing provides interface compatibility with the Transfer Carriage. The Perming/Deperming Coil and the Magnetic Mapping Track are used with the rails and must be considered for functional compatibility. The load capacity of the rails is the sum of a loaded fixture and Transfer Carriage.

3.2.4.2 Design Concepts. The railing consists of two pieces similar to that used for railways and a guide rail. The three pieces are in parallel and form a straight line (the guide rail being centered between the other two). The rails are mounted to crossties, which are mounted to the floor. Permanent limit stops are placed on the coil end of the railing. The railing is approximately 100 feet long.

3.2.5 Test Magnetometer Support (see Figure 5-2 of VC270FD100). The function of this item is to support the Test Magnetometer.

3.2.5.1 Design Requirements. The support provides interface compatibility with the magnetometer and is a rigid structure.

3.2.5.2 Design Concept. The structure consists of a three-legged platform and is mounted directly to the floor. The platform contains those fastening features necessary to hold the magnetometer rigidly in place. The height of the platform from the floor is approximately 12 feet.

4.0 SHIPPING EQUIPMENT

4.1 Spacecraft Shipping Containers

4.1.1 Spacecraft Shipping Cover (Item 28 of Figure 4-1 in VC270FD100). The cover will provide the necessary environmental protection to the fully assembled Flight Spacecraft (less Retropropulsion Unit) during shipment to KSC while on the Spacecraft Transporter. The cover will meet the following design requirements:

a. It will be capable of providing a barrier to ambient atmospheric conditions.

b. The cover will not contact the Flight Spacecraft at points where Spacecraft damage could result.

c. The cover will be constructed of electrically conductive material and will be provided with a grounding strap.

d. The cover will interface with the Spacecraft Transporter at the Flight Spacecraft tiedown points.

e. Ancillary equipment will be capable of providing a thermally and humidity controlled atmosphere within the sealed cover.

f. The cover will be capable of withstanding exposure to an outdoor unsheltered environment for a minimum of 30 days without degradation.

The cover will be constructed of a flexible electrically conductive plastic. A zipper will be provided to allow separation of the cover into two pieces. A bottom section will contain gasketed holes located to coincide with the Spacecraft Transporter at the Spacecraft tie-down points. The hat section will be supported by tubular bows which can be inserted in sockets mounted on the bed of the transporter. A self-powered recirculating environmental control system, capable of delivering the required conditioned air into the cover through a service connection, will be a separate but distinct part of the Spacecraft Transporter.

4.1.2 Solar Array Panel Shipping Container. This shipping container will provide the necessary environmental protection to spares of removed Solar Panels during shipment by common carrier between manufacturing and test sites. The container will meet the following design requirements:

a. It will reduce the induced and expected natural transportation environment to levels which will not damage the Solar Panels.

b. The container will be reusable.

c. The container will interface with the item at Panel-Spacecraft attachment points only.

d. The container will allow for safe and easy insertion and removal of the contents.

The container will be a two-piece, suitcase-type and will be constructed of lightweight, durable, rigid plastic. An internal shock and vibration mitigation system will be incorporated within the container between the container walls and the tie-down points of the item. Externally, provisions will be incorporated on the container shell to facilitate handling and tie-down.

4.2 OSE Shipping Containers. This generic term covers all containers required to provide safe transportation of the Subsystem OSE, STC, LCE, and AHSE Operational Support Equipment which may be shipped to various remote use points in support of the program. These containers will meet the following general design requirements:

a. They will provide the protection to OSE end items that is necessary to assure safe delivery of the equipment in a usable condition.

b. Container design will allow for reuse of the containers as dictated by program logictics.

c. Provisions will be made for mechanized handling and carrier tie-down of the containers.

In general, shipping containers for OSE will be of wood construction. Screws, rather than nails, will be used for container panel fastening to facilitate container reusability. Container bases will be skidded to allow four-way entry for fork-lift truck tines. Weight and volume of containers will be kept to a minimum consistent with the amount of protection required.

5.0 SPECIAL-PURPOSE TEST EQUIPMENT.

5.1 Weight and Center-of-Gravity Determining Equipment (Item 27 of Figure 4-1 in VC270FD100). Weight and center-of-gravity equipment are used in connection with the Spacecraft Assembly Fixture (Item 4 of Figure 4-1 in VC270FD100). The following is a functional description of the method which will be used to measure weight and center of gravity of the Spacecraft. The measuring equipment will enable two components of the center of gravity vector to be measured. The reference center will be the vehicle roll axis which is centered in Reference Plane A, with the roll axis parallel to local gravity. The weight and center of gravity system will be comprised of four load cells and an analog-to-digital computer providing weight and center-of-gravity readout.

The load cells will be located in the assembly fixture in such a manner that moments will be measured on either side of two lines at right angles to each other. The intersection of the two lines should coincide with the geometric roll axis. This is difficult to accomplish mechanically, but is relatively easy when done electrically. The electric axes will be adjusted to coincide exactly with the geometric axes of the vehicle. Calibration devices will be incorporated in the structure to permit calibration of each load cell without removing the Spacecraft from the stand. Tolerances and calibrations will be such that the final numbers which will be read out will be within ± 0.1% of the actual mass of the vehicle, and the center-of-gravity station number will be correct within ± 0.080 inch in each direction. Tare values of the vehicle supporting interface will be the electrical zero.

The difference between local gravity (approximately 32.157 min/sec^2 and standard gravity (32.174 min/sec^2) will be confirmed by reference through the Potsdam network, and a ratio will be incorporated in the computer to provide a force reading with reference to standard, rather than local, gravity. Means will be required to prevent false readings due to torsional or bending loads entering the load cells.

5.2 Free-Mode Test Equipment (Item 29 of Figure 4-1 in VC270FD100). Equipment used in connection with the free-mode test is comprised of lamp banks simulating sunlight and a means to support the Planetary Vehicle during the test. An inert rocket and a prime capsule will be mounted on the Spacecraft for failure analysis test. The PTM Spacecraft will be tested without rocket and capsule. The free-mode test will be implemented in the Spacecraft Assembly Fixture (Item 4 of Figure 4-1 in CII VC270FD100).

5.2.1 Design Requirements. Power required for the lamp banks will be 50 kilowatts in order to produce the equivalent of 1/3 sun (at 1 au) at the solar array. Excess heat energy from the lamps in each bank will be absorbed by gas or liquid coolant.

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

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

5.3 Thermal-Vacuum Test Equipment (Item 30 of Figure 4-1 in VC270FD100). The thermal-vacuum test will be conducted in two phases. The thermal model will be subjected to simulated space conditions while illuminated by simulated sunlight. The Flight Spacecraft will be subjected to simulated space conditions while thermal inputs will be supplied by heating plates held at a specific temperature.

5.3.1 Thermal Model Design Requirements. The thermal model will be placed in the 54-foot thermal-vacuum chamber using the Spacecraft Transporter for transportation and the

Spacecraft and Planetary Vehicle Lift Sling. An aluminum fixture will be mounted in the chamber which will provide 180-degree rotation to the thermal model.

5.3.2 Thermal Model Design Concept. The thermal model will be mounted to the fixture by using the Spacecraft-nose fairing interface. The fixture will be supported by trunnions, and will be counterweighted for rotation. Rotation will be effected by a hydraulic cylinder using silicone oil for the operating fluid. A small heating blanket attached to the cylinder will assure reliable operation under conditions of cold space. The pump for the hydraulic fluid will be mounted outside the chamber.

5.3.3 PTM Design Requirements. The PTM will be placed in the thermal-vacuum chamber using the Spacecraft Transporter for transportation and the Spacecraft and Planetary Vehicle Lift Sling.

5.3.4 PTM Design Concept. Heat inputs to the PTM will be by means of thermal plates supported in a light aluminum scaffolding erected inside the chamber. The PTM will be supported by an adjustable suspension which will use the interface ring of the Spacecraft and Planetary Vehicle Lift Sling and will permit leveling the Spacecraft. The scaffolding will have vertically movable sections to permit adjusting the thermal plates to conform to the PTM contour.

5.4 Vibration Test Equipment (Item 31 of Figure 4-1 in VC270FD100). A Vibration Test Fixture will be supplied to mate with the Spacecraft-nose fairing interface.

5.4.1 Design Requirements. Strength of the fixture will be such that a fully loaded Spacecraft will be supported uniformly, and deflections of the fixture will be minimized. Vibration inputs can be along the pitch, the yaw, or roll axes. Natural frequency in lowest mode of the fixture will be above 200 cps. An inert retro-rocket and the dynamic capsule simulator will be required.

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

5.5 Separation Test Equipment (Item 33 of Figure 4-1 in VC270FD100). A group of equipment is required to support operationally separable sections of the Planetary Vehicle with 5 degrees of freedom during separation of interfacing elements.

5.5.1 Design Requirements. Test equipment will include a dynamic capsule simulator, a Planetary Vehicle adapter, a massive support which simulates the booster (by JPL direction, the booster is assumed infinitely massive), a 1-3/4 inch steel cable to support the STM during test from a pivot 75 feet above the center of gravity of the STM, a divided arc with pointer, and a pivot for the cable at the center of gravity of the STM. (Auxiliary equipment used to measure and record motion of the Spacecraft during separation is not included in this description.)

5.5.2 Design Concept (See Figure 4-1 in VC270FD100). Both of the pivots, one fixed above and one mounted at the cg of the Planetary Vehicle, will be knife-edge pivots, arranged to form a universal joint. This will reduce the friction, always present in ball or roller bearings, to nearly nothing. The lower pivot will be adjustable in three directions to achieve maximum motion sensitivity for pitch or yaw motions. Roll motion will be permitted through twisting in the long cable. The knife-edge form of pivot is deemed best for this application because of its inherent ruggedness (10,000 lb/in. maximum load for a 90-degree edge) and its low-friction qualities, if hard tool steel is used.

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

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

5.6 Environmental Control Equipment. The function of this item is to provide a means for bringing the mounted electrical subsystems to any temperature within a set range and maintaining that temperature during the margin reverification test.

5.6.1 Design Requirements. Functional compatibility with the Spacecraft Assembly Fixture and electrical test equipment must be considered. Cleanliness of the Spacecraft shall not be degraded. The equipment will be used with the Spacecraft supported in the Spacecraft Assembly Fixture. The equipment will not touch the Spacecraft. The equipment will be capable of subjecting the subsystems to temperatures which are outside of the Flight Acceptance temperature limits by 15% of those limits expressed in degrees F.

5.6.2 Design Concept. In the installed position the enclosure will be in the form of a right circular cylinder, on end, enclosing the Spacecraft. The lower base of the cylinder will be open and will loosely interface with the floor. The upper base of the cylinder will be closed and will contain suspension points. The enclosure will be constructed of a flexible, low-heat-transmitting material. A skeleton structure will maintain the proper Spacecraft-enclosure clearance and yet permit the storage room requirement of the enclosure to be minimized. A mobile heat exchanger will be connected to the enclosure by flexible ducting to form a partially closed system. The exchanger will feature provisions for recirculating, temperature control, humidity control, and particle filtration of the system air.

5.7 Static Firing Test Equipment (Item 34 of Figure 4-1 in VC270FD100). Equipment for the static firing test is comprised of shipping equipment (necessary to ship the Spacecraft to the test stand), Spacecraft and Planetary Vehicle Handling Sling, (all of which have been described under AHSE Handling and Shipping Equipment) and the Static Firing Test Stand.

5.7.1 Design Requirements. The Static Firing Test Fixture mates with the Planetary Vehicle interface. It will resist both downward and upward loads imposed by the fueled Planetary Vehicle and by the force generated by firing. Measurement of the force will be accomplished by means of four load cells, each of which will make a record of the thrust. Greatest upward force will be less than 30,000 pounds. Greatest downward force will be a maximum of 20,000 pounds.

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

5.8 Acoustic Test Fixture (Item 32 of Figure 4-1 in VC270FD100). Equipment will be required to support the PTM in an acoustic chamber in such a manner that it will receive acoustic inputs in the same environment as encountered during powered flight.

5.8.1 Design Requirements. Equipment required for the acoustic test will be the Spacecraft and Planetary Vehicle Handling Sling, shipping containers for shipment to the acoustic chamber where the test will be conducted, a simulated capsule, a simulated shroud (GFE), and a fixture on which to mount the simulated shroud inside the acoustic chamber.

5.8.2 Design Concept. Tie-downs will be used between the simulated capsule and simulated shroud which will exert a force of 2.6g on the capsule in addition to its own weight. A plane-based aluminum fixture will be used to support the shroud and will simulate the shroud-booster interface. The tie-downs will be located at approximately 3-foot intervals around the periphery of the capsule. Strain gaged bolts will measure the tension applied to the 1/8-in tie-down cables.

5.9 EMI Test Equipment. Tests will be conducted in the Spacecraft Assembly Fixture (Item 4 of Figure 4-1 of CII VC270FD100). No additional AHSE will be required for EMI tests.

CII VC280FD105

MISSION DEPENDENT EQUIPMENT REQUIREMENTS AND FUNCTIONAL DESCRIPTION

SECTION

MISSION DEPENDENT EQUIPMENT REQUIREMENTS AND FUNCTIONAL DESCRIPTION

1.0 INTRODUCTION. The Mission Dependent Equipment (MDE) shall be designed to support the spacecraft mission requirements at the Deep Space Instrumentation Facility (DSIF) stations and the Space Flight Operations Facility (SFOF) at Pasadena, California. It shall consist of command encoding/detection equipment, command processing software, and telemetry demodulation equipment and decommutation software.

The predominant MDE design criteria shall be reliability and ease of calibration and maintenance. The MDE design shall make maximum utilization of, and be completely compatible with, the planned capabilities of the DSN for Voyager as defined in JPL document EPD 283. The equipment design shall conform to standard DSIF specifications in order to ensure mechanical, electrical and functional compatibility with the existing DSIF facilities. In addition the design of the equipment and software to be employed at the SFOF shall exhibit an equal degree of compatibility with the SFOF facilities as defined for the DSIF.

The DSIF includes a Tracking and Command Data Handling Subsystem which provides a general purpose computer facility (SDS920) for processing command and telemetry data. The Tracking and Command Data Handling Subsystem in conjunction with MDE must decommutate the telemetry data and prepare that data for distribution over the Ground Communications System (GCS). The Tracking and Command Data Handling Subsystem must also receive command messages from the SFOF, store the command messages until the time of execution, initiate transmission of commands to the spacecraft and verify proper transmission of the commands.

A similar condition exists at the SFOF which also includes a general purpose computer facility (IBM 7044/7094 system). This equipment in conjunction with MDE must analyze the telemetry data and prepare commands for transmission to the DSIF.

2.0 COMMAND MDE. The Command MDE consists of (a) the command modulator, (b) the command detector, and (c) the command processing software.

2.1 Command Modulator/Detector. The command modulator will provide a biphase modulated subcarrier at one and thirty bits per second compatible with the DSIF transmitter. The command detector will provide a means of demodulating the command subcarrier signals and verifying transmission validity in conjunction with the TCD computer.

2.2 Command Software. The command software will consist of programs for the TCD to allow generation of the sync pattern and to implement manchester coding of the command data.

2.3 Interface Description. The command MDE interface signals are shown in Table 2-1.

3.0 TELEMETRY MDE. The TLM MDE consists of (a) the telemetry demodulator, (b) the TLM Simulator, (c) the decommutation software. Identical hardware and software are used in the STC/OSE for the demodulation, simulation and decommutation functions.

TABLE 2-1. 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
COMMAND DETECTOR INTERFACES			
Interfacing Component	Function	Input/Output	Characteristic
General Purpose Computer	Command Data	Output	Digital
	Bit Sync	Output	Digital
Local Receiver	Command Signal	Input	Command Spectrum

3.1 Telemetry Demodulator. The Telemetry demodulator has the task of detecting and decoding a noise-corrupted modulated telemetry signal from ground telemetry receivers, magnetic tape recorders, or the spacecraft TLM simulator. The telemetry signal is a composite signal containing both data and synchronization information on either one or two subcarriers. For each subcarrier the demodulator output consists of serial and/or parallel binary data, bit-sync, word sync, and a 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. The incoming telemetry signal for each of the spacecraft modes is shown in the spacecraft TLM subsystem functional description.

The telemetry demodulator has two modes of operation: normal and self-check; the simulator is employed for the latter. Figure 3-1 shows the functional block diagram of the

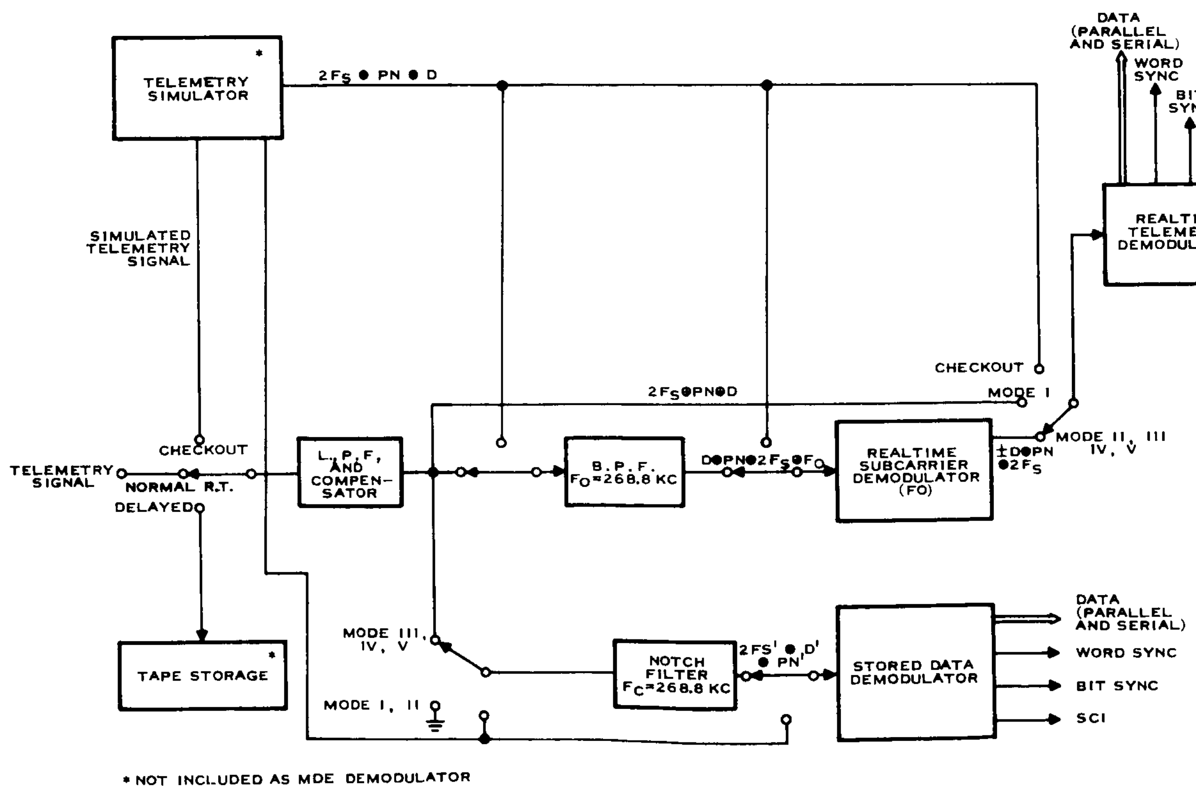

Figure 3-1. Telemetry Demodulator

telemetry demodulator. The input telemetry signal is sent through a low-pass filter (f_c - 537.6 kHz) and amplifier. 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 TLM demodulator.

The TLM signal is routed to the Real Time (RT) Telemetry Demodulator when receiving Mode 1 data. When receiving data Modes 2, 3, 4, and 5, the telemetry signal is routed through two additional filters, which separate the real time and stored data telemetry spectra. The output of a bandpass filter (f_o = 268.8 kHz) is sent to the Real Time Subcarrier Demodulator which removes the 268.8 kHz subcarrier and sends the resultant signal to the Real Time Telemetry Demodulator for data and sync detection. When receiving Modes 3, 4, and 5 data, the output of the Notch Filter (f_o = 268.8 kHz) is sent to the Stored Data Telemetry Demodulator.

3.1.1 Real Time Data Band Pass Filter (RT Data BPF). The RT Data BPF is used to mask out the stored Data TLM channel while passing the RT data TLM channel when frequency multiplexing is employed by the spacecraft TLM subsystem. The RT data BPF is centered at 268.8 kHz with a band-pass equal to or less than 26.88 kHz.

3.1.2 Stored Data Band Rejection Filter (Notch Filter). The stored data TLM channel employs a band rejection filter centered at 268.8 kHz to reject the RT data TLM channel. The band rejection enhances the stored data channel SNR.

3.1.3 Real Time Subcarrier Demodulator. The RT subcarrier Demodulator, Figure 3-2, acquires the RT subcarrier f_o (where f_o = 268.8 kHz) by locking a phase lockloop on the incoming subcarrier after the modulation has been removed by mixing and squaring. Once acquisition is obtained, the output signal, $e_s = \pm D \oplus PN \oplus 2f_s$, is sent to the RT TLM demodulator.

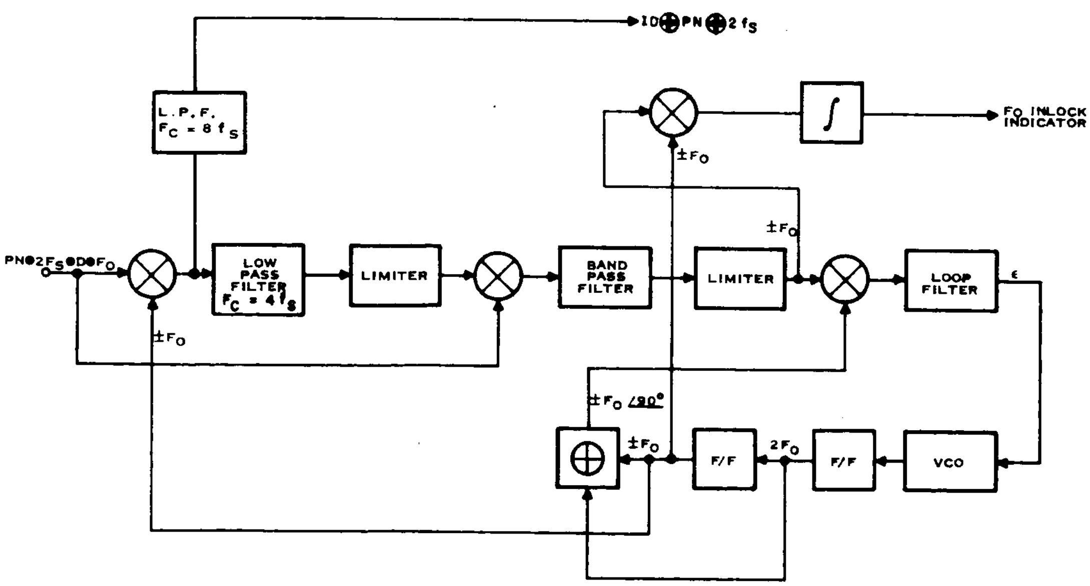

Figure 3-2. Real Time Subcarrier Demodulator

3.1.4 Real Time Telemetry Demodulator. The block diagram for the Real Time Telemetry Demodulator is shown in Figure 3-3. The incoming signal ($\pm$ D $\oplus$ PN $\oplus$ $2f_s$) is presented 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), NRZ-M data, bit sync, word sync and termination of the acquisition process.

Code slide-by, enabling acquisition, is accomplished by adding one $4f_s$ 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. NRZ-M data is matched-filter detected in the data channel after removing PN code and $2f_s$ clock from the bi-phase modulated signal. The NRZ-M data is converted to NRZ where it is presented to the output in serial or parallel form. Parallel data will be presented as eight bit words including seven bits for data and one bit for SCI. The NRZ-M to NRZ converter eliminates data ambiguity.

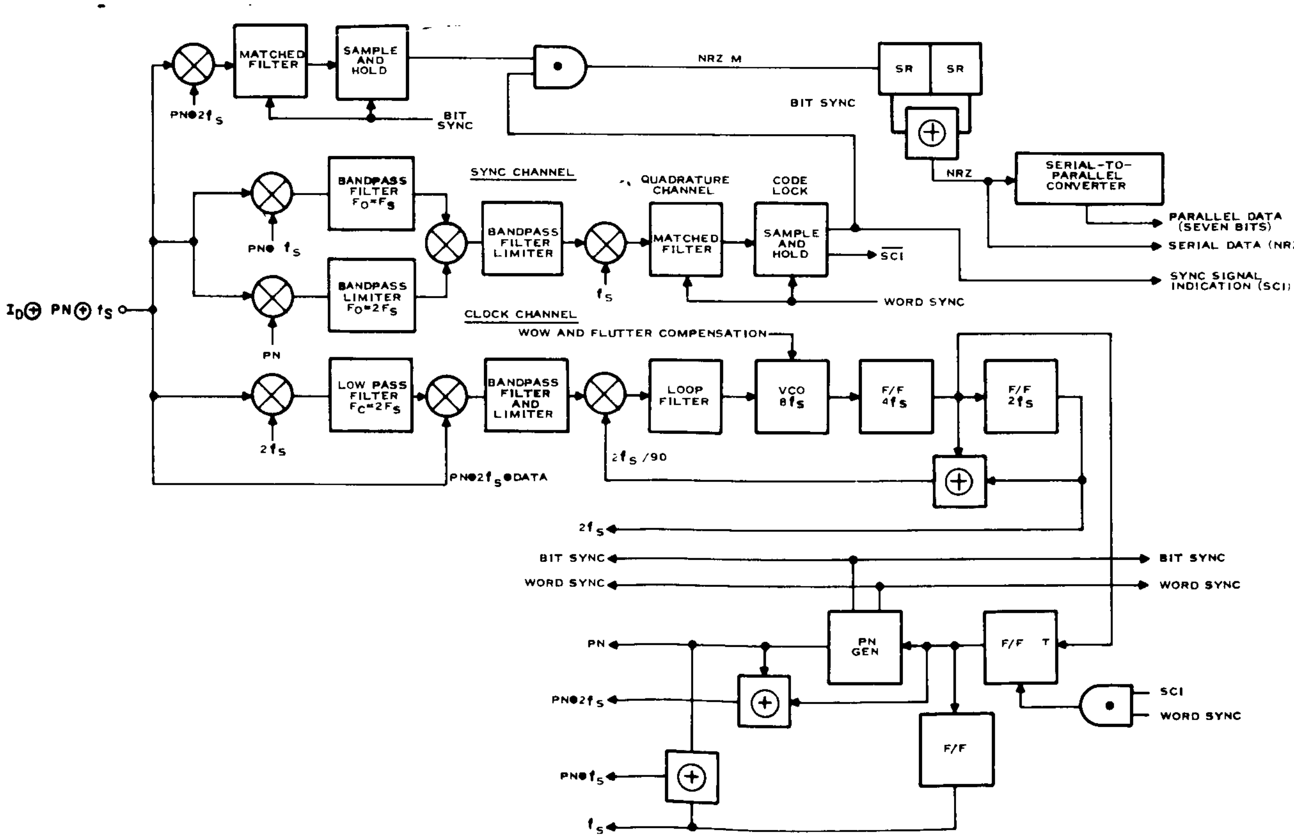

Figure 3-3. Real Time Telemetry Demodulator

After acquisition is complete, the loop filter time constants may be decreased to give better tracking performance by reducing the RMS phase error.

3.1.5 Stored Data Demodulator. The block diagram for the stored data demodulator is essentially identical to that for the RT TLM demodulator shown in Figure 3-3. Except for the necessary changes in the frequency-dependent components and the NRZ-M to NRZ converter, it is identical to the RT TLM demodulator.

3.2 Telemetry Simulator. The TLM simulator shown in Figure 3-4 functionally duplicates the spacecraft TLM subsystem in every respect in order to test and verify the MDE TLM Demodulator and Decommutator (TCD Software) operation without the use of a spacecraft TLM subsystem. The TLM Simulator generates simulated data where the simulated data contains: frame sync, data mode, medium deck sync, low deck sync and random data. The random data contains different words of known value for each channel. A degree of selection or programming of channel words is available as front panel control. The simulator will accommodate simulated stored data and RT TLM data generation.

The simulator output is a composite TLM signal as described in the spacecraft TLM subsystem and serial or parallel data. Noise may be added to check the demodulator performance.

3.3 Decommutation Software. Decommutation of real time telemetry data (and the near real time data collected and stored during a spacecraft maneuver) is accomplished with an

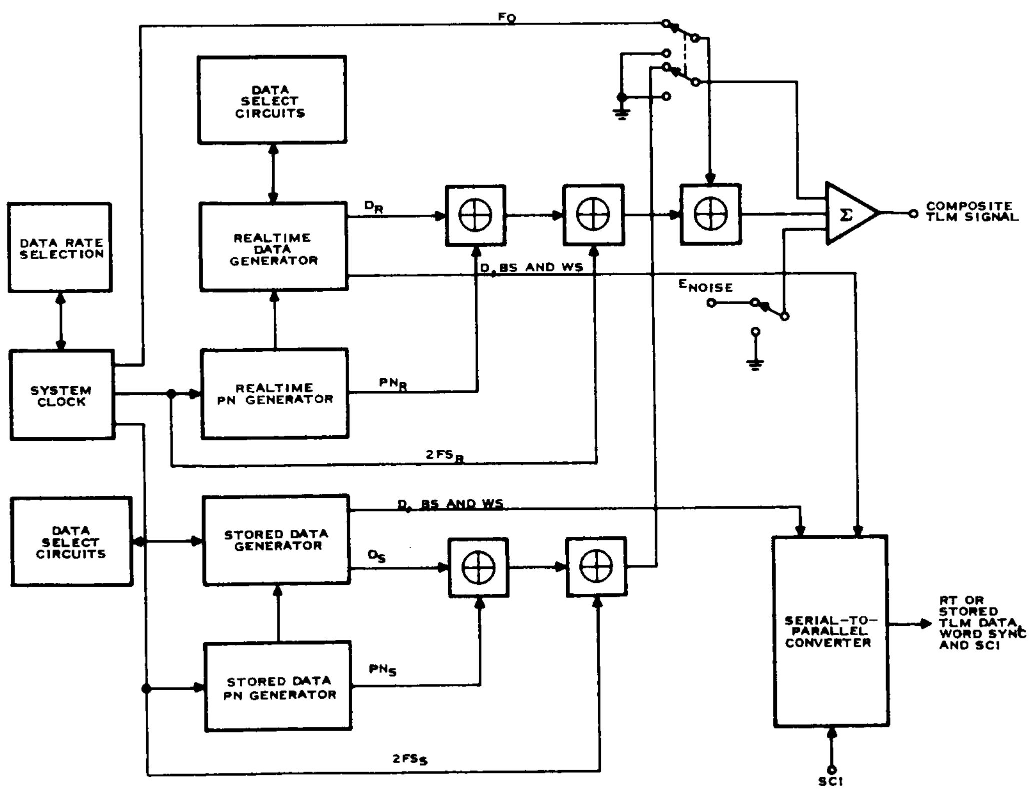

Figure 3-4. Telemetry Simulator

SDS 920 computer. Decommutation of the stored spacecraft science data will be performed by the science data users according to their own requirements. The input information is received from the demodulator or simulator in parallel form as a 24-bit computer word, consisting of seven data bits in the least significant positions and the SCI bit in the most significant position (remainder is unused). Word sync initiates a priority interrupt enabling the computer to read in the new data word by branching to the read instruction.

The following functions are included in the decommutation program.

a. Determine data mode and execute the necessary processing instructions at all data rates.

b. Generate and identify the data condition information as to the validity of received frame sync, data mode, medium deck sync, low deck sync and word checks for each channel of data. The word checks performed are limit checks or delta checks.

c. Synchronize and process the submultiplexed engineering data.

d. Format selected telemetry data for visual display, (including engineering split channels) or printout.

e. Channel address each of the data channels, including special engineering split channels.

f. Provide flywheel action for subcommutation program for frame sync, data mode, medium deck sync, and low deck sync words.

g. Format telemetry data to binary or BCD parallel output.

h. Tag each data channel with flag (check) word or DCI (Data Condition Indicator).

The output data is presented in 24-bit parallel form and routed to the appropriate user in a form determined by the user's need. In addition to the data bits and channel address, the output word may also contain: (a) mode identification, (b) out-of-limit indication, (c) data condition indication dependent on mode changes, frame sync condition, and subcommutation sync condition.

3.4 Decommutation Program. The decommutation program is basically the same as that described in the Phase IA - Task A final report (VB263FD106). The differences are mainly those resulting from the use of two frequency multiplexed data subcarriers (which makes separation of stored science data, and real time telemetry data through decommutation unnecessary) and the addition of deck sync words for each deck level in each engineering group.

3.5 Interface Description. The TLM MDE interface signals are shown in Table 3-1. The telemetry software is designed to accept the data inputs from the telemetry demodulator, and to format data for storage, transmission, and/or display. It 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.

TABLE 3-1. INTERFACE SIGNAL TABLE

Interface	Function	Characteristic	Comments
TLM Demodulator and Radio S/S OSE, TLM Simulator or MTR	Composite TLM Signal	0.3V to 3.0V rms	Subcarrier (s) biphase modulated with data and sync
Decommutator and TLM Demodulator, Simulator or MTR	TLM Data	Binary Parallel NRZ data	Synchronous with data/7
	TLM Word Sync	Binary RZ	Synchronous with data/7
	SCI	Binary NRZ	Continuous during phase lock
TLM Demodulator and Simulator or MTR	TLM data	Serial Binary NRZ data	At predetermined bit rate
	TLM Word Sync	Binary RZ	Synchronous with data/7
	TLM Bit Sync	Binary RZ	Synchronous with data

3.6 Physical Characteristics (MDE hardware only). The MDE hardware will be designed for 19-inch-panel mounts or drawers occupying one 75-inch-high NASA rack. 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 of operation or bit rates, modules are interchanged.

The MDE hardware including one TLM demodulator and simulator, blower, and power supplies will weigh 2000 pounds or less and consume less than 2000 watts of power when operating on 105-125 volts, 60 Hz - single phase input power.

To enhance system reliability and cost reduction SIC will be used throughout the system supplemented with conventional components where necessary. Use of SIC will also reduce the size of the system.

4.0 SFOF SOFTWARE. Generally, the mission software functions fall into four categories: (a) Event Generation, (b) Telemetry Processing, (c) Command Preparation and Verification, and (d) Engineering Data Analysis. Not all of the software in each category is mission dependent. In many cases the mission-dependent software will be a routine or subroutine of a larger, mission independent program.

4.1 Event Generation Software. These programs are responsible for (a) tracking and orbit determination, (b) control of midcourse and terminal maneuvering, (c) DSN contact scheduling, and (d) control of the capsule lander and payload operations. Of these only Item b, control of midcourse and terminal maneuvering, requires discussion in this specification. Items a and c are assumed to exist presently at JFL and require only minor modifications for VOYAGER use.

The mission-dependent Spacecraft Maneuver Program is required to calculate the vehicle orientation and engine burn parameters for midcourse maneuvering, Mars orbit injection, and Mars orbit correction. Program inputs include current flight path data, desired Mars orbit characteristics, and dual-spacecraft mission requirements.

It is assumed that certain existing mission independent routines will satisfy many of the computational requirements of the program. The basic requirements of the program are mission dependent, however, and these independent routines will be incorporated as subordinate entities.

4.2 TPS Subroutine. The Telemetry Processing Station (TPS) is responsible for conditioning high-speed telemetry data at the Space Flight Operations Facility for entry into the main computers. The primary function of the TPS is conversion of the incoming data into a format compatible with the 7288 Data Communications Channel of the IBM 7044 computer in real time.

4.3 7044 VOYAGER Input Routine. The IBM 7044 I/O Processor is a real-time multi-function complex responsible for the: (a) input processor, (b) output processor, (c) quick-look display processor, and (d) alarm monitor.

4.4 7094 Editor VOYAGER Subroutine. The 7094 Raw Data Edit Program (EDITOR) processes the raw data stored on the disc by the 7040 I/O Processor for use by other programs.

4.5 Command Processing Software. The functions of the command generation software are: (a) Command generation (7094), (b) Command message preparation and transmission (7094), and (c) SFOF/DSIF transmission verification.

4.6 Command Preparation Program. The purpose of this program is to assemble those commands required to perform a function established in the sequence of events.

4.7 Engineering Data Analysis Software. This family of software is intended to serve as a highly-flexible tool in assessing spacecraft status and performance throughout all phases of the mission.

In general, this program will function off-line, although certain portions of it may be initiated by real-time events. The following subroutines will be included in this program:

a. Vehicle Command Storage Program - This program is responsible for monitoring the execution of commands stored in the vehicle command storage. It will use as its initial input those commands stored prior to launch, and will receive notice of all subsequent commands transmitted.

b. Spacecraft Status Program - This program will provide comprehensive displays of current or past status on a subsystem-by-subsystem basis. These displays will indicate the engineering value of all telemetry points of the desired subsystem at the specified time.

c. Frame Word Slot Extractor Routine - This routine would enable the user to extract data values by word slot from the file data on the disc.

d. Means and Extremes Routine - This routine will compute the means and extremes of desired functions over a specified time interval.

e. Power Routine - This routine generates a report concerning the battery charge/discharge ratio over specified intervals.

f. Attitude Determination Routine - This routine is designed to analyze telemetry data in order to determine the attitude and rates of the spacecraft during specified intervals. It is intended, further, to compare these results with programmed and predicted values and display the results at the user area.