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Viking '75 Spacecraft Design and Test Summary

Volume III - Engineering Test Summary

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and Space Administration

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PREFACE

This publication, in three volumes, discusses the design of the Viking lander and orbiter as well as the engineering test program developed to achieve confidence that the design was adequate to survive the expected mission environments and to accomplish the mission objective. Volume I includes a summary of the Viking Mission and the design of the Viking lander. Volume II consists of the design of the Viking orbiter and Volume III comprises the engineering test program for the lander and the orbiter.

The material contained in this report was assembled from documentation produced by the Martin Marietta Corporation (now Martin Marietta Aerospace) and the Jet Propulsion Laboratory during the Viking spacecraft design and test program. Viking Project personnel contributing to the preparation of this publication and their area of contribution are as follows:

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Lander guidance and control design and test . . . Anthony Fontana

Lander propulsion design and test . . . . Anthony Fontana

Lander thermal design and test . . . . . . T. W. Edmund Hankinson

Viking orbiter detailed design information:

- Command and sequencing . . . . Carl R. Pearson
- Computer command subsystem . . . . Carl R. Pearson
- Flight data subsystem . . . . Carl R. Pearson
- Data storage subsystem . . . . Carl R. Pearson
- Visual imaging subsystem . . . . Carl R. Pearson
- Infrared thermal mapper subsystem . . . . Carl R. Pearson
- Mars atmospheric water detector subsystem . . . Carl R. Pearson
- Temperature control . . . . William A. Carmines
- Structure subsystem . . . . William A. Carmines
- Pyrotechnic subsystem . . . . William A. Carmines
- Cabling subsystem . . . . William A. Carmines
- Propulsion subsystem . . . . William A. Carmines
- Mechanical devices subsystem . . . . William A. Carmines

Test data and information . . . . William A. Carmines

CONTENTS

INTRODUCTION

The Viking Project was initiated in 1968 and was climaxed with the launching of two Viking spacecraft in 1975. The first landing on Mars was July 20, 1976, and the second landing was September 3, 1976. Each spacecraft contained an orbiter and a lander. The objective of the Viking Mission was to increase significantly man's knowledge of the planet Mars through orbital observations by the orbiter as well as by direct measurements made by the lander during Martian atmospheric entry, descent, and landing. Particular emphasis was placed on obtaining biological, chemical, and environmental data relevant to the existence of life on the planet at the present time, at some time in the past, or the possibility of life existing at a future date. Orbiter observations consisted of radio-science, imaging, thermal, and water-vapor measurements used to assist landing-site selection and the study of the dynamic and physical characteristics of Mars and its atmosphere. Lander direct measurements consisted of radio science; atmospheric structure and composition; landing-site imaging; atmospheric pressure, temperature, and wind velocity; identification of the elemental composition of the surface material; physical properties of the surface material; the search for evidence of living organisms and organic materials; and determination of seismological characteristics of the planet. The Viking scientific return was further expanded by the capability of simultaneous Martian observations from orbit and the surface.

This document, divided into three volumes, summarizes the design of the Viking lander and orbiter as well as the engineering test program developed to achieve confidence that the design was adequate to survive the expected mission environments and accomplish the mission objective. The engineering test program covered those aspects of testing prior to delivery of the Viking landers and orbiters to the John F. Kennedy Space Center. Most of the material contained in this document was taken from documentation prepared by Martin Marietta Corporation (now Martin Marietta Aerospace) and the Jet Propulsion Laboratory during the Viking spacecraft design and test program. This volume contains the engineering test program for the Viking lander and orbiter. All abbreviations used in this volume are defined in an appendix.

Use of trade names in this report does not constitute an official endorsement of such products or manufacturers, either expressed or implied, by the National Aeronautics and Space Administration.

LANDER TEST PROGRAM

The test program for the Viking lander was designed to be a systematic verification that the lander system complied with the technical and test requirements that were given in the Viking '75 Lander System Specifications. The tests were designed to provide necessary data on the functional interactions between the subsystems, the systems, the environment, and the AGE; to verify that the flight hardware met the performance and design requirements for anticipated mission environments; and to establish confidence, by environmental and functional tests, that each flight lander system was free from defects and was capable of performing the mission in the primary and planned alternate operating modes. The integrated tests accomplished on the Viking lander included qualification and flight acceptance tests of components, subsystems, and the lander system; integrated tests involving more than one subsystem; tests using major hardware models or special facilities; and major development tests that required project visibility because of their broad interest or impact. Three major types of tests, summarized in figure 1, were performed: development, flight acceptance, and qualification. Development testing included those tests accomplished during design and development to evaluate feasibility of the design for accomplishing its intended function. Flight acceptance tests were functional tests used to demonstrate adequacy of the hardware to meet expected flight environments. Qualification tests were conducted at conditions more severe than FAT conditions in order to determine if design and fabrication procedures were adequate to allow for expected variations in individual articles and environments.

Several lander hardware test models were developed to use as test beds for accomplishing some of the tests. These hardware models are described as follows:

The balloon-launched decelerator test vehicle was an entry-configured test vehicle that was balloon launched, fell freely, and was rocket propelled with provisions for flight-type decelerator subsystem, separation functions, and test instrumentation. This vehicle was used for functional qualification of the decelerator subsystem.

The lander antenna pattern test model was a 3/8-scale model used early in the program for tests of the UHF and S-band low-gain antennas to determine if there was any degradation of predicted patterns due to structural influences, reflections, and radiation for the various VLC configurations encountered during the mission. Full-scale model tests were performed at MMC for the UHF, S-band, and radar antennas to identify impedance matching problems and techniques and pattern perturbation due to structural interference and, also, to verify final pattern coverage and isolation predictions between communication and radar antennas.

The lander dynamic test model was a flight-type lander capsule structure with partial flight-type or equivalent propulsion lines and tanks and mass simulators for nonstructural hardware. It was used for structural vibration and acoustic tests, modal survey, pyrotechnic shock evaluation, and structural separation tests.

The lander static test model was a flight-type lander capsule structure that was used for qualification of primary structure under steady-state and low-frequency loads.

The proof test capsule was a complete lander capsule assembled from flight-type hardware. It was used for system level qualification tests, which included demonstration of flight system and AGE specification requirements, verification of flight article assembly and test operating procedures, verification of AGE

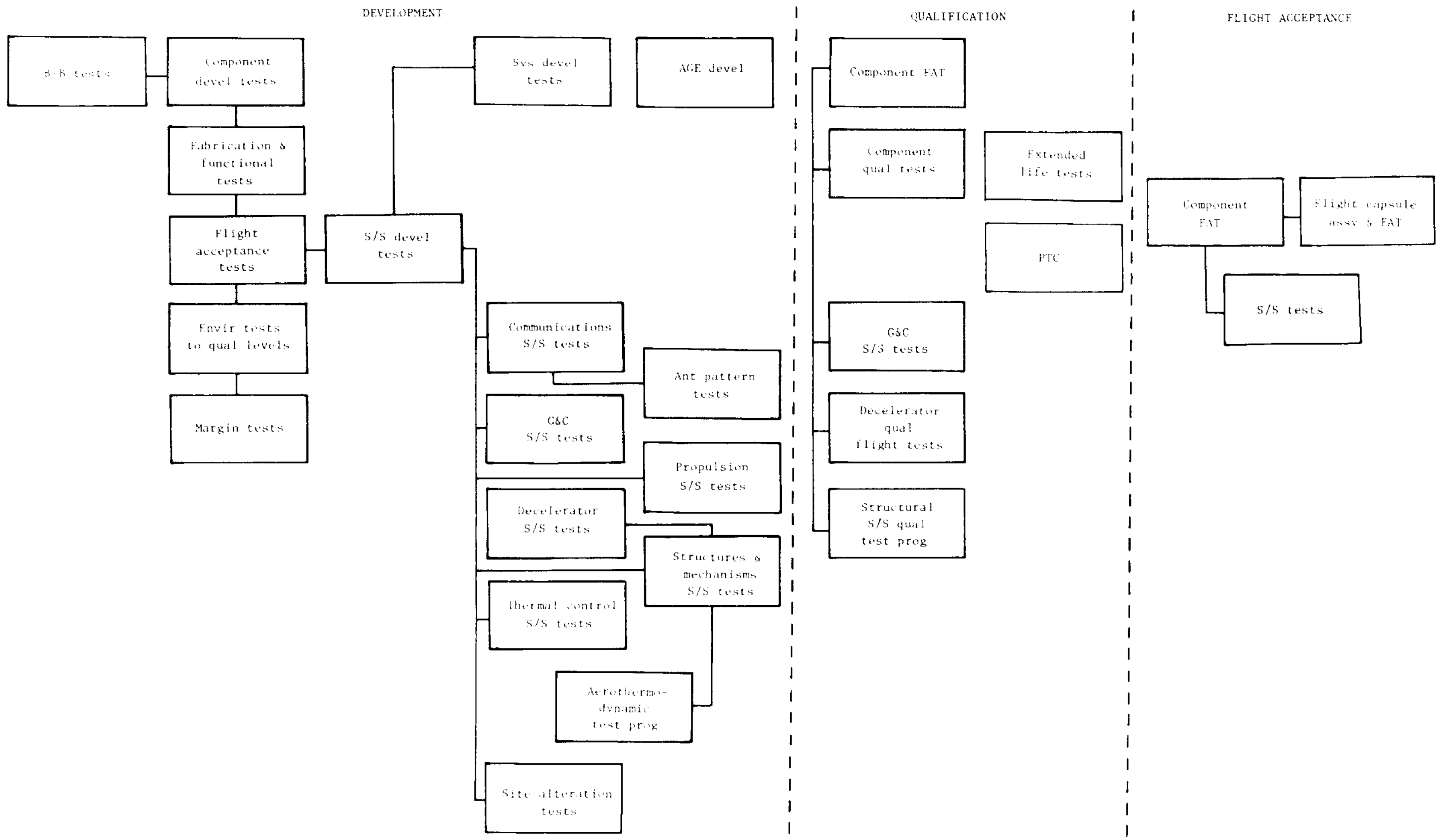

Figure 1.- Viking lander test summary.

software, demonstration of biological controls and terminal sterilization cycle, and demonstration of planned backup and alternative operating modes.

The structural landing test model was a 3/8 geometrically scaled model which was dropped at various velocities and attitudes to determine landing stability boundaries and rigid body acceleration loads.

The system test bed was a collection of the elements (facility, personnel, AGE, TSE, simulators, VLC components, etc.) required to perform subsystem development testing, interface validation, software development and validation, test sequence verification, subsystem tests which require a system configuration, checkout hardware testing, hardware change evaluation, system level support for PTC and flight capsule system testing and flight operations, alternate mode testing, and special tests for the Viking lander system.

The thermal effects test model was a full-scale model incorporating the development thermal control system and flight cabling. The thermal characteristics of flight equipment were simulated by thermal simulators. The TETM was used to verify the general design of the thermal control system, to verify or correct the thermal math models, to provide design data for the thermal control system, and provide input data for the thermal heat sterilization cycle.

Other models that were used during testing with the VLC or with the lander test models are described as follows:

The orbiter thermal effects simulator simulated the thermal characteristics of the orbiter including shadowing effects. This simulator with the VLCA was used during the thermal environmental tests of the spacecraft.

The orbiter dynamic test model simulated the dynamics of the Viking orbiter. The LDTM was mated to the ODTM with a flight-type VO adapter truss for spacecraft vibration tests.

DEVELOPMENT TESTS

The development tests were designed to meet the following requirements:

(1) Determine adequacy of design margins

(2) Identify failure modes and effects

(3) Determine time effects of varied stress levels

(4) Identify the effects of combinations of tolerances and drift of design parameters

(5) Identify the effects of combinations and sequences of environments and functional stress levels

(6) Determine safety parameters and functions

(7) Determine component interaction during subsystem operation

(8) Generate data to support the analysis

To meet these requirements, breadboard, prototype, and design and development units were tested for each lander component. Breadboard units which were fabricated from preliminary engineering design were tested to evaluate design feasibility, operating parameters, parameter variations, and thermal characteristics. Prototype units were tested to verify that the selected hardware design, as determined by breadboard tests, satisfied the lander requirements when fabricated in a flight configuration and subjected to operating and critical environments. DD units which were flight-type units were tested at the component, subsystem, and system level to satisfy the development test requirements.

Component Development Tests

In general, two development units of each component were constructed to carry out the component development tests. Both units were subjected to fabrication and functional tests and flight acceptance tests. After these tests, one unit was subjected to subsystem development tests and then assembled into the STB model for system level development testing. The other unit was subjected to environmental tests at qualification levels and margin tests and parametric variation tests as appropriate.

Fabrication and Functional Tests

Tests were made during fabrication at the subassembly level to insure proper operation at appropriate stages of assembly. A complete functional test was accomplished once assembly of the components was complete. These tests were used to verify all performance characteristics prior to the start of acceptance testing.

Flight Acceptance Tests

Flight acceptance tests were conducted during component development at flight acceptance environmental levels prior to performing subsystem and system tests or prior to performing more strenuous tests such as margin tests or tests at qualification levels. These tests consisted of functional and environmental tests which were designed to detect defects in materials and workmanship. Any defects found at this point were corrected prior to continuing to the next phase of the test program.

Environmental Tests

Environmental tests at qualification levels were conducted on one of the development units for each component. These tests were intended to simulate the mission environments with the addition of an appropriate test margin.

Mission environmental tests were selected for each component depending on its function during each of the mission phases of the Viking Program. Levels of testing for each component were selected as a function of location within the lander and of operational requirements. The tests performed are listed by mission phase in table 1.

TABLE 1.- ENVIRONMENTAL TESTS FOR VL

Mission phase	Test
Prelaunch	Heat compatibility Nonoperative pyrotechnic shock Electromagnetic compatibility Temperature humidity Propellant compatibility
Launch	Ascent pressure decay Launch acceleration Launch acoustic Launch vibration
Cruise and orbit	Thermal vacuum Corona and arcing
Deorbit, entry, and landing	Entry thermal Entry acceleration Entry acoustics Entry vibration Terminal descent vibration Landing shock
Landed	Surface thermal

Margin Tests

Margin tests were designed to provide additional data on component operation under environmental or functional conditions beyond those required for design. Any out-of-tolerance performance, degradation, or malfunction during these tests indicated potential limits to a range of design characteristics and was evaluated for possible weak points in the design. Tests were selected and applied according to the following criteria:

(1) Thermal margin tests were performed where it was anticipated that equipment usage during the actual mission might be extended beyond that for which it was originally designed as the result of revised mission requirements or the development of off-nominal thermal conditions. This consideration was applied specifically to equipment capable of being programmed during the Viking Mission.

(2) Performance margins, transient susceptibility, undervoltage or overvoltage operation, or contamination susceptibility tests were applied to evaluate designs.

(3) Vibration margin tests were conducted on those components required to function during launch, entry, or after landing.

(4) Extended life margin tests were run on those components with life limiting features.

Margin testing included both extended stress tests and extended life tests. The extended stress tests, which encompasses either extended level or extended cycle tests, were conducted at the conclusion of development testing on the design development articles. Extended life tests followed qualification or development testing as appropriate. The requirements for each type of margin testing are discussed in the following paragraph.

Extended stress tests included either extended functional or environmental level tests or repetitive environmental cycle tests. For either one, the stress was extended to specific limits or until the component failed, malfunctioned, or demonstrated performance beyond its design limits. Extended level tests designed to evaluate functional characteristics included variations such as input voltage, output impedance, operating fluid contamination levels, and transient amplitude and duration. Environmental stress tests included tests extending the specified dynamic excitation levels or the temperature extremes. For either one, increasingly more severe stresses were applied in discrete steps. In general, the required incremental increases in stress levels and the final limits included step voltage changes of 2 V/step to 10 V above and below design limits, changes in particulate contamination of 25-percent increments up to 150 percent of the maximum specified, step changes in vibration levels of 1.5 dB/step-up to 6 dB above qualification test requirements and step changes of 11° C/step (20° F/step) to 33° C (60° F) above and below design limits. Specific stress levels and limits were established for individual components. Repetitive environmental cycle tests involved the application of repetitive discrete environmental cycles (i.e., vibration and surface thermal) until the component failed or until twice the nominal specified period of exposure to that environment was achieved. Extended life tests involved operating the component for a period of time beyond the nominal operating life required in the component specification. This operation was performed under ambient conditions unless a specified environment was suspected of adversely affecting the life characteristics of the component. Tests continued until the component failed or malfunctioned or until twice the nominal specified operating life was achieved. Extended life tests were conducted as soon in the test program as practical on the selected component. Tests on the GCSC, the terminal descent rocket engine assembly, and the deorbit thrusters were conducted during component development. Tests on pyrotechnic initiators and pressure cartridges were conducted during development tests after long-term storage of these devices. All equipment operating time accumulated during prior testing was counted in satisfying extended life margin requirements.

Subsystem Development Tests

After the flight acceptance tests on one of the development units, selected components were assembled for subsystem development tests. The purpose of these tests were to identify and resolve subsystem design problems as early as possible. Since subsystem functional tests were performed as part of the system level STB tests, they are not included here. Only unique subsystem level tests are discussed. Special tests which may use components from several subsystems but not the entire VL system are also discussed.

Communications Subsystem Tests

Subsystem tests for communications included antenna, UHF compatibility, and rf compatibility tests. To measure the UHF and S-band LGA pattern coverage early in the development process, tests on a 3/8-scale model were performed. These tests were conducted with the various VL configurations that would be encountered during the mission to determine degradation and changes in patterns due to structural influences. In order to determine isolation between communication and radar antennas, a full-scale LAPTM was used. This test model was also used to obtain information on full-scale antenna patterns. Tests were conducted to verify the integrity of the UHF link under Earth ambient conditions. Lander RCE and orbiter RRS development hardware were tested under laboratory conditions. Tests using a development SBRA were conducted with DSN equipment to determine rf compatibility. In addition to conducting single carrier uplink with single carrier downlink and dual carrier uplink with single carrier downlink tests with VL hardware only, VL and VO hardware were tested together with the DSN in dual carrier uplink with dual carrier downlink rf modes.

Guidance and Control Subsystem Tests

Subsystem development testing was performed entirely by analysis and simulation. Advances in the state-of-the-art G&C analysis and simulation techniques allowed deletion of expensive free-flight air bearing and servo table tests from the test program. A block diagram of the subsystem development test simulation is shown in figure 2. This simulation was referred to as the VCMU. The lander flight computer (GCSC) was simulated by a separate computer which was capable of emulating the flight software execution on a one-for-one, bit-by-bit basis. The VCMU operated in two modes, the simulation mode (sim mode) and the hardware mode (H/W mode). The sim mode used mathematical models of the G&C components. The H/W mode used prototype or development G&C components and propulsion valves. Even though it was not specifically shown in figure 2, the VCMU could operate in mixed modes with any combination of real hardware, simulated hardware, and emulated computer.

Support equipment was provided for operation of the IRU, the RA, and the TDLR. The IRU was mounted on a platform which allowed manual positioning of the IRU in any fixed orientation. The IRU support electronics processed lander state variables (angular rate and linear accelerations) and introduced appropriate error signals into the IRU servo control loops. This approach produced

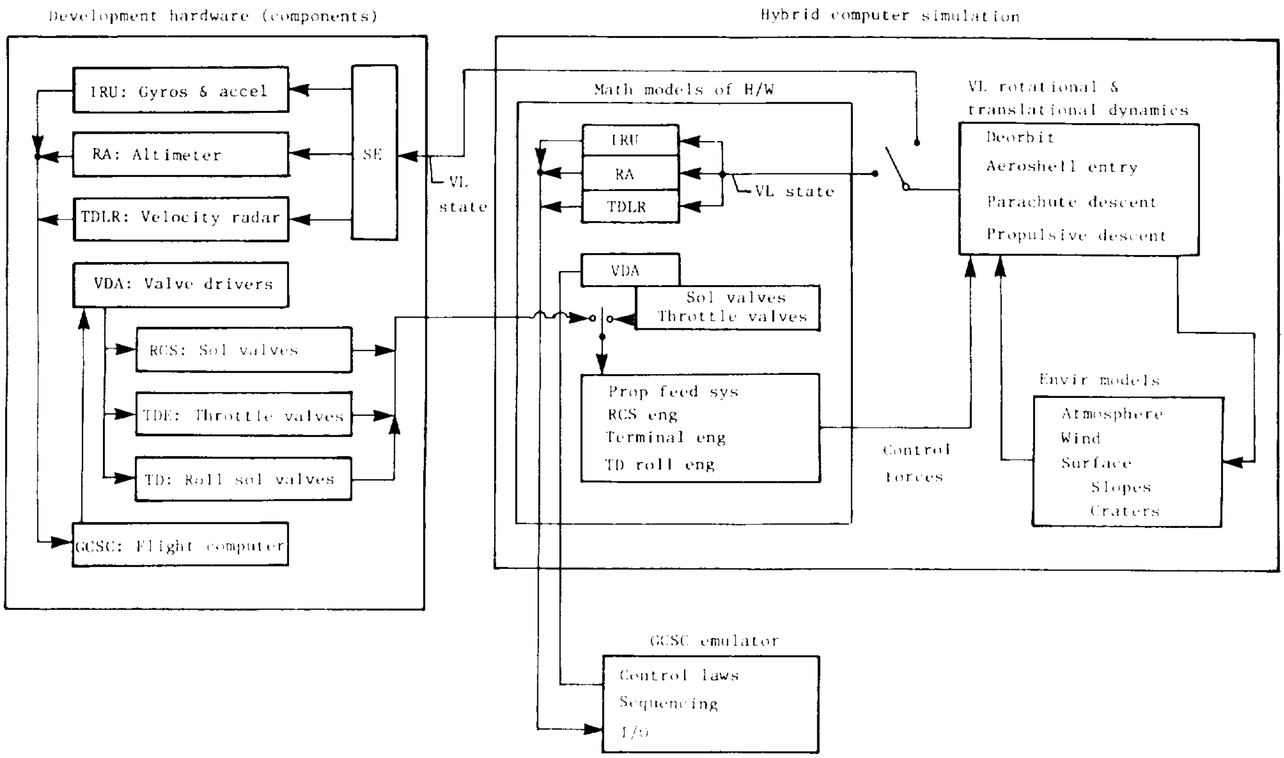

Figure 2.- Guidance and control simulation (VCMU).

proper dynamic response by the gyroscope and accelerometer torque rebalance loops without the use of a servo driven rate table or centrifuge. Steady-state calibrations preceded each block of H/W mode runs. Compensation for Earth rotation rate and gravity, IRU biases and scale factor, and support equipment bias was provided in the hybrid computer computation of lander state. The support electronics received the RA-transmitted signal and introduced appropriate time delays into the signal returned to the RA. The time delays were determined by the simulated altitude of the VL. Likewise, the support electronics introduced Doppler shift as a function of lander velocity state into the signals returned to each TDLR beam. The two-way interface between the RA and TDLR and their respective support electronics was via rf link using antenna hats. The G&C simulation (VCMU) was used for subsystem performance testing and for flight software development.

Subsystem performance testing.- Performance testing was performed in both the sim mode and H/W mode. A matrix of deterministic test cases explored boundary conditions established by error sources and environments. Specific performance tests were made to verify the linearized subsystem error analysis program. The VCMU runs propagated deterministic errors down the descent trajectory. The results were compared with the results from similar cases in the error analysis program. Special VCMU cases were also run to verify the sampled data stability analysis program. Loop gains were varied and transport lags were deterministically introduced and varied until instability occurred. The results were compared to predictions obtained from the stability analysis program.

A special G&C and propulsion interface test was made with the VCMU and development propulsion hardware. The propulsion hardware consisted of tanks, feedlines, and terminal descent engines mounted on a mock-up of the lander structure and located in a rocket motor firing facility. The VDA was located at the propulsion facility for this test. A laser digital data link carried throttle valve commands generated by GCSC to the VDA at the propulsion facility. Terminal engine thrust chamber pressure measurement was returned to VCMU via the laser link. The simulation of the propulsion system in the hybrid computer was replaced by the computation of engine thrust based on chamber pressure. The integrated test verified G&C and propulsion interface compatibility and performance during a simulated terminal descent.

Flight software development.- The emulated flight computer was initially used in the open-loop mode for non-real-time, bit-by-bit diagnostic testing of the flight code. Module level testing of the emulated software package was performed both open loop (trace mode) and closed loop (simulation mode). Integrated module testing was similar to the subsystem performance testing described in the previous section with the test cases being defined to stress the software and to verify sizing and timing analyses.

Radar Tests.- RA antenna tests were performed to determine the effect of lander structure and aeroshell configuration of the radiation pattern and gain characteristics. The tests were conducted on both the aeroshell antenna and the lander antenna with full-scale mock-ups of the aeroshell and lander structures. Similarly, velocity radar (TDLR) antenna pattern tests were performed. Radar cross-sectional data were obtained for the separated aeroshell and parachute base cover. The tests were conducted in an anechoic chamber with scaled models. The data were used to support RA false target discrimination analyses. Both the RA and TDLR were also mounted in an appropriate aircraft for airborne performance testing over terrain with varying degrees of roughness, surface slope, and reflectivities. The TDLR test included aircraft simulation of a typical terminal descent trajectory.

Propulsion Subsystem

Subsystem tests for the propulsion subsystem included open-loop RCS/deorbit test and open- and closed-loop terminal test. The RCS deorbit and terminal portions of the propulsion subsystem were operated open loop to demonstrate compliance with subsystem requirements. The terminal propulsion components were operated with the G&C subsystem to determine operating characteristics during a simulated terminal descent phase of the mission.

Structures and Mechanisms Subsystem Tests

Subsystem tests for the structures and mechanisms subsystem included 3/8-scale-lander drop tests, bioshield latch separation joint tests, ground, launch, and midfrequency vibration tests on the lander dynamic test model, bioshield heat compatibility tests, and special decelerator and aerothermodynamic tests. A 3/8-scale model of the lander was dropped onto a planar surface to verify stability and landing dynamics and development of landing-gear characteristics. Tests were conducted on a full-scale bioshield with a simulated lander structure to verify capability of the bioshield latch joint to contain gaseous N_2 pressure and to verify the separation function. In addition, these tests established a leak ratio, verified structural adequacy, and established assembly techniques. Low-level vibration tests were conducted with the LDTM to determine primary natural modes, frequencies, and damping for launch and terminal descent lander configurations. Midfrequency sinusoidal vibration tests were conducted on the LDTM attached to the VO adapter truss to simulate staging and spacecraft separation transients. These tests were conducted on development hardware at levels for qualification and flight acceptance to obtain preliminary response data, to develop techniques for later tests, and to assess dynamic effects on the propulsion subsystem components and HGA deployment mechanism assembly. The LDTM was also mated with the ODTM at JPL for spacecraft vibration tests. The LDTM/ODTM test configuration was subjected to vibration tests in three orthogonal axes to evaluate the effect of VO/VL interactions on the dynamic response at V S/C subsystem and component locations, to evaluate the adequacy of V S/C secondary structure, to evaluate the adequacy of component sinusoidal test levels, to evaluate the adequacy of PTC test levels, and to obtain data for comparison with analytical results. The precursor bioshield heat compatibility test was conducted to determine the thermal effects resulting from exposure to sterilization heat cycling at qualification levels.

A special test program covered development and qualification testing of the Viking parachute system. A combination of laboratory testing and flight testing of full-scale systems and elements was accomplished. Subsystem qualification was achieved through a series of functional flight tests with the BLDTV. Those tests necessary to provide performance data for analysis to verify that the design met the applicable design requirements were conducted. During the development testing, low-altitude drop tests were conducted to acquire data on parachute deployment and inflation, packing bag functions at deployment, and descent velocity for parachute and structural integrity verification. The drop tests were conducted on the complete subsystem to demonstrate min-max dynamic-pressure conditions with parachute deployment initiated by the mortar.

The aerothermodynamic test program used wind-tunnel and vacuum-chamber tests on various full-scale and subscale lander models to obtain data on the aerothermodynamic characteristics of the Viking entry vehicle. The primary objectives of the aerothermodynamic test program were to provide aerothermodynamic characteristics of the entry vehicle and lander configurations, consisting, in part, of the following information:

Static force and moment data for use in trajectory analysis and trajectory reconstruction

Dynamic stability data for configuration development, verification of controls design, stability analysis for BLDTV, and trajectory analysis

Pressure and convective heat-transfer data for direct design information, confirmation of design, support of entry and terminal phase science, and support of aerothermodynamic analysis

Static force and moment data as inputs for aerodynamic analysis of aeroshell-lander staging

Free-flight mass-scaled model parametric data for parachute base cover staging from lander

To provide aerothermodynamic characteristics of combined lander (entry vehicle + decelerator and lander + base cover + decelerator) consisting of

Drag performance data of parachute in wake of entry vehicle and in wake of lander + base cover for development and analysis

Parachute-alone drag performance

Wake surveys behind entry vehicle for performance analysis

Qualitative assessment of parachute deployment shock load

To provide data pertinent to systems design and analysis associated with the following items:

Influence and effects of terminal descent engine power on lander aerothermodynamics and on lander system thermal environment

Influence and effects of combined RCS deorbit engine thrust

Thermal control of lander components

Support of meteorological science

Support of structural dynamics analysis

Thermal Control Subsystem Tests

Development tests necessary to achieve the required level of confidence in the design and performance of the flight thermal control subsystem included component level, subsystem level, and special development tests. Component level tests have been discussed previously. Subsystem level tests utilized the TETM to evaluate the thermal behavior and control capability throughout the VLC. Where applicable, the TETM was combined with the OTES to determine the interaction between the VLC thermal control system and the orbiter. Tests verified the overall capability of the thermal design to meet mission requirements. Special temperature instrumentation was installed on the TETM to obtain the required data. Special development tests were conducted to evaluate the temperature distribution of the RTG's during cruise mode and the heat-transfer characteristics of the RTG in a Mars wind. Further thermal testing was performed during the qualification and flight acceptance system level test programs.

The primary purpose of the thermal subsystem tests using the TETM was to verify the VLC thermal design, to verify the analytical models used in design, to verify that the test equipment provided proper environments, and to verify that the thermal control hardware performed satisfactorily in the test environments. Secondary objectives included determination that the test techniques were adequate for thermal components including the thermal switch, heater thermostat systems, insulation, venting, surface coatings, and the prelaunch water-cooling system. The test lander used flight-type structure and flight-type thermal components. In general, all other components were simulators having the external geometry, thermal mass, surfaces, and coatings of flight components. Component power dissipation was simulated by heaters in each component. The RTG's were replaced by electrically heated thermoelectric generators (ETG's). All power to components was supplied and programmed from supplies external to the test chambers. Excess ETG-generated power was externally dissipated. The TETM Mars surface simulation test was conducted first as it was considered the most critical to the component designs. This test program was made up of two parallel efforts leading to two principal objectives: (1) development of the lander thermal control system and (2) development of Mars environmental simulation techniques applicable not only to development testing but to qualification and acceptance testing as well. With regard to the second objective, an iterative method of development was used, whereby the test techniques were established during the TETM program and were improved or updated during the PTC system level test program. The primary objectives of this test were to verify the lander thermal control system design, verify or correct the thermal mathematical model, provide thermal control design data, verify and/or update the test configuration mathematical model, investigate the freezing and thawing behavior of terminal descent propulsion residual propellents during the landed phase, assess lander steady-state heat loss and convection characteristics during a cold-soak test, evaluate surface sampler boom thermal effects under hot-case Mars surface conditions, verify and/or update Mars surface simulation test procedures, evaluate lander body thermal stress effects, and determine thermal effects on the flight configured electrical harness. The TETM was tested in three simulated environments: nominal environment (chamber-wall temperature at $89^\circ \pm 5.5^\circ$ C ($-190^\circ \pm 10^\circ$ F)) with nominal power duty cycle; cold extreme environment with minimum power duty cycle, maximum allowable deviation from subsolar latitude; hot extreme environment with maximum power duty cycle subsolar latitude. Tests were conducted in a 7.6-m (25-ft) thermal vacuum chamber. The lander was mounted on a gimballed ground plane simulator which rotated 180° providing angular relation to the solar beam simulating the Mars day from sunrise to sunset at night with the beam off. The ground plane simulator was temperature controlled to simulate predicted Mars surface temperatures. The chamber walls were temperature controlled to provide radiative and corrective heat sinks. The chamber pressure and atmospheric composition was controlled to simulate the hot and cold predicted conditions. The test was planned to bracket the hot and cold extremes expected on Mars. For the cold case, the chamber was not equipped to provide forced convection. The radiation surfaces of the ground plane simulator and the chamber wall were controlled to drive the lander surfaces to the analytically determined temperatures. No convection was assumed for the hot case. CO_2 at 2 torr (1 torr = 133.322 Pa) was used for the hot case and at 35 torr for the cold case. The ground plane simulator and chamber walls were controlled to temperatures calculated to provide the Mars hot environment. The solar beam intensity was increased to simulate the effect of sand and dust on the lander. Maximum power dissipation was used in the hot case.

Sterilization tests were performed next on the TETM in a temperature-controlled oven. The primary objectives of the TETM sterilization tests were to develop a heating cycle which would allow all VLC components to be subjected to the terminal sterilization requirements within the planned design time-temperature limits; determine temperature instrumentation requirements necessary to verify terminal sterilization temperature requirements; verify and update thermal mathematical models to provide supporting data for development of the terminal sterilization procedure; identify those areas of the VLC which required thermal hazard monitors; determine thermal effects on flight-configured electrical harness; and evaluate lander body structure thermal stress effects. The lander was enclosed along with the base cover and heat shield in the sterilization capsule. Overheating of the ETG's was prevented by a coolant loop coupled to the ETG's. The same loop helped speed the heating and cooling cycles. The ETG's were the only components with electrical power for these tests. The oven atmosphere was mostly nitrogen with oxygen limited to 2.5 percent. Oven temperature was increased linearly to 110° C (230° F) and reduced linearly after a 40-hour soak.

Cruise testing was then conducted in the thermal vacuum chamber. The objectives of the TETM interplanetary cruise solar vacuum tests were to verify the thermal design in the near-Earth and near-Mars environments; evaluate system and component thermal responses to natural or induced environmental transients to provide data necessary to verify, correct, or improve thermal design; provide test data which were to be combined with analytical predictions to develop or modify qualification test procedures and requirements; and determine thermal effects on flight-configured electrical harness. The test vehicle included the lander, heat shield, base cover, bioshield base, and a simulated orbiter (OTES) with interfacing truss. Hot and cold cruise conditions were obtained by using the maximum and minimum power profiles for the lander components and the orbiter interface. The chamber walls were at LN_2 temperature to simulate deep space. At the end of the cruise test a preseparation checkout was simulated.

Postseparation solar vacuum tests were conducted with the TETM in the thermal vacuum chamber. The objectives of the tests were to determine the capability of the thermal design to maintain VLC components within design temperature limits, provide test data which in conjunction with analytical predictions were to be utilized to develop or modify thermal-vacuum qualification test procedures and requirements, and determine thermal effects on the flight-configured electrical harness. The test vehicle consisted of the lander heat shield and base cover. The lander was hung with the base cover facing the solar beam. The solar intensity was adjusted to provide the amount of heat expected during coast. After a soak at cruise conditions, the power profile was provided to the components for the preseparation checkout followed by the postseparation power profile. This test simulated entry conditions for internal components only as no entry heat was imposed on the lander exterior. The short duration of entry makes the lander internal temperatures practically free of the entry heating.

The objectives of the cruise portion of the special development tests to evaluate the RTG's were to verify thermal analysis techniques and determine the temperature distribution of the RTG's during the cruise mode with the Mars surface windshield installed. A test article simulating the cruise configuration of the VLC was placed in a thermal vacuum chamber. Temperatures were monitored and recorded with the vacuum chamber pressure at 10^{-5} torr or less and the LN_2 cooled shroud operating. Characteristics of the RTG's under Mars wind conditions were determined during the aerothermodynamic test discussed in the section "Structures and Mechanisms Subsystems Tests."

Site Alteration Tests

Because of the possibility of altering or contaminating the Mars surface by the terminal descent system in the vicinity of the lander, special tests were conducted to determine these effects. The objectives of these tests were to determine the organic contamination effects on simulated Mars surface and atmosphere, determine the physical effects of exhaust gases on simulated Mars surface, determine the physical effects of exhaust gases on the lander, and conduct detailed investigation of thermal and chemical effects of exhaust gas impingement. Tests were conducted in a simulated Martian atmosphere using a prototype lander terminal descent engine and simulated Martian surface materials. A full-scale, one-third section of the lander body with terminal descent engine, propellant tank, landing leg, and footpad was used. Several terminal descent engine nozzle configurations were used to find the design with minimum site alteration impact.

System Development Tests

Following the successful completion of subsystem development testing the development components were utilized in system level development tests. The test requirements for the system level development tests were to evaluate potential alternative operating modes; review the results of prior testing to permit optimization of subsequent tests; perform all testing in accordance with approved procedures; perform all system level assembly and tests by trained and certified engineers and technicians; standardize all system level functional tests to the extent practicable to provide continuity and to permit correlation of test results between test phases and test articles; provide mandatory inspection points throughout the test procedure; maintain time and cycle records in test logs for devices, components, subsystems, and systems as applicable (time and cycle critical records were reviewed for appropriate disposition by the responsible organization when the operating time exceeded the maximum permitted in the applicable specifications); perform structural and component alignment to the requirements of VLS alignment criteria; and perform cleanliness requirements as specified by the VLC contamination control criteria. In addition, the following constraints were met before beginning any system level test of the lander or assembly:

The configuration of the lander STE and supporting software was verified

The configuration of the Viking lander system or an assembly was verified

The response of the lander to the test command sequence during system level tests was monitored by the STE. The following ground rules were used in establishing data analysis procedures:

(1) All lander engineering test data were evaluated in real time to assure that test parameters satisfied the lander system test and checkout requirements; science test data were evaluated after reconstruction and decommutation

(2) Both raw and processed data were recorded

(3) All variations from procedural requirements were recorded

(4) All data obtained were reviewed

(5) Success criteria for each test were derived from a review of the system specifications, system test descriptions, test outlines, and detailed test sequences

During lander system test operations, the following actions were taken when a problem occurred:

When the problem was of the nature that could create danger to personnel or damage to associated equipment, the test was stopped and the vehicle and STE were placed in a safe mode

Applicable trouble shooting steps and fault isolation processes were documented according to the Viking lander system quality assurance plan

If any failure or malfunction of the test article occurred, continuation of testing was determined by an investigation of the nature and cause of the failure or malfunction; the need for corrective action was determined and consequent retest requirements were established prior to resumption of testing

Retest was performed when the occurrence of problems required the replacement of a component or assembly

The system level development tests utilized the STB. The STB was a collection of the facility, personnel, AGE, TSE, simulators, and VLC components required to perform subsystem development testing, interface validation, software development and validation, test sequence verification, subsystem tests which require a system configuration, checkout hardware testing, hardware change evaluation, system level support for PTC and flight capsule system testing and flight operations, alternate mode testing, and special tests for the VLS. Figure 3 depicts the primary hardware configuration for the STB. The STB was divided into two functional test areas: the VCMU area and the STA. The VCMU area was designed to support VLS G&C subsystem testing and integrated system testing requiring G&C subsystem participation. The STA provided for all the VLC subsystem tests and for integrated system testing in conjunction with the VCMU. Guidance and control subsystem testing was performed independently in the VCMU. VLC subsystem and system electrical mechanical interfacing

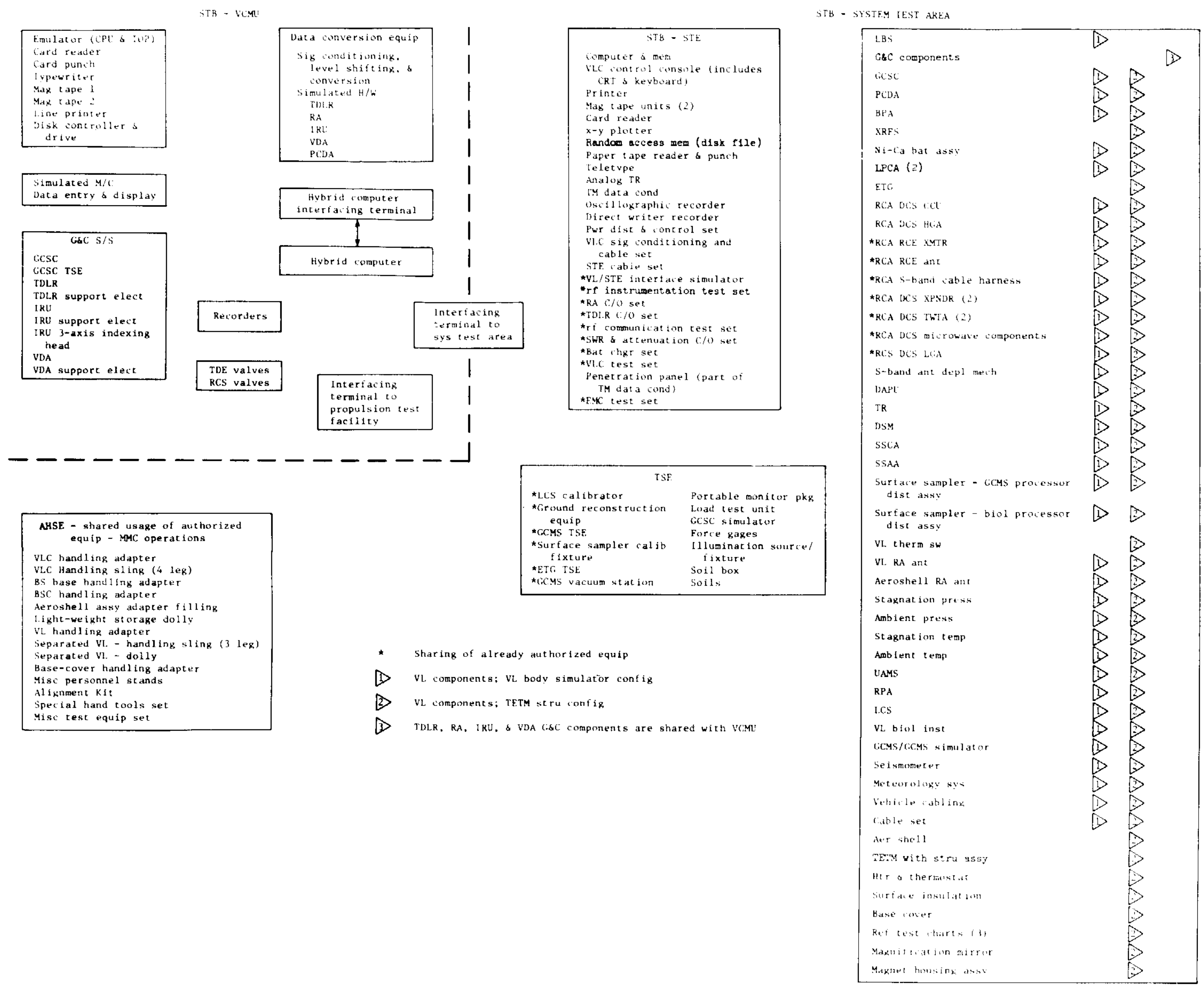

Figure 3.- General configuration of STB.

and functional compatibility tests, system integration and postlanding software validation tests, and system integration and special tests were performed with the VCMU operating in conjunction with the system test area facilities or with G&C components physically integrated into the STA. The objectives of the STB tests were

To validate VLC electrical and mechanical interfaces and functional compatibilities at the earliest point in time

To provide dynamic evaluations of the integrated guidance and control subsystem performance, including variation of design parameters

To evaluate components and subsystems in a partial or complete system configuration

To obtain power profiles of components, power system, and projected VL flight missions as represented by flight sequence and landed modes of operation

To support development, verification, and validation of GCSC postlanding flight software

To provide validation of STE and STACOP software and verification of test sequences

To provide verification of van set to VLC compatibility prior to van set/PTC integration

To obtain engineering information related to science instruments in a system configuration

To evaluate variations of system operating parameters

To evaluate selected system design modifications

To provide support for PTC and FC test operations

To provide support for flight operations during the FOS design period

To verify the transfer of functional operation for all redundancy in case of failure of either element of the redundancy

To verify seismometer response to lander mechanical operation as well as structural transmissibility

To provide for development, demonstration, verification, and test of an antenna pointing system

To verify alternate modes of operation

The VCMU configuration shown in figure 4 provided a capability that was oriented toward G&C subsystem development and test. The G&C subsystem

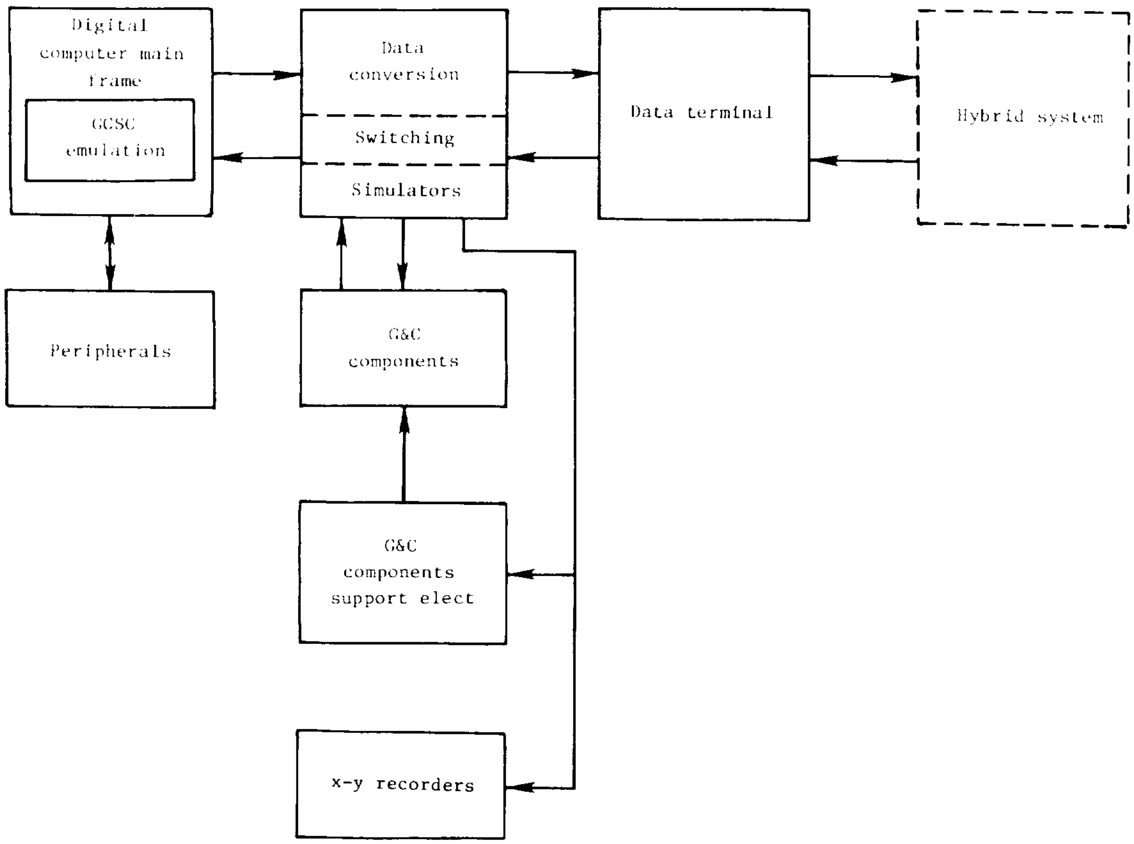

Figure 4.- Functional diagram of closed loop.

components were assembled into a functional configuration, interfaced with each other and with test support and simulation equipment as required to perform tests to meet G&C subsystem test objectives as previously discussed. The G&C components were interfaced electrically with the other VLC components located in the STA during certain integrated system phases. The VCMU configuration also supported propulsion subsystem RCS/deorbit open-loop and terminal descent open- and closed-loop functional firing tests. The VCMU included the following primary equipment:

General purpose digital computer system

Hybrid computer (support function only, not physically located in STB)

G&C development hardware and associated test equipment

Class I and class II simulation devices

Data conversion equipment and computer interfacing terminals

Interfacing terminals

Analog recorders

The general purpose digital computer system was used to emulate the GCSC during STACOP and flight software development and was used to control the VCMU system in the assessment of G&C subsystem performance. The hybrid computer was used for generating and simulating vehicle dynamic analog stimuli and responses as controlled by the main computer test program or by internal programs. A complete set of G&C subsystem development hardware and supporting TSE was provided. Class I simulation devices substituted for a real G&C component and could be interfaced with the GCSC or a simulated GCSC I/O interface. Class II simulation devices substituted for all non-G&C interfaces (PCDA, BPA, etc.) and interfaced with the GCSC and the main computer for control functions. Data conversion equipment and computer interfacing terminals provided all necessary hardware for data conversion, level shifting, and signal conditioning required for interfacing the various elements of the VCMU. Interfacing terminals were provided for interfacing the VCMU to the system test area and to a propulsion test facility. The x-y and x-t analog recorders were provided for recording dynamic analog signals. The VCMU also supported software development and verification to provide main computer, STACOP, and flight software on a time schedule adequate to support system test area operations, the total Viking test program, and flight operations. The software programs included VCMU master control programs, VCMU simulation control programs, hybrid simulation program, STACOP software, flight software, and software for GCSC/propulsion subsystem loop performance testing.

System and Subsystem Compatibility Tests

Prior to accomplishing system level STB tests, some subsystem verifications were made with the STB. System and subsystem compatibility tests, system and subsystem electrical and mechanical interfacing tests, and functional compatibility tests accomplished the following objectives:

- Perform mechanical form, fit, and function checks of the VLC components on a lander body simulator
- Verify VLC component, subsystem, and system electrical interfaces
- Verify STE-VLC interfaces
- Perform functional checkout of VLC components
- Verify component and subsystem design against design requirements
- Determine VLC component power profiles
- Verify VLC component design changes

Evaluation of the VLC component mechanical and electrical interfaces was accomplished during and after assembly of the hardware into a simulation of the landed configuration of the VLC. An LBS with lander cables was provided for mounting and interfacing the components. The LBS consisted of a simulated VLC top plate (equipment plate), simulated VLC side plates, modified to mount the aeroshell harness and components, and a three-leg adjustable mounting fixture. The structural configuration of the LBS represented the flight VLC to the extent that all components which normally were mounted on the lander body or sides were mounted on the same physical envelope as they were on a flight capsule. The simulated top and side plates were the same size and shape as the lander plates but were milled for weight simulation. The mounting provisions for the components mounted on the top plate were in the same relative locations as for the actual lander. The simulated side plates provided the same mounting locations for components as for the actual lander. The three legs (not simulating actual lander legs) were attached to the side plates and were capable of supporting the complete LBS, VLC components, lander cabling, and test harness. After all components were assembled on the LBS, hardware electrical and mechanical compatibility was verified by visual inspections and functional tests with and without the STE. Testing was accomplished without the use of RTG's or ETG's in this configuration.

Mechanical form, fit, and function tests.- During mechanical form, fit, and function tests, VLC components were mounted on the simulated lander body top plate and side plates. Observations and determinations of any obvious obstructions, interference, and alignment problems were made and corrective measures were implemented. Components not mounted on the top plate or side plates were analyzed for possible interference problems. Final STB mechanical form, fit, and function checks were performed when the components were mounted in the TETM structure. In this configuration, several subsystem tests were performed.

Power subsystem tests.- The power subsystem was tested for determination of component compatibility, parametric voltage with subsystem operation at lower and upper voltage limits, noise and ripple levels, transient voltage characteristics, voltage drop, power management, voltage regulation, charging capability (charge rate, current limiting, switching, etc.), input/output impedances, switching characteristics, overload capability, malfunction isolation capability, verification of pyrotechnic control circuits (energy storage and firing characteristics), power distribution and grounding checks at all vehicle cabling connections, and functional compatibility of power subsystem interface with other VLC subsystems.

Guidance and control subsystem tests.- G&C subsystem testing was conducted after being phased into the simulated lander body configuration for determination of compatibility of all G&C interfaces with other VLC subsystems, functional operation of the G&C subsystem in the system configuration, power profiles of G&C components, STE hardware interface verification and GCSC loading, STE software interface verification with the GCSC STACOP software, and test sequence control.

Telemetry and data handling subsystem tests.- The telemetry and data handling subsystem consisting of the DAPU, tape recorder, DSM, and the interconnecting wiring was verified for proper functional operation when connected into the flight cabling harness. Testing included functional operation of the telemetry subsystem as connected to the flight harness (DAPU blocks A and B were exercised), functional compatibility of telemetry interfaces with other VLC subsystems, functional operation of the telemetry subsystem in the system configuration including all DAPU modes, power profiles of telemetry subsystem components, and functional compatibility of TM/STE interfaces.

Communication subsystem tests.- The communications subsystem tests included determination of compatibility of all interfaces within the subsystem and with other VLC subsystems, functional operation of the communication subsystem in the system configuration, power profiles of communication subsystem components, and functional interfacing with the STE communications test equipment.

Thermal control subsystem tests.- The thermal control subsystem tests included determination of the compatibility of all electrical interfaces. STB-dedicated STE hardware and software were verified and interface compatibility with the VLC system established.

Science subsystem tests.- Surface sampler testing included determination of compatibility of surface sampler interfaces with other VLC subsystems and functional operation in the system configuration, including calibration, and power profiles. Each science instrument was tested for determination of compatibility of all interfaces with the VLC subsystems and instrument power profiles where possible.

System Integration and Software Validation Tests

Following completion of the system and subsystem electrical and mechanical interfacing and functional compatibility tests, system integration and software validation tests were conducted with the following objectives:

Verify interface and functional compatibility of STE with VLC system configuration

Validate STE and STACOP test software

Verify STE test sequences

Determine power system profiles in system level testing in conjunction with VLC system sequenced operation

Verify selected VLC component hardware design changes

Provide support for PTC operations

STE van set and portable STE equipment were interfaced with the VLC electrical system as mounted on the lander body simulator and in the STB. The interfaces were idential to PTC and flight unit interfaces. The simulator cable was a vehicle cabling set. The STE software and test sequences were loaded into the STE computer system and the STACOP software was loaded into the guidance, control, and sequencing computer by the STE. All VLC components or electrical simulators were connected to the flight harness. The data from these tests were analyzed and the test sequences and software were modified as required to meet preestablished test criteria. The operations were continued with tests repeated until interfaces and functional compatibility of the VLC hardware and STE hardware and software were established.

The STB-STE were reinterfaced with the VLC components and integrated testing continued after the van set was removed to support qualification tests. The remaining test sequences required for qualification tests using the PTC were verified.

Test requirements for checkout of the van set with VLC components included verification of all STE interfaces with the VLC, verification of the functional operation of the STE in its testing environment, verification of off-line software and data-file procedures, verification of STE self-test and diagnostic routines in the system test environment, validation of STE and STACOP test software, and verification of test sequences.

System Evaluation and Special Tests

The objectives of the system evaluation and special tests were as follows:

To verify VLC component and mechanism form, fit, function, and mechanical compatibility and interaction with each other and with a flight type structure

To validate flight software

To validate major changes in STE and STACOP software

To verify test tapes and procedures to be used on flight software to flight hardware testing and to verify major changes in test sequences

To determine power profiles

To verify selected VLC component hardware changes

To determine system operational characteristics with varying parameters

To determine system functional operation in alternate modes

To verify transfer to block redundant elements

To support PTC and FC test operations

To support flight operations

To develop operational techniques, demonstrate, verify, and test an antenna pointing system

To determine imagery/lander optical integration characteristics

To determine surface sampler/imagery interaction signatures

To determine the characteristics and record the effects of lander mechanical operations on the output of the seismometer

To determine the transmissibility of the lander between the footpad surface and the seismometer

To verify lander capability to support magnetic properties experiment

To complete power subsystem tests using ETG's

To verify selected lander capability to perform the physical properties experiment

To determine optimum method of sample delivery to XRFS

To support operating life requirements

This phase of STB operations consisted of those tests that were best accomplished in a flight-type configuration which provided a mechanical replica in form and dynamic response of the flight lander. The TETM was used as the primary structure for mounting the VLC component for these tests. The STB VLC components were removed from the lander body simulator and mounted to the TETM structure. The concept was to duplicate the flight lander to the maximum extent possible by using actual components or simulations. All VLC components not required or designated for STB tests were simulated in interface, form, and mass as applicable. STB-STE, TSE, and GFE was utilized to initiate, run, and monitor tests. ETG's were used to support integrated system and power subsystem testing. The STB-VCMU was utilized for simulations and emulations required for representing velocity and attitude references and flight dynamics.

Mechanical, form, fit, and function tests were conducted to verify the mechanical fit for each Viking lander component required to support the functional tests; verify the mechanical compatibility and interactions of the surface sampler, S-band antenna deployment mechanism, high-gain antenna assembly, and meteorology boom assembly; and verify camera alignment.

The VCMU operating in conjunction with the STB-STA and STB-STE was used to validate postlanding flight software. This STB configuration was used to support evaluation and validation of major changes to STE and STACOP software brought about by design changes or testing incompatibilities during system level testing of PTC and flight capsules. The STB was also used to support verification of test procedures and tapes to be used in flight software to flight hardware testing to be performed on each of the flight articles.

Parametric variation testing was performed in the STB to evaluate VLC components subsystem and system performance under varying power system voltage parameters. Signature characterisitcs were recorded. Power bus voltages were varied from the low to the high end of specification values and the effect on bus ripple, noise, and transients measured. Selected test sequences were run with the bus voltages at the extreme specification values. Selected VLC component electrical interfaces were degraded with resultant system performance evaluated.

Alternate operating mode tests were performed to evaluate the lander system in selected modes which may be used during a mission. Special test sequences and hardware changes were implemented to force the system to respond in a manner not normally attained in PTC and flight capsule testing. Selected alternate operating modes including battery failures, safety circuits, backup circuits, and automatic switchovers were verified to operate properly. In addition, selected combinations of these alternate modes were evaluated concurrently.

Redundancy transfer testing was performed to verify that all block redundant elements of the VLC operated per design and that transfer from one redundant element to the other could be accomplished in accordance with switchover design. Where design intent was for switchover if either element of the redundancy failed, both directions of switchovers were verified.

S-band antenna pointing tests were performed to verify and develop the lander system antenna pointing capability. The tests included antenna deployment verification tests; antenna drive step size, slewing step rate, and phasing tests; antenna drive mechanical stop and software program stop evaluation; antenna position telemetry data resolution and linearity with respect to commanded positions; alternate mode capability evaluation, that is, IRU failures, synchronization of antenna pointing programs with antenna positions, and ground update of pointing parameters; complete antenna pointing system (hardware and software) end-to-end performance tests using Earth and gravity and the ephemeris of the Sun or other celestial body; and end-to-end functional tests to evaluate slewing and stowage (parking) capabilities.

Tests were performed on the camera system/lander to determine imagery to lander optical integration characteristics and capabilities. The tests included photogrammetric and radiometric calibration with camera installed in lander camera systems locations; determination and recording of shadow profile and reflectance data; determination and recording of reference test illumination variation; and verification of camera near-field viewing capability including images of lander flight targets and magnets, leg strut gages, footpads, and surface below the descent engines as seen through the surface sampler mirror system. The lander camera and reference test charts were installed in the lander configuration. Calibration was verified at selected points. Camera viewing of the lander capsule and targets was performed using variations in illumination levels and angles. Data developed from these tests were recorded and used to assist with the interpretation of data returned from Mars during mission operations.

Surface sampler imagery interaction tests were performed to determine the low-resolution panoramic view and the high-resolution view obtained by the camera systems of the area accessible to the surface sampler. The capability of the imagery system to view the required surface sampler accessibility area was demonstrated in the lander configuration with the lander camera, surface sampler, and components or simulators accurately located. All components which might obstruct viewability were mounted on the lander configuration. A soil box was available for the surface sampler to perform trenching and other soil-surface sampler interactions within the camera system field of view. The surface sampler was positioned to permit imaging of its magnetic array through the magnification mirror.

Tests were performed to determine the effects of lander mechanical operations on the output of the seismometer. All components and devices which have mechanical operations were installed on the TETM. With the seismometer installed, all lander mechanical operations were performed singly and in selected combinations. Analog outputs from the seismometer and the digital output from the lander telemetry were recorded and analyzed to define signature characteristics for use in interpreting seismometer data returned from Mars. The signature tests were performed with the STB lander positioned on a hard surface and on a simulated nominal Mars surface. Ten soil samples were comminuted in the GCMS PDA during this test in support of physical properties investigations. A seismometer transmissibility test was performed which included verification that transmissibility of the lander between the footpad and the seismometer is greater than 0.8 for seismic level signals with frequencies less than 10 Hz and determination of transmissibility of the lander between the footpad surface and the seismometer for frequencies between 0.1 and 100 Hz.

The STB TETM model was positioned on a simulated Martian surface and was excited by ambient background noise. The surface in the immediate vicinity of the lander footpads, the footpads, and the mounting location of the seismometer were instrumented with sensitive geophones. Measurements recorded from the geophone outputs were analyzed to determine transmissibility. Measurements were made for at least two lander leg stroke positions, three soil types, three lander tilt positions, and with maximum and minimum extensions of the SSAA. Soil was at least 0.3 m (1 ft) deep on top of a concrete slab.

Tests were performed to verify lander capability to support the magnetic properties investigation and included the following:

The surface sampler magnets collected samples and the magnet cleaning device subsequently brushed off the samples to determine the efficiency of the brushing operation

Imagery of the surface sampler magnets through the magnification mirror was performed as part of the surface sampler/imagery interaction test

Imagery of the target magnets was performed as part of the imagery/lander optical integration test

ETG's were installed and connected to the STB lander configuration. Selected tests and test sequences were performed to verify that the hardware configuration was complete and all hardware was in an operating condition.

Tests were performed to characterize lander response to the physical properties of its environment as follows:

With the surface sampler boom extended to the maximum range in its horizontal plane, the vertical and horizontal bending fundamental frequencies and the twisting fundamental frequency were determined

With the surface sampler boom extended to one-half of the maximum range in its horizontal plane, the vertical and horizontal bending and twisting fundamental frequencies were determined

With the surface sampler boom extended to the maximum range in its horizontal plane, deflection of the collector head was measured over a range of upward and downward vertical loads sufficient to characterize empty and full collector head displacements in Martian gravity

Near-field imagery was performed as part of the imagery lander/optical integration test

Trenching and other soil-surface sampler interactions were performed as part of the surface sampler/imagery interaction test

Various tests were made to determine the optimum method of supplying "gravel" to the XRFS input tube.

EMC testing was performed in the STB in accordance with the following objectives:

Verify that special EMC instrumentation mates with VLS connectors

Determine susceptibility thresholds where data were not available from part or component tests

Verify that physical dimensions of the special EMC instrumentation is compatible with the VLS

Verify that special EMC instrumentation, procedures, and software are compatible with EMC test requirements

Identify system level interference problem areas

Noise was injected into the electrical system, and EMC sensor triggering levels were obtained to provide sensor levels of 6 dB. Mechanical and physical compatibility of electrical connectors and sensors were ascertained through fit tests at the appropriate interface. With the special instrumentation installed, the STB was cycled according to the preliminary EMC test procedure to validate the test methods planned for the VLS EMC demonstration test. During this period, special instrumentation was provided to measure and record VLC bus levels. No launch pad, launch vehicle, or orbiter rf sources were simulated and the lander rf system was operated in a closed loop.

STB-dedicated DD components were operated to assist in satisfying component operating life requirements. These tests were performed only if the operational hours on the test specimen during acceptance, development, qualification, and STB subsystem and system testing were insufficient to meet the operating life requirements.

AGE Tests

The major test activities that were performed on the electrical VL STE (fig. 5) are described. The categories of AGE covered by this test program are STE which is mounted in vans, STE which is portable, the associated software, LCE, and AHSE. Development and testing of the first-build STE was accomplished in parallel with the VLC flight system development and culminated in the interface of this STE system with STB. STB tests and STB/STE compatibility tests were described previously. The major phases of first-build STE testing were as follows:

(a) Laboratory development tests

(b) Individual equipment tests

(c) AGE installation and integration tests

(d) Lander/STE interface simulator tests

(e) STE/STB compatibility tests

Subsequent-build STE systems underwent test phases (b), (c), and (d) utilizing the test procedures developed and verified previously on the first build. Final verification of STE design and operation with flight-type hardware was demonstrated through use during the qualification tests of the PTC.

Electrical LCE was used for support of the VLC at LC 41 at ETR. The electrical LCE consisted of the launch coordinator's console, lander hardline interface signal conditioning set, and launch complex cable set (fabricated at ETR). The laboratory development tests and manufacturing tests described in the following paragraphs were applicable to LCE as well as to STE. After fabrication was complete, the LCE was functionally tested at MMC. Since the cable set was fabricated at ETR, a special test cable was utilized for this test. Then the equipment was packed and shipped to ETR for installation at the launch complex. A test was performed after installation and prior to usage of the LCE. Tests for AHSE items were conducted in conformance with requirements defined for each end item. System testing of the fluid and gas pressurization systems was part of these tests.

Laboratory development tests: The following tests were conducted in the Electrical AGE Development Laboratory at MMC. This laboratory contained general purpose electronic test equipment for fabrication and test of electronic assemblies.

Piece part tests: Initial AGE circuit design required that a search be made for piece parts suitable to the intended application. Screening of these parts included functional tests that were made to evaluate the basic performance characteristics of the part to applicable vendor specifications and parametric tests that were performed to evaluate the part for marginality and life under off-nominal or application-peculiar conditions. The results of these tests provided the basis for selection of a particular part or determined the need for additional development effort to acquire an acceptable part.

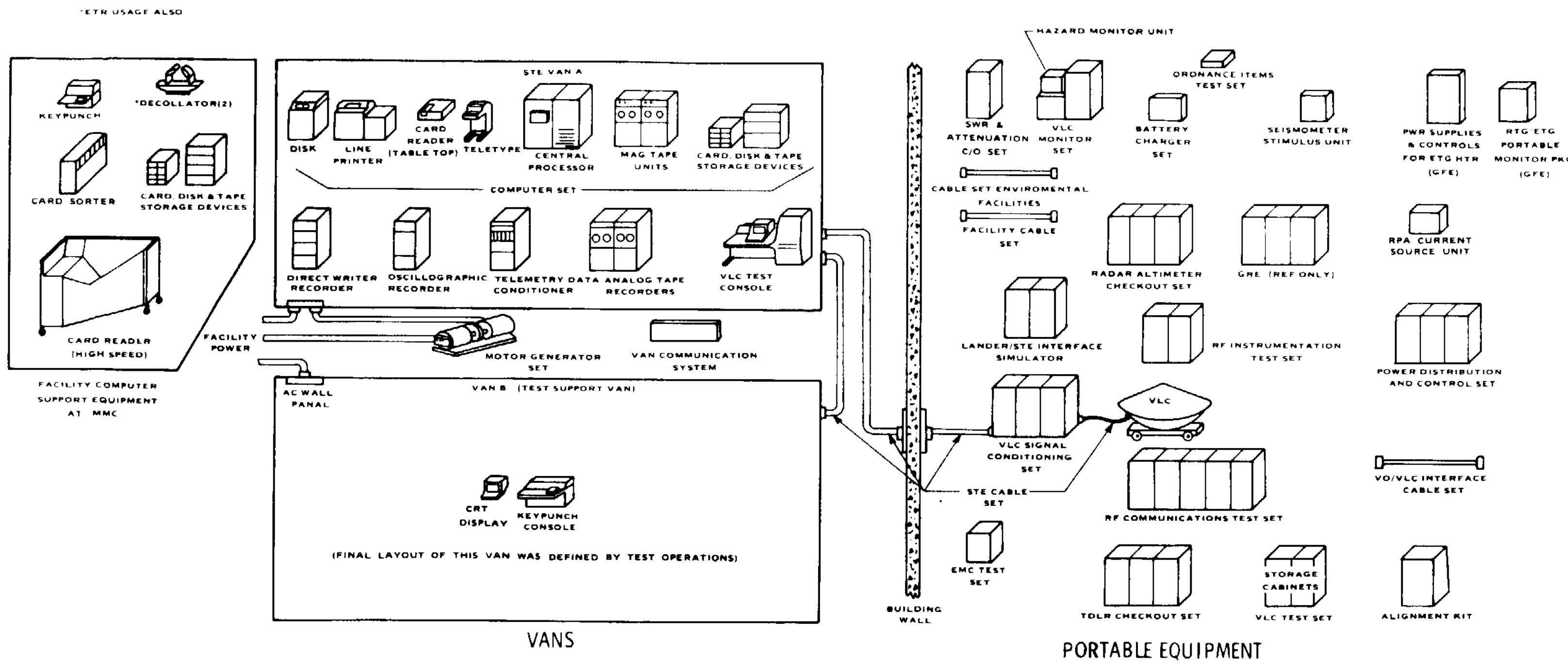

Figure 5.- System test equipment.

Circuit breadboard tests: In order to evaluate applicable circuit designs, tests were conducted on simple breadboards which provided preliminary information for circuit selection or design refinement. These tests included functional tests that were performed to determine and verify operating characteristics with nominal stimulus and environmental conditions, parametric tests that were designed to reveal deficiencies or margins when input or output conditions varied beyond the nominal design limits and to evaluate time and cycle performance, and temperature tests that included evaluation of circuit heat dissipation factors and circuit performance during thermal cycling at design temperature extremes.

Prototype printed-circuit board tests: Prototype PC board types were fabricated (or purchased) and tested except in those cases where the simplicity of the circuit, previous piece part testing, and design analysis proved it was not necessary. The PC board test specifications that were utilized for the testing of the production PC boards were written and validated on these prototype PC boards.

Component prototype tests: Component prototype tests were on the selected circuit designs as packaged in their module configurations. The purpose of these tests was to evaluate packaging compatibility with circuit performance. Functional, parametric, and temperature tests were performed as they were for the circuit breadboards in order to verify operating characteristics and design margins. Component prototype test objectives were extended in some cases to evaluation of basic mechanical component designs and modular packaging concepts.

Interface compatibility tests: There were two categories of interface compatibility that were verified during development testing:

Interfaces between certain STE items - Typical of this category was the computer interface unit which linked certain MMC-designed equipment with the STE computer set.

Interfaces between the VLC and STE - The VLC to STE interfaces consisted of discrete, digital, analog, and rf monitor or stimulus signal points. Tests were performed to evaluate isolation techniques, impedance matching, noise immunity, and other circuit compatibility characteristics. Interface mock-ups or breadboards were utilized to simulate VLC interfaces where required. Actual or anticipated cable lengths were used where practical. Interface test data from the STE computer set were compared with data from the STE computer set for software development. Typical tests performed were transient tests, rf transmission tests, and electrical and mechanical load tests.

Equipment item breadboard models: One of the more complex equipment items of the STE is the VLC test console; therefore, a breadboard model of this item was fabricated and tested. The completed VLC test console breadboard unit was connected to the software development computer set. A full range of integrated tests were conducted to verify both software and hardware. Design verifications included compatibility of the MMC data bus controller with the IOP data bus, discrete and time stimulus capability, discrete monitoring capability, capability of analog input unit to format and load digital words into computer memory, capability of the PCM input units to accept bit synchronizer outputs and perform all nominal functions on the serial data stream, capability of the time code reader generator and range time input unit to supply formatted range time to the computer, and capability of the digital I/O to provide digital stimulus words and acquire incoming digital words or data. Small test sequences were written utilizing the Viking test language. These were translated by the off-line software system and then were executed utilizing the normal controls and displays on the test console. This accomplished a major portion of the integration between the MMC interface hardware and the on-line software system.

Item build and test: Testing of modules, components, chassis, racks, and cables were by the most cost-effective method determined on an individual basis. Automatic testing was used when quantity and type of test warranted. The test procedures were validated during first article test. A brief description of the types of tests performed during the various stages of fabrication and assembly are as follows:

Printed-circuit board tests - All PC boards fabricated received a functional performance test to verify proper operation.

Continuity tests - Continuity and insulation tests were performed on equipment wiring to verify proper continuity of wiring and connections and the insulation resistance between conductors and from conductors to ground.

Ground isolation - Checks were performed to verify the isolation of all circuit returns from equipment structure or from different ground return systems as required by engineering drawings. Ground return to internal and external ground reference points were checked.

Power application - Power was applied to the equipment and checks were made to assure the absence of shorts or opens. Voltages and various distribution and load points were checked to verify design values.

Functional tests - Functional tests were performed to verify equipment in all normal operating modes. External interfaces were simulated where practical and outputs were verified for proper operation.

<u>STE software verification</u>: At all points in the software testing process, functional compliance with specifications for all aspects of the software were verified. Two basic check points occurred in the verification of the software during unit test and integrated system test. Unit checks were made before the various pieces or components of the entire system were brought together. Special temporary software was required to generate the inputs to the unit in test, to create the calling structure, and to log the outputs for later verification. All input variations and operating situations were simulated, and timing and interrupt considerations were included. The purpose of this test was to verify the logic process and the input and output fidelity of the software unit. Integrated tests were conducted upon completion of unit tests with the various system components integrated into a single operational system. Two checkout aspects existed in the integrated tests. The first was a mutual compatibility test which assured that storage allocations, communication buffers, and priorities were properly assigned. The second checkout phase involved software-

generated and breadboard inputs simulating the functions of the intended hardware. Also included were temporary mapping routines which allowed detailed postrun analysis of the logic and process flow during the run.

STE integration and system tests: After the vans had all of the STE equipment items installed, the vans and selected portable equipment items were moved to the STE integration and systems test area. The vans were connected to the portable equipment with an STE cable set. The integration and systems tests verified facility power, the computer set, the van communications system, the power distribution and control set, the VLC test console, the direct writer recorder, oscillographic recorder, telemetry data conditioner, analog tape recorders, the VLC signal conditioning set, and the rf and rf radar checkout sets. Tests were designed to meet the following objectives:

Verify that all interconnections were properly mated and that the electrical grounding was correct and functionally verify all stimulus, monitor, power, and control lines between the STE and the LIS

Verify the capability of the STE to provide digital, analog, and discrete stimuli of the type normally required during VLC test operations

Verify the capability of the STE to monitor PCM, discrete, analog, and digital data of the type normally monitored during VLC test operations

Verify the capability of the STE to provide power under the load conditions normally experienced during VLC testing

Verify the capability of STE computer set and on-line software system to control the various STE equipment, acquire and process data in real-time, display appropriate information to the test personnel, hold and recycle to selected points in the test sequence, and respond to manual requests or control inputs from the VLC test console

Train system test personnel to operate and maintain the STE

Exercise portable AGE which interfaces with other STE in the same manner as during vehicle tests to demonstrate that control and data occur as expected

Verify PCM I/O performance by using either a test tape which simulates selected vehicle data or portions of recorded STE receiver output from a previous test.

AGE environmental test: Confirmation of the AGE environmental requirements were assured by examination of hardware design, verification of prior usage, component level test, or actual demonstration through usage in system test operations of the TETM, LDTM, and PTC test models. The test equipment exposed to PTC and flight lander environments required environment compatibility evaluation prior to PTC and flight lander testing.

QUALIFICATION TESTS

Qualification tests which are sometimes called type approval tests were designed to meet the objectives stated previously. These tests fullfilled the following requirements:

(1) Components/assemblies were qualified at the highest assembly level identified as a remove and replace item from the flight vehicle. Wherever components were packaged together for removal and replacement as an assembly, environmental qualification was made at this level. All new designs were qualified by testing. Existing designs were qualified by prior test data together with detailed analysis of any design modification.

(2) Prior to qualification testing, all hardware was subjected to flight acceptance testing.

(3) Qualification test levels (environmental and functional) were sufficiently higher than flight acceptance test levels to provide confidence that the flight equipment would perform within specification after exposure to flight acceptance testing and mission environments.

Hardware was reviewed to determine if requalification was required as a result of changes in design, changes in manufacturing source or process, a more severe mission environment, or operating condition. To meet these requirements two flight qualification units were constructed for each component. One of these units was subjected to components flight acceptance and qualification test levels and then utilized for extended life tests. For the structures and mechanisms subsystem components, units were used immediately following flight acceptance tests for subsystem tests. The second unit for each component was assembled into the proof test capsule for system level qualification tests following flight acceptance tests on these components.

Component Flight Acceptance and Qualification Tests

Prior to being used for subsystem or system tests, the qualification components were exposed to flight acceptance tests. For those units that were exposed to component qualification test levels, the flight acceptance test levels were integrated into the component qualification tests. This method reduced overall test time for each component and facilitated scheduling of test facilities. Component qualification test and margins relative to flight acceptance levels are described as follows:

The qualification environmental tests were selected for each component based on its type of construction and mission phase operating requirement. Where a particular environment occurred during more than one mission phase, tests were designed to envelope the appropriate levels and durations. The following paragraphs define the criteria for each qualification environmental test and, if appropriate, the equivalent flight acceptance test level.

Heat compatability: Nonoperating high-temperature exposures were applied to all components to demonstrate their ability to function after exposure to the temperatures utilized in the final lander sterilization prior to launch and are as follows:

Flight acceptance test	Qualification test
40 hr cycle at 112° C (233° F)	Three 54-hr cycles at 123° C (254° F)
	Five 40-hr cycles at 123° C (254° F)

Pyro shock: This qualification test simulated the effects of pyrotechnic shock on components located in close proximity to pyrotechnic devices. The pyro shock propagation characteristics through the structure were utilized to generate normalized spectrum and levels for each component. The tests were performed by shaped vibration spectra or actual pyro firings in a test bed. A minimum of three shocks were performed in each direction of the three major orthogonal axes.

Electromagnetic compatibility: Broadband conducted interference, narrowband rf susceptibility, transient susceptibility, magnetic field susceptibility, or high rf field susceptibility tests were performed to assure that the component design was compatible with the lander system specification.

Ascent pressure decay: A test was performed to simulate with margin the dynamic decrease in external pressure expected during launch on those components that included thin-wall vented compartments.

Launch acceleration: This test evaluated components having spring-mass or unpotted assemblies which were not exposed to the higher level entry decelerations or landing shock test.

Launch acoustics: Launch acoustics testing was performed on components mounted on the aeroshell or bioshield surface and exposed to the launch acoustic excitation. The qualification margin was 6 dB relative to FAT levels and the exposure time was 5 min relative to FAT's 1 min.

Launch vibration: This test was performed on all components subjected to structurally transmitted launch vibration. The test was performed in each of the three orthogonal axis and consisted of sine-wave and random vibration. Qualification margins were 1.5 for sine-wave amplitude and 4 for sine-wave sweep duration and 4.5 dB for random vibration and 5 for random vibration duration.

Cruise thermal vacuum: Cruise thermal vacuum simulated the cruise environment and was applied to all components required to operate during and after the cruise mission mode. The test consisted of a pressure of 10^{-5} torr with temperature cycling between high- and low-temperature extremes plus a high-temperature dwell. Performance was measured during the transient period as well as each steady-state dwell temperature. The cycles and levels were as follows:

Flight acceptance test	Qualification test
Three temperature dwells of 4 hr each between high and low temperatures 14° C (25° F) less than qualification. A dwell of 34 hr at the high FAT temperature	Six temperature dwells of 4 hr each between the qualification high and low temperatures A dwell of 238 hr at the high qualification temperature

Corona and arcing: All components which operated above 50 V were tested to demonstrate the absence of corona and arcing at pressures ranging from Earth ambient to 10^{-5}-torr vacuum to a 15-torr simulated Martian atmosphere.

Entry thermal: This external component simulated the entry heating and convective cooling transient during Martian atmospheric entry. The component was stabilized at the high entry thermal initial temperature at a pressure of 1 torr and then subjected to a radiation environment that linearly ramped at 2° C/sec (3.6° F/sec) to a maximum of 167° C (300° F). Total exposure to the thermal radiation was 530 sec. The convective cooling consisted of injection of precooled nitrogen that increased the pressure to 5 torr in 120 sec and an ambient atmospheric temperature of -101° C (-150° F). The test was performed three times during component qualification and once during FAT.

Entry acoustics: Entry acoustics testing was performed on external or surface-mounted components in which acoustic excitation exceeded the level of structurally transmitted vibration. The qualification margin was 6 dB relative to FAT and five times FAT duration.

Entry and terminal descent vibration: This test was performed on all components required to operate during or after entry and terminal descent. Qualification margins were 4.5 dB above FAT random levels, 1.5 times FAT sinusoidal levels, and five times FAT duration.

Landing shock: This test was performed on all components required to function after landing. The shock pulse consisted of a 30g, 22 msec half sine wave applied three times in each direction of the three orthogonal axes.

Surface thermal: This test simulated operation in a 5-torr Martian atmosphere consisting of 90 percent carbon dioxide and 10 percent argon. Qualification consisted of 10 cycles of exposures to 4 hr dwells at the lowest and highest temperature extremes. Qualification margins were 14° C (25° F) and 5 temperature cycles.

Extended life tests: For those components that did not receive extended life tests during development testing or those components that had major changes as a result of the development program, extended life tests were conducted following component qualification tests. The GCSC received extended life tests during both development and qualification. The surface sampler gear motors received were functionally tested during qualification, and the parachute pressure cartridge were functionally tested after long term storage on qualification units.

Subsystem Tests

The only subsystem tests that were conducted during qualification were for the structures and mechanisms subsystem and the G&C subsystem. These tests also included the special decelerator qualification flight tests.

Structures and Mechanisms Subsystem

An LDTM acoustic launch and entry phase qualification test was conducted to evaluate the dynamic response characteristics of the LDTM and the associated hardware when subjected to acoustic spectra experienced during the launch and entry phases, to confirm secondary structure capability to sustain critical design ultimate acoustic loads without failure, and to, assess dynamic effects on propulsion subsystem components. This test consisted of acoustic excitation of the LDTM and the associated hardware for the launch and entry phase environments, including the low-frequency entry acoustic environment. The complete LDTM was mated through the VL-VO adapter truss to a test fixture. The c.g., mass, and inertia of equipment, including propellants, were simulated. The bioshield cap was removed during the low-frequency entry acoustic exposure.

Drop tests using the LDTM in the terminal landing configuration were conducted to meet the following objectives:

To qualify landing system structure to ultimate landing dynamic velocities

To verify analytical dynamic loads and impact shock at selected equipment locations

To assess dynamic effects on propulsion subsystem components

To assess dynamic effects on HGA deployment mechanism assembly

The LDTM was dropped with selected touchdown parameters to determine dynamic load conditions. A subsequent drop test was performed on the LDTM at ultimate dynamic load for qualification. Qualification of the landing system was accomplished jointly by using test data derived from this test and the LSTM landing configuration qualification test.

Pyrotechnic shock environment and aeroshell qualification separation tests using the LDTM were conducted to determine the shock environment induced by pyrotechnic devices that were used for aeroshell separation and mortar firing, to determine the degree of attenuation caused by passage of the shock through structural members and joints, to determine the shock environment induced by the simultaneous deployment of the landing legs, to assess dynamics effects on propulsion subsystem components, to demonstrate separation dynamics, and to demonstrate HGA deployment mechanism and determine the shock induced by the pin puller pyro. The test consisted of installing the aeroshell and mortar pyrotechnic devices on the LDTM, firing the devices, and measuring the high-frequency acceleration response at selected locations. Other shocks were evaluated during separation tests and during component testing. Separation of the aeroshell was performed with necessary local temperature effects simulated and complete separation was accomplished without any residual contact between structures. Measurements of aeroshell separation dynamics were made. The landing leg pyro pin pullers were fired simultaneously and the legs were deployed. Pin puller pyro shock and leg deployment shock were measured. The HGA pin puller was fired and the antenna was deployed after the base-cover separation test.

A lander-orbiter separation qualification test using the LDTM was conducted. The test objectives were to demonstrate the functional performance of the lander-orbiter separation event, to demonstrate the separation dynamics, and to define pyrotechnic shock spectra. The LDTM separation interfaces were incorporated into test fixtures representing the VO and VL. The test article was supported to simulate the effects of reduced gravity. Separation event parameters demonstrated were the separation dynamics, structural integrity, and pyro shock spectra and loads.

A bioshield cap separation qualification test using the LDTM was conducted to demonstrate the functional performance of the bioshield cap separation, to demonstrate the separation dynamics, and to define the pyrotechnic shock spectra. The bioshield cap was supported to simulate the effects of reduced gravity where significant. Separation event parameters demonstrated were the separation dynamics, structural integrity, and pyro shock spectra.

A base-cover separation qualification test was conducted to demonstrate the functional performance of the base-cover separation, to demonstrate the separation dynamics, and to define pyrotechnic shock spectra. The base cover was suspended in a manner to simulate reduced gravity at base-cover separation from the VL. Parameters demonstrated were the separation dynamics, structural integrity, and pyrotechnic shock spectra.

Launch cruise, entry, and landing configuration qualification static structural tests were conducted using the LSTM. The test objectives for the launch configuration were to demonstrate the structural integrity of the bioshield for the combined pressure, load, and thermal launch phase environment; to demonstrate the structural integrity of the base cover and aeroshell for the launch environment; and to demonstrate the structural integrity of the VLC for critical combined acceleration and random vibration loads during launch. The test consisted of several conditions which simulated launch design loads. These included pressure cycling of the bioshield then pressurization to ultimate pressure; application of limit and ultimate pressure, temperature, and inertia loads to the base cover and aeroshell; and application of simulated pressures, loads, and temperatures at limit and ultimate conditions at critical mass points of the VLC.

The test objective for the cruise configuration was to demonstrate the structural integrity of the bioshield and base cover for critical loads and temperatures during cruise and orbit. Separation loads on the bioshield cap and inertial loads on the bioshield base were simulated. Limit and ultimate conditions were imposed and critical thermal stresses on the base cover simulated.

The test objectives for the entry configuration were to demonstrate the structural integrity of the aeroshell, base cover, and lander body during the aeroshell phase of entry; to demonstrate the structural integrity of the separation guide and rail structure; to demonstrate the structural integrity of the base cover for positive dynamic pressure during the parachute phase of entry; and to demonstrate the integrity of the parachute support structure and lander fittings for mortar and parachute loads. Limit and ultimate loads were applied during this test as follows:

Critical pressures, loads, and temperatures were applied simulating the aeroshell entry phase

Critical guide and rail loads and temperatures were applied to simulate separation conditions

Critical pressures and temperatures on the base cover were applied to simulate the parachute phase condition

Critical loads, pressures, and temperatures were applied to simulate mortar and parachute deployment condition

The objective of the landing configuration test was to demonstrate the structural integrity of the lander body and landing legs for selected critical loads during landing. Critical inertial loads not certified by the ultimate dynamic load drop test condition were applied at mass points and reacted at the leg to lander interface. Limit and ultimate dynamic loads were imposed.

Fracture toughness and threshold stress intensity evaluation of Viking pressure vessel alloy was conducted. The test objective was to determine the fracture toughness and threshold stress intensity parameters required in order to assure a reliable fail-safe performance of the propellant tank vessels during the Viking Mission. Fracture toughness was established and the threshold stress intensity was evaluated for the long time service requirements of the Viking Mission. Test conditions closely paralleled the actual mission environment and included the effects of propellant, temperature, and pressure.

Tests were conducted to demonstrate the structural integrity of mechanisms and mounting structure not qualified at the component level for loads during launch, landing impact, and Mars surface operation. Loads were applied to surface sampler acquisition assembly, lander camera mast, HGA assembly deployment mechanism, surface sampler distribution assemblies, meteorology deployment mechanisms, and landing leg assembly. Those areas shown to be critical by analysis were instrumented, and loads, stresses, temperatures, and deflections were monitored and recorded during testing.

Tests were conducted to demonstrate that the landing leg assembly met the maximum performance and design requirements. After assembly test and heat compatibility exposure of the LDTM landing leg assemblies were completed, a series of leg deployment and main strut down-lock tests were performed. Deployment tests included fixture-mounted precursor mechanical deployment verification and LDTM-mounted pyrotechnic deployment qualification. Qualification of the landing leg assemblies included LDTM drop tests at sufficient velocities to demonstrate that the landing system met design requirements.

The approach that was taken to qualify the Viking ablation heat-shield materials was dictated to a great extent by the duration of the environmental exposures to which these materials were to be subjected prior to entering the Martian atmosphere. Briefly, these environments were prelaunch storage, sterilization, and cruise, orbit, and descent vacuum exposure. To determine aging effects of these exposures on the material performance, tests were conducted periodically on material specimens which were preconditioned both in the individual environment and in all environments sequentially. The overall heat-shield test objectives were to establish heat-shield nominal properties and design margins, to define design details in areas which were not amenable to straight forward analysis, to verify the thermal and thermostructural integrity of the heat-shield design as released, and to conduct final qualification tests on the heat-shield materials.

Functional flight qualification tests of the Viking decelerator system were conducted as the BLDT program at the White Sands Missile Range in the summer of 1972. The prime objective of these tests was to verify the satisfactory operation and performance of the full-scale Viking decelerator in a simulated Mars environment and in the wake of a full-size Viking entry vehicle. The general qualification objectives of all the BLDT flights were as follows:

To verify that the mortar provided sufficient velocity to support full deployment of the parachute

To verify that ejection from mortar fire through line stretch, bag strip, and initial full inflation was relatively smooth and free of canopy "dumps" or other discontinuities

To verify that the canopy maintained a relatively stable drag shape after initial inflation and canopy area oscillation phase was over

To verify that sufficient drag performance was produced to support Viking Mission requirements of terminal velocity and aeroshell separation; this requirement referred to quasi-steady-state drag and did not apply to the highly dynamic pulsations that occurred during the parachute opening process

To demonstrate that the parachute had an adequate structural margin to sustain maximum opening loads for the Viking Mission and maintained an essentially damage-free condition through the deceleration phase

To demonstrate that vehicle oscillations were less than or equal to ±25° amplitude in quasi-steady-state descent with no wind when analytically extrapolated to Mars conditions

To demonstrate that vehicle attitude rates were less than or equal to 30 deg/sec in quasi-steady-state descent with no wind when analytically extrapolated to Mars conditions

In order to provide the velocity and atmospheric density equivalent of the Mars parachute deployment conditions, the BLDTV was lifted to approximately 36.6 km (120 000 ft) in the Earth atmosphere beneath a large balloon system. The BLDTV was similar in size and shape to the Viking entry vehicle. Once at the proper altitude over the White Sands Missile Range, the BLDTV was boosted by rocket motors to the proper test conditions of Mach number and dynamic pressure.

Three test points were originally selected to bracket the range of possible Mars deployment conditions. Test 1 (vehicle designation AV-1) was to demonstrate performance and structural integrity at deployment conditions that were in excess of the maximum Mars effective dynamic pressure and in excess of Mach number equal to 2.0. The first test vehicle overshot its intended deployment dynamic pressure by about 23 percent because of vehicle damage incurred during launch. Although the parachute was deployed successfully, damage was sustained in two of the gores. The test was subsequently ruled unsuccessful and its objectives reassigned to a fourth test vehicle (AV-4). Test 2 (AV-2) was to demonstrate performance at deployment conditions in the transonic region and at a dynamic pressure lower than could be experienced on Mars. Test 3 (AV-3) was to demonstrate performance at deployment conditions representing a velocity that was less than the Mars envelope and a nominal dynamic pressure.

The desired test conditions of dynamic pressure and Mach number or velocity occur at Earth altitudes in the 43-km (140 000-ft) altitude range. A combination of balloons and rockets were employed to reach the desired test altitude for each test. Four tests were conducted: two at supersonic conditions and one each at transonic and subsonic conditions. The supersonic and transonic tests required propulsion units built into the test vehicle to reach the desired Mach number. The typical powered flight mission sequence is shown in figure 6. The subsonic test did not need propulsion units but involved simply a free-fall drop from the balloon.

The test vehicle was physically similar to the Viking entry vehicle except for the protruding rocket motor nozzles required on the powered vehicles. The test vehicle weighed approximately 857 kg (1890 lb) at decelerator deployment. On each of the flights on-board instrumentation included forward and aft looking cameras, bridle leg tensiometers, rate gyros, and accelerometers.

Guidance and Control Subsystem

Qualification testing of the guidance and control subsystem was performed via the G&C simulation known as the Viking controls mock-up. The VCMU was described in the section "Guidance and Control Subsystem Tests." The only

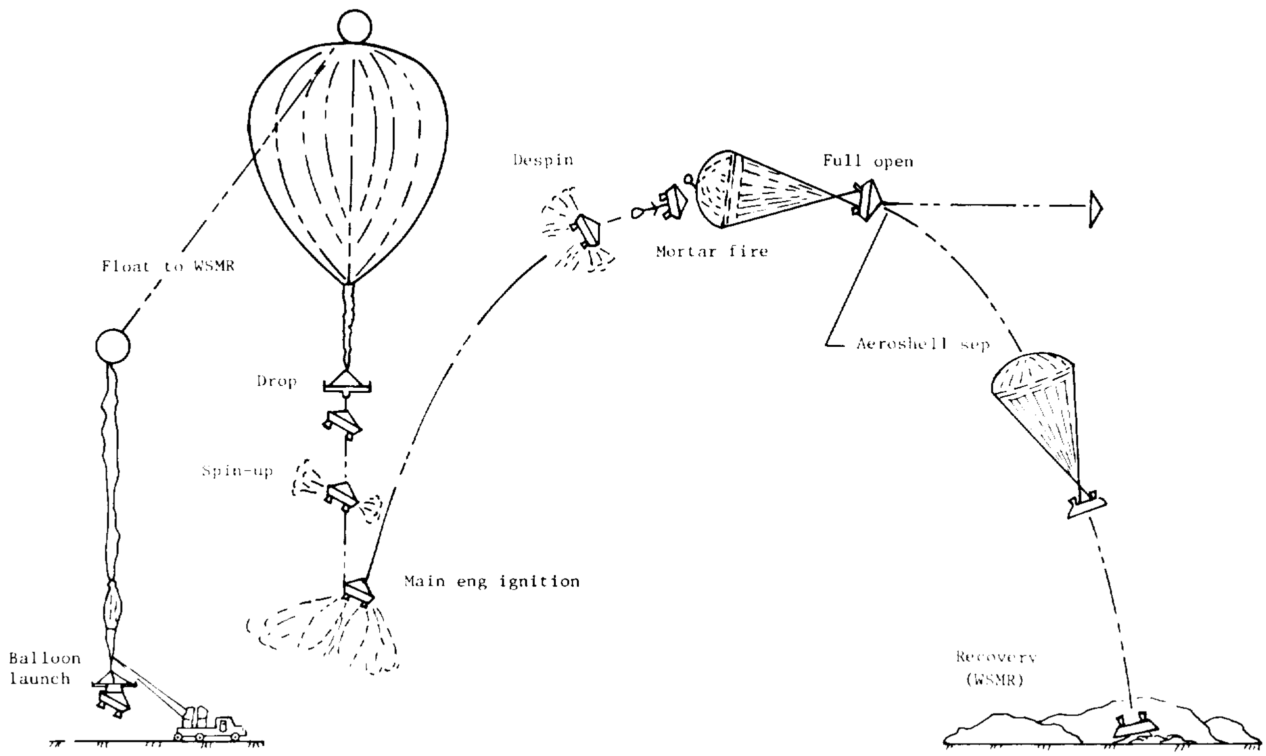

Figure 6.- Typical BLDT mission sequence.

significant change to the VCMU was the replacement of the prototype and development hardware with flight-type hardware. Flight-type hardware referred to the design that evolved after component environmental qualification testing. In most cases, the qualification component was used in the VCMU after completion of environmental qualification tests. The subsystem performance test matrix used for development testing was revised to take advantage of a better definition of the boundary conditions. Selected cases from the revised test matrix were run under strict configuration control.

System Qualification Tests

Qualification tests at the system level were accomplished utilizing flight-type hardware in the PTC. The PTC was a complete lander capsule designed to demonstrate that the VLC system met the performance and the design requirements under anticipated mission environments with required margins; to verify the VLC assembly and test operating procedures; to verify AGE, STE, and software; to demonstrate that the terminal sterilization cycle met planetary quarantine requirements; and to demonstrate all planned backup and alternative operating modes. Beginning with capsule assembly and functional test, the PTC test sequence closely followed the planned mission sequence. VLC environmental tests consisted of qualification level testing. Selected planned backup or alternative modes developed during STB testing and defined as part of the space flight operations plan were exercised during the appropriate test sequence. Figure 7 is a flow diagram of the PTC test program. The major tests identified in this flow are discussed in subsequent paragraphs. A series of electrical and mechanical functional tests were performed on VLC systems. Tests consisted of assembly tests, subsystem verification tests, prelaunch/preseparation checkout, capsule subsystem verification tests, and special tests.

Assembly tests: During the VLC assembly phase, functional tests were accomplished to verify proper hardware assembly and subsystem performance. It was during this assembly test phase that total access to checkout connectors was possible; thus, the most detailed functional testing could be performed. The objectives of the VLC assembly tests were to verify that the VLC components and subsystems were properly assembled, aligned, and calibrated and to establish baseline operating parameters to support the ascending level of tests. The lander body, with the necessary portion of the removable test harness installed, was mounted on a fixture which permitted access to the inside of the internal compartment. All components were installed except the RTG's. Pyrotechnic simulators were used in place of live ordnance items. Flight harness and removable test harness were not interconnected until it was verified that the power at each distribution point was of correct amplitude and polarity. The STE was verified prior to electrical mating with the lander system. Special instrumentation was used when necessary to provide additional test data, hazard monitoring data, and calibration data. During assembly of the VL the assembly tests and functional verification tests were performed. After assembly of the VL, it was inverted in the AHSE fixture for the purpose of detecting and removing any foreign materials.

Power subsystem electrical continuity, single-point ground, and input power were verified and power profiles were obtained. To obtain power profiles, the vehicle was powered from the STE through the orbiter interface and loaded in a sequential manner by turning individual components on or off using GCSC commands. Input bus voltages from the STE were varied from high to low level of the specification in at least three steps. Analog records of vehicle voltage and current variations were obtained.

Communications subsystem VSWR and insertion loss tests were conducted. Telemetry and instrument calibration checks, and BER and antenna positioning tests were run. In order to determine the transmitting and receiving characteristics of the lander S-band and UHF systems, the VSWR, attenuation, and line loss between the transmitter/receiver and the appropriate antenna including all cabling were measured. Power output at the S-band HGA and the UHF LGA was measured after all VSWR tests were completed. An end-to-end verification was performed after installation of flight telemetry temperature instrumentation during vehicle assembly. VL telemetry temperature data were compared with ground instrumentation measurements to verify telemetry data accuracy. The BER of the lander command detectors was measured. The command detectors were operated with an input from the rf test set consisting of idle sequence. The output was compared with the input bit stream for 30 000 to 50 000 bits and the extreme value theory was applied to verify that the BER was 10^{-5} or less. A functional test of the antenna pointing system equipment and mechanism was performed to verify proper antenna positioning. Verification of proper response

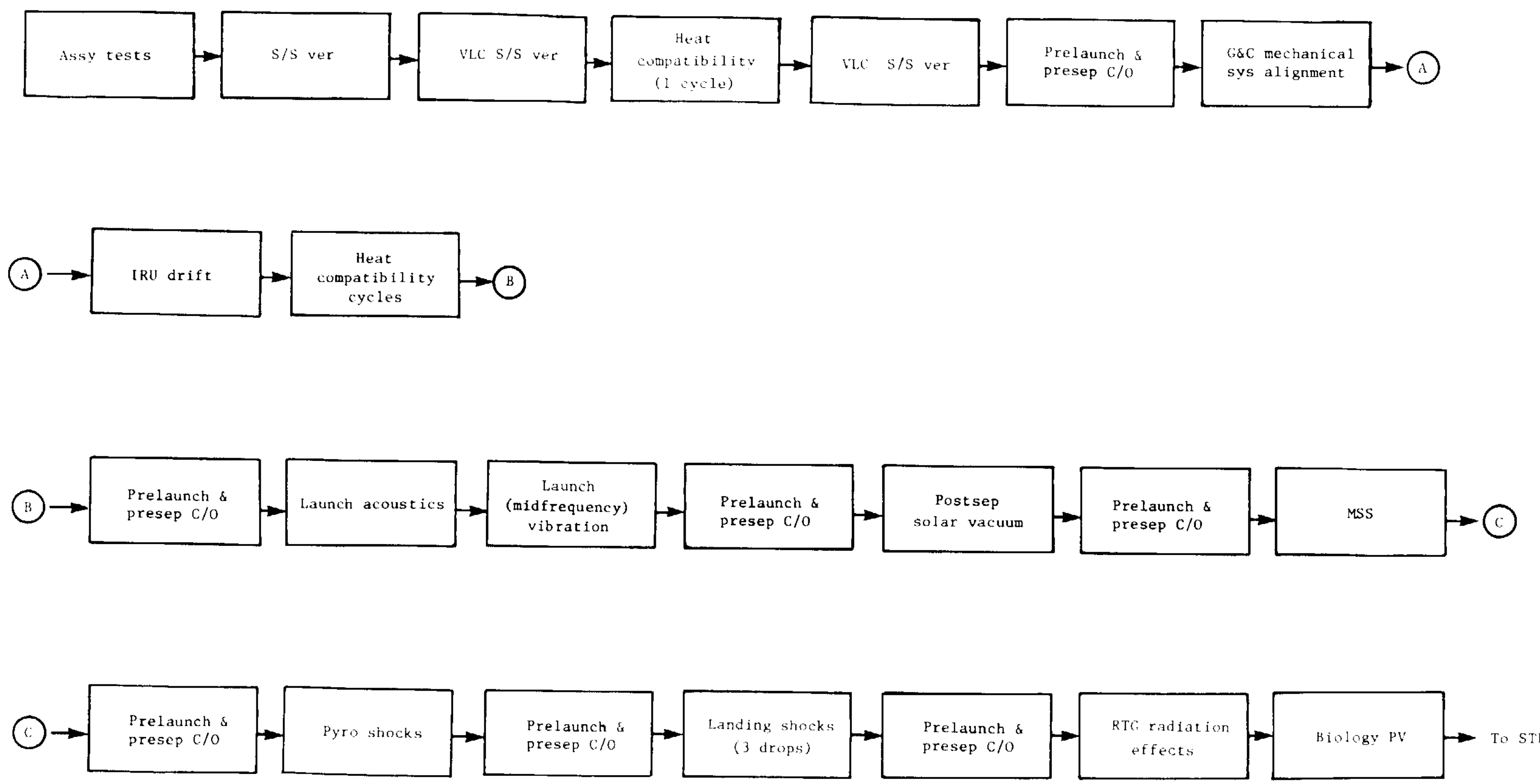

Figure 7.- Test flow of PTC.

of the GCSC and IRU inputs and response of the antenna drive system as commanded by the GCSC was accomplished. The antenna pointing system was commanded through preselected positions of azimuth and elevations to demonstrate the positioning of the S-band HGA by commands from the GCSC, pointing accuracy of the antenna, and the maximum displacement of the antenna. Guidance and control subsystem single orientation drift tests and four position orientation drift and misalignment tests were conducted on the IRU, and G&C alignments with vehicle references were verified. Two basic types of single orientation drift tests were performed for evaluation of IRU characteristics. Evaluation tests with orientation unknown consisted of comparing the preenvironmental sensor outputs against limits after taking into consideration Earth rotation rate and gravity vectors. IRU outputs taken in four different orientations relative to Earth rotation rate and gravity vectors were analyzed to derive alignment and parametric measurements of scale factor and bias, accelerometer triad misalignments, gyros g-sensitive drift, and mass unbalance terms. Components which are required to be aligned precisely in order to accomplish the mission objectives were mechanically verified to be installed properly relative to applicable vehicle reference.

Propulsion system pressurization and leak checks were made on the deorbit and terminal propulsion subsystems prior to and after lander capsule environmental testing. These subsystems were pressurized to operating levels and checked for external leakage with mass spectrometer techniques. Pyrotechnic circuits were verified after pyro simulator installation. Functional tests were performed to verify pyrotechnic circuit continuity and energy levels from firing circuits, safe-arm relays, and simulated pyro initiators, as well as to verify the time correlation of pyro event actuation in response to GCSC to LPCA command signals. The minimum all fire energy delivered by the LPCA capacitor banks was also verified. A low level voltage (5 V dc max) functional verification test was performed as a poststerilization checkout of the pyro.

Structures and mechanisms deployment and calibration checks were made, a bioshield leak check was made, and vent valve operation was verified. All mechanisms which would extend or deploy from a stowed position were actuated to determine proper operation and to verify proper latching. Pin pullers and other retention devices were manually released to permit movement of the mechanism. Proper mechanical alignment was verified after deployment. The bioshield was pressurized to launch pressure and checked for leakage with pressure decay or mass spectrometric techniques. The bioshield was pressurized to the upper extreme of the specification with the vent valve constrained to prohibit venting. The valve was then released and the pressure venting rate verified. The bioshield was vented to the lower extreme of the specification with the gas makeup system disconnected. The makeup system was then connected and verification that the pressure was raised to specification requirements was made.

The vehicle weight and center of gravity were determined mechanically for each lander configuration. The data obtained were used to verify mass properties predictions. All components were either installed or simulated by using mass models. No gases or propellants were loaded to simulate the effects of sloshing and improper commodity orientation. Propellant effects were added by using modeling techniques.

Subsystem verification: Following completion of the assembly tests, subsystem verification tests consisting of the functional tests were performed on each subsystem. Objectives of these tests were to integrate components and subsystems with the lander. The subsystem verification test consisted of a series of electrical and electromechanical functional tests. Prior to the physical assembly of the capsule, a complete subsystem verification test was performed using the CACES. This preassembly test provided optimum component access and troubleshooting capability during subsystem verification as well as ascertaining subsystem functional integrity before physically mating the lander, aeroshell, base cover, and bioshield. In general, operating modes, power consumption, output data, and alignments were verified for each subsystem where applicable.

Prelaunch and preseparation checkout: This test was accomplished between environmental tests primarily to determine if the flight components were operating prior to subjecting them to the next environment and check survivability of previous environment. It was intended to be a rapid, gross health check and as similar as possible to the "on-pad" and flight prelaunch and preseparation checkout. This test consisted of the following six parts:

GCSC A/DAPU A battery charge test

GCSC A/DAPU A prelaunch/preseparation checkout

GCSC B/DAPU B prelaunch/preseparation checkout

RAE verification

TDLR verification

LPCA verification

Capsule subsystem verification: A total integrated test was performed on the encapsulated VL configuration. This capsule subsystem verification test was similar to the subsystem verification test previously discussed. The test was modified as necessary to be compatible with vehicle configuration and available test data interfaces. Some functions could not be verified in the capsule configuration because of loss of some data interfaces.

Environmental tests: The purpose of environmental tests performed on the VLS was to verify that all systems operated properly when subjected to the stresses created by extremes of temperature, vacuum, pressure, and the mechanical stimulation caused by acoustic, vibration, and shock energy. Prior to, during, and following each exposure to environmental stresses, the applicable subsystem verification and/or prelaunch/preseparation checkout were performed to verify that all systems were operating properly, to establish a data baseline, and to determine if system degradation had occurred as a result of the environmental exposure. Functional tests performed during the period of exposure consisted of those operations which were part of that portion of the planned mission sequence. However, in addition, other components were powered and monitored to detect defective hardware or incipient failures. A final series of mechanical verifications including leak checks, alignments, and weight and c.g. determination were performed following completion of all phases of the environmental program. These tests on the PTC qualify the system to required environments. The environmental tests conducted during PTC qualification are as follows:

The lander capsule was subjected to heat compatibility tests to evaluate, demonstrate, and verify that the system was capable of withstanding terminal sterilization. During these tests, the PTC was exposed to a dry nitrogen atmosphere such that the cycle followed a thermal profile measured at a specified location within the VLC. The cycle continued for a time prescribed by planetary quarantine and based on assayed bioburden. Chamber atmosphere and the VLC effluent gas contained no more than 2.5 percent oxygen and 0.5 percent other gases by volume, and 0.097 percent water. The dry nitrogen atmosphere was circulated about the VLC to provide uniform heating or cool down at a maximum rate of 50° C/hr (90° F/hr). The VLS was nonoperating (with the exception of the UAMS ion pump) during these tests but was functionally tested at the conclusion. The UAMS ion pump was continuously operated during the heat compatibility testing and terminal sterilization.

Prior to and after exposure to the sterilization environment, a capsule subsystem verification or prelaunch/preseparation test was performed. These tests took place after installation in the oven and all special instrumentation was installed. Postheat compatibility tests verified mechanical alignments and subsystem calibrations. The IRU four-position and single-position assembly drift tests were conducted after the first heat cycle. These tests were discussed previously under "Assembly tests." The PTC was exposed to three heat compatibility cycles. The first of these was a flight-level exposure with the chamber atmosphere at a temperature of $110 \pm 1.6^{\circ}$ C ($230 \pm 3^{\circ}$ F) for 50 hr. The remaining two cycles were qualification-level exposures with the chamber atmosphere at a temperature of $121 \pm 1.6^{\circ}$ C ($250 \pm 3^{\circ}$ F) for 50 hr. Bioshield internal pressure was maintained between 1.2 and 1.7 kPa (0.18 and 0.25 psig) during exposure to the sterilization environment. A bioshield leak check and stiction test were conducted after the last cycle to insure the integrity of the bioshield and verify that the base and cap would not stick together.

Acoustic tests were conducted to simulate the effects of the acoustic environment for determining the broadband response of the lander system during launch and entry. For these tests the lander assembly was installed in the reverberant acoustic facility at MMC and interconnected with the STE necessary to accomplish appropriate functional tests. Adequate instrumentation including microphones and triaxial accelerometers were installed on or within the lander assembly to permit verification of required acoustic levels and measurement of dynamic response of the lander assembly to the acoustic excitation.

The launch acoustic test for the complete lander capsule, including the bioshield, mounted on a fixture which simulated the VL-VO adapter truss was exposed to the acoustic field derived from the launch environment. The components including ETG's and batteries were installed with pyro simulators used in place of live devices. The VLC with the bioshield cap pointing upward was mounted on a test adapter truss which was attached to the floor of the acoustic facility. The IRU was powered and the telemetry system was operating. All pyrotechnic events which would occur during ascent were activated during the acoustic stimulation period. All power was applied from external sources through the VO interface.

The entry acoustic tests were conducted with the separated lander system (lander within the aeroshell) suspended in a lateral free mode with the aeroshell down. The system was operated through a compressed time entry sequence during this test. All lander components except the bioshell base and cap were installed with pyrotechnic simulators used. All G&C subsystem components were powered and the telemetry system was operating. All pyrotechnic and solenoid valve functions which would occur during entry were exercised during exposure to the acoustic field. All power was supplied from the internal bus using either ETG's or ground sources.

Launch sine-wave vibration tests were conducted to produce mechanical excitation in the lander capsule for verification of functional compatibility during and after launch transients. The test environment was based on an enveloped response to launch transient excitation in the frequency range at 5 to 200 Hz. Longitudinal and one axis of lateral excitation was applied at the base of the VL-VO adapter truss. The vibration inputs applied at the adapter truss were controlled such that the specified levels were not exceeded at specific points on and within the VLC. Inputs were limited so that no point exceeded the booster powered flight design structural load and no component was subjected to sinusoidal levels beyond which it had been qualified. All lander components were installed with ETG's, batteries, and pyrotechnic simulators used. Initialization and warm-up of components which needed to be operating were performed by using external checkout power. Internal battery power was used during vibration tests. A prelaunch/preseparation checkout was performed before and after completion of the launch vibration test.

Pyrotechnic shock tests were conducted and consisted of a series of pyrotechnically actuated operations, separations, and deployments which paralleled the mission sequence. These tests evaluated the effect of the various pyrotechnic devices on the system operation as well as demonstrating the actual function of the pyrotechnically actuated mechanisms. The VLC was instrumented with strain gages and high response accelerometers to monitor the induced shock environment. The test series involved six test stages with flight-type pyrotechnic devices being installed prior to each stage as necessary to accomplish the requisite pyrotechnic events. The six stages were poststerilization through injection, bioshield cap separation, VLC separation, deorbit/entry, aeroshell separation, and landed. Separation hardware was suspended as necessary to offset gravitational effects or caught in nets to preclude drop damage. A prelaunch/preseparation test was conducted before and after the pyrotechnic shock tests and between the deorbit entry and aeroshell separation states. During the tests, the pyrotechnic devices were fired using selected portions of the subsystem verification test. Power and commands were supplied from sources which simulated mission operating modes; for example, prelaunch devices were fired by STE power and command, and landed devices were fired from the internal bus and by GCSC command. The lander capsule was attached to a VL-VO adapter truss which was connected to the facility floor. The bioshield cap was counterbalanced by using bungee cord to effect separation from the bioshield base. The separated lander was supported in a lateral free mode with the base cover and parachute support module counterbalanced. The landed configuration was placed on the facility floor with the legs extended and locked. All stowable, pyrotechnic-released mechanisms were allowed to deploy normally.

Landing shock (drop) tests were conducted with the lander oriented and dropped from a height necessary to achieve a velocity of 3.4 m/sec (11 ft/sec) at the center of gravity at impact. The lander was dropped three times on a rigid surface with the orientation angle selected to produce worst-case dynamic loads. The lander was oriented such that initial impact occurred on a different landing leg each time. A prelaunch/preseparation test was performed to verify system integrity prior to and following the landing shock test. During the drop and at impact, the lander system was functioning in its landing mode. Prior to the first drop and subsequent to the last drop, mechanical alignments were verified. Deployable booms were released from the stowed position manually. Power was supplied from the internal bus by using ETG's or external sources. The lander had the legs extended and locked and all deployable mechanisms stowed during the drop.

A postseparation solar vacuum test was conducted with the VLC without bioshield mounted in the solar-vacuum chamber. The VLC was suspended with support adapters and bridle attachments. The postseparation environmental conditions of 1×10^{-5} torr or less, -160° C (-320° F) wall temperature and 549 W/m^2 (174 BTU/hr-ft^2) solar intensity were simulated. A prelaunch preseparation checkout was performed after stabilization at the worst-case thermal conditions under the required vacuum. Proper performance was demonstrated during thermal transients. Power was supplied from the internal bus by using ETG's during all mission simulation portions of the prelaunch/preseparation checkout.

A Mars surface simulation/science end-to-end test was conducted to qualify the VL thermal control subsystem under extreme hot and cold Mars model diurnal cycling consuming maximum and minimum lander power, respectively. In addition, the SEET objectives were

To verify the operation of the complete lander subsystems and ground support system from acquisition of a surface sample to analysis, interpretation, and reporting of the data by the Science Teams; all elements of VL flight operations were included

To identify any existing problems for prelaunch resolution

To familiarize the Viking Scientists and Flight Operations Personnel with total operation of the lander system

The MSS/SEET program was performed in four phases: 5 nominal days (diurnal cycles); 3 hot days, 3 nominal days, and 4 cold days of MSS. This program was derived based upon the availability of the GCMS. Due to a GCMS failure, five diurnal cycles of testing, designated as SEET A, were conducted without the GCMS. A mass simulator was used in place of the GCMS and GCMS/PDA. The remainder of the science instruments were operated during these 5 days. A biology thermal simulator was used throughout the test in lieu of a biology instrument, and the seismometer was retained in the caged mode. The GCMS and

GCMS/PDA were not available at the completion of the SEET A testing; therefore, the MSS/SEET hot diurnal cycles were conducted. After three hot cycles, the thermal objectives were satisfied; thus, the test could be terminated without the scheduled fourth cycle. At this time, the GCMS and the integrated GCMS/PDA were installed with the "blind" (unknown) sample preloaded in oven 2. After installation into the PTC and the PTC reinstallation into the SSL vacuum chamber and nominal environmental conditions reached, SEET B was started. This phase of the test was designed to produce organic analyses of the blind sample at both 200° C and 500° C, to deliver an end-to-end sample to the GCMS by the surface sampler and analyze at both temperatures, to deliver and analyze a sample in the XRFS, and to complete physical and magnetic properties investigations. Samples were delivered to the GCMS and XRFS and an organic analysis of the blind sample was performed at 200° C during the first day of the test. Due to a system-induced GCMS failure, this test phase was completed without conducting the 500° C GCMS organic analysis. The four MSS test cold days were then performed.

A biology performance verification test was conducted to demonstrate the operation of the biology instrument (except soil analysis) and its compatibility with the VLS; to demonstrate the compability of the biology instrument system test results with the results of component tests previously conducted; to demonstrate the operation of the Viking lander in the RTG radiation environment; to demonstrate that the RTG radiation environment in the lander configuration was within predicted levels, and to demonstrate the end-to-end operation of the GCMS and the XRFS instruments with soil delivery and analysis. The biology performance verification test was the first major integrated VL activity including operation of the biology instrument. This test was performed under a pressure of 4 torr to permit operation of the biology instrument without limitations imposed by Mars environment design operation incompatibilities with Earth ambient environment. The Viking lander equipment compartment was thermally stabilized and controlled through RTG coolant loop flow. Although the chamber shroud was temperature controlled to stabilize the atmospheric temperatures, the ground plane simulator was not used and the solar simulator provided only an illumination function during imaging and surface sampler activity.

The Viking RTG's were installed and, together with the PTC batteries, provided VL power throughout this test duration. The Viking lander (Mars surface configuration) hardware was operated through various landed mission modes to demonstrate functionality of that hardware in the RTG radiation environment and to provide a quantitative measure of the influence of the RTG field on known susceptible hardware. Lander hardware activity was scheduled around the preplanned biology instrument sequence of events such that lander component influence on the biology performance could be detected.

The SEET (previously discussed), while basically satisfying all major test objectives, did impose a retest requirement for end-to-end soil delivery by the surface sampler to the GCMS and the XRFS. This soil was delivered to and analyzed by those science instruments during the first 4-torr test day prior to initiation of the biology instrument activity.

The biology PV/SEET retest program was performed at three test chamber conditions: Earth ambient, vacuum (<50 μm of mercury pressure), and Mars surface pressure of 4 torr. At Earth ambient condition, the lander was installed in the test chamber and a blind (unknown) soil sample was handloaded in the GCMS PDA and sealed in oven 1 of the GCMS. After the test chamber had been evacuated to a pressure of 50 μm, the GCMS and the biology instruments were vented and evacuated of air. A GCMS bakeout of 3 hr was then performed, followed by a GCMS column conditioning. The test chamber was, at this time, backfilled with carbon dioxide gas to a Mars surface pressure of 4 torr. GCMS organic analyses of the blind soil sample at 200° C and 500° C and XRFS calibrations were performed successfully. Surface sampler operations were then started by ejecting the collector head shroud and delivering soil from the soil box to the biology and GCMS PDA's and the XRFS instrument. At the end of the biology analysis tests, two additional GCMS organic analyses at 500° C were conducted successfully in ovens 2 and 3.

FLIGHT ACCEPTANCE TESTS

The flight acceptance test program consisted of component flight acceptance tests and flight capsule tests. These tests were performed to verify that flight components and flight lander systems complied with design specifications and to establish confidence that all hardware was free from defects.

Component Flight Acceptance Tests

Three units of each component were constructed for flight. Two units were launched and one unit was maintained as a spare. Prior to assembly into a lander for flight systems tests, each component was subjected to functional and environmental tests at flight acceptance levels. These component flight acceptance tests were defined in the section "Component Development Tests." In general, of those tests defined, heat compatibility and vibration tests were performed on all components. Cruise thermal vacuum, entry thermal, or surface thermal tests were performed depending on which of these included environments in which the particular component was to operate.

Subsystem Flight Acceptance Tests

Acceptance testing of the guidance and control subsystem was performed via the G&C simulation VCMU. The VCMU is described in the section "Subsystem Development Testing." The VCMU used flight-type hardware as was the case during subsystem qualification testing. Subsystem performance testing and flight software testing were combined at the acceptance test level. The combined tests validated the flight software.

After final assembly of the flight landers, a test was performed on each lander to verify proper G&C and propulsion subsystem phasing (verify correct polarities). The lander structure was physically isolated from the facility electrical ground and all ground support equipment was disconnected. A single hardwire interface was retained for emergency shutdown. Test control and data return were provided via the flight rf link to the communications and telemetry and data handling subsystems. These isolation precautions were taken to insure that the lander operation was in no way dependent on the ground support equipment. The performance of the phasing test required data base changes to the flight computer load; no flight software code changes were required. Data base changes were simply changes in numerical values stored in the computer, for example, time between events, and initial state variables. The following phasing test description is based on an understanding of the mission sequence of events and of the G&C design. (See page 64 of ref. 1.)

The objective of the test was to exercise all control loops, all control modes, and the critical information flow paths through the flight software. Earth rotation rate and gravity served as the test stimuli. The lander sat on the floor of a test facility; it was turned in azimuth and tilted slightly relative to the local vertical. The desired lander orientation was analytically determined, physically controlled to a small tolerance, and precisely measured to provide accurate knowledge of orientation. Accurate knowledge of the lander orientation, the test site latitude, and the value of gravity at the test site, allowed pretest computation and prediction of control system response to the stimuli as a function of test time and the mission sequence of events.

The test began at separation of the lander from the orbiter and continued until lander touchdown. Time between mission events was compressed so that the lander "flew" the 3 1/2 hr descent trajectory in about 8 min of test time. The first desired attitude matrix $[A_d]$ (which normally defines an attitude maneuver in preparation for the deorbit burn) was set to identity, and the attitude control deadband was reduced in size. The lander orientation was defined so that accumulation of Earth rotation rate caused deadband crossings in two axes, and the appropriate RCS engine commands were transmitted from the computer to the VDA. This was state 1: positive pitch and positive yaw engines "on." The RCS feedlines were pressurized downstream of the tank pyrovalves, and flow indicators were installed at each engine nozzle. Motion of the flow indicators was recorded by cameras; a clock was placed in the field of view of each camera. Verification of proper computer command phasing was obtained from rf telemetry. Verification of proper solenoid valve phasing was obtained from the motion-picture film. Both telemetry data and motion-picture data were time tagged so that the predicted time of deadband crossing could be accurately confirmed. Telemetry and flow indicator monitoring was continued throughout the test.

Since the lander position was fixed, attitude error continued to grow with Earth rotation rate and time, and the positive pitch/positive yaw engines remained on as the deorbit sequence was entered. The acceleration of gravity was negative (opposite in polarity) relative to the acceleration normally produced by the deorbit engine thrust. Thus, the accumulated velocity change ΔV was negative and was always less than the required ΔV-time contour. Engine commands were accordingly generated to turn on all pitch and yaw deorbit engines in an attempt to acquire positive ΔV and to satisfy contour requirements. However, attitude control had priority over velocity control and the positive pitch and positive yaw engines were now commanded off in response to the inverted attitude control logic. This was state 2: negative pitch and negative yaw on and positive pitch and positive yaw off.

The lander tilt was carefully defined relative to the composite of gravitational acceleration and the centripetal acceleration due to Earth rotation. Therefore, the velocity steering computations were driven by sensed lateral accelerations which gradually accumulated and approximately halfway through the deorbit sequence reached sufficient magnitude to override the integrated Earth rotational rate attitude error and reversed the attitude control command polarities. This was state 3: positive pitch and yaw on and negative pitch and yaw off. The termination of the deorbit ΔV accumulation was triggered by a squared function in the flight software so that the negative gravitational acceleration still initialized the deorbit burn cutoff function. At deorbit termination the second desired attitude matrix $[A_d]$ was called. This $[A_d]$ was defined to command an attitude maneuver that exercised a different combination of attitude commands; this was state 4.

At this point, the following end-to-end phasing has been verified:

(1) Normal deadband crossing - from gyroscope-sensed Earth rotation rate through attitude control logic to engine actuation

(2) Velocity-change control with inverted attitude conntrol logic - from accelerometer sensed axial gravity component through thrust modulation logic to engine actuation

(3) Velocity steering attitude control augmentation with inverted attitude control logic - from accelerometer-sensed lateral acceleration through velocity steering logic to engine actuation

(4) Attitude maneuver - from a desired matrix through attitude error computations to engine actuation

Item (4) was repeated using the remaining available desired attitude matrices, the entry pitch-over maneuver, and the parachute roll maneuver to verify all possible combinations of attitude control phasing for both the RCS and TDS.

From entry to touchdown, the acceleration of gravity was of the proper polarity to simulate the aeroshell deceleration, parachute deceleration, and the terminal propulsive deceleration. The initial velocity at entry was redefined to provide a reasonable descent time. The flight program proceeded through the normal navigational and guidance computations; events such as parachute deployment occurred at the scheduled altitude and the associated pyrotechnic commands were routed to a panel of visual indicators mounted on the lander. The panel of event indicators was recorded on film as a function of time for later correlation with pretest predictions. At terminal engine ignition, the warm-up positions of the three throttle valves were defined to verify proper signal transmission from the computer through the VDA to the throttle valves. At terminal descent contour intercept, the terminal engines throttled-up to the axial limits. Throttle limiting occurred because the test was set up to provide velocities greater than those defined by the descent contour for the computer altitude. The test was terminated by mechanical engagement of the touchdown switches in one landing leg.

During the phasing test, the radars (RA and TDLR) were powered on at appropriate mission times. The radars did not lock and their data were disregarded by the navigator routines. The radars did not have dual polarities; therefore, there was no compromise to the phasing test.

The lander gyroscopes were calibrated in-flight relative to the orbiter celestial sensors. Verification of proper phasing between the respective sensors was accomplished in two steps. First, the proper phasing between the orbiter celestial sensors and orbiter gyroscopes was verified during orbiter assembly by comparing the respective sensor outputs during physical motion of the orbiter relative to simulated solar and stellar targets. Second, during final checkout of the mated orbiter and lander on the launch pad, the proper phasing of orbiter gyroscopes and lander gyroscopes was verified by using Earth rotation rate inputs.

Flight Capsule Tests

Two flight Viking lander capsules were assembled and tested prior to being shipped to KSC for final testing, spacecraft assembly and tests, and launch. After completion of all component development and qualification tests, component level flight acceptance tests, AGE functional tests, and the corresponding PTC tests, each lander was tested as shown in the flow of figure 8. In addition,

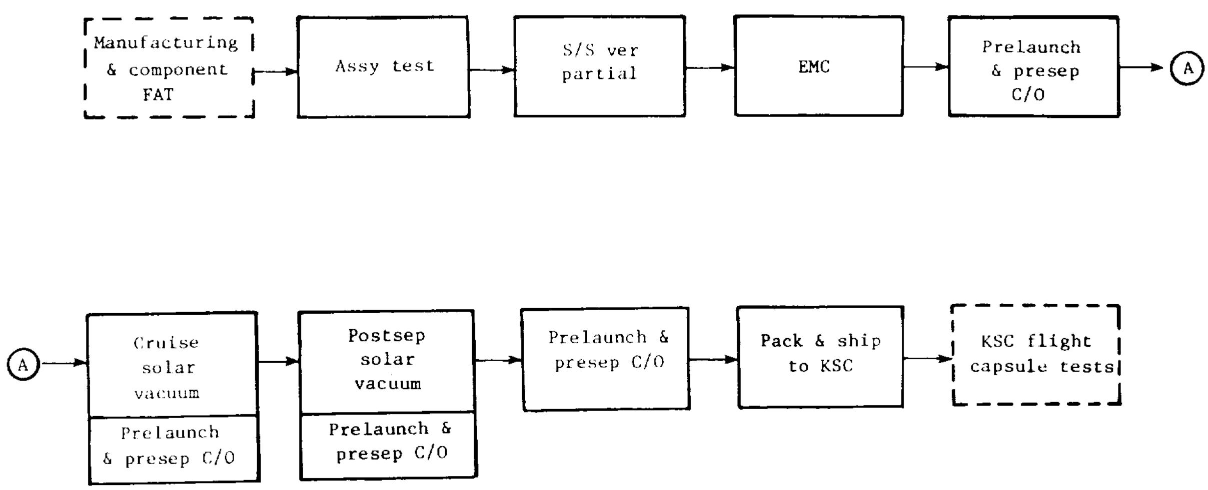

Figure 8.- Test flow of flight capsule.

lander 1 received system level EMC tests. The majority of the tests included in this flow were defined in the section "System Qualification Tests." Those tests not defined there are described as follows:

<u>Electromagnetic capability test</u>: EMC tests were conducted only on lander 1 to demonstrate compliance of critical VLS circuits with established EMC acceptance criteria. Electromagnetic compatibility of the Viking lander system was demonstrated for each lander configuration in accordance with the EMC control

plan. Special interrogating and monitoring equipments were installed to obtain data to ascertain compliance with acceptable performance margins. Functional testing of the mission sequences was performed to exercise specific subsystems in modes which created potential worst-case conducted and radiated noise environments. No attempt was made to determine transient signatures of monitored circuits nor was noise injected from external sources. The STE simulated the orbiter interface including power system source impedances and digital and discrete interfaces for the capsule configuration test. Vehicle bus voltage noise and ripple were indicated.

Interplanetary cruise solar vacuum test: The lander flight capsules were subjected to a solar vacuum environment which permitted simulation of interplanetary cruise and separation to entry periods. These tests simulated worst-case thermal conditions, including transients which were predicted during this period. The test evaluated and verified the capability of the thermal control system of the lander capsule to maintain temperatures within design specifications, to evaluate thermal switch operation at the system level, and, in addition, to demonstrate proper operation of lander subsystems during and after exposure to the environment. Thermocouples were installed in the VLC monitor temperatures. The lander capsule as mated to the OTES was installed in the solar vacuum chamber in an inverted position with the lander below the OTES. The spacecraft -Z-axis was illuminated such that the OTES was between the solar source and the VLC. The mounting mechanism to which the vehicle was attached was designed to minimize heat transfer between the vehicle and the fixture and to minimize shadowing of the lander by the fixture. The environmental conditions for near-Earth (max thermal conditions) and near-Mars (min thermal conditions) were simulated during this test. Electrical power input to the VLC was monitored and controlled during these tests to assure generation of the requisite self-generated heat. All functional duty cycles were biased to provide worst-case conditions.

TEST RESULTS

The successful completion of the test program outlined in this report contributed significantly to the success of the two Viking landers. The test program showed that the lander design could operate under the conditions in excess of those expected for the Viking Mission. As the test program proceeded, some design defects and inadequate manufacturing processes were discovered and corrected. This report of the test results does not address all problems encountered but concentrates on typical ones that were considered especially significant because of difficulty in finding solutions. The problems are discussed within the section for the phase of testing in which the problem first occurred.

Component Tests

During component development, significant problems occurred in several components either during breadboard, development, or qualification tests. Some of these problems continued to plague the test program through all phases. The problems discussed in this section were deemed significant enough to be included on the Viking Project top ten problem list for several months. Since most of these problems occurred early in the development cycle, most were corrected before subsystem or system testing commenced. Therefore, the component test program was very successful in screening most component design problems early in the development process. This procedure allowed some flexibility in modifying downstream testing when schedules became very critical.

Radar altimeter: During breadboard testing several design problems were discovered with the main problem being the inability to meet the low altitude specification. Although the problem appeared to be explainable at the breadboard level, component development and G&C subsystem testing were started with a relaxation of the lower altitude limit capability of 38 m (125 ft) versus the 30.5 m (100 ft) expected. The STB system level testing was also started with a breadboard rather than a development unit. Even though the development testing schedule was impacted, all problems were solved through design modifications and the RA successfully completed component development tests.

Lander camera system: Although the cameras had several minor problems during development testing the major problem discovered early in the development process was the inability to obtain photo sensor arrays (PSA) that would meet specifications. After attempting many modifications to the PSA subcontractor's design and processes, it was determined that another supplier was required if the development schedules were to be met. Although delivery of the PSA's continued to be a concern, the cameras completed development testing with only minor design changes required.

Upper atmosphere mass spectrometer: The UAMS had problems during development testing because of inadequate electronic design and mechanical designs that failed to meet vibration and leak test specifications. These problems caused delays in development and qualification testing. Although the major electronic design problems were solved during development, leaks due to bad feed-throughs and welds continued to be a problem through flight article acceptance tests. Another significant problem in the design of the UAMS was the development of the cutter assembly used to expose the UAMS inlet through the lander aeroshell. Several design approaches were tested before one was obtained that would pass component qualification tests.

Surface sampler boom assembly: During the first year of the surface sampler development, the boom assembly development schedule was significantly impacted by MDA failures. Failures occurred in nearly every part of the MDA including brushes, bearings, gears, and the torque limiter. After extensive study of the failures and causes, Viking Project management decided to procure MDA's from a different supplier. Selection of a new supplier resulted in a problem with delivering surface sampler components for lander system tests. The prototype boom had to be used for early phases of the STB tests and the qualification unit was delivered after the start of PTC testing. Several other minor problems surfaced during component flight acceptance and qualification testing of the surface sampler but these problems were corrected with eventual successful completion of all tests.

Inertial reference unit: Development of the IRU encountered problems with failures during functional testing of the early development units. These problems included hybrid electronic circuit failures, improper assembly of flat cables and machining errors. These problems were overcome with changes in designs and procedures but with delays in deliveries for system level test. In addition, during component qualification tests, endstones that supported the thrust loads of the rotating parts of the gyro were cracking during warm-up from cold temperatures. Also, at these cold temperatures, the formation of bubbles in the flotation fluid caused flex lead failure and/or damage. Both problems were solved by the addition of auxiliary IRU case-mounted heaters to be sure temperatures were above minimum during flight.

Seismometer: The seismometer development unit failed functional tests following pyrotechnic shock exposure. Although the electronics operated correctly, all six sensor flexure pivots were broken and all three sensor coils were open. The failures were determined to result from inadequate caging of the sensor. After a redesign of the caging mechanism and after instituting new alignment processes for the sensor units, the seismometer successfully passed the shock and vibration environments.

Flight software: As a result of simultaneous development of the VCMU laboratory, flight software verification, and support software development for system tests, problems arose in the development and verification testing of the flight software. In addition, a large number of late changes were made to work around hardware problems that were discovered during lander system testing. In order to have a tested version of the flight software ready for system tests at KSC and for launch, the verification test program was shortened and some planned interim versions of the software were not tested or released.

Biology: Early in its development, the key science instrument on the lander for life detection had over 30 major problems open against it. The more severe problems included hybrid circuit designs, valve designs, overall instrument thermal design, and excessive sensor noise. Even though considerable progress was made in solving these problems during development, several instrument builds had to be eliminated to maintain schedules. Because of this and the problems that continued to appear during component testing, an instrument was not available for system level testing until the end of the qualification level PTC testing. Problems continued to occur during these tests. Design changes were incorporated into the flight units, but these units were not available for system level tests on the flight landers at MMC and had to be integrated with the flight landers at KSC.

Guidance, control, and sequencing computer: The GCSC which was the brain that directed the activity on board the lander encountered problems very early in its development. Although most of these early problems did not occur during testing, they did eventually impact the test program. The early problems were delays due to design changes resulting from the numerous interfaces with other lander components, difficulty in laying out very high density printed-circuit boards required to meet size constraints, and problems encountered with developing 0.5-μm (2 mil) plated-wire memory. Of these problems, the last one proved to be the most difficult to solve. These problems also impacted the DSM since the same memory elements were used in its components. To solve the problems encountered, changes in manufacturing techniques and testing procedures were developed with the aid of NASA and MMC. The size of the GCSC memory was reduced from 20 000 to 18 000 words early in the development. In order to meet the schedule for a development unit for system testing in the STB, component flight acceptance testing was waived and a unit with external core memories was delivered. A second development unit with known bad memory locations was delivered for subsystem testing in the VCMU. A unit with one-half core and one-half plated-wire memory had to be used for PTC testing. Because of the severe memory development problems, a parallel development of core memories that could be integrated with the GCSC processor was instituted. However, with new processes for plating wires and keepers for the memories, the 0.5 μm (2 mil) wire memory development was on the way to being solved. The parallel development of a core memory was carried to the point of having a unit sterilized and qualification tested. Additional problems were encountered during flight acceptance testing of flight units. However, these problems were not serious, and all GCSC and DSM flight units were eventually delivered in time for flight lander builds.

Gas chromatograph mass spectrometer: The GCMS experienced a significant number of problems during its design, development, and testing phases. The more significant of these problems involved the following areas:

The MS analyzer and the GCMS plumbing experienced numerous leaks. The instrument operation relied on a low internal pressure ($<10^{-6}$ torr in the MS analyzer). Therefore, in-leakage of atmosphere, Earth or Martian, could not be tolerated. Numerous different sources of leakage were uncovered and design or manufacturing procedure changes incorporated during the development and testing phases.

Chemical conversion of volatile organic material was detected within the instrument. The instrument was intended to identify organic compounds which could be volatilized from a surface sample. The requirement was to transport volatilized hot organic compounds to the MS analyzer for detection. All chemically active surfaces, including metals, had to be removed from the sample path or deactivated. Also, late in the testing phase, it was determined that metal in the sample in combination with the primary carrier gas (hydrogen) would result in catalytic conversion of the organic compounds. In the spring of 1975, a separate CO_2 gas supply was added to replace hydrogen as the gas present in the oven during sample heating.

Corona and arcing were detected in the MS high-voltage electronics and analyzer feed through connectors during special corona testing of the MS assembly. The high-voltage potting material, application procedures, and design geometry were changed to alleviate the problems.

After corrective action as described for these problems, the GCMS successfully passed its required testing.

GCMS processor distribution assembly: During qualification testing of the GCMS PDA, a problem with the shuttle block mechanism occurred. The shuttle block which moved surface material from a load position to either an instrument delivery position or a dump position was jamming. After unsuccessfully trying several modifications to the existing design, a new design was incorporated.

Because of this problem and schedule problems with the PDA cleaning process, flight PDA's were delivered to KSC for integration with the flight landers.

System Test Bed

The STB test program was very valuable in discovering system interface problems early enough for incorporation of changes into flight hardware. It also was very useful in discovering flaws in test procedures and test software that were used during PTC and flight article system tests. In addition, many problems with the STE were discovered and fixed prior to use of the STE with the PTC and the flight articles. One of the biggest problems encountered during STB testing was inadequate test sequences. Test sequence development was designated as a top ten problem until a procedure for proper design review and testing of sequences was incorporated in the development process. The initial phases of integration of hardware in the STB disclosed typical interface problems between components and between the lander and STE. An example of this was a problem with decommutation of telemetry data by the STE. Because of the numerous telemetry formats employed by the lander and the variability of data words within a format, the STE software for decommutating these data was usually not working properly. This made it difficult to review test data. It was very late in the STB test program before the software could successfully decommutate all lander telemetry formats. Some typical problems encountered during STB testing are listed in table 2. Many of the hardware problems encountered during STB testing resulted from having to use development hardware with known deficiencies or simulators because components originally designated for use on STB were not available.

TABLE 2.- TYPICAL STB PROBLEMS

Component	Failure	Cause
DAPU	Incorrect format 2 data	Incorrect clock phase to DSM
GCSC	Memory failed to load	Overheating and wiring errors
STE	Cannot decommutate TM data	STE software errors
RPA	Incorrect data	Output wires reversed
DCS CCU	DCS command detector in-lock discretes reversed	Wiring error
SSCA	Collector head rotation reversed	Drive motor wired backwards
BPA	Float charge circuit inoperable	Float charge resistors failed due to incorrect sequence
LPCA	Pyro no fires	Missing or incorrect pyro simulators

Proof Test Capsule

The PTC test program provided valuable operational experience as well as demonstrating that the design was qualified for its intended mission. Initial subsystem integration was delayed by the late delivery of components to the PTC. Temporary hardware substitutions were made by using development components and functional simulators. The initial component integration disclosed typical problems of wiring design errors and unexpected operational modes that were easily corrected and also incorporated in the flight hardware build. An example was a DAPU/DSM conflict where in some modes the DAPU filled the DSM starting address for the next write sequence with 1's causing the DAPU to sense a "DSM full" during the next write sequence and revert to stand-by mode. The flight DAPU design was modified. The DSM also had an unexpected mode of continuous read or write operation that could be entered if the DAPU interface drivers were off and the power bus and enable signals were on. The flight DSM design was modified. Heat compatability tests demonstrated that the lander capsule was capable of surviving the terminal sterilization temperature cycle. The tests disclosed a few problems shown in the following table that had been

Component	Failure	Cause
Camera 1	Intermittent black pictures	Worn elevation motor brushes (nonflight brush holders)
	Jerky azimuth scan motion	Azimuth tachometer was noisy (nonflight config)
IRU	X- and Y-axis accel bias shifts	Failed capacitors (nonflight part) and a shorted PC board (nonflight)
GCSC	Loss of indirect addressing due to sep of PC board from the plated through hole	Nonflight PC board
DSM	Inoperative	Open plated through hole, failed sense digit transformer and bad bond in a hybrid resistor diode (all nonflight config)
TWTA	Power supply over current trip-out caused by a gassy tube	Tube dropped during assembly
RAE 2 (DD unit)	Two transistors and resistors failed	Worst-case analysis identified marginal design, flight units corrected

discovered during failure modes and effects analysis or component qualification tests but the flight component corrective action had not been retrofitted into the PTC test hardware.

Launch acoustic tests verified the ability of lander components to operate during and after exposure to qualification level broadband excitation. Overall PTC acceleration responses were less than the LDTM by a factor of 2 to 3 which verified that most component qualification levels were conservative. One exception was the UAMS response which exceeded its qualification level. The UAMS was the only component damaged by the acoustic excitation. Appropriate changes were made to the UAMS qualification requirements.

Launch vibration qualification tests verified the lander ability to survive launch vehicle staging and separation transients. PTC damping was again higher than the LDTM with UAMS RCS/deorbit, and terminal descent engine response higher than LDTM. No failures were observed. Postseparation thermal vacuum tests simulated the mission phase from preseparation checkout through the first post-landed UHF link under hot worst-case conditions. The test demonstrated that the UHF thermal dissipation was greater than anticipated resulting in expected flight temperatures 24.4° C (42° F) higher than component FAT levels. To alleviate the concern, the thermal mass of the UHF heat sink was increased, the UHF FAT and qualification temperatures were modified, and landed relay link operating times were reduced to control the UHF maximum operating temperatures.

Pyro systems functional tests were performed to demonstrate pyro initiated separations and deployments as well as to induce flight-type shock environments. All 79 pyrotechnic events were successful. Propellant loading valve wiring errors were found and corrected; the equipment module staging connector retraction bracket was redesigned as it prevented positive separation; and the stagnation temperature transducer installation methods were modified as the initial deployment was unsuccessful.

Three landing shock drop tests were successfully performed with pitch and yaw positions selected to produce worst-case anticipated landing shock conditions.

Mars surface simulation tests demonstrated landed mission science and engineering in nominal, as well as worst-case hot and cold environments. (A simulator was used for the biology instrument.) The test sequence contained 5 days of nominal Martian conditions, 3 days of hot environments followed by 3 nominal days and 4 cold days. Flight team personnel participated in test data analysis to provide realistic training and demonstration of software and procedures. The tests demonstrated the lander was qualified to function in the Martian environment. An operational error caused failure of the GCMS during the test; therefore, it was not possible to demonstrate surface sampler digging, delivery to the GCMS, and GCMS blind sample analysis. A decision was made to accomplish the GCMS objectives in combination with the biology instrument system tests in a separate PTC Mars surface simulation tests. The subsequent biology planetary verification tests did verify that the lander, including the biology and GCMS experiments, was qualified to perform the Viking Mission.

Flight Acceptance Tests

No major problems were encountered during flight acceptance tests. Some failures occurred during component and system level tests but none were due to component design inadequacies. Because of late deliveries of some components, system level tests for the flight landers had to be started with development hardware or simulators in the place of some flight components. In addition some components had already been designated for incorporation during final buildup and test at KSC prior to sterilization. As a result of the success during PTC qualification tests, no system level vibration tests or acoustic tests were conducted on the flight landers prior to shipment to KSC. EMC tests were conducted only on flight lander 1, but both flight landers were subjected to assembly tests and thermal vacuum tests. Some component failures were encountered during these tests resulting in rework of some components. Table 3 summarizes the failures encountered during flight acceptance testing. None of these problems were deemed significant enough to delay delivery of the flight landers to KSC. Several of the damaged components were repaired and reinstalled at KSC or were replaced with other flight components.

TABLE 3.- FLIGHT ARTICLE TEST FAILURES

Component or S/S	Failure	Cause
	VL-1	
Pyrotechnic	Inadvertent firing of staging connector	Incorrect test procedures
VDA	Excessive VDA current due to valve simulator short	Valve simulators not properly designed
RAE	RAE 2 failed to switch to modes 2 and 3 (false lock)	Stray signal reflections from lander
DAPU	Cruise and presep data IF failed	Most probably caused by improper facility ground and inadequate cable-up procedure
Pressure sensors	Abnormally low reading on some measurements	Sensors failed, possible UAMS corona problem
UAMS	Corona discharge	Operated during environment it was not designed for
	VL-2	
Pyrotechnic	Short in harness	Incorrect mounting procedure
BPA	High current during communication test	Damaged cable
IRU	Cover heater miswired	Incorrect drawings
VDA	Incorrect power monitor reading	Internal part failure
	Short in TDE power monitor	No protection for disconnected cable
TDLR	Noise burst	Frequency drift in unit
Pressure instruments	Marginal data	Nonflight units
UAMS	Out-of-tolerance electron accelerating voltage measurement	

During EMC testing some interfaces appeared to be below the required margins. More extensive testing was recommended with qualification hardware, and after analysis and additional STB testing, most margins were deemed to be acceptable. However, in some cases, different cable routings and constraints on operating sequences were recommended to reduce noise problems.

During thermal vacuum testing, the results from lander 1 and lander 2 were comparable. All component temperatures were within flight acceptance levels except during the preseparation checkout through landing phases of the mission. Recommendations were made to reduce the power consumption during this period to bring the few components affected back within the flight acceptance levels. One problem that occurred during thermal vacuum testing on lander 1 affected several components. The UAMS ion pump was powered during the Mars surface environment and since it was not designed to operate in that environment, corona discharge resulted. Because of possible damage to other components, the command control unit, the RPA, the pressure instruments, and the DAPU were scheduled for removal and inspection before reinstallation at KSC. Table 4 is a list of

TABLE 4.- FLIGHT COMPONENTS INSTALLED AT KSC

Flight lander 1	Flight lander 2
Biology	Biology
*Command control unit	Biology PDA
Camera duster	GCMS
*DAPU	GCMS PDA
GCMS	GCSC
GCMS PDA	Meteorology (partial)
Meteorology (partial)	RA
*Pressure sensors	Stagnation sensor
RA	UAMS
*RPA	
UAMS	

*Items inspected as result of UAMS corona problem.

all flight components that were installed at KSC due to either failures or nonavailability. Those marked with an asterisk are the items inspected as the result of the UAMS corona problem. Once the flight landers were reassembled at KSC, similar tests to those performed at MMC were conducted to verify that new components or repaired components were acceptable for flight.

ORBITER TEST PROGRAM

The test program for the Viking roriter was designed to provide an orderly sequence of tests, consistent with the objectives of the Viking '75 Master Integrated Test Plan, which would lead to the type approval of the proof test orbiter and flight acceptance of the flight orbiter systems. Specific test program objectives were to verify design, to verify performance in mission modes, to verify compatibility with mission environments, to verify compatibility with other mission systems, to demonstrate design margin for internal and external noise and other mission environments, to provide reference baseline data for normal system operation, to provide a source of qualified spare orbiter subsystems to support the two flight systems, to provide trained personnel for system operations, and to develop a mission support data processing capability.

The system level TAT and FAT programs success was highly dependent on a comprehensive development test program. The formal TAT program established confidence that the design was qualified for its designated flight mission by the verification of design margins at both the subsystem and system levels. The verification of design margins also built confidence and provided evidence that flight equipment would not be degraded by less severe environmental exposure during FAT. The formal FAT program provided certification of flight acceptability at the subsystem and system level when exposed to operating conditions and environments in excess of expected flight requirements. Development tests were those component, breadboard, subassembly, assembly, and subsystem level tests required to support preliminary and final design decision for flight equipment, assembly and handling equipment, and test program design.

DEVELOPMENT TESTS

Several models of the orbiter subsystems and systems were fabricated for use in development and qualification tests. The electronics hardware development testing consisted of design verification testing, component and module tests, subassembly tests, and subsystem tests. The design verification tests were conducted on breadboard level hardware and consisted of subsystem level, computerized, and support equipment monitored functional tests. These tests verified the adequacy of the design to meet its functional and interface requirements. Voltage and temperature margin tests were also conducted at this level. In addition to the normal component screening and qualification tests, a number of special component and module qualification tests were conducted. These included tests such as special vibration testing of the relay units for the CCS output units to verify the adequacy of shock isolation to eliminate induced relay closures.

As early as possible in the design schedule (at least prior to the S/S critical design review), development tests were conducted to identify any design requirements not met by the S/S. Also identified were overall link prediction errors, any S/S interface and DSN incompatibilities, possible EMC/RFI, multiple rf carrier, modulation interference problems, potential parameter changes for increased optimization, and anomalies encountered. In the development test area, some of the test requirements were satisfied by data from previous missions, that is, RFS/MDS (TMU only)/LGA from Mariner Mars 1971 project and RFS/XTXS from Mariner Venus/Mercury 1973 project, and the data were not reverified. However, in other areas where major modifications were made or new S/S design was required, complete development test data were obtained, that is, MDS (CDU), PWRS, CCS, FDS, ACS, ARTCS, DSS, HGA, VIS, IRTMS, MAWDS, RRS, RTS, and RAS. In addition, all multiple unit operational characteristics for both the VO and DSN/VMCCC had to be evaluated as early as possible in order to adequately assess and insure satisfactory performance due to the fact that the multiple S/C configuration had never been previously required. For example, in the antenna S/S's area a special 1/5-scale model of the VO with the scaled VL bioshield base attached was used to develop the RAS UHF relay antenna. Initially a log conical spiral design was proposed to be used, and tests as well as patterns were made using the 1/5-scale model. (See fig. 9.) Later it was

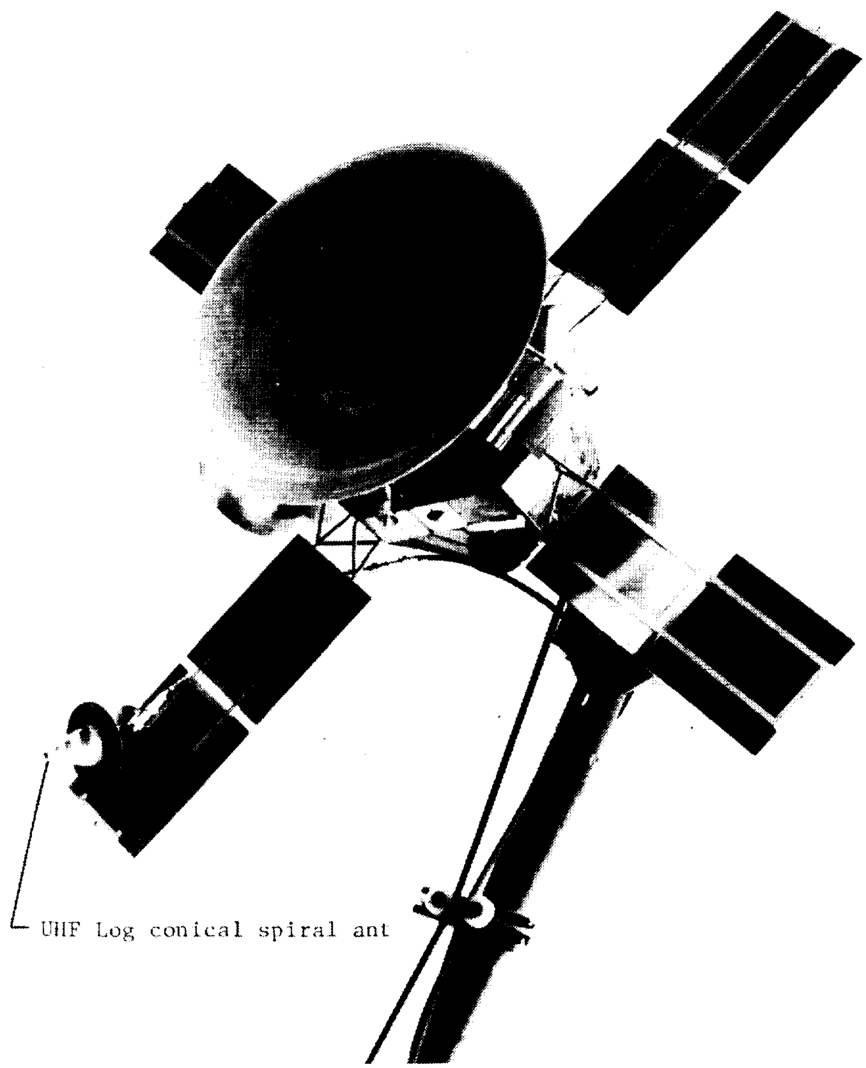

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Figure 9.- 1/5-scale antenna model.

determined that the resonant quadrifilar helix design would provide better gain and polarization characteristics over the required pattern; therefore, it was selected and completely tested with the 1/5-scale antenna test model.

The full-scale VO ATM (fig. 10) was fabricated from framing material and chicken wire to produce a lightweight assembly that could be supported on a specially built antenna test pattern rotating stinger and platform assembly to obtain S-band antenna radiation patterns for the LGA. The LGA patterns were required as soon as possible to be used by ETR in the initial acquisition and and tracking of the S/C immediately after launch and for the other ground tracking stations when the S/C was near Earth. These patterns were taken at 2115 MHz and 2295 MHz in increments of 2° clock angle and include complete 360° cuts in cone angle. The patterns were taken with the VO in the postseparation solar panels deployed configuration, and both left- and right-hand circular polarized radiation patterns were recorded. The HGA patterns were also both right- and left-hand circular polarized and were made at the design frequencies of 2115, 2295, and 8415 MHz. The patterns were of the HGA by itself and could be assumed correct or unaffected by the surrounding VO or VL components and structure as long as the pointing operational constraints provided for the HGA were observed.

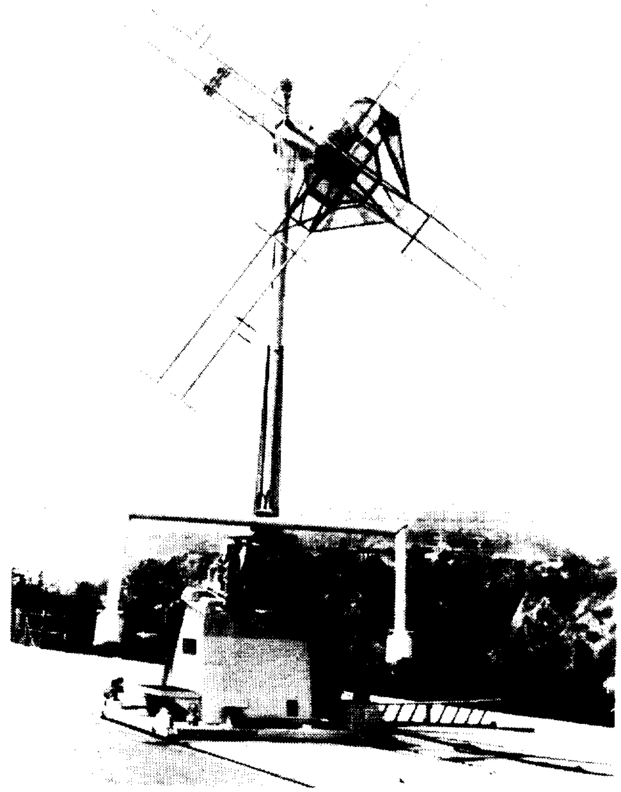

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Figure 10.- Full-scale VO ATM.

Subassembly level tests and subsystem level tests were conducted to verify that functional, voltage and temperature margin, and dynamic environmental requirements were satisfied. In addition, a number of subsystems were subjected to rf compatibility tests. A very limited life test was conducted on certain components in PWRS and DSS. The power life test consisted of an extended test of the flight equivalent batteries, simulating anticipated charge/discharge cycling of the mission. The DSS life test was a multimission simulated test of flight equivalent bearings and drive belts. The flight software development testing consisted of simulation testing of each routine, and then the flight software was used for all system level tests and a number of special purpose software tests. The initial software tests were conducted in a simulator and also in the breadboard hardware and support equipment monitored tests. The majority of flight software validation testing was conducted in the system environment by mission-like standard and nonstandard sequences which were part of the system test procedures.

The ASTM was configured from a welded steel bus that provided the correct mechanical interfaces for the propulsion support structure, the V S/C A attachment, a propulsion module structure assembly consisting of flight-like tanks, and mass simulated rocket engine and control assemblies. The ASTM was heavily instrumented with strain gages and fitted with straps to provide load attachments. As an assembly, the ASTM was used as a static modal test article (figs. 11, 12, and 13) to verify early estimated stiffness characteristics of the orbiter mathematical model. The ASTM propulsion module subassembly (fig. 14) was used in separate tests to provide modal data when mounted to a rigid fixture and exercised to design limit loads with suspended modal shakers. Spacecraft handling equipment test fixtures and test plans and procedures were validated or modified as required based on ASTM test experience.

A structurally rigid mass simulation model of the VL was fabricated and used extensively in orbiter modal and vibration tests. The loads model of the VO included the RL which MMC later replaced with their flexible lander to form the S/C model. These orbiter tests and analytical configurations were identical.

A surface tension propellant management system was chosen early in the design phase of the propulsion system. The propellant tanks were designed to

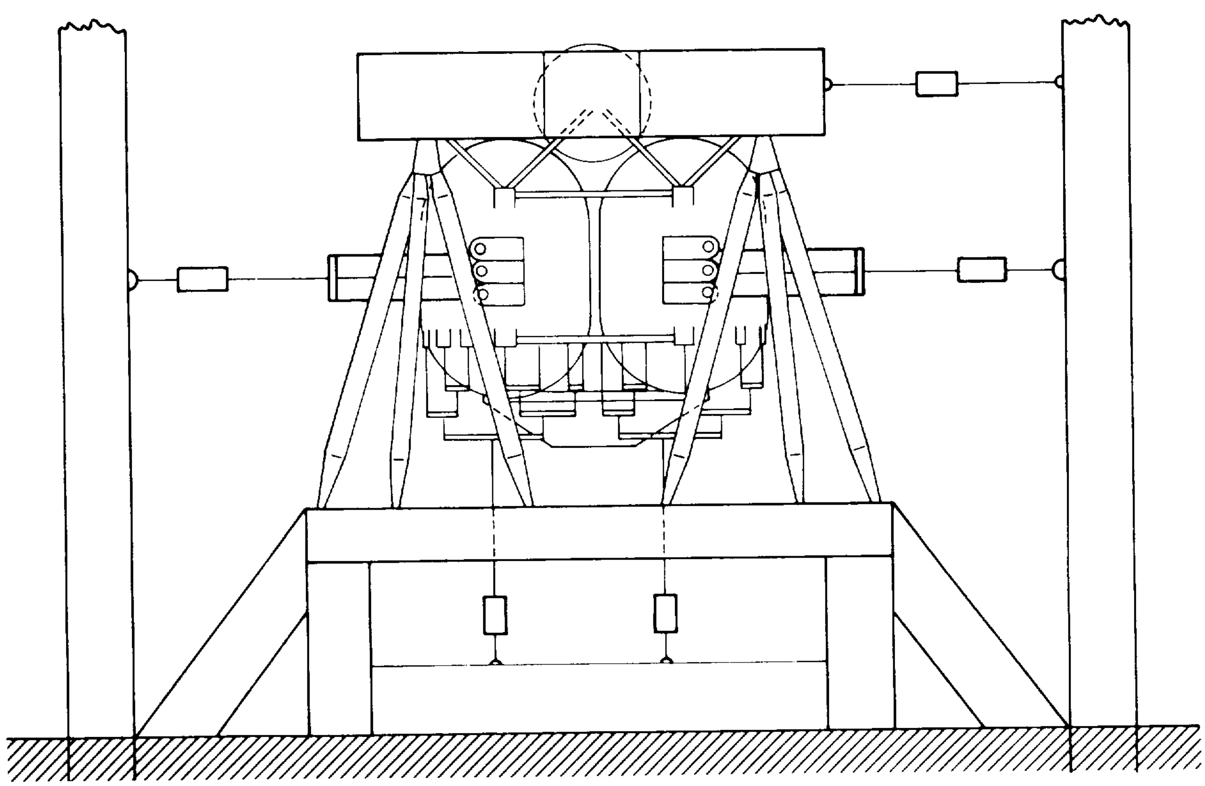

Figure 11.- ASTM VO static load test configuration.

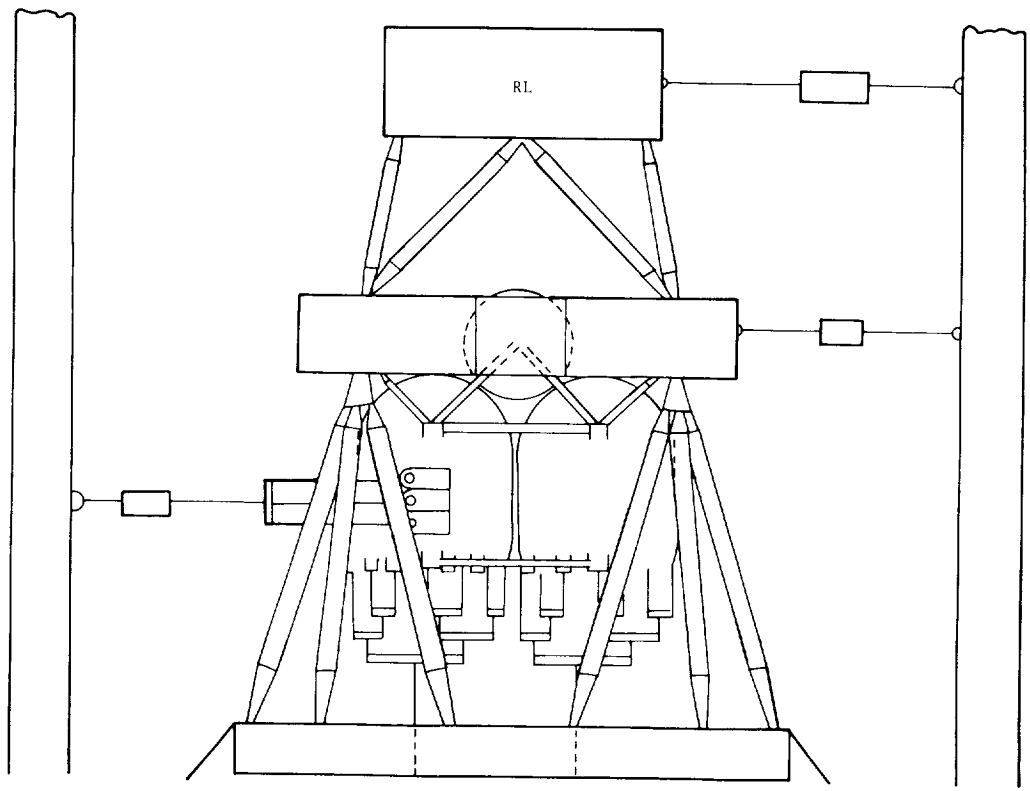

Figure 12.- ASTM VO/RL static load test configuration.

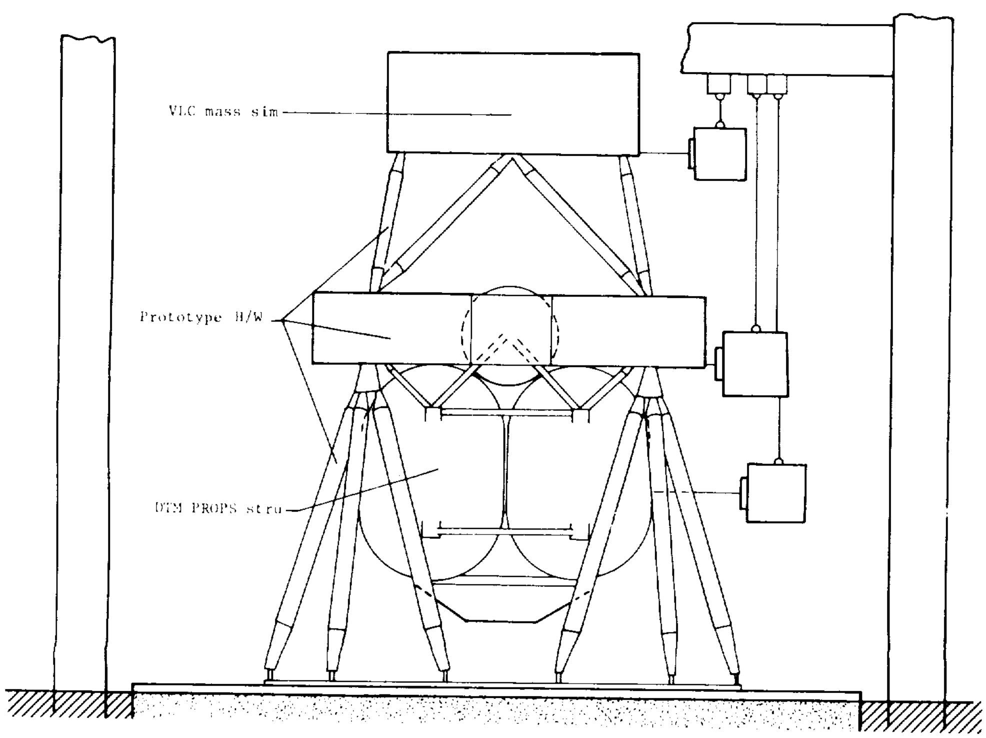

Figure 13.- ASTM VO/RL modal tests configuration.

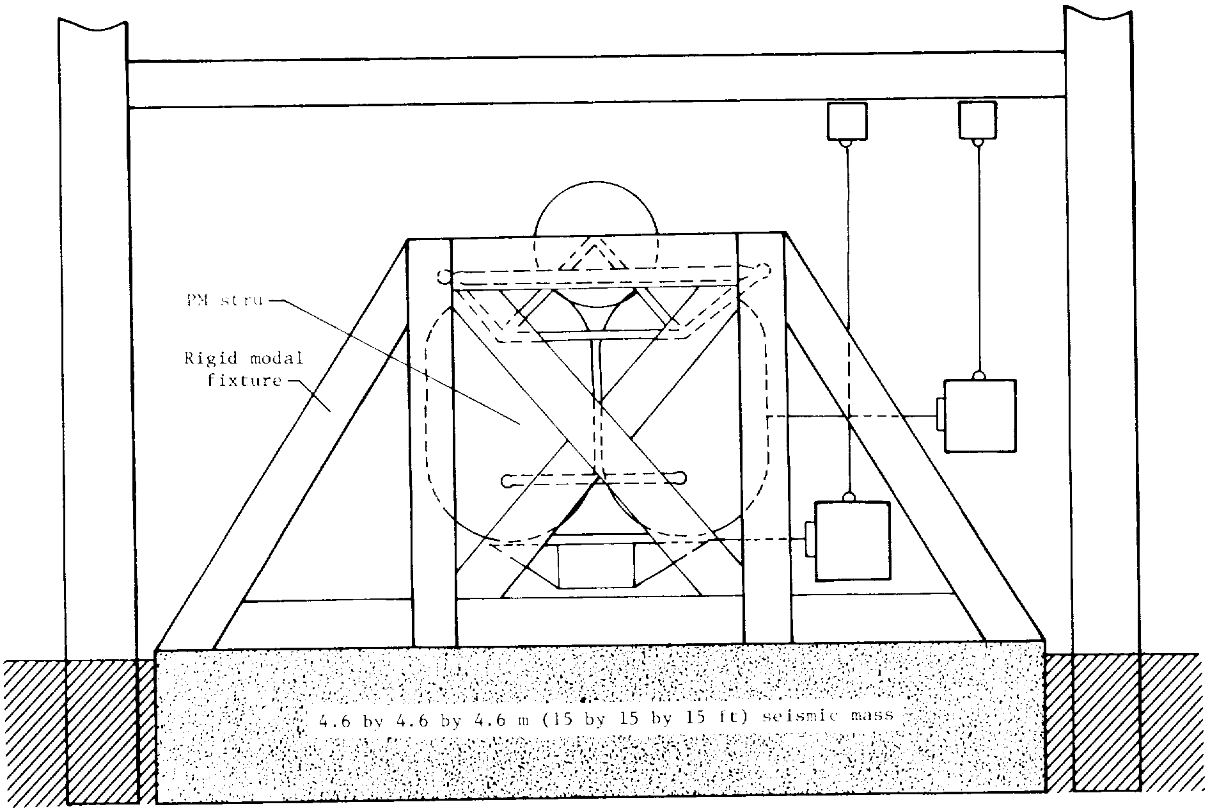

Figure 14.- ASTM PM modal tests configuration.

accept a reasonably large variety of configured surface tension devices. A series of surface tension design concepts were evaluated both analytically and experimentally. Full-size devices were applicable for limited tests such as propellant slosh, structural integrity, and configuration compatibility. It was also necessary to build several special purpose scaled models. The surface tension devices were scaled and installed in scaled plastic tanks to provide visual and photographic observations of test results. The primary concern was ullage bubble control in a 0g environment to assure propellant available at the exit end of the propellant tank at time of engine ignition and during the burn. The most viable test method among the many considered was the free-fall drop test. In order to evaluate the various mission events affecting propellant management, it was necessary to build models ranging from 1/32 to full scale to provide correlation with analysis, scaling parameters, and tests. Special development tests were conducted for design verification of installation and removal, fill, drain and slosh, vibration, steady acceleration, and shock.

The propulsion B/B and ETM were designed and fabricated to verify all functional aspects of the PROPS. The B/B was configured from development Viking hardware and some surplus Mariner Mars '71 hardware. Extensive special instrumentation was employed to evaluate component and assembly operations and interactions. The B/B was installed in a vacuum environment test facility and hot fired. More than 9400 sec of engine burn time was accumulated including three full duration MOI burns and 23 total burns. The ETM was the first propulsion evaluation of Viking prototype hardware. The ETM was subjected to full functional performance including hot firings similar to the B/B. At this point in time, it was adequately demonstrated that the Viking propulsion design was fundamentally sound.

The ODTM was fabricated from structural flight-type hardware to physically and dynamically simulate the Viking orbiter for the low-frequency range. The ODTM provided a test bed whereby a total inertially simulated spacecraft was assembled, starting from the Centaur transition adapter and including the V S/C A, the ODTM, the VLCA, and the RL. With this configuration, it was possible to conduct very meaningful static, modal, and sine vibration tests. These tests were required in order to verify the estimated dynamic characteristics of the mathematical model of the orbiter structure used in the loads analysis and in analysis for establishment of structural qualification test levels. They also aided determination of open-loop transfer functions, establishment of flight instrumentation location, autopilot interaction, and determination of dynamic deflections. In addition, these tests verified the ability to predict analytically the dynamic response and loads in the ODTM.

In the course of hardware design and development of Viking primary structure, a number of independent component tests to failure were performed. As an example, four separate aluminum tubular members were failed in compression. These included two members of the orbiter bus upper plane truss and one member from each of the V S/C A and VLCA. The test to failure was primarily done to gain early confidence in the various strut designs prior to more complex tests involving the total S/C. The secondary objective was to verify the mode of failure, verify ultimate capability, and check material allowables.

The HGA assembly, including the reflector dish and support structure, was exposed to modal and sine vibration tests in both the stowed and deployed position. The tests were conducted to verify the mathematics model and to determine if the structural criteria used in the design of each part were met. Another test objective was to obtain natural frequencies, mode shapes, and modal damping. The HGA was mounted on a rigid mass and subjected to three-axis sinusoidal vibration subsequent to a modal survey.

The scan platform structure, latch system, actuator support mechanism, and inertially simulated science instruments were fixture mounted and modal tested in both stowed and deployed configuration. The objectives were to experimentally determine the structural and dynamic characteristics of the scan platform and support structure for verification and/or modification of the analytical model. Instrumentation located on the test article provided data on natural frequencies, mode shapes, damping behavior, linearity, modal forces, and clock and cone stiffness.

The objectives of the separation and deployment tests were to evaluate the ability to predetermine VO parameters significant to separations, to predict separation characteristics using these parameters, and to verify design and operation of unlatching and deployment mechanisms for antennas, solar panels, and the scan platform. Selected configurations were tested and correlated to analysis. There were no full-scale separation tests conducted for either the S/C to LV or VLCA to VO separations. It was a project decision to rely on extensive conservative analysis supported by component or subassembly tests. Tests were run to measure force and deflection characteristics of a VLCA tripod assembly and dynamic characteristics of a free tripod when released from a deflected position. Separation springs were tested for force constants and alignments. Spring matching and alignment was critical from the standpoint of clearance during separation. Electrical separation connector drag and influence during separation were determined by extensive simulated separation tests where parameter effects were measured. The separation release devices were explosive nuts. Considerable testing was performed at temperature extremes with single and dual squib firings as well as simultaneous single and dual squib firings. Deployment tests were performed to verify the design of release, hinge, torque limiting, and damping devices by suspending them on cables and counteracting the effects of Earth gravity. This allowed deployment tests at near flight conditions. Solar panels are held in a stowed position by fly-away release rods. Functional and environmental testing of the release rod and solar panel hardware and boost dampers that stabilize the release rod during launch was conducted both at the device level and also at the VO system test level. The RAS release and deployment, HGA release, and scan platform release devices were qualified through extensive environmental and functional testing.

In general, thermal control development tests were made to verify experimentally the analytical predictions of overall orbiter thermal performance and to determine experimentally certain parameters which were not accommodated analytically. The tests, as discussed subsequently, were performed by using a special OTCM. (See fig. 15.) Tests were conducted in a space simulator having high-vacuum, cold-wall, and solar simulation capability. The heat dissipation in the orbiter, the environment provided, and the configuration were varied during the test program as means of achieving the objectives. The OTCM was a full-scale model of the Viking orbiter. It was built to provide the correct conductive and radiative heat-transfer paths but did not contain operating electronics; instead, resistive electrical heaters were used to simulate the heat dissipation of the electronics. The lander thermal simulator was used with the OTCM as appropriate. The thermal control tests were divided into two segments called TCM I and TCM II. TCM I used approximately 240 hr of space simulator time and was broken into three phases with chamber breaks between phases. The specific objective of TCM I was to demonstrate the adequacy of the orbiter temperature control design by evaluating the assemblies conservatively with and without the lander simulator during propellant warm-up which occurred shortly after launch and lasted 5 days, perihelion cruise, MOI, Mars aphelion cruise and orbit, and Mars solar occultation. A conservative margin was introduced by combining either a maximum or minimum expected solar intensity with a maximum and/or minimum internal power dissipation. A total orbiter bus power consisting of all maximum or minimum electronic bay power, regardless of the operational mode in which each would occur, was used. Total scan platform and appendage item power was accounted for. The specific objective of TCM II was to conduct an in-depth evaluation of the thermal control design and to bracket all mission conditions. Design changes required, based on TCM I findings, were incorporated prior to TCM II tests. TCM II used approximately

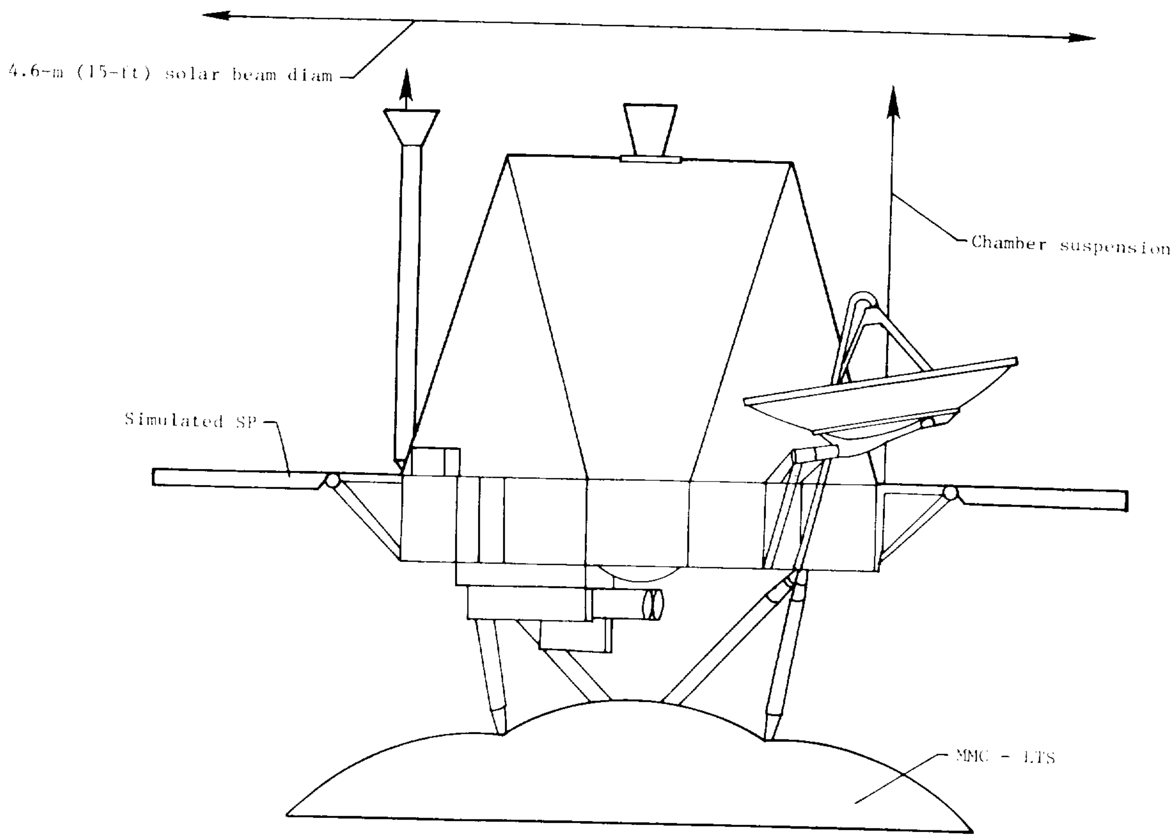

Figure 15.- VO thermal control model configuration.

640 hr and was divided into three phases: orbiter with lander simulator, orbiter alone, and solar panel and relay antenna but no orbiter.

Because of the optical instruments on the orbiter, it was extremely important to eliminate or reduce all stray-light reflections. Tests were made to determine the relative amounts of stray light and evaluate the effects on celestial sensors and science instruments. Because the external surfaces of the OTCM realistically simulated the surface finishes and geometry of the flight vehicles, it was used for the stray-light mapping test.

The LDTM and ODTM were assembled in a launch configuration at JPL and used to verify form fit and structural dynamics. The data obtained from spacecraft level development dynamic tests were used to verify the validity of the analytical techniques which were used to predict V S/C responses. The testing of the LDTM and ODTM covered forced sine vibration and alignment. The configuration was as shown in figure 16. The LDTM and ODTM were mated using VLCA, V S/C A, and VTA and coupled to external vibration sources. Dynamic instrumentation used during separate LDTM and ODTM tests was augmented by additional sensors located on the VLCA and near the VO/VLC mechanical interfaces. Test instrumentation cables were suspended in a manner which minimized mechanical

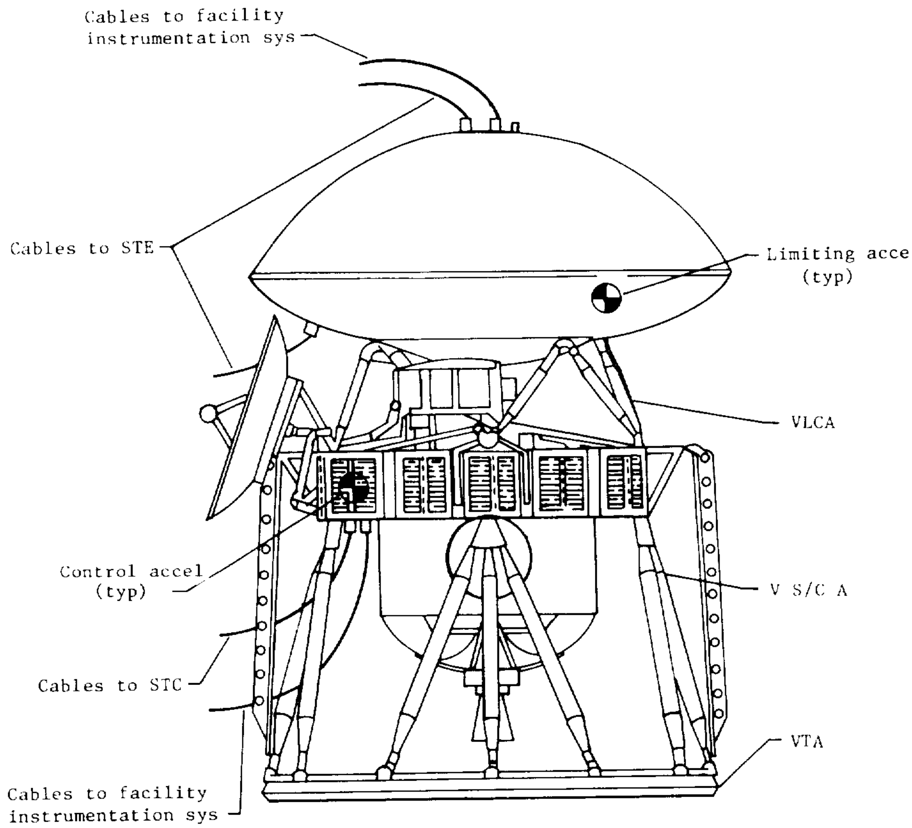

Figure 16.- Dynamic test configuration for LDTM and ODTM

attenuation and amplification effects. The LDTM bioshield was pressurized. The objective of the forced sine vibration test was to produce mechanical excitation to demonstrate integrity of the secondary structure and equipment mounting and determine dynamic interactions. The environment was based on predicted response to launch excitation in the range from 5 to 200 Hz. Longitudinal excitation was applied through the V S/C A and VTA. Points of application for lateral excitation were on the orbiter bus. The vibration inputs measured on the orbiter bus primary structure were controlled such that a specified level was achieved at one or more locations. The inputs were limited so that no point on the VO or VLC primary structure exceeded the design structural load. Spacecraft alignment verification test objectives were to verify the basic VL/VO alignment geometry in the LDTM/ODTM stacked configuration. The basic alignment geometry about the spacecraft X- and Y-axes were verified by optical alignment measurement between the VLC bioshield at the VLCA/VLC field joint (station 200) and the VO sun sensor mounting surfaces. The basic alignment geometry about the spacecraft Z-axis was verified by optical alignment pins on the orbiter bus. An error analysis was prepared which defined the resulting end-to-end measurement uncertainty. Means were also provided to check VLC alignment between S/C station 200 and VLC plane A (IRU mounting surface) as a backup or additional verification. The test was considered a success because the measured values agreed with the predicted values within ±9.0 arc-min.

The final structural development test was a static ultimate test to qualify the primary structure not previously qualified to ultimate loads. The ODTM was configured as a spacecraft with the static test fixture simulating lander loading geometry, VLCA, PM, V S/C A, VTA, and Centaur truss adapter. External loads were applied directly to the bus, scan platform, solar panel outriggers, static lander, and propulsion module. The resulting structural internal loads and deflections were monitored and verified by strain gages and linear sensors. Initially a series of loads were induced to verify the influence coefficients used in the mathematical model. The mathematical model was used to determine critical loads for use in the static qualification tests. These loads were then applied to the ODTM in a series of qualification tests to demonstrate the capability of the orbiter structure to sustain mission loads with margin.

QUALIFICATION TESTS

Following development tests, the orbiter assemblies and subsystems were built and qualified as appropriate to FA and TA levels, functionally and environmentally. They were then delivered to the SAF at JPL for systems assembly and functional test prior to the start of the formal environmental tests. Figure 17 summarizes the required system level tests in the sequence planned for TA and FA.

The Viking orbiter system design was environmentally qualified for its intended flight mission by the implementation of a formal environmental test program. Type approval, flight acceptance, and life tests were the formal environmental test requirements of the Viking orbiter project. A TAT of the PTO was performed, and each flight orbiter was FA tested as a system. Each flight orbiter assembly was FA tested prior to its installation on the orbiter or use as spare hardware. The environmental testing was performed in conformance with the general and detail environmental test specifications, the general environmental test standard, and the detail environmental test procedures. In accordance with Viking orbiter project policy, the Orbiter Environmental Requirements Engineer was responsible for defining the rquirements, technical auditing, and appraisal of the orbiter system environmental test program. It was his responsibility to periodically evaluate the status of the environmental test program and to advise the Orbiter Spacecraft Manager of the compliance of the various system elements with the environmental requirements. Table 5 includes all environmental tests that were part of the formal qualification and FAT program.

The principal objectives of the system and assembly level environmental tests are as follows:

System test objectives were to establish confidence that the orbiter system design would function as required under exposure to flight mission environmental conditions and to establish confidence that the flight orbiter quality would be representative of the quality of design verified by the TA system test and would, therefore, be flight acceptable.

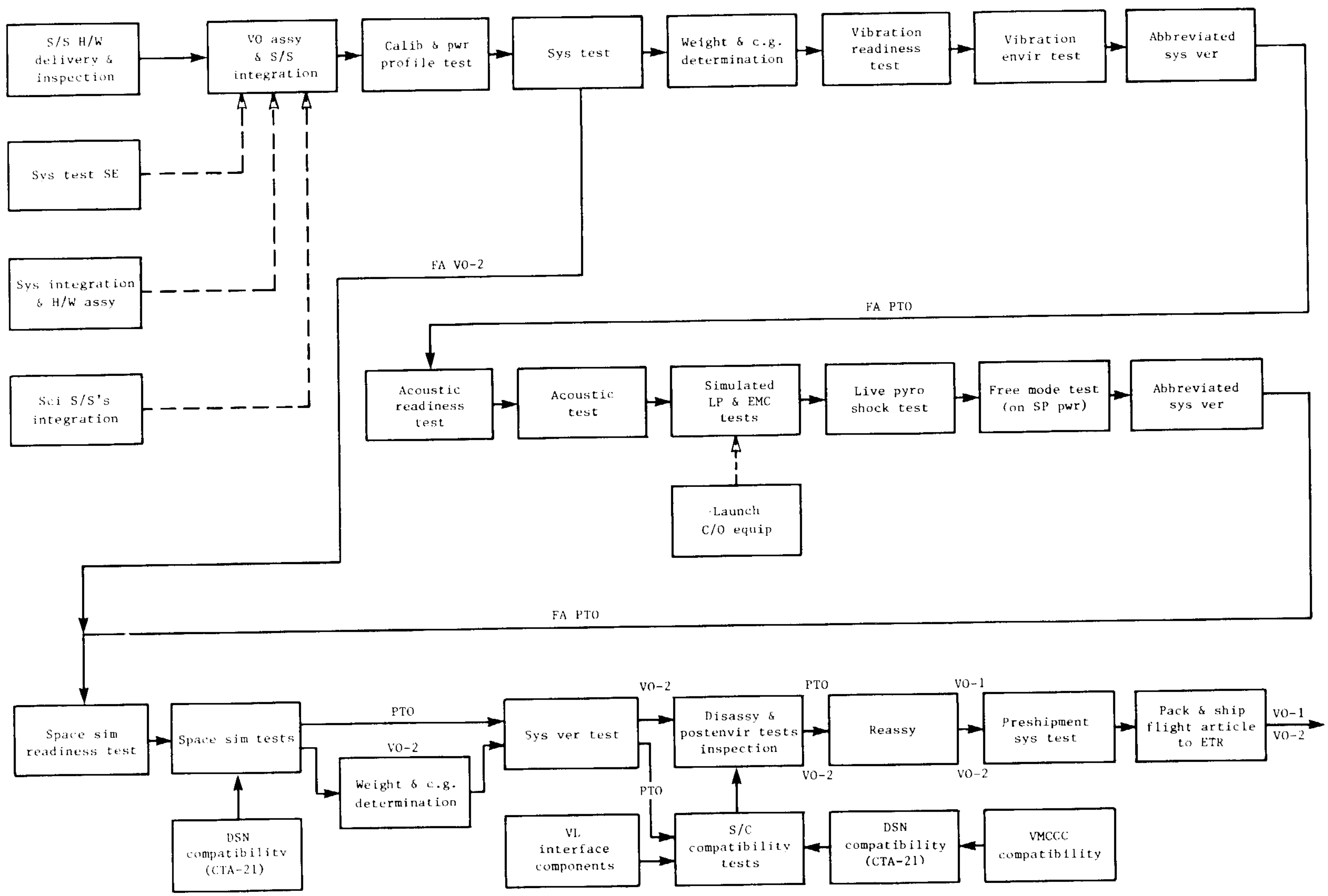

Figure 17.- Flow diagram of VO system tests.

TABLE 5.- ENVIRONMENTAL TESTS FOR VO

	Subassy level	Assy level	Sys level	S/C level	Assy level	Sys level
Test type	TA	TA	TA	TA	FA	FA
Ground handling of equip pkg drop		✓				
Transportation vibration		✓				
Ground op humidity		✓				
On-pad explosive atmosphere		✓				
EMI of S/C: Int & ext sources		✓	✓	✓		✓
Launch phase vibration		✓	✓	✓	✓	✓
Launch phase acoustic noise		✓	✓	✓		
Launch through orbit op therm vacuum		✓	✓	✓	✓	✓
Int shroud press transients in launch phase		✓				
Pyro induced shocks		✓	✓	✓		✓
Static acceleration of LV during powered flight		✓				
Therm shocks during mission events		✓				
Electron and/or proton radiation during cruise & orbit		✓				
Cold weld of mech	✓					
RTG radiation				(a)		

[a]Not a qual test; VOS was subjected to brief RTG environment.

Assembly test objectives were to establish confidence that each orbiter assembly design would function as required under exposure to all environments the assemblies would encounter as individual units or as part of a spacecraft and to establish confidence that each flight orbiter assembly quality was representative of the TAT verified design quality and that the assembly would function as required under environmental conditions less severe than TAT requirements but more severe than expected mission environment conditions.

Type approval tests were the normal orbiter equipment environmental qualification tests required on certain Viking orbiter subsystems and at the PTO system level. TAT demonstrated the design adequacy of the equipment and included appropriate performance and margin tests and other similar tests as defined in the test specifications. The equipment was considered environmentally qualified upon satisfactory completion of the TAT's at both the subsystem and system levels. Satisfactory completion of FAT prior to the start of TAT was desirable but not mandatory.

The Viking orbiter system level TAT program was determined by an evaluation of the various critical environments having significant environmental interactions within the orbiter. The subsystem level TAT's consisted of environments imposed on the equipment through system interactions plus those environments imposed on the equipment only at the assembly level prior to orbiter installation. The FAT program was based upon similar considerations as the TAT program; however, the FA environmental tests selected at both the system and subsystem level were those which tended to yield the maximum information about the flight acceptability of the equipment.

The propulsion subsystem received vibration, acoustic, thermal, pyro shock, and some functional testing as part of the formal PTO system level tests. The PROPS TA model was assembled from flight qualified components. After assembly, the PROPS was subjected to proof pressure test and examined for evidence of deformation, yield, or damage during the test. Proof tests were followed by external leak tests. The leak tests were performed on braze joints, service valves, engine valve fittings, and pyro valve squib cavities. Vibration, acoustic, and thermal testing of components and assemblies were completed prior to assembly of the PROPS TA model. The PROPS TA model was installed in the vacuum chamber for the hot fire testing; after this installation, full functional and leak tests were performed to confirm specification compliance. Such things as regulator lock-up pressure, check valve cracking pressure, solenoid valve response time, relief valve cracking and reseat pressure, and engine valve response time were verified within specification. The hot fire tests included subjecting the PROPS TA model to two simulated mission duty cycles. There were a total of 34 separate engine firings ranging in duration from 0.38 sec to 2444.93 sec and yielding a total cumulative burn time of 6700 sec. Performance checks such as vacuum thrust level, vacuum specific impulse, instantaneous thrust chamber pressure, average mixture ratio gimbal actuator force, engine valve peak soak back temperature following an MOI burn (important because of concern for engine valve seat material deformation), and actual versus calculated use rate for fuel, oxidizer, and helium pressurant gas were made for specification compliance. There were some areas of concern following these tests and they were further evaluated. Examples of these areas were events such as gas system pyro valve slam-start accompanied by check valve chattering and unfavorable system pressure which could potentially rupture a burst disk, predicted versus actual fuel use, fuel latch valve inadvertently opening during a pyro event, latch valve position indication switch failure, and fuel feed line pressure sensor "zero shift" of 2 percent; none of these areas proved to be critical. The PROPS TAT program added to the experience already gained in the PROPS B/B and ETM tests mentioned earlier and demonstrated that the PROPS design was fundamentally sound and would meet all mission requirements.

SYSTEM TESTS

Objective

The VO system test program provided an orderly sequence of tests and operations leading to the type approval of the PTO and flight acceptance of the flight VO systems and spare assemblies. A series of compatibility and interface tests of the VO/VL, the VO/VL/LFOS, and the VO/VL/DSN/VMCCC were conducted, and secondary accomplishments including procedural development and personnel training were also attained.

The formal system test demonstrated the VO performance as an integrated system. Within the limits of the test configuration and the Earth-based environment, the system test provided a comprehensive and detailed exercise and evaluation of all parameters and functions of the VO. The system tests provided the following functions:

Exposed unexpected subsystem interactions and anomalies

Exercised all major elements of the system and demonstrated their ability to meet the applicable design requirements in the system environment

Provided a system performance history

Verified system performance in mission modes

Verified system compatibility with other Viking systems

Demonstrated system functional margin for the internal noise environment

Assisted in developing mission support data processing

All system level tests included both parts of block and functionally redundant elements. Tests with the PTO included both TA elements and spare elements in the block-redundant S/S's. The mixture of spares and TA hardware provided for a complete VO system for design verification and also for FA of the spare S/S's by operation in the system. When the spare S/S's were delivered, they were substituded for the TA S/S's to obtain the maximum system operating time on the spare hardware.

Design and Operations Philosophy

The VO system test equipment design was based on the use of the required S/S test equipment capabilities rather than construction of a new system test equipment complex. Test capabilities were developed as computer aided, manpower intensive, manually controlled operations. No automation of system test procedures was used. No major software effort was required to support the test program other than the telemetry processing and CCS command generation software which carried over to the flight operations.

Skilled manpower permitted quick adjustment of test to troubleshoot, define, and then work around anomalies without significant delays. Meticulous record keeping permitted a recovery of by-passed test elements without excessive recycling of tests already completed. Test team manpower maintained continuous communications with the cognizant divisions, facilitating quick response when problems required expert support.

Implementation

All system test operations were performed by the VO test teams. At the beginning of the system test program, two complete test teams were required initially to provide around-the-clock support of the PTO testing in the space simulator and then to cover the activities taking place in SAF - the PTO undergoing VL compatibility and special tests, VO-2 S/S's being verified and integrated, and the VO-3 bus arriving and being assembled. (See fig. 18.) However, after completing a very successful PTO test program and observing the exceptional quality of the hardware used, a decision was made to use the PTO as a flight system to accomplish a recommended cost reduction option. This action eliminated the requirement for VO-3 system testing and reduced the necessity

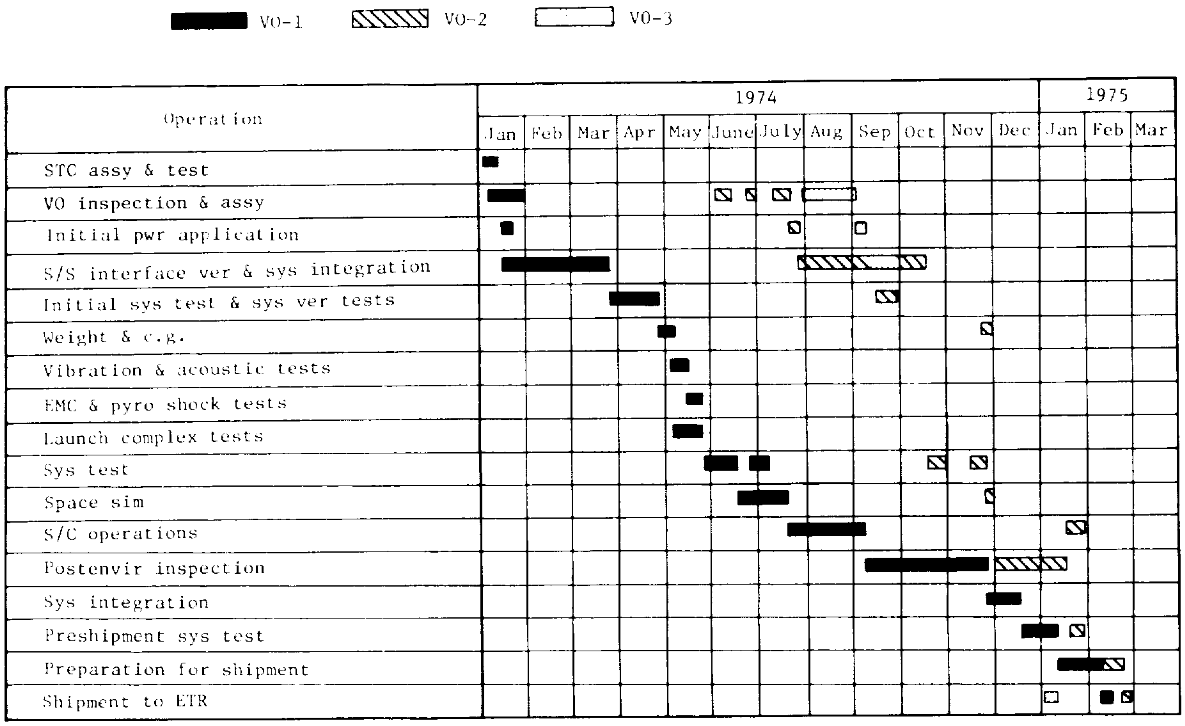

Figure 18.- VO system test calendar of events at JPL.

of 2 test teams to 1. The reductions allowed electrical tests to be performed on one VOS by the single electrical team while mechanical work was simultaneously performed on the second VOS. Consequently, VO-2 became the second flight unit and VO-3 S/S's were designated as spares. Additional cost reductions were realized in VO-2 testing including one less system test, no vibration, acoustics, or pyro shock, and reduced space simulation test time.

System Test Complex

System testing and data analysis were performed by a standard array of S/S and system test equipment called the system test complex. The S/S test equipment elements of the STC were capable of providing failure isolation to the replaceable spares level through a combination of processed telemetry, direct and umbilical access, and data from other support equipment. Figure 19 illustrates the STC and interfaces for system test setup.

Mission and Test Computer

MTC provided computer processing to handle and record the large mass of data produced by system testing. In real time, the MTC presented a system overview, sifted all data to expose unpredictable anomalies or malfunctions and provided the means of correlating seemingly unrelated events. In nonreal time, recorded data were reprocessed and presented in many other ways as requested.

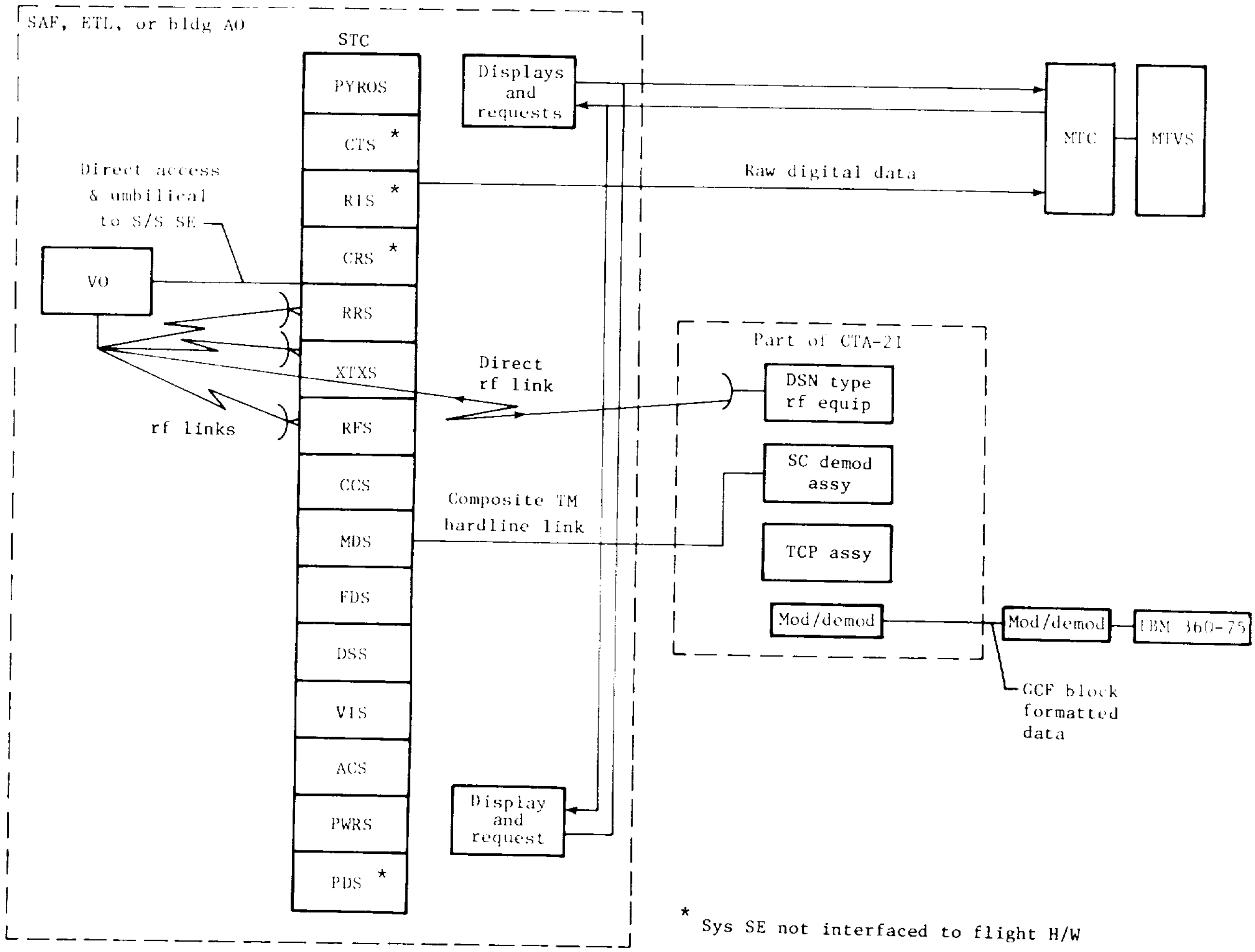

Figure 19.- System test complex configuration.

Critiques and Reviews

At the end of each test phase on each system, such as, a system test or the space simulation test, a critique was held by the Operations Manager. The Test Chief, his team, and a representative of each technical discipline were required to sum up the test results and report to the group. A board consisting of the LaRC Test Manager, the VO System Engineer, and the Operations Manager reviewed the results and recommendations and concurred in action required to continue into the next phase. Critiques served as management milestones between major reviews.

PTO/VO-1 System Tests at JPL

Delivery and Inspection

As each individual VO S/S or hardware was received, and before it was installed or tested, it was inspected and certified by SAF quality assurance personnel. The VO cable basically consisted of an SEC harness on the aft end of the VO, an upper ring harness, and interconnecting cables to each bay and was tested at installation. The electronics assemblies were installed in as logical a sequence as possible as they became available, either in breadboard or prototype form. Pyrotechnic equipment, scan platform, science instruments, guidance sensors, and mechanical devices were installed and individually exercised in preparation for system testing. Not installed at this time were solar panels, radio antennas, PROPS, and RCA. Instead, solar panel and propulsion simulators were used, and the antenna outputs were hard-wired into the SE.

Initial Power Application

The first power turn-on occurred as soon as all harnesses and the PWRS were installed and checked. All connectors were open-circuit tested to ensure that the correct voltage appeared at the proper pins, and then a simulated S/S dummy load was applied to verify harness resistance. Various power parameters were also verified and initial VO external power turn-on then occurred.

Subsystem Interface Verification and System Integration

As each S/S was installed, a check was made of its interface impedances and grounds. Following initial power turn-on, S/S interfaces were checked as the S/S's were delivered and installed. The installation was accomplished in order of increasing complexity, progressing from simple loops of two S/S's only, to more complex multisubsystem loops. The VL interface duplicator substituted for the VL during this testing. After basic integration of the PTO with the STC to the level of system test capability, each S/S was tested according to its own JPL procedure. Because of hardware availability problems and the inability to exercise all VO operating modes early in the testing phase, elements of the test procedures were exercised out of order. As new assemblies were substituted for prototypes (fig. 20) applicable portions of the integration procedures were repeated. All S/S integration procedures were completed by the end of this phase.

Initial System Test and System Verification Tests

The SRT verified the major VO system functions and the system SE configuration (direct access connections and external stimuli, VIS collimator alignment, ACS light hoods, gas jet pressure switches, etc.). Power profile and benchmark data were gathered for comparison with like data from future SRT's. Six system level verification tests were performed to verify correct S/S calibration and alignment before completion of the initial system test which preceded environmental testing. The six tests were

(1) VO block tests validated flight software as to proper initial conditions, constraints, final state, and response

(2) Power profile and transient measurements were taken for each S/S to verify tolerances

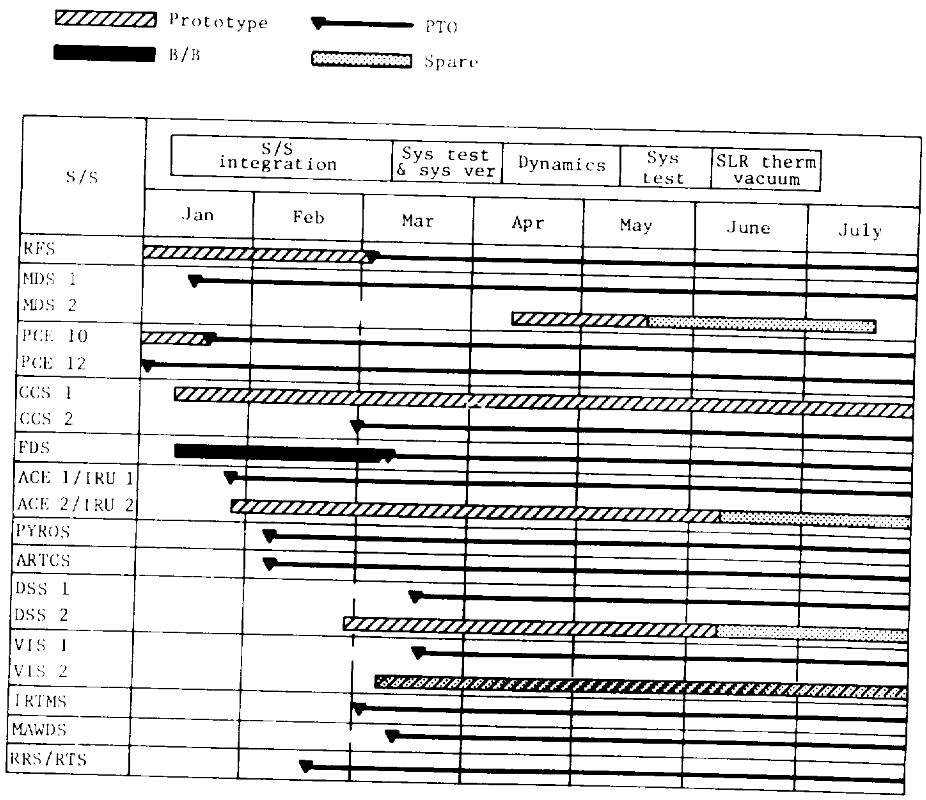

Figure 20.- PTO system (H/W) content versus time.

(3) Solar-panel mechanical deployment was tested as close as feasible to flight configuration to verify proper functioning and to compare with DTM test results

(4) Telemetry calibration was verified as to agreement with data sheets, proper output, and FDS programmability of all engineering telemetry measurements

(5) Science instrument alignment with respect to other instruments and with respect to the scan platform coordinate system was accomplished by use of a collimator and mirrors

(6) ACS calibration and mechanical interface verification was done with measurements of the mechanical alignment offset of the various ARTCS actuators

No major problems were found during this phase of testing.

Weight and Center-of-Gravity Measurement

Mechanical buildup of the PTO for the weight and center-of-gravity measurement began according to the appropriate JPL procedure and continued until the measurement was completed. The measured weight was within 4.5 kg (10 lb) of the predicted weight.

Environmental and Launch Complex Tests

Following completion of VO vibration tests, another series of vibration tests with a VLMS installed on the VO was carried out. The PTO was returned to the transporter (fig. 21) and rolled into the acoustics test chamber for

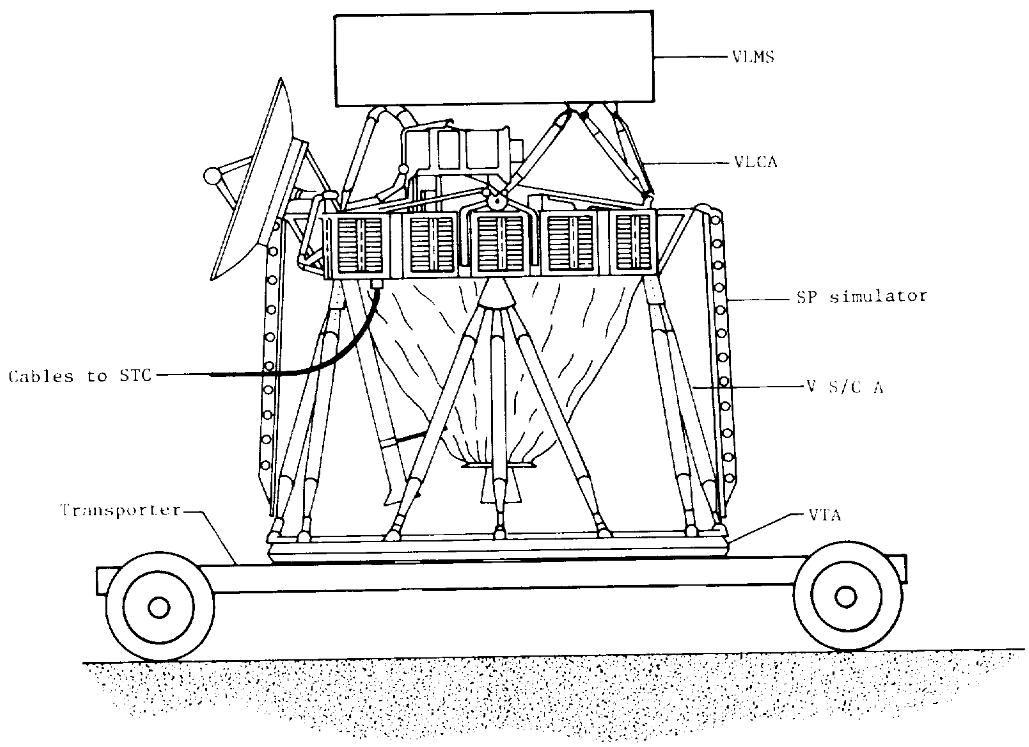

Figure 21.- VO/VL acoustic test configuration.

acoustics, EMC, and pyro shock tests. The vibration and acoustics tests were completed, and the launch complex tests were performed in conjunction with the other tests in this phase. This test configuration was identical to the launch pad configuration, and the LCET was used to support all tests in this phase. The vibration and acoustic tests caused the performance of some S/S's to vary during the tests. However, these S/S's returned to normal operation after the testing, and analysis indicated that no corrective action was required. For example, the RFS receiver LO drive decreased by about 0.3 dB during vibration testing. Variations of this type have been observed in similar RFS designs on prior programs and are not considered significant. Another example was the DSS, which was unable to maintain bit sync lock on the recorded data during vibration and acoustic testing. The problem was caused by tape speed variations of ±20 percent in the DTR, which was the typical performance of the DTR during vibration.

Electromagnetic Compatibility and Pyrotechnics Shock

EMC and pyro shock tests were performed immediately after the acoustics tests, and the PTO was moved back to SAF. At the conclusion of the environmental tests the PTO appeared to be in good condition. The test configuration for ACS, CCS, and MDS included only the PTO half of each S/S, with the other half reserved as flight spare and not installed for the TAT's. The MAWDS was not powered at all, as is normal for the prelaunch configuration. Performance of the LCET equipment and remote control elements for on-pad activity was satisfactory and trouble free. The VOS had 523 running hours on it. Two significant problems occurred during this phase of the tests as follows:

(1) During the EMC test, the TM channels for the PROPS tank pressure decreased when the VL relay radio simulation source was turned on and transmitted. The problem was traced to pressure sensors which were sensitive to the 381 MHz transmitting frequency. Special tests revealed no damage to the sensors and, as data from these channels were not used during operation of the VL relay radio, the only action taken was monitoring of the problem during subsequent tests.

(2) The quantitative leak test exposed leaks in the RCA high-pressure module by using a leak-detecting liquid solution. The fill-valve handle cavity and isolation valve stem in bay 11 were leaking. Both valve assemblies were disassembled and microscopically inspected. Small metallic particles were found on the O-ring sealing surfaces but no damage was evident. After ultrasonic cleaning, reassembly (replacing the fill valve O-ring with a new O-ring), and tests, no further leaks occurred. The bay 3 fill-valve handle cavity had a torn silicon rubber O-ring. As a result of review with the hardware vendor, more specific assembly instructions were incorporated and leak testing was required six times before launch.

System Test Before Space Simulation

Major reconfiguration had to be accomplished before the next system test and the space simulator test. All development test model and prototype hardware was removed and replaced, that is, VIS A, CCS B, CDU A, TMU B, ACS B, DSS B, PWRS, and PROPS. Several S/S's required rework, that is, FDS, RFS, MDS, and MAWDS. Six new S/S's were integrated during this phase, that is, MDS, ACS, IRTMS, PWRS, DSS, and PROPS. The system test demonstrated that the VO performance was satisfactory. A complete functionally redundant system was assembled, and all new and reworked hardware performed well. Minor problems encountered in the DSS, IRTMS and FDS were analyzed and successfully corrected. The PTO had completed approximately 670 hr of system testing at the conclusion of this phase.

Space Simulation Test

The PTO was moved to the 25-ft Space Simulator Facility for the space simulation test. A number of operations were performed during the 29 days that the PTO was in the space simulator; some of these operations are as follows:

(1) A detailed system test demonstrated satisfactory functional performance while operating in a simulated space environment.

(2) Solar vacuum tests verified the temperature control design, and no thermal control problems were encountered in normal FA and TA thermal modes. However, the simulated solar occultation after TAT cold temperature was not performed because some S/S's were at their allowable lower limits. The system was cooler than the temperature control model predictions, indicating lower total power dissipation. During the TAT hot temperature, the HGA was configured for the near-Earth condition and ran hot as expected (near 93° C (200° F)).

(3) Parallel testing for compatibility with CTA-21 (the DSN compatibility test area facility) was about 90 percent complete when the test ended.

At the conclusion of the space simulator testing, the PTO had operated over 1086 hr of which 380 hr were in vacuum. Significant problems encountered during this phase were as follows:

(1) Battery 1 cells were giving unusual readings at the PWRS SE. Two cell monitor fuses were found to be blown, and a suspect scanner card was removed from the SE. Many hours of testing could find nothing wrong with the card but, since a spare card performed correctly, the suspect card was replaced.

(2) SEC 4 reached a mechanical stop before arriving at its electrical null, preventing the slew from terminating. Buildup of mechanical tolerances was a probable cause of the problem. The software was changed to restrict the SEC from motion to either extreme which prevented a reoccurrence of the problem.

(3) DSS DTR A performance during a TA cold test degraded and prevented data recovery. The servo loop would not maintain tach lock due to a very high 30-Hz component in the loop. The DTR gradually recovered after 2 hr of operations as the temperature rose about 3° C. Analysis revealed the transport damper was not damping the normal transport resonance at both high and low temperatures due to variations of the spring and damping constants of the rubber damping material over temperature. To assure an adequate temperature margin, the preferred cruise temperature was changed to 16 to 27° C (60 to 80° F) and, at near 10° C (50° F), a warmup time of up to 5 hr was specified.

(4) During phase 2 SVT, the MAWDS optics heater was on 85 percent of its duty cycle, too high a percentage to provide adequate heater power margin for the scan platform in colder cone/clock positions. The duty cycle was improved to 60 to 63 percent by increasing the heater on the MAWDS side of the scan platform structure from 2 W to 4 W.

Viking Spacecraft Testing at JPL

At the end of space simulator testing, the PTO was moved back to SAF. The next 2-month period of operations included completion of the CTA-21 compatibility tests, PTO/VL hardware integration, and other compatibility tests. The VO system generally performed properly and in specification for all phases of this series. Operating time of the PTO increased to 1323 hr. The original test plan had the spacecraft test lander in SAF for integration of VO, VL, and DSN and VMCCC via CTA-21. Schedule problems precluded the complete VL tests at JPL, and a substitute test series using the PTO and all VL interface S/S's was conducted instead. All VO and VL functions related to their electrical, DSN, and VMCCC interfaces were verified. VO compatibility with the VL RTG's was verified with some margin. All functional interfaces between the VO and VL were verified to be within specifications except the VO/VL command modulation was not in specification. Subsequent analysis showed that false acquisitions caused by the modulation were inherent in the system, predictable, and avoidable with judicious selection of flight operating conditions.

PTO Special Tests

Following VL integration, the PTO test program was reviewed and several special purpose and failure-mode tests were performed to verify the design. The VO performed satisfactorily with the following exceptions:

(1) During EMC testing, the IRTMS showed a susceptability to the UHF and S-band emission frequencies by an offset in all detector outputs. The primary cause of the interference was rf radiation entering into the instrument aperture and being picked up on the detector to hybrid-preamplifier wires. S-band interference was eliminated by grounding the detector center taps, making a twisted triplet of the three wires between each detector and its associated preamplifier, and closing the gap at the base of all telescopes with a silver-filled epoxy fillet. The requirement that the IRTMS be immune to the UHF rf emmission was waived, because the S/S's would not be required to operate simultaneously when in close proximity.

(2) An MTCF picture recorded during the DCT 2 test showed a tooth-shaped pattern in the lower-left corner. The VIS vidicon had a manufacturing defect in its photosurface and was replaced.

(3) Inspection of the IRTMS revealed a wire with cut copper strands and insulation. A test connector mounting at the inside of the back housing cover had chafed several wires contained in the wire bundle, although no shorts could be found. The chafed wires were repaired and steps taken to insure no recurrence of the problem.

(4) After initializing the CCS with an update and enabling the spacecraft separation routine with a command, a manually applied short to the Viking separation lines caused both CCS processors to enter the error routine. The software operated in such a way that it attempted to store a word into protected memory, resulting in a hardware error interrupt. The launch hold reset routine was altered to prevent the reoccurrence of failure.

(5) The PTO drew excessive current from the external source when the switch on the breadboard booster regulator was placed in the 64-V position during overvoltage testing. A broken wire in the booster regulator permitted its output voltage to rise to approximately 82 V dc. All S/S elements on board the

PTO at the time of the overvoltage test were reviewed and all components that were overstressed were replaced.

Postenvironmental Inspection

The PTO was moved from the test stand to the work stand, where an extensive disassembly and inspection began. All S/S's were cleaned, certain new spare S/S's replaced TA articles, and the orbiter, now designated VO-1, was reassembled in preparation for a final system test. No significant problems were reported during this phase.

Preshipment System Test

With the VO-1 assembly completed, a month of system integration testing followed, during which no major problems developed. Abnormal activity was noted on the LGA while the RFS was driving the HGA, but insufficient data were recorded for analysis. Since the problem occurred but once, the only action taken was to request the MTC to record raw data for analysis should the problem reoccur. The formal system test proceeded smoothly to completion with only the optional MOI restore test being omitted. Radiated emissions and susceptibility tests were run in parallel with the system tests with no new anomalies. The MAWDS detector 5 output was abnormal for a period during the system test caused by non-orbiter-generated 381-MHz interference in SAF. In any case, the interference would not be a problem in flight, and the requirement that the MAWDS not be susceptible to the 381-MHz rf signal was waived. VO-1 had accumulated a total of 1510 hr of operational time at the end of the test.

Preparation and Shipment to ETR

After the VO-1 passed the final system test and was approved for shipment, it was partially disassembled and inspected. Certain delicate items (ACS/IRU's, HGA, etc.) were put in their own special shipping containers, while the VO was mounted on its transporter. Two CCS connector pins, one in the output unit module and one in the cable harness, were found damaged during disassembly. The pins were offset and the connectors pushed back during a previous assembly. The pins and connectors were reworked and tested. Then, VO-1 and SE left JPL for ETR.

VO-2 System Tests at JPL

Delivery and Inspection

The VO-2 bus was received at SAF and installed on a work dolly, the scan platform positioned, the IRTMS purge lines routed, and the upper ring harness installed. The bus was then transferred to a system test stand. Assembly continued intermittently while the team members were involved with the PTO system tests and space simulation. Power-off checks preparatory to power turn-on were made, with initial power turn-on occurring 6 days later by using a borrowed test battery from VO-1. Nearly 6 hr of operating time elapsed with no major problems.

Subsystem Interface Verification and System Integration

Integration of the S/S's began with the PWRS prior to initial power application and continued with other S/S's as they became available. Several S/S's had minor problems, including the FDS, DSS, HGA actuator, MDS, ACS, and IRTMS; however, they were all corrected without excessive delay.

Initial System Test

The initial VO-2 system test was begun after several days of system readiness testing. A complete system test was performed and lasted for 102.2 hr. The only significant configuration compromise was the PTO RFS unit substituted for the flight unit. VO performance was very satisfactory with the following few exceptions:

(1) The CCS issued 2 erroneous commands out of 11 after updating the CCS flight program for the VLC preseparation sequence, performing that sequence, and activating the ACE power changeover routine. The launch hold reset support macro commands, which had not been updated, were changed to update the erroneous commands.

(2) MAWDS detector 5 offset values were too high during the first part of the intensity calibrate operation. The same decrease in instrument responsivity had occurred twice before and was thought to be due to deposits on the head, cold lines, and detector housing. The deposits were polymers from outgassing of the conformal coating. The MAWDS was disassembled, cleaned, reassembled, and the grids were reset to compensate for reponsivity variations between detector elements. Also, the procedures for MAWDS SVT chamber operation were modified to allow for a pump purging cycle to eliminate outgassing products prior to cooldown.

(3) VO command block 1.8, engine venting, of the block dictionary failed to set the PSU common prior to issuing the (8C) command. (For a description of the commands (number and letter in parentheses), see table 2 in ref. 2.) The sequencing error in block 1.8 was then corrected.

(4) At various times during VO-1 and VO-2 testing, MAWDS operation was affected by an unknown disturbance. In one disturbance, the MAWDS operation experienced an uncommanded wavelength scan, and in another disturbance an unplanned raster reset and an A/PW multiplexer step. It was determined these problems were caused by VIS turn-off transients which disturbed the MAWDS logic. The order of instrument turn-offs was changed so that the MAWDS was turned off before the VIS when both were turned off. Another change disabled suspect BCE circuits in the flight hardware, thus reducing MAWDS sensitivity to the VIS interference.

System Verification Tests and System Test

VO performance was satisfactory during the system interface timing verification, the S/S integration phases, and the system test which followed. All parts of the system test were completed in 104 hr, and the total VO-2 operating time was 494 hr. A system radiated emissions test and a radiated susceptibility test were run in parallel with the system test. No unexpected radiation or susceptibility was encountered, and the system did reproduce the known (and waivered) susceptibilities of VO-1. The known susceptibilities are as follows:

(1) An anomaly previously reported (three times) on the PTO and a total of three different times on VO-2 involving FDS memory errors was corrected with an engineering change to an inherent design deficiency.

(2) During CCS integration, PWRS transferred from the main to the standby power chain, and the CCS failed to issue the expected safing sequence commands. This failure occurred with new CCS regulated power supplies which required a longer processor delay than the old supplies. An authorized change increased the processor delay by a more than adequate margin.

(3) At VO turn-on, the XTXS rf output was 10.8 dB down from normal and required 38 min to reach its normal value. The unit was sensitive to momentary loss of its 19-MHz input because of a misaligned ×5 multiplier. The XTXS vendor performed realignment.

(4) The Canopus tracker did not operate in a closed-loop mode. The arm of the test stimulus hood was hanging up in one position for unknown reasons. Both the motor and gear-head assemblies were replaced with new units. The arm was removed and all mating surfaces cleaned and checked for burrs. The hood was reassembled and found to still hang up very slightly but not unacceptably so; therefore, the unit was designated as a spare.

(5) IRTMS full-frame and science decommutation data indicated loss of the sign bit when the VIS was off. A previous FDS logic modification to correct a MAWDS A/PW data problem inhibited the sign bit during a MAWDS A/PW read. An additional change corrected the FDS design to reset the sign bit inhibit after each A/PW read without dependency on another S/S.

(6) The ACS failed to respond to a manual command to control power to the CC buffer drivers and DC matrix. A design error bypassed the power bus switch, rendering the command useless. Isolating diodes were added to CC drivers so that the power bus could only receive power through the switch.

(7) During power-off continuity checks on the 30-V converter preload, a Zener diode was found shorted to the VO chassis. Replacement of nonflight diodes for flight parts had just taken place, and an improperly used lock washer pierced the mica washer insulating the diode from the chassis. The lock washer was replaced with a flat washer, the mica washer was replaced, and the diode installed properly. The other diodes were also inspected and reinstalled where necessary.

Space Simulation Test

Program limitations prevented a complete space simulation test as performed on VO-1, but the following tests were performed:

(1) Complete SVT with 195 hr under vacuum, and the VO system encountering only a few major problems described in the next paragraph.

(2) The CTA-21 compatibility test was run concurrently with space simulation. Although not all test criteria were met, there was no evidence of incompatibility between the VO-2 and the DSN.

(3) An attempt to measure the frame sync error rate threshold of VL relay data using analog tapes recorded at MMC (Denver) was not successful. Many reasons were responsible, including test equipment, improperly initialized data, and lack of VMCCC time.

(4) A weight and center-of-gravity measurement was accomplished following space simulation with expected results.

(5) A contamination control bioassay showed that VO-2 was within launch allowable limits.

At the conclusion of space simulator testing, VO-2 had operated a total of 708 hr. Significant problems encountered during SVT testing were as follows:

(1) The RFS SE rack lost both uplink and downlink when the RFS was connected to the LGA during the system readiness test. A coaxial connector center pin was missing, leaving the RFS TWTA 1 output driving an unterminated cable. The center pin used in this type of coaxial connector was threaded and had caused similar problems in prior programs. Based on subsequent performance of the TWTA's and previous tests run on similar units, no damage was likely to have occurred. Personnel were alerted that this type of connector failure could happen, and steps were taken to insure it would not recur.

(2) The MAWDS wavelength servo lock did not achieve lock at power turn-on. Lock failure was due to the electromechanical chopper not starting at low FA temperature. The chopper did start after a slight warm-up of the head. Modifications to the tuning fork chopper drive/pick-up coil eliminated the temperature dependence of the circuit.

(3) The XTXS output varied 1.6 dB. A coaxial cable running over the output isolator was found to have a broken shield braid, and a connector on the input isolator had lost the gold plating from its center pin. Replacement of the cable and the isolator corrected the problem.

(4) During a ranging performance test with a static Doppler offset, excessive range count was measured. The cause of the anomaly was not clear but was thought to be due to a nonstandard CTA-21 configuration which overloaded a signal line. DSN personnel were informed of the situation, and care was taken to avoid the problem in future testing.

Postenvironmental Inspection

Disassembly and inspection began shortly after VO-2 was moved back to the SAF. S/S's requiring rework were returned to their cognizant areas. All S/S's were cleaned and reassembled for final testing.

Preshipment System Test

VO-2 inspection was completed and a complete system test was performed, interrupted by two additional tests:

(1) One day was set aside for three major end-to-end tests with VO-2, CTA-21, and VMCCC. These tests were the DCT 2, VL DAPU data interface verification, and an end-to-end in orbit pass.

(2) Several special adverse-condition tests were also run in order to simulate failure modes and exercise CCS recovery routines.

The most significant problem encountered during the system test phase was the failure of the FDS to perform properly. When initial power was applied to the S/C, the FDS SE and MTC could not sync on the data out of the FDS. The problem was caused by the cleaning process used on the FDS power converter combined with the fact that the power converter transistors were not conformally coated. FDS subassemblies were inspected for sufficient conformal coating of transistors, and the FDS cleaning process was changed to use Freon TE instead of alcohol. Since the power converter failure overstressed the FDS memory, the flight units were reassigned.

The ARTCS cone actuator did not move or draw power. Two touching connector pins caused a short and blew two redundant fuses. No other damage or overstressing occurred. The bent pin was straightened, removing the short, and the fuses were replaced. The area was conformal coated and dip daubed.

The ACS SE counter indicated a change in accelerometer pulses when the ACS was in the all-axes inertial mode and a positive turn command was being executed. The anomaly was traced to a distortion in the 6.2-V output from the IRU power transformer from which the accelerometer clock frequency was derived. Two unused inverters in the clock circuitry were put to use to make the circuit insensitive to the distortions.

Power was turned off after VO-2 had operated 106 hr during the preshipment system test for a total accumulated operating time of 821 hr.

Preparation and Shipment to ETR

After the VO-2 completed the final system test and was approved for shipment, it was partially disassembled and inspected. Certain delicate, critical items were put in their own special containers, while the VO was mounted on its transporter. No major problems developed during this time, and the VO and its SE were shipped to the ETR.

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June 20, 1980

APPENDIX

ABBREVIATIONS

ACE	attitude control electronics
ACS	attitude control subsystem
AGE	aerospace ground equipment
AHSE	assembly, handling, and support equipment
A/PW	analog to pulse width
ARTCS	articulation control subsystem
ASTM	assembly and structural test model
ATM	antenna test model
ac	alternating current
accel	accelerometer
ant	antenna
assy	assembly
B/B	breadboard
BCE	bench checkout equipment
BER	bit error rate
BLDT	balloon-launched decelerator test
BLDTV	balloon-launched decelerator test vehicle
BPA	bioshield power assembly
BS	bioshield
BSC	bioshield cap
bat	battery
biol	biology
bldg	building
CACES	capsule assembly cable extender set

APPENDIX

CC	coded command
CCS	computer command subsystem
CCU	command control unit
CDU	command detector unit
C/O	checkout
CPU	central processing unit
CRS	central recorder system
CRT	cathode-ray tube
CTA	compatibility test area
CTS	central timing subsystem
calib	calibration
c.g.	center of gravity
chgr	charger
cond	conditioner
config	configuration
DAPU	data acquisition and processing unit
DC	discrete command
DCS	direct communications system
DCT	data compatibility test
DD	design and development
DSM	data storage memory
DSN	deep space network
DSS	data storage subsystem
DTM	dynamic test model
DTR	digital tape recorder
dc	direct current

demod	demodulation
depl	deployment
devel	development
dist	distribution
EMC	electromagnetic compatibility
EMI	electromagnetic interference
ETG	electrically heated thermoelectric generator
ETL	environmental test laboratory
ETM	engineering test model
ETR	Air Force Eastern Test Range
elect	electronics
eng	engine
envir	environment
equip	equipment
ext	external
FA	flight acceptance
FAT	flight acceptance test
FC	flight capsule
FDS	flight data subsystem
FOS	flight operations system
G&C	guidance and control
GCF	ground communications facility
GCMS	gas chromatograph mass spectrometer
GCSC	guidance, control, and sequencing computer
GFE	government-furnished equipment
GRE	ground reconstruction equipment

APPENDIX

HGA	high-gain antenna
H/W	hardware
htr	heater
IF	intermediate frequency
I/O	input/output
IOP	input/output processor
IRTMS	infrared thermal mapper subsystem
IRU	inertial reference unit
inst	instrument
int	internal
JPL	Jet Propulsion Laboratory
KSC	John F. Kennedy Space Center
LAPTM	lander antenna pattern test model
LaRC	Langley Research Center
LBS	lander body simulator
LC	launch complex
LCE	launch complex equipment
LCET	launch complex equipment trailer
LCS	lander camera systems
LDTM	lander dynamic test model
LFOS	launch and flight operations system
LGA	low-gain antenna
LIS	lander STE interface simulator
LO	local oscillator
LP	launch pad
LPCA	lander pyrotechnic control assembly

LSTM	lander static test model
LTS	lander thermal simulator
LV	launch vehicle
MAWDS	Mars atmospheric water detector subsystem
M/C	master control
MDA	motor drive assembly
MDS	modulation/demodulation subsystem
MMC	Martin Marietta Corporation (now Martin Marietta Aerospace)
MOI	Mars orbit insertion (time engine ignition)
MS	mass spectrometer
MSS	Mars surface simulation
MTC	mission and test computer
MTCF	mission and test computer facility
MTVS	mission and test video system
mag	magnetic
max	maximum
mech	mechanism
mem	memory
min	minimum
misc	miscellaneous
mod	modulation
ODTM	orbiter dynamic test model
OTCM	orbiter thermal control model
OTES	orbiter thermal effects simulator
op	operation
PC	printed circuit

PCA	pressurant control assembly
PCDA	power conditioning and distribution assembly
PCE	power conditioning equipment
PCM	pulse code modulation
PDA	processing and distribution assembly
PDS	power distribution subsystem
PM	propulsion module
PROPS	propulsion subsystem
PSU	pyrotechnic switching unit
PTC	proof test capsule
PTO	proof test orbiter
PV	planetary verification
PWRS	power subsystem
PYROS	pyrotechnic subsystem
pkg	package
press	pressure
prog	program
prop	propellant
pwr	power
pyro	pyrotechnic
qual	qualification
RA	radar altimeter
RAE	radar altimeter electronics
RAS	relay antenna subsystem
RCA	reaction control assembly
RCE	relay communication equipment

RCS	reaction control system
RFI	radio frequency interference
RFS	radio frequency subsystem
RIS	remote input subsystem (used in MTC)
RL	rigid lander
RPA	retarding potential analyzer
RRS	relay radio subsystem
RTG	radioisotope thermoelectric generator
RTS	relay telemetry subsystem
ref	reference
rf	radio frequency
SAF	spacecraft assembly facility
SBRA	S-band radio assembly
SC	subcarrier
S/C	spacecraft
SE	support equipment
SEC	solar energy controller
SEET	science end-to-end test
SLR	solar
SP	solar panel
SRT	system readiness test
S/S	subsystem
SSAA	surface sampler acquisition assembly
SSCA	surface sampler control assembly
SSL	space simulation laboratory
STA	system test area

STACOP	System Test and Checkout Program
STB	system test bed
STC	system test complex
STE	system test equipment
SVT	solar vacuum test
SWR	standing-wave ratio
sci	science
sep	separation
sig	signal
sim	simulation
sol	solenoid
stru	structure
sw	switch
sys	system
TA	type approval
TAT	type approval test
TCM	thermal control model
TCP	telemetry and command processor
TD	terminal descent
TDE	terminal descent engine
TDLR	terminal descent and landing radar
TDS	terminal descent system
TETM	thermal effects test model
TM	telemetry
TMU	telemetry modulation unit
TR	tape recorder

TSE	test support equipment
TWTA	traveling wave tube amplifier
temp	temperature
therm	thermal
typ	typical
UAMS	upper atmosphere mass spectrometer
UHF	ultrahigh frequency
VCMU	Viking control mock-up
VDA	valve drive amplifier
VIS	visual imaging subsystem
VL	Viking lander
VLC	Viking lander capsule
VLCA	Viking lander capsule adapter
VLMS	Viking lander mass simulator
VLS	Viking lander system
VMCCC	Viking Mission control and computing center
VO	Viking orbiter
VOS	Viking orbiter system
V S/C	Viking spacecraft
V S/C A	Viking spacecraft adapter
VSWR	voltage standing-wave ratio
VTA	Viking transition adapter
ver	verification
WSMR	White Sands Missile Range
XMTR	transmitter
XPNDR	transponder

XRFS	X-ray fluorescence spectrometer
XTXS	X-band transmitter subsystem

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1. Holmberg, Neil A.; Faust, Robert P.; and Holt, H. Milton: Viking '75 Spacecraft Design and Test Summary. Volume I - Lander Design. NASA RP-1027, 1980.

2. Holmberg, Neil A.; Faust, Robert P.; and Holt, H. Milton: Viking '75 Spacecraft Design and Test Summary. Volume II - Orbiter Design. NASA RP-1027, 1980.

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16. Abstract

This publication, in three volumes, discusses the design of the Viking lander and orbiter as well as the engineering test program that was developed to achieve confidence that the design was adequate to survive the expected mission environments and to accomplish the mission objective. Volume I includes a summary of the Viking Mission and the design of the Viking lander. Volume II consists of the design of the Viking orbiter and Volume III comprises the engineering test program for the lander and the orbiter.

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