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Protoflight Photovoltaic Power Module System-Level Tests in the Space Power Facility

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ABSTRACT

Work Package Four (WP04), which includes the NASA Lewis Research Center and its contractor Rocketdyne, has selected an approach for the Space Station Freedom (SSF) Photovoltaic (PV) Power Module flight certification that combines system-level qualification and acceptance testing in the thermal vacuum environment: the "protoflight-vehicle" approach. This approach maximizes on-the-ground verification to assure system-level performance and to minimize risk of on-orbit failures.

This paper addresses the preliminary plans for system-level thermal vacuum environmental testing of the protoflight PV Power Module in the NASA Lewis Space Power Facility (SPF). Details of the facility modifications to refurbish SPF, after 13 years of downtime, are briefly discussed. The results of an evaluation of the effectiveness of system-level environmental testing in screening out incipient part and workmanship defects and unique failure modes are discussed. Preliminary test objectives, hardware configuration, support equipment, and operations are presented.

I. INTRODUCTION

The PV Power Module can be viewed as a space vehicle that will provide the initial power capability during the baseline Space Station Freedom (SSF) assembly and as an integral part of the station Electric Power System (EPS), which is described in Section II.

In order to accomplish successful integration, assembly, and on-orbit operations of the PV Power Module, the Space Station Program (SSFP) verification requirements dictate that on-the-ground verification activities prior to launch shall be maximized to ensure system performance, especially where the risk of on-orbit failures of critical systems/subsystems is involved [1]. The history and rationale for WP04's selection of the protoflight approach to the PV Power Module flight certification to meet the SSFP requirements are described in Section III.

The importance of performing system-level environmental tests of the PV Power Module cannot be overlooked and the closest means to simulate actual space environment is the thermal vacuum test. An in-house study of the effectiveness of the system-level environmental tests was used to the system-level environmental tests was used to evaluate the technical and programmatic merits of conducting the thermal vacuum test at the NASA Lewis Space Power Facility (SPF). Unique failure modes and defects typically found during system-level thermal vacuum test of space vehicles were also reviewed and are described in Section IV.

Finally, this paper addresses WP04's preliminary plans to perform the system-level thermal vacuum test of the Protoflight PV Module in SPF in 1994, which are discussed in Section V. This section describes SPF test objectives, test configuration, test facility refurbishment status, and potential test scenarios.

II. WP04 ELECTRIC POWER SYSTEM PROGRAM CHALLENGE

WP04's primary role is to provide the end-to-end Electric Power System (EPS) architecture for the SSF, including photovoltaic and solar dynamic power generation, storage, management, and distribution to the user interface. Responsibility for the end-to-end system design and integration together with the design, development, test, engineering (DDT&E), and production of all the system hardware and software is included. The current EPS baseline delivers a net average of 75 kWe to the user (after distribution losses) at the completion of Phase I and 125 kW at the completion of Phase II, and is designed for growth to 300 kWe net average as the station usage increases. The SSF Program phases are described in [2].

The station configuration at the end of Phase I, also known as the "November 16, 1987 baseline", is shown in Fig. 1. This baseline consists of two Solar Power Modules (SPM's) located outboard of the alpha gimbal, each with two PV Modules and their respective structural integration hardware (i.e., six 5-m truss bays, transition structure, and utilities). Each PV Module consists of: (a) two solar array assemblies (b) two beta gimbal assemblies, and (c) one integrated equipment assembly (IEA), containing the energy storage assembly (ESA), the thermal control assembly (TCA), and the PV electrical equipment assembly (EEA). The primary power distribution and EPS control hardware is located inboard of the alpha gimbal and distributed throughout the station. Table I contains all the hardware outboard and inboard of the alpha gimbal and their respective primary functions [2].

The WP04's EPS Program represents a great challenge, considering the PV Power Module complexity and the SSFP driving requirements. The PV Power Module will be the largest on-orbit photovoltaic power generation application in space to-date. Each PV Power Module utilizes two 37.5 kWe (net average) solar array wings with dual flexible, deployable blankets, multiple Individual Pressure Vessel (IPV) Ni/H_2 battery cells contained in the energy storage assembly, and an active two-phase ammonia thermal control assembly with erectable heat-pipe radiator for thermal rejection of approximately 5.7 kWe. Each solar array assembly is populated with approximately 32,800 planar silicon solar cells operating at 160 Vdc. Each energy storage assembly contains 450 IPV Ni/H_2 cells that operate at pressures of 900 to 1000 psig and voltage of 120 Vdc. The SSFP requires that the batteries and solar arrays satisfy long life requirements of 5 and 15 years, respectively, while being subjected to true environmental degradation due to Low Earth Orbit (LEO) plasma, micrometeoroids, and space debris. In addition, the SSF safety requirements for two fault tolerance drive the Power Module design redundancy to maintain the power needed to support critical life functions under off-normal or contingency conditions.

III. THE PROTOFLIGHT APPROACH

The EPS Master Verification Program consists of six phases: (a) development, (b) qualification, (c) acceptance, (d) installation, assembly, and checkout (IACO), (e) prelaunch checkout, and (f) on-orbit testing. A flow diagram of the PV Power Module Master Verification Program is shown in Fig. 2. This program will consist of on-the-ground testing to verify the PV Module on-orbit configuration, performance, operation, and interfaces [3]. Test results will be used to correlate module-level analyses performed previously. As shown in the figure, the integrated PV Module test in SPF is performed in parallel with lower-level hardware qualification and acceptance testing.

The protoflight approach to the PV Module thermal vacuum test was initially proposed by the WP04 contactor as part of the Phase C/D, SSFP Fabrication Phase. This approach, which meets the requirements of MIL-STD-1540B, utilizes flight hardware which has already undergone extensive component and subsystem acceptance testing. It is generally described as "a test conducted on flight hardware at acceptance test levels for qualification durations" [5]. Testing is tailored to ensure hardware is not overstressed and that margins still exist on design life. In this approach, a combination of flight hardware and simulators can be utilized instead of a dedicated test article. The simulators replace critical hardware that could be damaged under 1-g test conditions. Post-test hardware refurbishment can be performed, if required [4].

IV. SYSTEM-LEVEL THERMAL VACUUM TEST EFFECTIVENESS

Thermal vacuum tests are generally conducted as an acceptance test on space vehicles to verify thermal control, demonstrate flight worthiness, and provide environmental stress screening of incipient defects introduced during flight hardware production. The objective of the screening is to detect material, process, and workmanship defects that respond to thermal vacuum and thermal stress conditions.

The protoflight vehicle approach will precipitate infant mortality failures caused by system interactions, thereby increasing overall system reliability. The SPF test will demonstrate that the design and manufacture of the mechanical, electrical, and thermal subsystems are adequate to survive environments slightly more severe than those anticipated during life without degradation. Also, the test will contribute to the final acceptance of the PV Module because workmanship problems that escaped lower-level screens can be corrected before final delivery to the launch site and integration with the Shuttle.

Considerable data exists to establish the effectiveness of thermal vacuum testing at the system-level and lower levels of assembly. A summary of the results of a literature review conducted to identify the unique failure modes of the thermal vacuum environment follows.

1. Reliability Concepts

The rate of failure for components and subsystems is commonly characterized by the reliability "bathtub curve", which divides failures into three categories or regions: initial failure or infant mortality, random failures, and wearout region [6]. A profile of a representative failure curve is shown in Fig. 3.

During the early lifetime of a component, a large number of initial defects exist because of weaknesses such as poor insulation, weak or incorrect parts, bad assembly, and poor fits. These are commonly referred to as infant mortality failures. During the middle operational period, fewer failures occur, but they are intermittent or unpredictable. These failures can be bad solder joints, open transformers, metal defects or moisture problems, and are called random failures. Finally, as equipment reaches old age and deteriorates, there is a region of rising failure rates known as the wearout region. Acceptance tests at any level of assembly are intended to operate the hardware through the infant mortality failure region.

Although data from the technical literature generally shows a continual drop in the observed failures from lower-to-higher levels of testing, increases in the number of failures are typically observed at the start of systems-level testing and the start of orbital operations. These increases are attributed to defects introduced because of the interactive effects between components, system integration processes, and the disruptive or transient stresses present during launch [7]. Typical profiles of failure rate curves for different levels of test and operation are shown in Figure 4.

2. Types of Failures

System-level thermal vacuum testing is considered to be highly perceptive in revealing infant mortality workmanship, design, and hardware flaws which respond uniquely to the thermal vacuum environment, or which escaped detection during lower-level testing. Electrical/electronic equipment and high-voltage subsystems are considered to be the most susceptible to failures during thermal vacuum testing.

A comparison of the types of failures attributable to acoustic, thermal vacuum, and thermal cycle environments is shown in Table II [8]. The effects unique to the vacuum environment include corona and arcing, multipacting (secondary emissions), and potential outgassing problems. Corona discharge is associated with high-voltage subsystems and connectors under vacuum pressure conditions, while multipacting can occur in vacuum cavities found in switches, coaxial cables, and connectors [9]. The instability of nonmetallic materials, particularly adhesives, in a vacuum environment can cause outgassing, contamination and deposition, weight loss, and mechanical property changes. Additional effects identified with the vacuum environment include lubrication changes (i.e., galling), differential pressure displacements which can cause delamination of printed circuit boards, and changes in conduction paths [10].

Electrical and electronic components with many piece-parts or integrated circuit boards smaller than 400 cm^2 are considered to be particularly sensitive to the thermal vacuum environment because thermally-induced expansion and contraction of piece-parts is a common cause of failure [11]. Certain failures may only be evident at temperature extremes when physical displacement resulting from different thermal coefficients of expansion is at a maximum. Vacuum-induced expansions and differential pressure displacements can cause failures in piece-parts that would not be revealed by thermal cycling tests alone [10].

In addition, the thermal vacuum environment is able to stress mountings, cabling, interface connectors, tie-downs, and other interconnectivity hardware. Problems with faulty thermal joints and other manufacturing defects are revealed when thermal conduction paths are changed. Also, since the system thermal vacuum test is the main verification test of the thermal control system, problems with heaters, insulations, and thermostats are detected by the test [12].

A Lockheed Missiles and Space Company (LMSC) study [13] based on 49 spacecraft found that the majority of thermal vacuum failures were associated with defects in parts (51 percent), workmanship (21 percent), design (15 percent), and process/control (13 percent). Out of a total of 462 systems acceptance test failures, 126 (27 percent) were detected during the thermal vacuum testing. The Aerospace Corporation compiled an extensive database on the types of defects that caused failures during systems acceptance testing [14]. A summary of the data for four programs (B, C, D, and F) with a total of 39 spacecraft is shown in Table III. The data indicates that the thermal vacuum test was particularly effective in detecting defects in parts, wiring, contamination, arcing/corona, pressure leakage, workmanship, and defective components, which were not exposed by the other systems acceptance tests. Out of the total of 76 thermal vacuum failures, the largest number of defects were for defective components (16 percent), incorrect/broken/shorted wiring (11 percent), corona arcing (6.9 percent) and contamination (6.9 percent).

A comparison was made by LMSC between identical component and system-level acceptance test failures that occurred on 19 spacecraft [15]. It was found that 44 percent of the defects were being screened out at the component level and that 56 percent were being screened out at the systems-level. A number of factors is believed to be responsible for the detection of component-level failures at the system-level rather than the component-level [16]:

(1) System interactions, including possible incompatibilities of design changes not present during lower-level testing

(2) Cumulative effects of additional testing time and thermal cycles

(3) Continuous monitoring at the system-level detects intermittents

The systems thermal vacuum test is especially important for testing the first of a series of spacecraft because electrical and mechanical design integration defects, such as cabling errors and mechanical interferences, typically occur. This is the first time that the entire integrated system, including harnesses, mounting brackets, and thermal control equipment, are exposed to the thermal vacuum environment as an entire spacecraft.

3. Test Effectiveness

A summary of the test effectiveness of systems thermal vacuum acceptance testing in screening defects for a number of multi-spacecraft programs is shown in Table IV. The data shows a great deal of scatter, with the percentage of thermal vacuum failures ranging from 3.3 to 87 percent of all the failures detected during systems acceptance testing. The average number of failures also varies widely from program to program, ranging from 0.4 to 5.8 failures per satellite for thermal vacuum tests.

An early LMSC study on the effectiveness of spacecraft acceptance te: ıg found that the most perceptive tests were the ,ırst-turn-on t‹·ts of electrical and mechanical equipment and the thermal vacuum test [16]. Based on a data set of 38 spacecraft, 99 of 307 verified failures (32 percent) were detected and corrected during thermal vacuum testing. This corresponded to an average thermal vacuum test failure rate of 2.6 failures per spacecraft. LMSC's 40 percent reduction in the number of system test failures was attributed to a number of factors, including improvements in inspection, low-level testing, parts screening, and manufacturing processes. A previously mentioned LMSC study conducted on a data base of 49 spacecraft showed a lower effectiveness (126 out of 462 failures detected, 27 percent), but recommended that consideration be given to increasing acceptance testing time.

By contrast, some test programs have shown that the acoustic or thermal cycle environments are more effective screens. For example, two Air Force projects with a total of 62 spacecraft attributed only 5 out of 152 acceptance test failures (3.3 percent) to the thermal vacuum test [17]. It was concluded that testing over the widest possible range of temperatures was a more effective screen of workmanship and design defects at the system-level. Most of the test program literature reviewed, however, showed that the thermal vacuum test is much more effective than other acceptance environmental tests, such as acoustic, thermal cycle and thermal burn-in, in screening workmanship and parts defects. Results can vary from program to program due to differences in design maturity and development risk of the spacecraft equipment tested.

The elimination of all system thermal vacuum test would significantly increase program risk. For example, LMSC conducted a failure analysis on 71 spacecraft and estimated that the number of on-orbit failures would have increased from 118 to 537 if acceptance environmental tests were eliminated. It was also estimated that the number of spacecraft expected to fail prior to the end of their orbital lifetimes would have increased from 1 to 17 [18].

V. PROTOFLIGHT PV MODULE THERMAL VACUUM TEST

1. Test Objectives

The objective of the SPF test of the PV Power Module is the performance and design validation of its fully-integrated thermal, mechanical, electrical, energy storage, and controls subsystems under realistic space environmental conditions. The test is currently planned to be conducted on only the first PV Module because a higher level of confidence is needed for the mission-critical first element launch (FEL). Subsequent PV Modules will be verified by analysis and similarity, even though consideration has been given to testing all flight PV Modules. Successful completion of the test will establish engineering confidence in the overall design, operational readiness, and safety of the PV Module.

Currently, this test is envisioned to include verification of the full and partial-power performance of the electrical and energy storage subsystems during normal, off-normal, and contigency operations, including startup, peaking, transient, and shutdown events. Functional testing will be conducted to verify the power split between batteries and the main inverter units (MIU's), battery charge/discharge algorithms, effects of peaking and rapid load changes, and system response to changes in input-output power levels. Analytical models and software algorithms for the Orbital Replacement Units (ORU's), the assemblies, and the system will be verified (i.e., solar array sun-tracking capability, battery charging and discharging schemes, fault simulation and recovery). Additional functional tests will be conducted to verify operation of the solar array mast and mast drive mechanisms, powered mating of the alpha gimbal to the PV Module, and solar array containment box latching devices. Hardware interface checkout during the PV Module assembly into the test chamber, verification of the electrical interfaces of the PV Module with the station data management system (DMS kit), and the PV controller operations during hardware build-up will also be performed.

This thermal vacuum test will verify the overall thermal performance of the baseline thermal control assembly after integration with the rest of the PV Module assemblies. Thermal interactions and heat transfer parameters will be verified to the maximum practical extent among the integrated ORU's, the radiator, condenser, coolant pump units, utility plates, controls, and interconnect plumbing (i.e., cables and connectors). Special attention will be given to the subsystem's ability to ensure that the required ORU-temperatures on the IEA are not exceeded during normal and off-normal operations and that the battery baseplate temperatures are maintained within a specified range under normal and faulted operations.

As discussed in Section IV, system-level thermal vacuum tests are intended to expose hidden design and/or workmanship flaws which were not detected during lower-level environmental testing. The effectiveness of the vendor-piece-part environmental stress screening can be evaluated. The SPF test will also stress mountings, cabling, and connectors, which are subject to attachment errors, degrading tolerance stack-ups, and incompatible thermal expansion coefficients.

2. Test Hardware and Configuration

The protoflight PV Module thermal vacuum test will be accomplished with a combination of protoflight hardware and simulators to minimize the risk of flight hardware damage under 1-g test conditions. The test hardware and test configuration reflect the November 16, 1987 baseline, as described in Section II. The test hardware and support equipment requirements are listed in Tables VI and VII.

Current plans are to use flight hardware assemblies for the beta gimbal, the integrated equipment assembly structure with the complete energy storage, electrical equipment, the thermal control assemblies.

The PV Module critical dimensions and preliminary test configuration in SPF are shown in Figs. 8 and 9, respectively. All the energy storage and electrical Power Management And Distribution (PMAD)-source equipment will be installed in replaceable ORU boxes which are mounted on standard utility plates. Eight utility plates, which support four ORU's each, will be attached to the IEA structure. The baseline thermal control radiator will be of flight-quality, and will be structurally supported at the condenser/IEA interface by a test support fixture. Electrical and fluid interconnects, cables, cable trays, and connectors will be of flight-quality. Only one set of beta gimbal and solar array simulator is needed for this test. Both sets of solar array and beta gimbal are identical and the other set can be verified by similarity as a cost effective approach [19]. The thermal control assembly performance will be verified by orienting the radiators horizontally so that no significant pressure head develops. Elevation changes within an ORU box utility plate heat pipes will also be avoided.

Electrical and/or mechanical simulators will be used for the solar arrays, the alpha gimbal, station truss structure, and the tie-down support mounts and brackets to avoid damage under 1-g test conditions. The solar array simulator will resemble a stowed solar array with the containment box partially opened. This mechanical simulator will duplicate the solar array/beta gimbal dynamic interface. The containment box masses and torsion tube simulate the array torsional response. The strong back and cables offload the array weight from the gimbal interface. The solar array electrical simulator, located in the support equipment trailer outside the test chamber, will be connected to the hardware via a test chamber feed through. The alpha gimbal simulator is expected to have a station configuration roll ring assembly for transmission of power and data to the PV Module during testing. It is not considered essential to use the alpha gimbal bearing, housing, or drive mechanism to accomplish the test objectives [19]. Truss-bay simulators representing two-and-a half bays of station truss will provide structural attachment points for the IEA and beta gimbal mountings.

The structural support for all the assemblies will be provided by a large test fixture or assembly work platform. The assembly work platform will be mounted on a series of vacuum-compatible rail tracks that will use the existing rails to move the test configuration from the assembly area to the test chamber. The present configuration is oriented to minimize cable runs to the two support equipment trailers that will be located outside the test chamber. These trailers will contain dc and ac power sources, electrical simulators, loads' bank, and a control console with the data acquisition system. These support equipment trailers will also be used at different locations for the PMAD development tests, IACO tests, and others. A Layout of the support equipment trailers is shown in Fig. 10.

3. Space Power Facility Description

The Space Power Facility (SPF), which is managed by NASA Lewis, is the world's largest space environmental simulation chamber. It was selected because it is the only facility large enough to accommodate the PV Module test article and its support structure in as-close-to-flight configuration as possible [20].

SPF was built during the late 1960's in the NASA Lewis Plumbrook Station in Sandusky, Ohio (Fig. 5). The facility was designed to accommodate any type of electric power generating system up to 15 MW, electric and chemical propulsion systems, and nuclear power generation systems. The facility was operational in 1969 and a series of tests were successfully performed before being placed in standby mode in 1975. During these 6 years, SPF was used to conduct operational tests of a 9 kW closed Brayton cycle space power system, separation tests of Skylab and Titan/Centaur Shrouds, development base heating tests of an Orbiter scale-model, flight qualification of space experiment hardware, and cloud physics research [21]. From 1979 to 1985, NASA Lewis leased the facility to Garrett Corporation for manufacturing of gas centrifuges under the Department of Energy (DOE) contract. During Garrett's occupancy, extensive modifications were made to the test chamber and its associated equipment.

During the past two years, the facility was restored to its original condition, under a DOE/NASA interagency agreement. Modifications were made to the test chamber, vacuum system, and the assembly and disassembly areas. A number of structural and mechanical subsystems underwent cleaning, repair, and/or reinstallation. Pumpdown tests performed in late 1988 achieved vacuum levels of 10^{-5} torr, and minor chamber leaks were discovered. NASA presently plans to eliminate the test chamber leaks to improve the chamber vacuum level of 10^{-7} torr. In addition, NASA will evaluate and correct failure modes that produce backstreaming and establish cleanliness level of the test chamber and disassembly area. Future plans also include complete rehabilitation and reactivation of the liquid nitrogen (LN_2) system in FY 91 to provide adequate thermal environment for the PV Module. The capabilities and repair status of the test chamber and its major subsystems are summarized in Table V [22].

SPF consists of three main areas: (a) test chamber, (b) assembly and shop area, and (c) disassembly area. A plan view of SPF and a cross section of the test chamber are shown in Fig. 6 and 7. Three sets of standard gage railroad tracks run through the main areas.

The SPF test chamber consists of a vacuum-tight, hemisphere-shaped aluminum vessel of 100-ft-diameter by 122-ft-maximum height. It is surrounded by a vacuum-tight heavy concrete enclosure for nuclear shielding and internal pressure containment of up to 8 psig. The structure is also designed to withstand atmospheric pressure externally with 25 mm Hg between the test chamber and the concrete enclosure. The chamber is constructed of type 5083 aluminum, which is clad on the interior surface with 1/8-in.-thick type 3033 aluminum for corrosion control. The aluminum chamber vessel is designed for 2.5 psig external pressure and internal pressure of 5.0 psig. It is capable of environmental testing in a vacuum between 10^{-5} to 10^{-7} torr. The chamber floor is flat and heavily braced to withstand a total load of 250 lb/ft^2 (200 tons). The chamber is equipped with a 20-ton remotely-controlled polar crane. It also provides for a test article envelope of 90 ft-diameter by 100-ft-height [21]. Other chamber features include: an 8 by 8-ft airlock and a full complement of welded penetrations for electrical and control wiring, cooling water, and the vacuum system. Two concrete doors with 50 by 50-ft openings provide access to the assembly and disassembly areas. Double-door seals are used to prevent chamber leakage.

The assembly and shop areas are located adjacent to the test chamber and their sizes are 75(width) by 150(length) by 80(height)-ft and 50(width) by 150(length) by 40(height)-ft, respectively. Both areas are equipped with overhead-bridge cranes (25- and 10-ton, respectively).

The disassembly area is also located adjacent to the test chamber, opposite to the assembly area and its size is 70(width) by 150(length) by 76(height)-ft. It contains a remotely-controlled overhead bridge crane and can be used for disassembly and packaging-for-shipment of the test article. This area's internal surface is epoxy-coated, with extensive ventilation and contamination control.

The vacuum system consists of 32 high-vacuum oil diffusion pumps that are 48-in.-diameter, liquid nitrogen-baffled, electrically heated, and each with a capacity of 43,000 liter/sec. These diffusion pumps are mounted in the chamber floor. Two 5-stage roughing trains of mechanical pumps provide additional pumping capacity. The test chamber can be evacuated from atmospheric pressure to 10^{-6} torr in approximately 18 hr. The diffusion pumps can not normally be activated until a pressure of 10^{-3} torr has been reached [22].

The facility has a 40-ft-diameter by 40-ft-high cold wall in storage that is capable of cryogenic temperatures down to -300 °F [22]. It is divided into four quadrants, each with its own GN_2 supply and return lines (22 cooling zones available) which penetrate the aluminum floor. The LN_2 is supplied by two on-site storage vessels of 217,000 and 28,000-gal capacity. NASA is presently considering a larger cold wall design, which utilizes the existing design, to support the PV Module tests.

The facility has also in storage a 7-MW quartz heater which consists of 1-kW tungsten lamps currently configured in a 16-ft-diameter by 57-ft high heater arrangement. In addition, a 400 kW (1° 19' collimation) arc lamp with mirrors is available for solar simulation.

SPF has a facility control room (828 ft^2) where all the facility operation panels (controls for vacuum and cryogenic systems) reside and a test control room (2526 ft^2). The instrumentation system provides about 900 hard lines from the chamber to the instrument room in the basement of the test control room. The hard lines consist of thermocouple, bridge-type, pot-type, signal monitor, high-frequency, high-power, and low-power circuits.

Other SPF utilities include: potable and demineralized water systems, sewage-treatment system, cooling tower, and compressed-air system.

4. Test Scenarios

According to the MIL-STD-1540B guidelines for thermal vacuum testing of spacecraft flight systems, testing will consist of at least four complete hot and cold cycles at the maximum predicted orbital rate of temperature change, with at least an 8 hr thermal soak at each temperature extreme of the cycle. The chamber pressure shall be maintained at 1×10^{-4} torr or less [23]. The MIL-STD-1540B environmental test requirements for system-level thermal vacuum testing are summarized in Table VIII.

During chamber pumpdown, selective components may be monitored for corona and multipacting, or secondary emissions. An example where these types of phenomena could be expected to occur is in the case of the batteries, which will be launched fully charged in the Orbiter [23]. The rate and quantity of outgassing may also be monitored to assure that the PV Module and test equipment does not degrade system-level performance. The SPF chamber temperature will be controlled so that during the hot or cold temperature extremes, at least one ORU on the IEA will be at its protoflight design temperature [24]. The exact protoflight temperature levels are still subject to discussion, but could be as much as 11 °C greater (less) than the maximum (minimum) predicted on-orbit temperature extremes [19]. The temperatures for critical components will be recorded in real-time to prevent overtesting. During the thermal vacuum cycles, the batteries are expected to undergo limited cycling at temperature ranges not to exceed -5 to +25 °C. The Depth-of-Discharge (DOD) will range up to 35 percent for most cycles, but several low C-rate 80 percent DOD contingency tests may be performed [19].

Potential test scenarios are still under discussion, but are expected to fall within the following areas [24]:

(a) Non-operational hot and cold soak tests to simulate unpowered storage modes. Hardware would be subjected to the expected temperature extremes with functional tests performed before and after exposure to determine effect on performance.

(b) Operational hot and cold soak tests where the hardware would be subjected to temperature extremes while powered on. Operational status would allow electrical intermittents, such as relay chatter and thermally-induced shorts, to be detected.

(c) On-orbit configuration tests performed at nominal environments to verify functional performance. Details are described previously.

(d) Hot and cold start-up tests where the hardware is soaked at a temperature extreme until stabilized, and start-up performance is then verified.

The environmental tests will be preceded by functional tests conducted at ambient pressure and temperature conditions. The test will be designed to determine the baseline functional parameters of all the electrical, energy storage, thermal, and mechanical subsystems and to determine performance deficiencies prior to introduction of environmental stresses on the PV Module [23].

During at least one temperature cycle, MIL-STD-1540B requires that thermal equilibrium be established at both hot and cold temperature extremes to allow verification of performance of thermostats, heat pipes, electric heaters, and the control authority of the active thermal system [23].

Test durations shall be sufficient to test all orbital operational conditions and all equipment functional modes including redundancy [23]. Redundancy checking will be performed under operational conditions so that the number of life cycle/load exposures by each redundant circuit in the thermal control and electrical subsystems are approximately equal [24].

Currently, the SPF testing window is scheduled for 6 months prior to hardware refurbishment as needed and delivery to the contractor plant for the PV installation, assembly, and checkout (IACO). Hardware refurbishment is essentially a "health determination" and performance rebaselining procedure. Battery refurbishment will consist of: inspection and insulation test, touch-up and repair as needed, two standard capacity and voltage test cycles, self-discharge test, and final inspection and insulation test [19]. Final PV Module delivery to the launch site will occur three months prior to the first element launch (FEL) in FY 1995.

VI. CONCLUDING REMARKS

WP04's selected approach to the protoflight PV Power Module Thermal Vacuum Test will provide engineering confidence in the capability of PV Power Module to perform as an integrated system which can be assembled and verified in a ground test facility. Programmatically, this approach meets cost and schedule constraints. The history of the system-level environmental tests for space vehicles demonstrates the value of this test in lowering the risk of on-orbit failures. Problems detected during system-level thermal vacuum testing, such as corona/arcing, multipacting, electrical shorting, outgassing, and delamination, are generally significant in nature. These problems should have been discovered at lower testing levels, and if not corrected, could be considered potential on-orbit failures.

The NASA Lewis Space Power Facility is the only facility large enough to accommodate the baseline PV Power Module in its most representative flight configuration. The NASA/DOE effort to restore SPF to its original condition is complete. Current and near term plans will support complete facility readiness in FY 1994.

The information obtained from this thermal vacuum test will be essential to establish confidence in the operational readiness and safety of the PV Power Module. The data will help verify analysis modeling tools and assumptions made earlier during the development phase, help debug operational contingency procedures for on-orbit failure scenarios, and assist during training of astronauts and operators. It is expected that a first-of-its-kind database will be established for use in future on-orbit anomaly investigations, future hardware development, and systems performance modeling of later-assembly configurations.

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TABLE I. - SOLAR POWER MODULE HARDWARE AND PRIMARY FUNCTION (NOVEMBER 16, 1987 BASELINE)

(a) Solar Power Module (SPM) Hardware

Hardware name	Quantity	Primary function
I. Photovoltaic module	2	Provide 37.5 kWe net average power
A. Inboard	1	Provide 18.75 kWe net average power
B. Outboard	1	Provide 18.75 kWe Net average power
1. Solar array assembly	a2	Generate high-voltage dc power during daylight portion of the orbit to supply the ac power demand to recharge the batteries
2. Sequential Shunt Unit (SSU)	a2	Control and regulate array voltage to approximately 160 Vdc
3. Beta gimbal assembly	a2	Provide beta pointing and tracking for the solar arrays in response to commands from the Photovoltaic Controller (PVC)
4. Integrated Equipment Assembly (IEA)	a1	Structural mounting for the ESA, the EEA, and the TCA
a. Energy Storage Assembly (ESA)		15 battery assemblies that operate with five BCDU's to meet station power requirments during eclipse, start-up ops, and loss of local or station-wide power
b. Electrical Equipment Assembly (EEA)		Condition and control station electrical power
1. dc Switching Unit (DCSU)	a2	Switch power from the SSU to MIU during sunlight; switch power from the battery BCDU to the MIU during elcipse eclipse
2. Battery Charge/Discharge Unit (BCDU)	a5	Condition battery charge and discharge power based on commands from the PVC
3. Main Inverter Unit (MIU)	a2	Convert dc regulated source power to 440 Vac, 20 kHz, 1-phase power
4. Main Bus Switching Unit (MBSU)	a2	Transfer 440 Vac, 20-kHz, 1-phase power to a PDCA, in the inboard PV Module
5. Power Distribution and Control Unit (PVCU)	a2	Reduce the voltage to 208 Vac and provide ac control power to the SSU, PV pump unit, BCDU, MIU, and DCSU; Provide ac power to the PVC for beta gimbal motor power and PV array deployment power
6. Photovoltaic controller (PVC)	a2	Provide communications between the PMC and the PV module functional controllers
c. Thermal Control Assembly (TCA)	a1	Actively cools the ESA and EEA to a dedicated radiator for rejection to space
II. SPM integration hardware		Integrate the major PV Module assemblies with each other and the rest of the station
A. Truss structure		Six 5-m truss bays to provide the primary SPM structure on each side of the alpha gimbal
B. PV equipment strut set		Provide the transition structure to mount the IEA's to the truss structure
C. Beta gimbal transition structure		Provide the structure to mount the beta gimbals to the truss structure and to support the solar array assemblies
D. Utility trays		Provide the electrical and data interface between the PV module components and the station interfaces
E. EVA translation rails		Allow translation of EVA crewperson along the integrated truss assembly
F. Orientation lights		Provide the necessary lighting for adequate SPM viewing by EVA crewpersons or cameras

(b) Distributed electric power system hardware

Hardware name	Primary function
I. Alpha gimbal	Provide transfer, via roll rings, of 440 Vac power from the outboard MBSA (in SPM) to the inboard MBSA (on the integrated truss assembly-ITA) and dc control power the battery BCDU to the EPS ORU's inboard of the alpha gimbal; allow bidirectional data transfer between the PVC and the PMC
II. Inboard Main Bus Switching Assembly (MBSA)	Place distribution power (440 Vac) onto the station feeder network (upper and lower 25-kW ring feeders for each resource node)
III. Truss feeders	Follow the station truss struture (providing 440 Vac to truss-mounted PDCU's), which reduce voltage from 440 to to 208 Vac (utility grade power)
IV. Power distribution and control assembly-truss (PCDA-truss)	Provide utility-grade power to the pallet user interface and truss payloads
V. Transformer units (TU's)	Provide single-point grounding and reduce voltage from 440 to 208 Vac
VI. Node switching unit	Receive 208 Vac from the (NSU) TU and place it onto a feeder network
VII. PDCA Modules	Provide utility grade power to the rack user interface in the resource nodes and lab and hab modules
VIII. NSTS power converter unit (NPCU)	Each NPCU receives utility-grade power from the module mounted PDCU's and transfers NSTS-to-station EPS power and vice versa
IX. Power management controller (PMC)	Each PMC receives 208 Vac power from the module-mounted PDCA. PMC manages, coordinates, and controls the distributed EPS hardware

[a]Quantity per module.

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TABLE II. - ENVIRONMENTAL TEST OBJECTIVES (Ref. 8)

Failure conditions	Environmental tests		
	Acoustic	Thermal vaccum	Thermal cycling
Launch environment performance	X	---	---
Component vibration response	X	---	---
Electrical intermittence	X	X	X
Orbital environment performance	---	X	---
Thermal control	---	X	---
Corona/arcing	---	X	---
Multipacting	---	X	---
Material outgassing	---	X	---
Latent defect/ failure propagation	X	X	X
Thermal stress effects	---	X	X
Integration hardware verification	X	X	X

TABLE III. - NUMBER OF FAILURES DETECTED DURING SPACECRAFT SYSTEMS ACCEPTANCE TESTING

	Program B 16 tests		Program C 5 tests		Program D 13 tests		Program F 5 tests		Combined programs net of 30 tests		
	Total	Thermal vacuum	Total	Thermal vacuum	Total	Thermal vacuum	Total	Thermal vacuum	Total	Thermal vacuum	Percent of total
Mounting broken/loosened	04	01	01	--	03	--	03	--	11	01	9.1
Part broken/defective/shorted	22	08	11	11	32	10	08	02	73	31	42.5
Wire broken/shorted/incorrect	18	06	02	02	16	--	01	--	37	08	21.6
Solder joint cold/broken	03	--	--	--	05	01	01	--	09	01	11.1
Contamination short/other	05	02	--	--	07	03	01	--	13	05	38.5
Adjustment changed	03	--	01	--	16	03	--	--	20	03	15.0
Connector shorted/open	01	--	03	--	22	01	--	--	26	01	3.9
Arcing/corona	01	01	--	--	04	04	--	--	05	05	100.0
Pressure leakage	03	01	--	--	--	--	02	--	05	01	20.0
Component defective	01	--	07	07	11	05	07	--	26	12	46.2
Workmanship	08	02	02	02	04	--	--	--	14	04	28.6
Design	07	01	02	02	08	--	02	--	19	03	15.8
Unknown	01	01	--	--	03	--	--	--	04	01	25.0
Total	77	23	29	24	131	27	25	02	262	76	34.4

TABLE IV. - SUMMARY OF THERMAL VACUUM ACCEPTANCE TEST EFFECTIVENESS FOR VARIOUS SATELLITE PROGRAMS

Company	Period	Number of satellites	System acceptance failures			Average failures per test			Author	Ref.
			Total	Thermal vacuum	Percent	Acoustic	Thermal cycling	Thermal vacuum		
Lockheed	1970-1974	46	508	113	22	1.29	----	2.46	Brown (1975)	15
Lockheed	1970-1977	38	307	99	32	0.96	----	2.28	Brown (1978)	16
TRW	---------	6	152	5	3	1.33	5	0.83	Krausz (1980)	17
Lockheed	1970-1980	49	462	126	27	1.94	----	2.57	Smith (1981)	13
Aerospace	---------	37	537	300	56	1.1	----	2.0	Laube (1982)	25
Boeing	---------	3	49	11	23	0.63	----	3.67	McDaniel (1984)	26
Aerospace	---------	60	231	116	50	0.71	2.79	1.93	Hamberg (1988)	08
Aerospace	1970-1974	16	77	23	30	0.63	----	1.43	Laube (1983)	14
Aerospace	1977-1979	5	29	24	83	1	----	4.8	Laube (1983)	14
Aerospace	1970-1981	13	131	27	21	0.77	1.46	2.08	Laube (1983)	14
Aerospace	1970-1981	5	25	2	8	1	1.2	0.4	Laube (1983)	14
Rockwell	---------	5	76	29	38	0.4	----	5.8	Wasserman (1981)	27

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TABLE V. - SPF CAPABILITIES AND REPAIR STATUS

System	Capability	Status
Vacuum	10^{-5} torr	Expect 1975 levels after operation to 10^{-7} torr
Vacuum Pump System	32 of 48 in. diffusion pumps LN_2 baffled Roughing train 5 stages 2 trains	Operational
Test Instrumentation	No data acquisition No data reduction Hard lines check out	930 pairs of hard lines from chamber to instrument basement
Facility Power	34.5 kV redundant source	Operational
Test Chamber Power	7 MW	Available
Facility Cryogenic	LN_2 Storage: 217,000 gal 28,000 gal	22 supply and return zones available. 217,000 gal dewar not refurbished
Cold Wall	None installed in chamber	40 ft diameter by 40 ft high 4 quadrant cold wall in storage
Cold Wall Cryogenic System	4 of 22 zones installed to test chamber floor	Compressors, feed systems not checked out
Quartz Lamp Heaters Housing	7 MW quartz lamp heater	1 kW lamps installed in 2 of 8 ft radius by 57 ft high
Solar Simulation	400 kW - 1° 19' collimation Arc lamp with mirror	Lamp and mirrors in storage
Utilities Water	Domestic and fire protection 15 000 gal storage	Operational
Cooling Tower	8000 GPM 120 GPM makeup	Operational
Natural Gas	4 in. - 20 psig feed 850 SCFM	Operational
Service Air	650 CFM compressor with 170 CFM backup	Operational
Test Chamber	100 ft diameter by 122 ft high Aluminum chamber inside 130 ft diameter by 132 ft high concrete chamber 50 ft by 50 ft doors	Operational 20 ton polar crane operational
Disassembly Area	70 ft wide by 150 ft long 76 ft high Concrete structure epoxy coated 20 ton crane	Restored
Assembly Area	75 ft wide by 150 ft long, 88 ft wide Standard superstructure 20 ton crane	Restored
Shop Area	40 ft long by 150 ft long, 34 ft high 10 ton crane	Restored Equipped with light machine shop power equipment and tools
Cold Wall Cryogenic System	4 of 22 zones installed to test chamber floor	Compressors, feed system not checked out
Quartz Lamp	7 MW quartz lamp heater	1 kW lamps installed in 28 ft radius by 57 ft high housing in storage

TABLE VI. - PV MODULE TEST HARDWARE IN SPF

Beta gimbal assembly
Solar array assembly simulator
Integrated equipment assembly
  Energy storage subsystem
  Electrical equipment and controls subsystem
  Thermal control subsystem
Integration hardware
  Interconnecting harnesses, cables, and cable trays
  Tie-down and supporting structure mounts and brackets (simulators)
Alpha-joint simulator
Truss structure simulator

TABLE VII. - PV MODULE TEST SUPPORT EQUIPMENT IN SPF

PV GSE	
Battery simulator Solar array simulator PV IEA handling set and dolly Beta gimbal handling set	Radiator handling set (GFP) PV protective covers set Radiator supply cart
PMAD GSE	
ac power source dc power source Control power simulator dc load bank Programmable ac load	Roll ring simulator Motor simulator PMC simulator DAC/control console PMAD equipment trailers (2) Standard package ORU handling set
Test Support Equipment	
Thermal conditioning set (cold walls) Cable set Instrumentation multiplexer Instrumentation set	Structural support set Assembly platform Solar array mechanical simulator Minor test equipment (racks, power supplies, tape recordes, etc.)

TABLE VIII. - MIL-STD-1540B ENVIRONMENTAL TEST REQUIREMENTS FOR SYSTEM-LEVEL THERMAL VACUUM TESTING (Ref. 23)

Thermal vacuum test	Qualification	Acceptance
	Functional performance 10^{-4} torr or less 10 °C beyond maximum predicted extremes Control component temperature to achieve margin where feasible 8 Cycles - 8 hr at extremes Functionals on first and last cycles Electrically operating/monitored	Functional performance 10^{-4} torr or less Maximum predicted extremes Control component temperature to achieve maximum/minimum predictions 4 cycles - 8 hr at extremes Functionals on first and last cycles Electrically operating/monitored

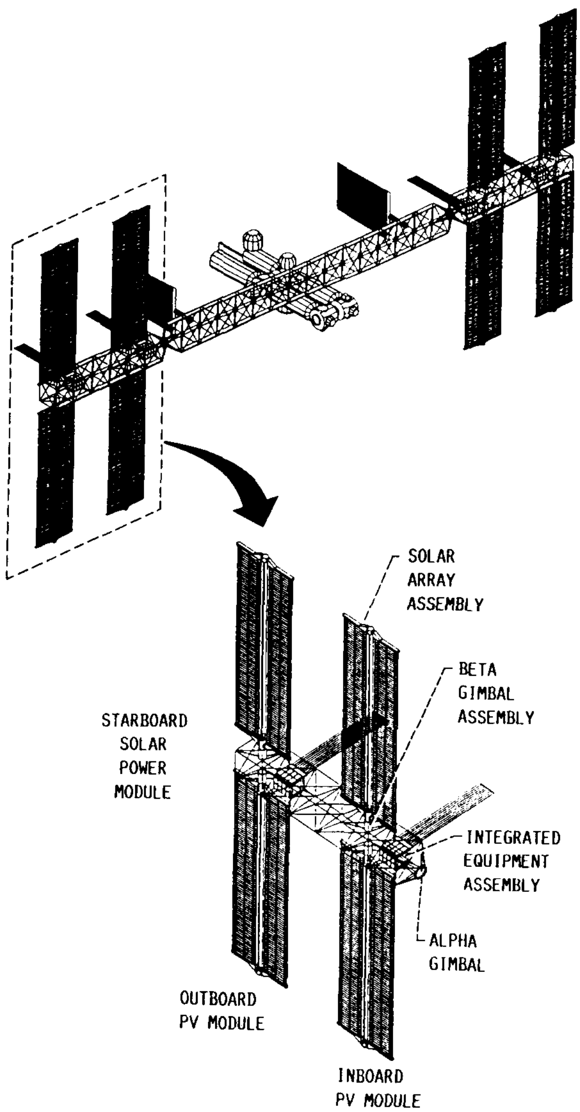

FIGURE 1. - BASELINE SPACE STATION FREEDOM CONFIGURATION AT THE END OF PHASE I (NOVEMBER 16, 1987 BASELINE)

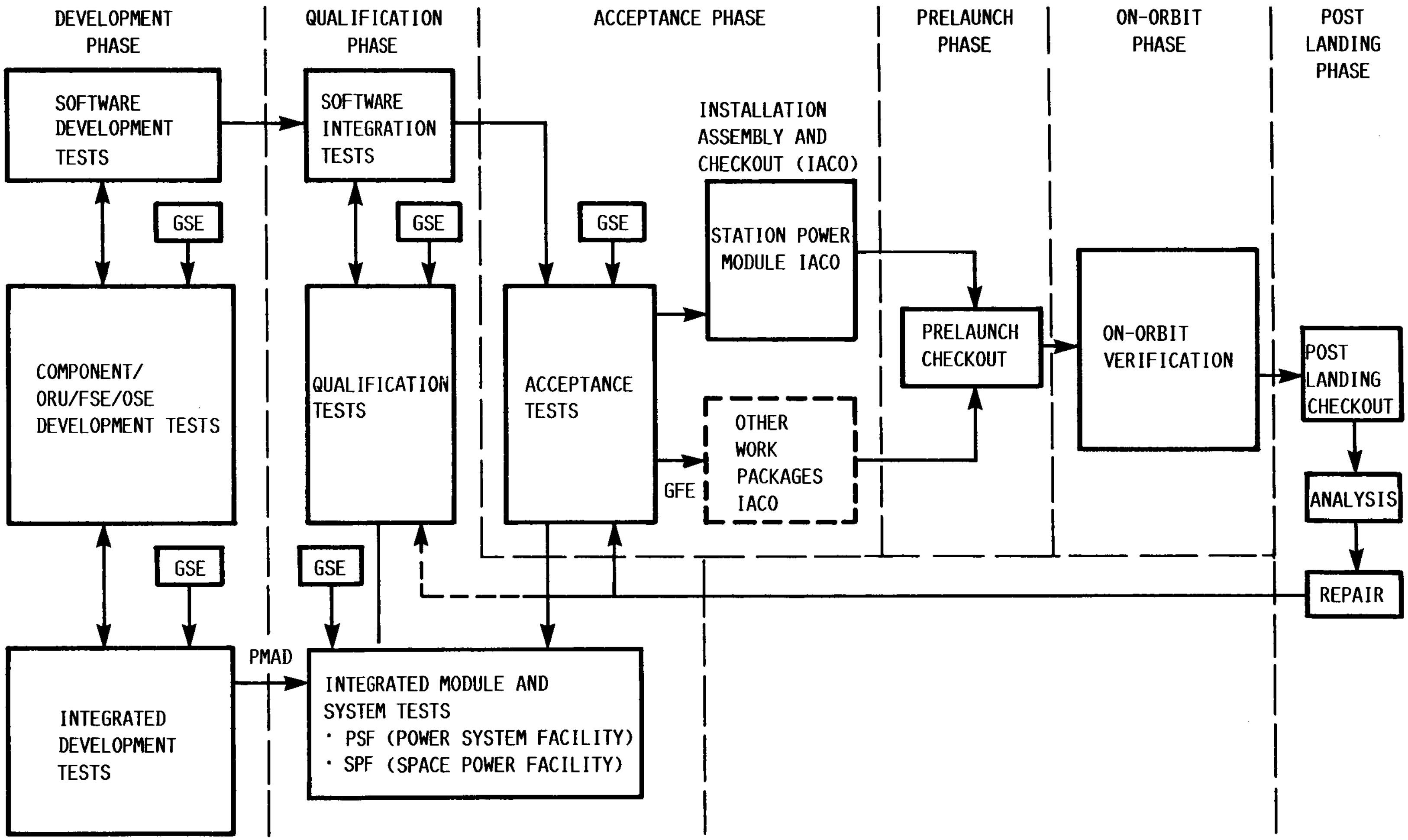

FIGURE 2. - PV POWER MODULE MASTER VERIFICATION PROGRAM FLOW DIAGRAM.

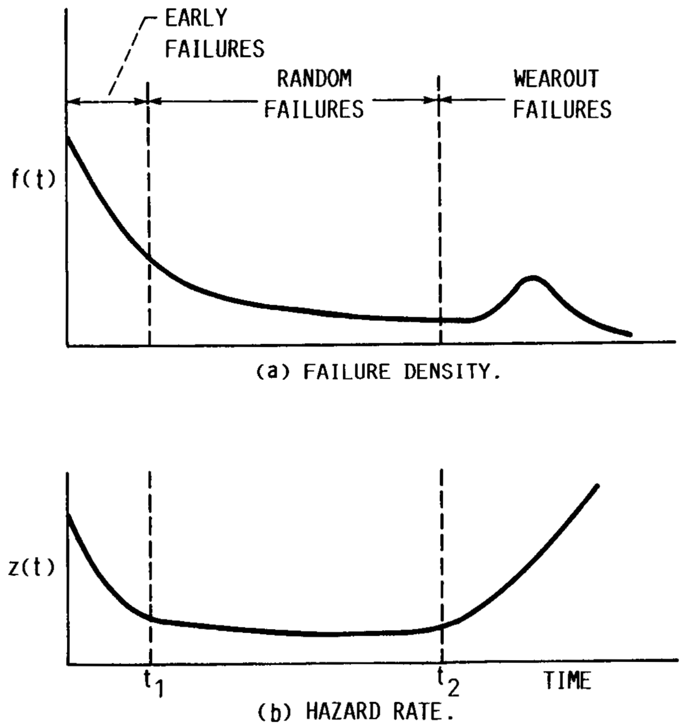

FIGURE 3. - REPRESENTATIVE FAILURE CURVE (REF. 6).

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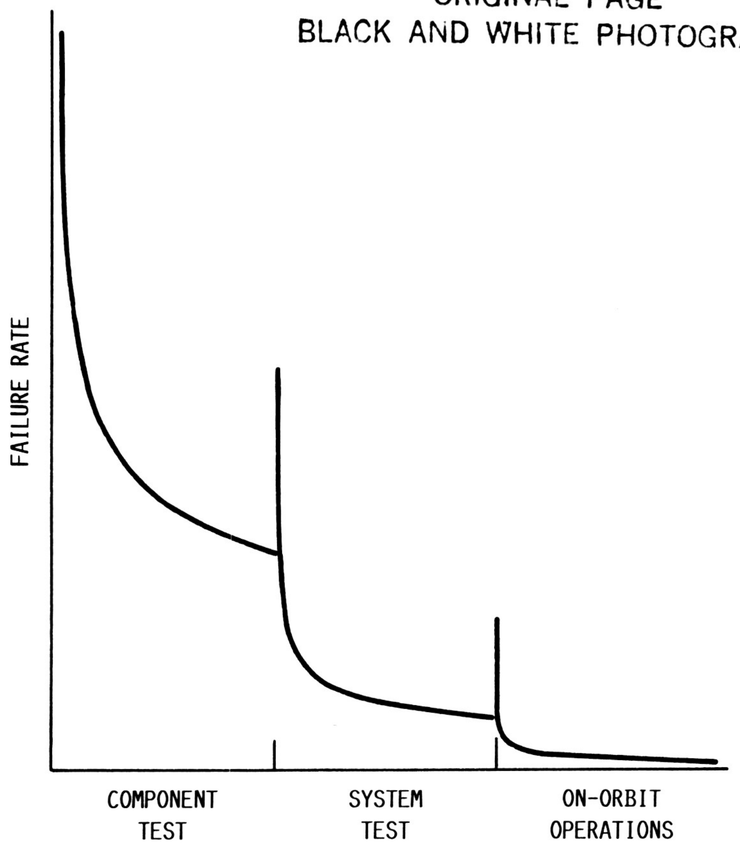

FIGURE 4. - TYPICAL FAILURE RATE PROFILES FOR DIFFERENT TEST LEVELS AND PHASES (REF. 7).

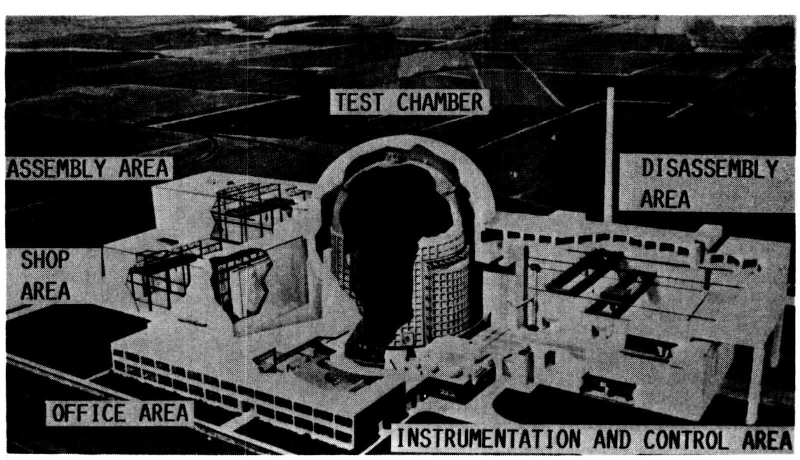

FIGURE 5. - SPF CUTAWAY VIEW.

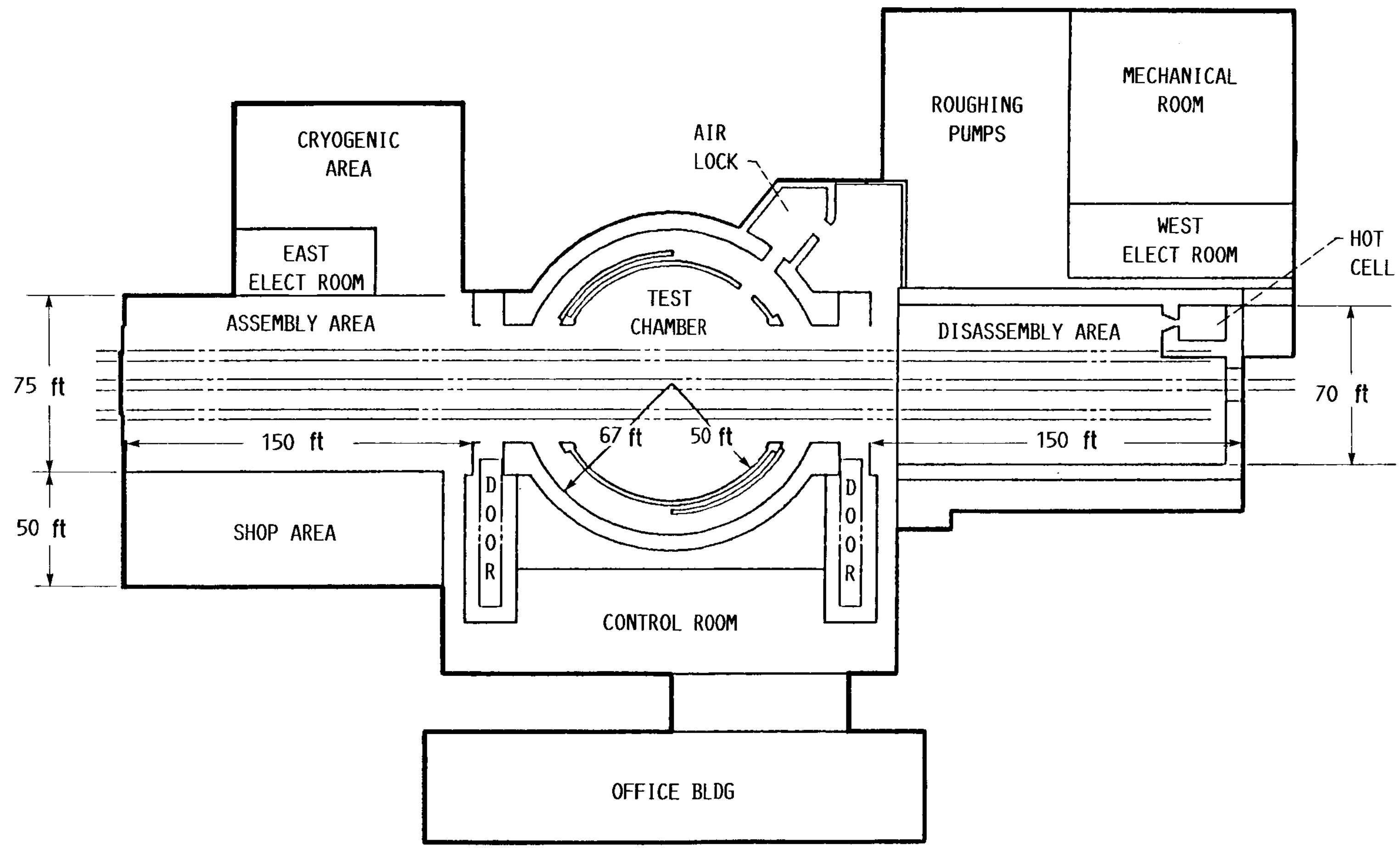

FIGURE 6. - SPF PLAN VIEW.

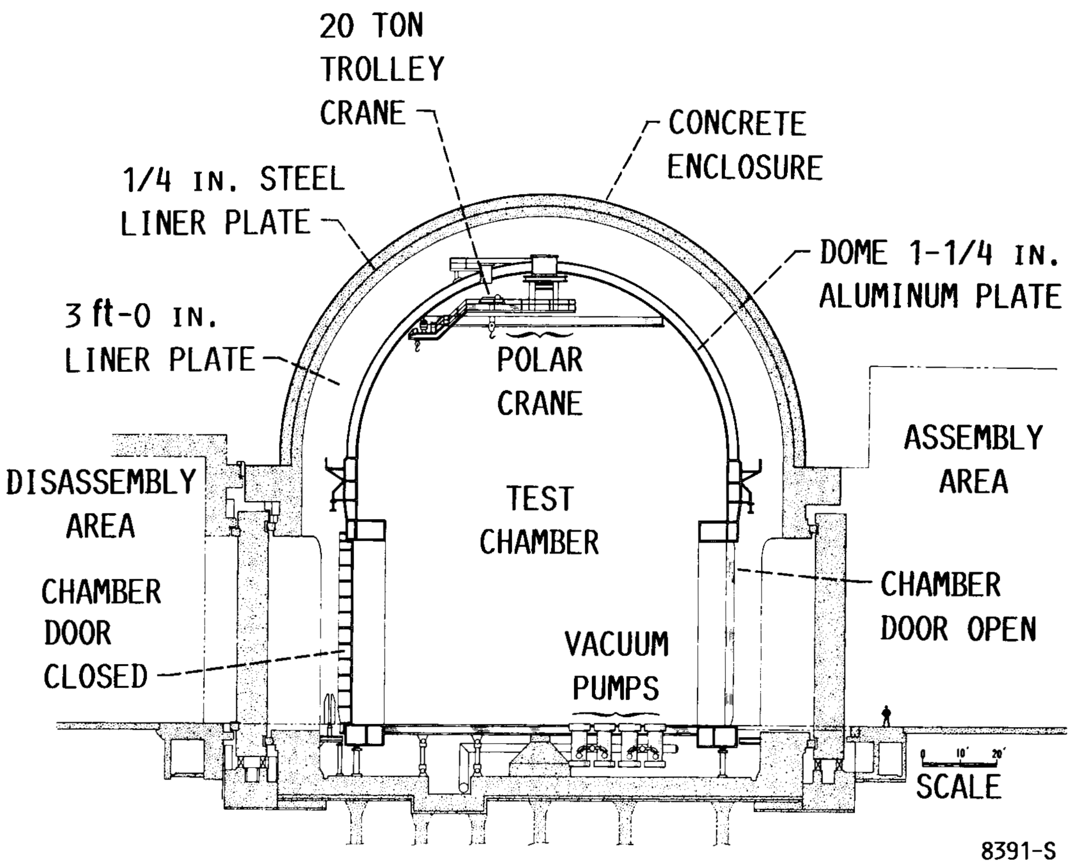

FIGURE 7. - SPF TEST CHAMBER CROSS-SECTIONAL VIEW.

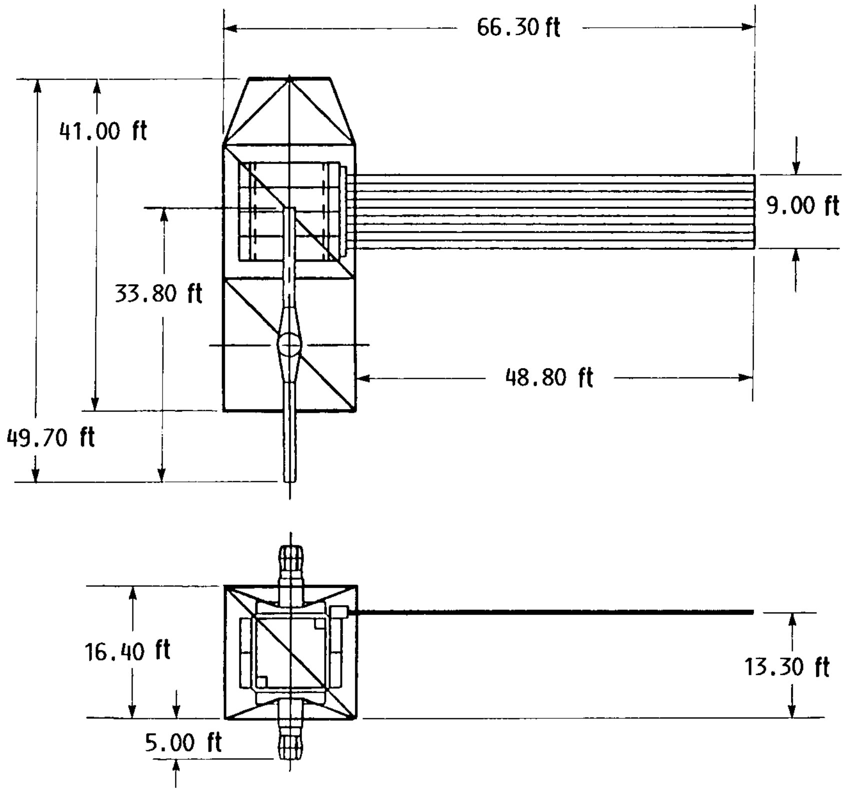

FIGURE 8. - PV MODULE CRITICAL DIMENSIONS (NOVEMBER 16, 1987 BASELINE).

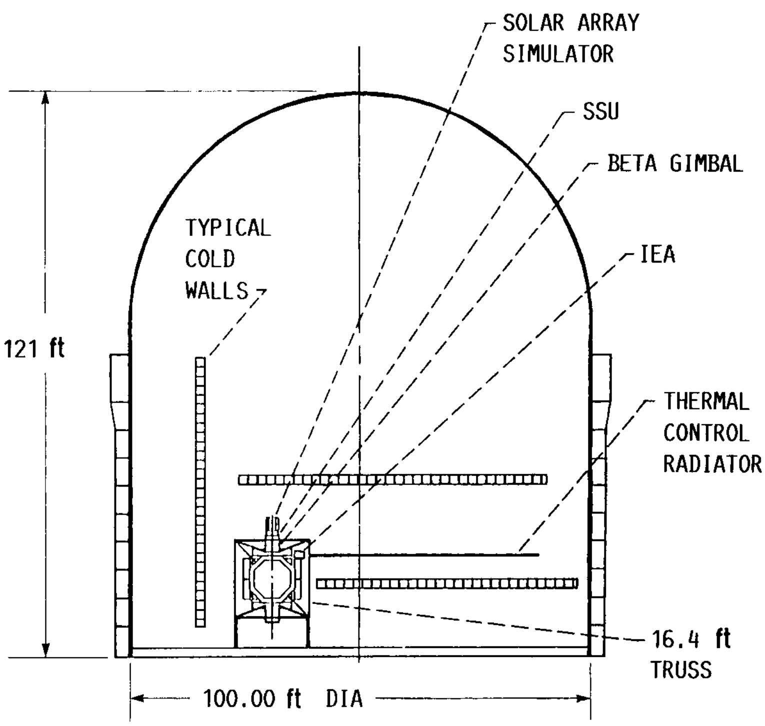

FIGURE 9. - PV MODULE PRELIMINARY TEST CONFIGURATON IN SPF.

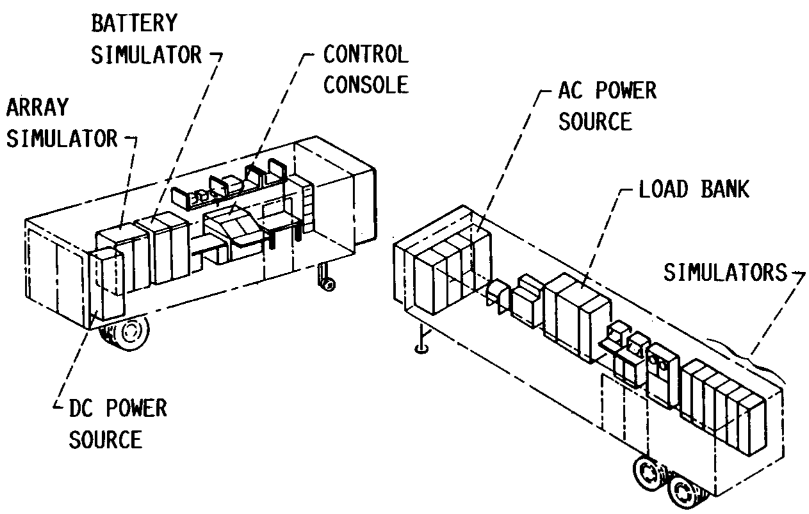

FIGURE 10. - PV MODULE TEST SUPPORT EQUIPMENT TRAILERS.

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16. Abstract

Work Package Four (WP04), which includes the NASA Lewis Research Center and its contractor Rocketdyne, has selected an approach for the Space Station Freedom Photovoltaic (PV) Power Module flight certification that combines system-level qualification and acceptance testing in the thermal vacuum environment: The "protoflight-vehicle" approach. This approach maximizes on-the-ground verification to assure system-level performance and to minimize risk of on-orbit failures. This paper addresses the preliminary plans for system-level thermal vacuum environmental testing of the protoflight PV Power Module in the NASA Lewis Space Power Facility (SPF). Details of the facility modifications to refurbish SPF, after 13 years of downtime, are briefly discussed. The results of an evaluation of the effectiveness of system-level environmental testing in screening out incipient part and workmanship defects and unique failure modes are discussed. Preliminary test objectives, test hardware configurations, test support equipment, and operations are presented.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
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