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APOLLO EXPERIENCE REPORT - CERTIFICATION TEST PROGRAM

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16. Abstract

A review of the Apollo Spacecraft Certification (qualification) Test Program is presented. The approach to devising the spectrum of dynamic and climatic environments, the formulation of test durations, and the relative significance of the formal certification test program compared with development testing and acceptance testing are reviewed. Management controls for the formulation of test requirements, test techniques, data review, and acceptance of test results are considered. Significant experience gained from the Apollo Spacecraft Certification Test Program which may be applicable to future manned spacecraft is presented.

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APOLLO EXPERIENCE REPORT

CERTIFICATION TEST PROGRAM

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SUMMARY

The Apollo Spacecraft Certification Test Program was designed to ensure vigorous testing of the flight hardware in simulated flight conditions before flight. To accomplish this test program, various approaches were studied, and an integrated test and analysis approach using a limited amount of hardware to provide engineering confidence resulted. Statistical demonstration of the reliability of the hardware was abandoned as an approach because of impracticality and fiscal considerations. Every test anomaly was carefully evaluated, and recurrence control was initiated to ensure early maturity of the hardware. The certification test approach used in the Apollo Spacecraft Program is applicable to future manned spacecraft, but caution should be exercised in tailoring the certification test program to new applications.

INTRODUCTION

Ground and flight tests were of significant importance in the development of the Apollo spacecraft hardware. In this report, the Apollo spacecraft certification testing portion of the Apollo test program is discussed.

The two basic categories of the Apollo ground test program are described as design development testing and acceptance checkout testing. Preprototype, prototype, and production hardware were used during design development testing and are described generally as design feasibility, design verification, and certification testing, respectively.

The purpose of design development testing was to ensure that the hardware design was adequate for the performance of specified functions for the time and under the spectrum of "worst case" environments that were expected for a like piece of hardware during the combined ground and flight life of the hardware. Development test hardware was used solely for testing and not for flight.

Acceptance and checkout testing was intended to ensure that the manufactured hardware had no latent defects and that the equipment conformed to functional specification requirements. The program consisted of functional and, in many cases, environmental testing of the hardware at the manufacturing plant, functional testing at the prime contractor's facility before installation into the spacecraft, and a logical and thorough series of ambient functional subsystem and system checkout tests after hardware installation in the spacecraft at the prime contractor's facility. Many of these exhaustive series of tests were repeated at the NASA John F. Kennedy Space Center (KSC), culminating in the final countdown tests before lift-off. This complete range of hardware acceptance and checkout tests was conducted only on hardware that was to be flown.

Although valuable experience was gained in all the test areas, the scope of this report is restricted to the final portion of the design development testing (certification testing). The following facets of the certification test program are described in detail.

1. Establishment of the test requirements, environments, durations, and levels of exposure

2. Management controls before, during, and after testing

3. Knowledge gained and results obtained from the program

In addition, a brief discussion of the potential role of certification testing in future manned spacecraft programs is presented.

REQUIREMENTS

At the beginning of the Apollo Spacecraft Program, three primary test concepts were studied to determine the optimum method for the certification of the design of the spacecraft hardware. The first test concept considered was a statistical demonstration test program that would provide the rigor associated with demonstrating the reliability of the hardware statistically and would produce a significant amount of test data and test experience on production hardware. This approach, however, would require an increase of at least 15 to 20 times the number of test articles with attendant large costs and schedule implications.

The second test concept considered was a non-time-oriented design-limit test program in which production hardware would be used to demonstrate the capability of the hardware to withstand the severity of the flight environments increased by a factor of safety. A limited amount of test data would be obtained rapidly, and the tests would require a limited amount of test hardware. However, no statistical confidence in the reliability of the hardware would be produced, and the time-oriented sequential exposure of the hardware to the levels and durations of the environments expected in flight would be omitted.

The third test concept considered was an integrated test and analysis program in which two production units of hardware would be used, one unit for design-limit testing and the second unit for mission-life testing. This program would not require an excessive amount of hardware and would demonstrate the capability of the hardware to withstand the full mission spectrum of environmental magnitudes and durations with adequate margins of safety. More test articles would be required during the testing of certain critical items such as those in the propulsion system and during the testing of high-usage hardware such as switches and relays. Although a statistical demonstration of the reliability of the hardware would not be obtained, this approach would provide a significant degree of confidence in the design. The acceptance test would then provide the necessary confidence in the quality of the flight hardware.

The third test concept was selected for the certification test program. Testing was performed at the line replaceable unit (LRU) and at selected higher levels of assembly to ensure equivalent certification testing of the entire spacecraft. Hardware levels of assembly were tested, ranging from switches, dials, and meters to mechanical and electrical subassemblies, gyros, and computers. Also included in the testing were valves, tanks, and complete engines. More than 700 certification tests for the command and service module (CSM) and more than 500 certification tests for the lunar module (LM) were required to be completed successfully before flight. Complete subsystem and vehicle-level tests were included in the program to demonstrate the design capability of the interfaces between hardware elements (more than 125 tests for the CSM and 175 tests for the LM).

The establishment of certification test requirements to which the Apollo spacecraft hardware was to be exposed was a lengthy and, in some cases, an iterative process. The design environments were known, but some of the level and duration requirements were developed over a period of time. For example, development of the definition of the vibration environment spanned several years as knowledge of the actual flight vibration environment increased. Environments that were considered for each LRU included vibration, temperature, shock, acceleration, corrosive contaminants, vacuum, salt spray, sand and dust, humidity, oxygen, and pressure. All the environments to which the hardware would be exposed during acceptance tests, spacecraft assembly, transportation (both ground and air), spacecraft checkout, launch-site preparations, launch, and mission (including earth orbit, translunar and transearth coasts, entry, and landing) were considered. The hardware was then tested in a logical sequence to the most critical level of these environments.

The selection of the items to be exposed to a particular environment at a particular level and duration was determined on an item-by-item basis. This selection was made: (1) by assessing the type of hardware construction, the location of the hardware in the spacecraft, and the expected duration of hardware exposure to a particular environment; and (2) by determining whether the hardware was operational before, during, and after exposure. Conservative levels for environmental testing were selected to provide an adequate margin above the environments expected during flight. At the same time, care was taken to prevent the use of unrealistically severe levels.

After the environments and durations were determined for a given unit of hardware, the certification test requirements (CTR's) were formally documented for approval at the NASA Manned Spacecraft Center (MSC) before testing. All the CTR's developed for one given mission were known as a certification test network.

Vibration levels to which the Apollo spacecraft (CSM and LM) are subjected during the launch and boost phases of flight were primarily acoustically induced. Data on acoustic levels were obtained from wind-tunnel tests and from Little Joe II, early Saturn I, Saturn IB, and Saturn V flight tests. These data were used in vibroacoustic ground tests on full-scale LM and CSM structural test articles to certify the structure of each and to determine the vibration response levels at equipment locations within the vehicle. The vibration levels determined from these tests were used for testing individual components. As additional data became available from the LM and CSM vibroacoustic vehicle tests, necessary adjustments were made to the existing vibration test requirements.

Similar care was taken in the development of the exposure levels for the other ground and flight environments. Unrealistic environmental requirements could penalize hardware design and require extensive retesting because hardware certification tests were conducted concurrently with the installation of like items of hardware into flight vehicles. Hardware failures during unrealistically high environmental exposure levels could have resulted in unnecessary design changes and a resultant waste in funds and schedule time. As a supplement to the certification test program, a selected number of off-limits tests were conducted on certification hardware at higher environmental exposure levels to gain a better understanding of the actual hardware limitations. Failures during this type of testing were assessed from this standpoint and resulted in few design changes.

Ideally, an item of flight hardware would be subjected to one environmental acceptance test, one prelaunch checkout, and one mission. Practically, however, almost all the flight hardware could be subjected to more than one environmental acceptance test as a result of reacceptance after repairs or modification following the initial acceptance. The certification testing was thus arranged to demonstrate the capability of the hardware to withstand at least five environmental acceptance tests and still perform properly in tests that were equivalent to one complete prelaunch checkout and two complete missions. It was considered that a total of five acceptance tests was a practical upper limit of refurbish and retest cycles, and that testing for two full mission durations would give a practical and acceptable measure of performance margins.

The sequence in which the environments were applied to the test article approximated the sequence in which the flight article would be subjected to these environments during the ground and flight life of the article. The range of environmental exposure levels for Apollo hardware is shown in table I.

A typical set of test requirements for one piece of electronic equipment, the LM S-band transceiver, is shown in table II. After this piece of hardware had been tested successfully at pressures lower than 1×10^{-5} torr, it was determined that the equipment is required to operate at higher pressures. The S-band transceiver is located in the aft equipment bay of the LM on the outside of the pressure cabin but inside the thermal insulation and skin of the LM. The pressure in this region was determined to be in the range of 10 to 1×10^{-3} torr. A decision was made to retest the transceiver in the higher pressure range in which some electronic equipment was known to be sensitive to the corona discharge phenomenon. The test was conducted, and failures attributable to corona were corrected by pressurizing the transceiver. The validity of the design change was verified by subsequent testing. This example, in which the most severe environment for the equipment was not the most severe absolute environment (i.e., less than hard vacuum), also demonstrated the necessity for adjusting the requirements based on current knowledge of the operating environment.

TABLE I. - RANGES AND ENVIRONMENTS OF SPACECRAFT COMPONENTS

Basic test type	Test level	Environmental conditions
Vibration	0.015 to 14 g^2/Hz	Ground phases Transportation and handling Static firing and acceptance test Flight phases Boost phase Space flight phase Entry phase
Acoustics	67 to 165 dB	
Acceleration	Up to 20g	
Shock	To a maximum of 78g	
Temperature	-250° to 60° F	Natural conditions Transportation, ground handling, storage Sheltered Earth parking orbit to preentry phase Postlanding Induced conditions Transportation, ground handling, storage, checkout Sheltered Boost phase Earth parking orbit to preentry phase Lunar landing Entry phase
Corrosive contaminants	1 percent salt spray	
Oxygen	95 percent dry oxygen at 5 psia	
Humidity	Relative humidity at 5 psia and 60° to 90° F	
Thermal vacuum		
Temperature	-300° to 270° F	
Pressure	As low as 1×10^{-5} torr	

TABLE II. - SET OF CERTIFICATION TEST REQUIREMENTS

FOR THE LM S-BAND TRANSCEIVER

Test conditions	Levels and exposure	Remarks
Acceptance	--	To be conducted in accordance with applicable procedure.
Temperature[a]	-20° F for 6 hr 15° F for 6 hr 110° F for 6 hr 145° F for 6 hr	Equipment nonoperating during test. Tests to be conducted in accordance with applicable procedure.
Vibration	Launch and boost[a] Random: 2-1/2 min in each of the 3 mutually perpendicular axes for 7-1/2 min 20 to 2000 Hz — 0.005 g^2/Hz Sinusoidal vibration frequency shall be swept logarithmically from 5 to 100 to 5 Hz at 3 octaves/min for 1 sweep cycle along the X-axis, Y-axis, and Z-axis. 5 to 10 Hz — 0.2 double amplitude 10 to 100 Hz — 1.0g	Equipment nonoperating.
	Lunar ascent and descent[a] Random: 21 min in each of the 3 mutually perpendicular axes for 63 min Z-axis 10 to 20 Hz — 12 dB/octave 20 to 100 Hz — 0.01 g^2/Hz 100 to 120 Hz — 24 dB/octave 120 to 150 Hz — 0.45 g^2/Hz 150 to 180 Hz — -24 dB/octave 180 to 2000 Hz — 0.005 g^2/Hz X-axis and Y-axis 10 to 20 Hz — 12 dB/octave 20 to 100 Hz — 0.01 g^2/Hz 100 to 120 Hz — -12 dB/octave 120 to 2000 Hz — 0.005 g^2/Hz Sinusoidal vibration frequency shall be swept logarithmically from 5 to 100 to 5 Hz at 1/2 octave/min for 1 sweep cycle along the X-axis, Y-axis, and Z-axis. 5 to 30 Hz — 0.03 double amplitude 30 to 100 Hz — 1.4g	Equipment operating.

[a]To be followed by an operational test conducted in accordance with applicable procedure.

TABLE II. - SET OF CERTIFICATION TEST REQUIREMENTS

FOR THE LM S-BAND TRANSCEIVER - Concluded

Test conditions	Levels and exposure	Remarks
Acceleration	X-axis: 7.4g Y-axis and Z-axis: -- Duration: 3 min	Equipment nonoperating during test. To be conducted in accordance with applicable procedure.
Thermal vacuum[a]	Pressure: 1×10^{-5} torr Temperature at root of flange: 35° F for 48 hr	Equipment nonoperating for cold soak.
	Followed by thermal vacuum profile 35° to 135° F 10 to 1×10^{-4} torr	Thermal vacuum profile to be performed twice with simultaneous operation of primary mode and secondary mode.
Lunar landing shock	Shock pulse, sawtooth, 15g peak, 11 ± 1 msec rise, 1 ± 1 msec decay, 1 pulse/direction for a total of 6 pulses	Equipment operating during test. To be conducted in accordance with applicable procedure.
Sea air humidity	The test shall be conducted in accordance with applicable procedure except as follows: temperature, 90° F (+5°); salt concentrate, 1 percent (+0.5 percent, -0.0 percent); chamber humidity, 85 percent (+15 percent, -10 percent); exposure, 24 hr; method, 2 min/hr. Operational test to be performed on the assemblies in accordance with applicable procedure.	Equipment nonoperating during test.
Additional vibration[a]	Random: 5 min in each of the 3 mutually perpendicular axes for 15 min 20 to 80 Hz 3 dB/octave 80 to 350 Hz $0.067\ g^2/Hz$ 350 to 2000 Hz -3 dB/octave	Equipment operating.
Electromagnetic interference (EMI) plus integration and checkout after acceptance test and before launch	Ambient conditions: each electronic replaceable assembly (ERA) shall be operated for a minimum of 250 hr in accordance with operational test procedures.	This condition simulates the total operating time accumulated on each ERA from point of shipment to contractor through all checkout and acceptance testing before launch at KSC. EMI to be conducted in accordance with applicable procedure.

[a]To be followed by an operational test conducted in accordance with applicable procedure.

MANAGEMENT CONTROLS

The multitude of certification tests to be conducted, the numerous locations across the country at which the testing was done, and the large number of persons involved necessitated a thorough management control system. Although the successful development of spacecraft hardware cannot be reduced to a specific formula, a series of specific requirements was used to manage the certification test program. Those requirements considered significant are as follows.

1. Testing of the hardware to demonstrate design capability

2. Use of production hardware

3. Testing of units at the highest practical level of assembly

4. Use of two test articles for design-limit testing and mission-life testing

5. Use of natural and induced environments

6. Use of combined environments when practical

7. Testing of all redundant paths

8. Performing acceptance testing before certification testing

9. Use of certification by similarity to eliminate test duplication

10. Use of analysis to supplement testing

11. Testing at higher levels of assembly and at vehicle-level phases to demonstrate interfaces

12. Thorough understanding of all anomalies

13. Use of positive corrective action for anomalies and retesting as appropriate

14. Recertification after design, process, and manufacturing changes

15. Successful completion of all certification tests before flight

First, the use of testing as the primary method for the demonstration of hardware capability under environmental stress was undoubtedly the key to the success of the certification test program. An attitude of "proof through testing" was dominant in the management of the test program.

The use of production hardware, whereby the test article was produced under the same design manufacturing processes and controls as the flight hardware, ensured that the minor, and sometimes subtle, design or process changes (from which new failure modes can be introduced) were adequately tested.

Units were tested at the highest practical level of assembly to ensure the discovery of as many of the interface problems as possible. Although this procedure was often dictated by the level of assembly at which a particular manufacturer produced hardware, additional higher level-of-assembly tests were conducted. For example, the display and control panels and the consoles in the spacecraft cabin were tested environmentally as complete built-up assemblies even though similar individual instruments and control devices on the panels previously had been subjected to separate certification tests.

Where possible, two test articles were used, one for design-limit testing and the other for mission-life testing, to give the assurance of an adequate margin of safety for environmental exposure as well as for operating time and operating cycle.

The use of natural and induced environments assured that all possible environmental factors were considered and imposed on the hardware. For example, the corrosive contaminants, oxygen, and humidity (CCOH) tests imposed on hardware located in the cabin combined requirements that simulated the manned crew compartment atmosphere.

When practical, combined environmental exposures were used. The CCOH testing and the combination of thermal cycling and vacuum testing are examples of environments that were combined most frequently.

Demonstration testing of all the redundant paths in the equipment was required. The functional testing of hardware, to detect intermittencies while the hardware was exposed to an environment, was performed to the greatest possible extent.

Acceptance tests were performed on the certification hardware before certification testing was begun. This testing sequence provided assurance that the test hardware was free of manufacturing defects, and that the hardware was subjected to the same total envelope of environmental exposure to which the actual flight articles would be subjected.

Because the use of certification by similarity was permitted in the guidelines, duplication of testing was eliminated. For hardware common to the LM and the CSM, tests were conducted to the levels at which the environments in those areas were the most severe. As a result, cost savings were realized.

The use of analysis to supplement testing was common, but the substitution of analysis for testing was permitted only when testing was impractical or impossible. For example, numerous tests were conducted on the LM landing gear to demonstrate adequate structural design margins. These tests included 16 drop tests of a structural test article and five drop tests of a flight-configured LM to simulate the more critical loading conditions on the total vehicle. However, because of the numerous combinations of landing velocities, attitudes, and angular rates resulting in different sets of landing stability dynamics and load inputs to the structure, it was necessary to perform the primary certification for these two factors by analysis and to obtain point confirmation by test procedure. More than 40 000 kinematic loads cases and 17 000 computer simulations of the landing dynamics and loads were analyzed.

Tests at the higher level-of-assembly and vehicle-level phases of the certification test program were likewise exhaustive for the demonstration of the interfaces and the interacting effects of the hardware of a given module. The vibroacoustic testing of the entire LM and CSM ground test spacecraft, the land and water impact tests on the command module, the LM drop tests, the thermal vacuum tests of the full-scale LM and CSM vehicles, and the full-scale launch escape system tests conducted at the White Sands Test Facility added thousands of ground test hours to flight-configured hardware.

A thorough understanding of all anomalies was another enforced ground rule. The concept of a random failure was unacceptable to management. It was acknowledged that hardware failures were caused by discrete flaws in design, manufacturing, or procedures, and the function of the personnel responsible for the hardware was to understand the cause of any anomaly and to ensure that the particular problem did not recur in flight.

Positive corrective action for all anomalies and retesting to ensure the adequacy of the corrective action were performed in virtually all cases. The corrective action could be one of hardware redesign, hardware manufacturing, hardware quality control improvement, or procedural change.

Some typical problems and resolutions are as follows.

1. The suit fan stopped running after shock test. A main-bearing failure was suspected. A failure analysis showed that the bearings were contaminated from manufacturing procedures. In addition, the grease used was not suitable for this application, and improper fan-to-housing clearances were used. The rotor was not balanced correctly, and the fan housing was of insufficient strength. These problems were corrected by design and manufacturing process changes, and retesting verified the adequacy of these changes.

2. An excessive rate of oxygen depletion from one of the oxygen tanks was experienced after a 15-hour thermal stabilization period. This depletion was caused by the scrubbing action of the insulation during vibration, resulting in a vacuum degradation and excessive oxygen boiloff. Because this action would have an effect on long-duration lunar missions, a vacuum-ion pump was installed to improve the vacuum retention qualities of the tank. Retesting verified the adequacy of the redesign.

3. During waveform and percent modulation tests on an inverter modulation test at 30-volt direct current input, erratic frequency and voltage outputs were noted. Transistor pairs were not matched for gain. The problem was corrected by matching the gains of transistor pairs; this corrective action was proved to be effective.

4. During the testing of a fuse assembly for the LM, numerous open or intermittent circuits were observed. These failures were caused by the heat of soldering the fuse (small thermal mass) to the wafer terminals (large mass). This heat was being conducted down the fuse leads and was softening the fuse cement or melting the fuse element (or both). This failure condition was also prevalent during acceptance testing of a subsequent generation of production hardware.

Original attempts to correct this problem by substituting a different fuse and a wider wafer separation to provide more positive heat sinking were not successful. The final corrective action, which was proved successful by retesting, resulted in changing the assembly process from soldering to a machine-controlled welding operation.

The successful completion of all required preflight certification testing has been accomplished for all Apollo spacecraft flown to date. In addition, MSC approval of all certification requirements, changes, and test results was required.

The normal approval cycle required the prime contractors to submit the certification test results to MSC as soon as the findings were available, allowing MSC and the contractor to review the test results simultaneously. If the contractor approved the test results, with or without reservations, as being adequate to certify the hardware for flight, these findings were documented formally and forwarded to MSC for approval. The approval of the MSC Subsystems Manager and MSC Reliability and Quality Assurance was required.

In cases where certification testing or retesting occurred just before a launch date or a critical test at KSC, MSC personnel would witness the test and review onsite raw test data, and a Qualification Site Approval would be granted, if appropriate. This process permitted immediate certification of the hardware for the mission, pending the review of the formal test report from the contractor.

RESULTS

The results of the certification test program are considered to be both qualitative and quantitative. Included in the qualitative results are the following.

1. The use of failure modes and effects analysis was helpful in understanding the criticality of the hardware being tested and deciding whether failure could involve crew safety, affect mission success, or just be a nuisance in flight. This knowledge also was important to the decisionmaking process during corrective action procedures after the occurrence of a hardware failure. This analysis technique was not limited to certification testing but was equally useful for the entire ground test program.

2. Certification testing at the highest practical level of assembly did not eliminate the need to qualify and conduct screening and burn-in tests on electrical, electronic, and electromechanical (EEE) piece parts. Controlled EEE piece parts were necessary if the certification test results were to be applicable to identical flight hardware.

3. Teardown inspection after test completion provided the capability for the determination of incipient failures that were not detected with the test instrumentation.

4. Because of the inherent lag in the preparation of test reports, onsite review of raw test data was required when successful test completion was a constraint to an imminent flight.

5. Although the certification test program demonstrated design capability under environmental exposure, the program was not 100 percent effective in exposing all

design deficiencies. The environmental acceptance tests that were conducted on flight hardware for the primary purpose of detecting manufacturing flaws also resulted in the detection of some design problems, particularly in the designs that were difficult to manufacture.

6. Certification by similarity required exhaustive review for acceptability. Likewise, when certification was based on analysis rather than test, as much, if not more, review effort was required.

Quantitative results of the certification test included the following.

1. Certification testing, environmental acceptance testing, and actual flight were considered the three major areas in which Apollo hardware is exposed to environment. Considering only those design-related failures found in production hardware, 68 percent were found during certification testing, and 30 percent were found during the environmental acceptance testing (vibration and thermal cycling) of the flight hardware. These results illustrate the fact that some design weaknesses were hard to find, with a few remaining undetected during the design and development phase and the certification testing, and were finally discovered in the acceptance testing of the follow-on production hardware. Many of these problems were producibility type problems incapable of detection with the few units being tested. The combination of certification and environmental acceptance testing kept the number of design problems to a minimum during flight.

2. Certification testing also exposed 45 percent of the failures attributable to poor workmanship, although the certification hardware totaled considerably less than 45 percent of the production hardware. The primary reason for this is that discovery of the workmanship problems during early testing of production hardware resulted in the implementation of corrective actions and controls, thus precluding manufacturing defects in follow-on production hardware.

Additional quantitative results are evident from a representative distribution by environment of the design failures that occurred during certification testing. Of all the environments to which the hardware was exposed, more than three-fourths of the failures were experienced in the vibration, thermal vacuum, and temperature-cycling environments.

CONCLUDING REMARKS

One possible recommendation for the design of a certification test program emphasizes dynamic environmental exposures. As previously mentioned, numerous failures that were attributed to hardware design were detected during tests in the launch vibration environment. Although Apollo spacecraft hardware was designed on the basis of a single launch vibration cycle, future space vehicle hardware may be required to survive numerous launch and reentry cycles. Because the major cost of conducting subassembly vibration tests is associated with the test setup and data-reduction time, rather than with the actual vibration exposure, the equivalent of numerous launch vibration cycles should be conducted on the equipment once it is set up. By this means, the hardware design would be exposed to those vibration environments that have been proven by experience to be most likely to uncover design deficiencies.

The unique certification requirements for the long-duration space missions are governed by the extremely long times involved, in some cases as many as 10 years. Although the equipment is subjected to only a single launch environment, which can be tested, the long operational time in the space environment for equipment cannot be simulated in short-time testing with any degree of confidence. To date, no well demonstrated method exists for extrapolating compressed-time testing techniques for equipment that will undergo extended exposure to either pressurized or thermal vacuum environments.

These factors and the need for ready accessibility to as much hardware as possible for replacement capability may result in the requirement that all possible hardware be located within the pressurized, or at least the pressurizable, confines of the spacecraft. Thus, the hardware systems within the space vehicle should probably be designed to allow for the inflight repair and replacement of failed hardware with spare hardware carried on board or supplied from earth for near-earth space operations.

One possible approach for demonstrating the design of hardware for long-duration missions would be to use near-earth long-term space vehicles as inflight certification test sites. The data obtained from those test sites, could then be used, along with supplementary data from ground tests, to extrapolate performance to somewhat longer times in space. This approach would require an early hardware design freeze so that the results of the certification tests would remain valid for the later mission hardware.

Only a few of the possible approaches that could be applied to the unique hardware certification requirements of future programs have been outlined. To date, no definitive decisions have been made. Many of the Apollo Spacecraft Certification Test Program guidelines can be applied to ensure that the hardware launched as a part of the U.S. space program is adequate for mission performances.

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