THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

MIL-HDBK-340 (USAF)
01 JULY 1985

MILITARY HANDBOOK

APPLICATION GUIDELINES FOR MIL-STD-1540B; TEST REQUIREMENTS FOR SPACE VEHICLES

NO DELIVERABLE DATA REQUIRED
BY THIS DOCUMENT

FSC1810

DISTRIBUTION STATEMENT A Approved for public release; distribution unlimited.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

DEPARTMENT OF DEFENSE
Washington, D.C. 20301

MIL-HDBK-340 (USAF)

Application Guidelines for MIL-STD-1540B,
Test Requirements for Space Vehicle

1. This military handbook is approved for use within the Department of the Air Force, and is available for use by all Departments and Agencies of the Department of Defense.

2. Beneficial comments (recommendations, additions, -deletions) and any pertinent data which may be of use in improving this document should be addressed to:

USAF Space Division, SD/ALM
P. O. Box 92960
Worldway Postal Center
Los Angeles, CA 90009 - 2960

by using the self-addressed Standardization Document Improvement Proposal (DD Form 1426) appearing at the end of this document or by letter.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

FOREWORD

General military specifications and military standards are top-level DoD standardization documents. They are prepared in accordance with DoD directives to establish a formal corporate memory of good practices and requirements. In some cases they are referenced in government regulations to impose internal compliance on government organizations. More typically. these DoD standardization documents are intended for reference in contracts as compliance documents for imposing the good practices and other stated requirements on contractors. Although these roles are broad, many facets of engineering technical data that are important to program success are not appropriate for inclusion in these formal DoD standardization documents. These added facets of information may be documented in technical reports or, as in this case, in military handbooks.

This military handbook is intended to document additional facets of engineering technical information pertinent to the requirements stated in MIL-STD-1540B, "Test Requirements for Space Vehicles." As a technical reference, this handbook should provide the basis for achieving a consistent technical approach for tailoring MIL-STD-1540B requirements, where appropriate, and may also provide the bases to justify deviations or alternative approaches where they are appropriate. Each major subsection of this handbook addresses a subject taken from MIL-STD-1540B. Remember that the information included herein is for general guidance; it need not be followed if it does not accommodate the requirements of the program. In the case of difference between this handbook and the requirements of MIL-STD-1540B, the requirements of MIL-STD-1540B should take precedence.

Some guidance regarding format, presentation, and organization of material in this military handbook seems advisable. The handbook has the same organization as a military standard, i.e., the first three sections are: Section 1, Scope, Section 2, Referenced Documents, and Section 3, Definitions. Sections 4 through 12 generally follow the sequence of material in MIL-STD-1540B, although two or more subjects or paragraphs of MIL-STD-1540B are often linked and discussed in one major section or subsection of this handbook. Thus, the section, subsection, and paragraph numbers of this handbook do not correspond to the paragraph numbers of the standard. However, exact references are given in the handbook to the corresponding paragraph numbers of MIL-STD-1540B.

(Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

FOREWORD (Continued)

For the convenience of the users of MIL-STD-1540B, Table XXI in Section 13 of this handbook provides a cross-reference from the primary MIL-STD-1540B paragraph numbers to the corresponding paragraph numbers of this handbook.

Each major subsection of this handbook addresses a subject area of interest. Each subject area is organized into three major paragraphs. The first paragraph is titled “Standard Criteria,” and it quotes the text of the MIL-STD-1540B paragraphs which are discussed. This allows the reader to use this handbook without constant reference to the standard, making it easier and more efficient to use. Also. for the convenience of the reader, the text quoted from MIL-STD-1540B is printed in italics to distinguish it from the text of the handbook. The second major paragraph is titled “Rationale for . ...” and it contains background information such as the purpose or reasons for the subject area requirements in the standard. The third major paragraph is titled “Guidance for Use of . ...” and it contains the information intended to aid the reader in the detailed application of the MIL-STD-1540B requirements for that subject area.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Continued)

PAGE

FIGURES

TABLES

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

CONTENTS (Concluded)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 1

SCOPE

1.1 GENERAL

This handbook provides additional information pertaining to the test requirements of MIL-STD-1540B, "Test Requirements for Space Vehicles." This handbook includes information Only on those test requirements for which additional explanations and guidance have been developed beyond that given in MIL-STD-1540B. Section 13 of this handbook provides an index and a cross reference from MIL-STD-1540B paragraph numbers to the corresponding paragraph numbers of this handbook. Further information and additional sections may be developed and added to the handbook in future revisions.

1.2 PURPOSE

This handbook was written to provide explanations and guidance to the users of MIL-STD-1540B. The information presented herein is intended to aid in the formulation and review of the detailed test requirements for space vehicles including the tailoring of MIL-STD-1540B requirements into specific program specifications or contracts.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 2

REFERENCED DOCUMENTS

2.1 GOVERNMENT DOCUMENTS

2.1.1 Specifications, Standards, and Handbooks. Unless otherwise specified, the following specifications, standards, and handbooks of the issue listed in that issue of the Department of Defense Index of Specification and Standards (DoDISS) Specified in the solicitation form a part of this standard to the extent specified herein.

SPECIFICATIONS:

STANDARDS

Military

MIL-STD-810	Environmental Test Methods and Engineering Guidelines
MIL-STD-1522	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
MIL-STD-1540B	Test Requirements for Space Vehicles

2.1.2 Other Government Documents, Drawings, and Publication . The following other Government documents, drawings, and publications form a part of this standard to the extent specified herein.

NASA S-69-1117	Leakage Testing Handbook

(Copies of specifications, standards, handbooks, drawings, and publication required by contractor in connection with specified acquisition functions should be obtained from the contracting activity or as directed by the contracting officer.)

2.2 ORDER OF PRECEDENCE

In the event of a conflict between the text of this handbook and MIL-STD-1540B, the MIL-STD-1540B requirements shall take precedence.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

SECTION 3

DEFINITIONS

The definitions of terms used in this handbook are the same as in MIL-STD-1540B. Additional terms are as defined in MIL-STD-1522.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

SECTION 4

APPLICATION OF TEST REQUIREMENTS

4.1 APPLICATION TO OTHER VEHICLES

4.1.1 Standard Criteria. Contents of Paragraph 1.2 of MIL-STD-1540B (the intended application of the standard), are as fOllows:

> 1.2 APPLICATION
>
> The tailored application of these test requirements to a particular space program Is Intended to assure a high level of confidence in achieving a successful space mission. This standard is Intended for use in the procurement of space vehicle hardware. including space vehicles and airborne support equipment that remain in the space shuttle orbiter during orbital flight, as well as orbital satellites.

4.1.2 Rationale for Application of Test Requirements. The test requirements specified are a composite of those tests currently used in achieving successful military space missions. MIL-STD-1540B therefore established a standard test baseline applicable to all space vehicles. However, it is intended that the test requirements for use on a particular space program should be tailored to the specific vehicle or project, considering the realistic environmental life cycle, design complexity, state of the art, mission criticality, and acceptable risk. Of course, any program may find it revealing to make comparisons of its planned test program to these established baselines, regardless of the contractual requirements.

4.1.3 Guidance for Application of Test Requirements to Other Vehicles. In addition to the use of the stated baseline test requirements for military space vehicle and airborne support equipment, the tests in MIL-STD-1540B are stated in terms of design and operating environments. That means that the test requirements often can be applied directly to other types of vehicles, or they can be easily modified to apply to other types of equipment requiring high reliability. In particular, the qualification and acceptance test requirements for components of space vehicles (Paragraphs 6.4 and 7.3 of MIL-STD-1540B) often are directly applicable to components of other types of vehicles. For example, the component test baselines usually can be applied directly to testing components

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

of missiles, launch vehicles, and injection stages. The space vehicle tests (Paragraphs 6.2 and 7.1 of MIL-STD-1540B) may also be tailored for testing of launch Vehicles and injection stages. The major considerations in these cases are the differences in the environmental life cycle and the service life of the different vehicles. For example, the service life of an injection stage is from several minutes to several days versus only minutes for expendable launch vehicles, while the service life of a military space vehicle may be 10 years or more.

4.2 QUALIFICATION BY SIMILARITY

4.2.1 Standard Criteria. MIL-STD-1540B does not directly address criteria for the qualification of items by similarity; however, it does provide the standard test baselines for comparison.

4.2.2 Rationale for Qualification by Similarity. The continued production and use of items designed for space vehicles of one program on space vehicles of another program is of interest to every program office. Not only are the design, tooling, and qualification costs eliminated for subsequent programs, but the continuing usage of the same item increases the confidence in the item's reliability. Of course, to accommodate specific requirements of another program, it may not be possible to use the same exact item, so there may be changes required in the item or in its testing. If those changes are within reasonable bounds, then qualification of the revised item by similarity should be considered.

4.2.3 Guidance for Qualification by Similarity

4.2.3.1 Component Criteria. If component "A" is to be considered as a candidate for qualification by similarity to a component "B" that has already been qualified for space use, then all of the following conditions should apply:

a. Component "A" should be a minor variation of component "B." Dissimilarities will require understanding and evaluation in terms of weight, mechanical configuration, thermal effects, and dynamic response. Minor design changes involving substitution of piece parts and materials with equivalent reliability items can generally be tolerated. Design dissimilarities resulting from addition or subtraction of piece parts and particularly moving parts, ceramic or glass parts, crystals, magnetic devices, and power conversion or distribution equipment should be given priority attention in the evaluation.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

b. Components “A” and “B” should perform similar functions, with "B" having equivalent or greater operating life with variations only in terms of performance such as accuracy, sensitivity, formatting, and input-output characteristics.

c. Components “A” and “B” should be produced by the same manufacturer using identical tools and manufacturing processes.

d. The environments encountered by component “B” during its qualification or flight history should have been equal to or more severe than the qualification environments intended for component "A."

e. Component “B” should have successfully passed a post-environmental functional test series indicating survival of the qualification stresses.

f. Component “B” should have been a representative flight article.

g. Component “B” should not have been qualified by similarity or analysis.

4.2.3.2 <u>Criteria for Other Items</u>. In some cases, the item to be qualified by similarity is not a component but is some other level of assembly, such as a subsystem. In that case, the criteria for the item to be qualified by similarity would be the same as though the item were a component (see Paragraph 4.2.3.1).

4.2.3.3 <u>Partial Testing</u>. It is recognized that in some cases, where all the criteria in Paragraph 4.2.3.1 are not satisfied, qualification based on engineering analysis plus partial testing may be permissible. In this case, negotiation between the contracting officer and the contractor may result in an abbreviated testing program satisfactory for qualification of the component or item in question. The acceptability of qualification by similarity should be documented by test reports, drawings, and analyses. This justification or proof of qualification should be prepared in data packages and submitted to the contracting officer as required by the contract. The contracting officer usually has the final decision as to the acceptability of qualification by similarity, and the burden of proof of qualification is the responsibility of the contractor.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 5

DEFINITION COMMENTS

5.1 DESIGN ENVIRONMENTS

5.1.1 Standard Criteria. Contents of Paragraphs 3.8, 3.9, 3.12, and 4.2.2 of MIL-STD-1540B (definition of design environments, design margins, and requirements for tolerances) are presented in order to provide guidance regarding the interaction of these parameters and their effect on the test requirements of space vehicles and components.

3.8 DESIGN ENVIRONMENTS, SPACE VEHICLE

The design environments for a space vehicle are the composite of the various environmental stresses to which the space vehicle must be designed. Each of the design environments for a space vehicle is based upon:

a. The maximum and minimum predicted environments during the operational life of the space vehicle, plus

b. An environmental design margin (see 3.12) that Increases the environmental range to provide an acceptable level of confidence that a failure will not occur during the service life of the space vehicle.

3.9 DESIGN ENVIRONMENTS, SPACE VEHICLE COMPONENTS

The design environments for space vehicle components are the composite of the various environmental stresses to which the space vehicle hardware components must be designed. Each of the design environments for a space vehicle component is based upon:

a. The maximum and minimum predicted environments during the operational life of the component, or for temperature. a standard thermal range between -24 deg C and +61 deg C when the predicted range is less severe, plus

b. An environmental design margin (see 3.12) that increases the environmental range to provide an acceptable level of confidence that a failure will not occur during the service life of the component (see 3.37).

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

3.12 ENVIRONMENTAL DESIGN MARGIN

An environmental design margin for an item is an Increase in the environmental range used for the design (and for the qualification testing) of an item to reduce the risk of an operational failure. It may include Increases In the maximum levels, decreases in the minimum levels. and Increases In the time exposure to the extreme levels. The environmental design margin is intended:

a. To accommodate differences among qualification and flight units due to variations in parts, materials, processes, manufacturing. testing, and degradation during useage;

b. To incorporate the allowable test condition tolerances;

c. To avoid qualification test levels that are less severe than the acceptance test ranges or operating ranges;

d. To help assure against fatigue failures due to repeated testing and operational use.

Unless otherwise specified. the test condition tolerances allowed by this standard are assumed to be incorporated In the environmental design margin. For example, space vehicle items are designed, unless otherwise specified, to thermal environments 10 deg C higher and 10 deg C lower than the maximum predicted thermal ranges (see 3.25). This 10 deg C environmental design margin includes a ± 3 deg C tolerance for acceptance test conditions and a ± 3 deg C tolerance for qualification test conditions.

Unless otherwise specified, space vehicle items are also designed to acoustic noise and random vibration environments that are 6 dB above the maximum predicted levels. This 6 dB environmental design margin for acoustic noise and random vlbration includes a ± 1.5 dB tolerance In the overall level (Integrated root mean square value over the total frequency range of the test spectrum) for acceptance test conditions and a ± 1.5 dB tolerance for qualification test conditions.

When the qualification or acceptance tests are controlled using test condition tolerances with magnitudes less than specified herein (3 deg C or 1.5 dB), the

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

environmental design margins (10 deg C or 6 dB) may be reduced accordingly. For example. If qualification and acceptance acoustic tests were both controlled to ± 1.0 dB, the design margin would be 5 dB instead of 6 dB. If larger test condition tolerances are allowed, then the design margins would be Increased accordingly.

Other environmental design margins applicable to space vehicle Items include 6 dB for shock, 6 dB for sinusoidal vlbration, a factor of 2 for launch or injection acceleration, and a factor of 1.25 for maximum acceleration of deployed components on a spinning space vehicle,

Another element of the environmental design margin is the the the Item is exposed to the design environmental levels. An increase in exposure time or number of cycles over that expected in operation is usually specified for vibration and acoustic design environments to increase confidence that wearout or fatigue failures will not occur. of course. the environmental design margins may be changed to either higher or lower levels, or to longer or shorter exposure times, depending upon specific program requirements and allowable risk.

4.2.2 Test Condition Tolerances

The test condition tolerances allowed by this standard shall be applied to the nominal test values specified. Unless otherwise specified, the following maximum allowable tolerances on test conditions shall apply.

Temperature	± 3 deg C
Pressure	
Above 1.3×10^{2} pascals (1 Torr)	± 10 percent
1.3×10^{-1} to 1.3×10^{2} pascals (0.001 Torr to 1 Torr)	± 25 percent
Less than 1.3×10^{-1} pascals (0.001 Torr)	± 80 percent
Relative Humidity	± 5 percent
Acceleration	± 10 percent

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

Vibration Frequency	*± 2 percent*
Sinusoidal Vibration Amplitude	*± 10 percent*
Random Vibration Acceleration Power Spectral Density	
20 to 500 Hz (25 Hz or narrower)	*± 1.5 dB*
500 to 2000 Hz (50 Hz or narrower)	*± 3.0 dB*
Random overall grins	*± 1.5 da*
Sound Pressure Level	
1/3 Octave Band	*± 3.0 dB*
Overall	*± 1.5 dB*
Shock Response Spectrum (Q = 10)	
1/6 Octave Band Center Frequency Amplitude	*± 6 dB with 30 percent of the response spectrum center frequency amplitudes greater than nominal test specificati on*
Static Load	*± 5 percent*

5.1.2 Rationale for Definition of Design Environments. The environmental levels to which an item should be qualified are the same as the design environmental levels for the item. These design environmental levels are typically based upon the maximum and minimum predicted environmental levels for an item during its operational life plus the appropriate environmental design margin. The maximum expected extremes of the operational environments are defined in Paragraphs 3.8 and 3.9 of MIL-STD-1540B. A standard operating thermal range of -24 deg C to +61 deg C is usually used for components when the maximum predicted operating range is less severe. The environmental design margins specified are primarily intended to incorporate the allowable test condition tolerances and to accommodate any differences among production units. The environmental design margins are also intended to assure qualification test levels that are more severe than the maximum operating ranges that can occur in flight and help assure against performance degradation and fatigue failures due to repeated acceptance testing and operational use. For example, the 10 deg C environmental design margin specified in MIL-STD-1540B makes the standard thermal design range for components from -34 deg C to +71 deg C. This standard design range for space components is similar to that used for aircraft subsystems and therefore should not impose unusual design problems in most cases. In addition, this standard design range encourages the development of standard

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

modules, provides a very revealing test screen for defective components, allows components to be moved to other locations on a spacecraft without affecting qualification, and may allow the use of a qualified component on other spacecraft without requalification.

5.1.3 Guidance for Application of Design Environments. Environmental acceptance tests of space vehicles and components are intended to aid in detecting workmanship and material problems and to verify proper functioning during exposure of the items to environmental levels equal to the maximum extremes predicted during their operational life. The difference between the environmental levels used for acceptance testing and for qualification testing are therefore the specified environmental design margins. Where qualification equipment is used for flight, the standard environmental design margins should be reconsidered, as discussed in Section 11 of this handbook.

As used in MIL-STD-1540B, environmental design margins may be interpreted to be the same as qualification test margins. The margins are added environmental exposure in amplitude (e.g., temperature level or vibration amplitude) and, in the case of dynamic environments, exposure duration. The design margin is intended to diminish the risk of operational failure due to manufacturing variations in flight hardware which might produce less resistance to failure than the qualification specimen. Also, the margins assure that the hardware will be flightworthy following repeated acceptance tests, should they be necessary, and they ensure the capability for retest necessitated by rework and repairs without the risk of fatigue failure. If qualification hardware is to be used for flight, consideration should be given to the fatigue life as it relates to the design margin.

To assure that minimum design (qualification) test margins are maintained, consideration is given to the effect of test tolerances on margins. The contribution of tolerances to the margin determination process is described in detail in Paragraph 3.12 of MIL-STD-1540B. The nominal margins generally specified may be decreased as long as test tolerances are commensurately tightened. The tolerances in Paragraph 4.2.2 of MIL-STD-1540B permit variation in test amplitudes from specified values in recognition of generally attainable test control capabilities. The interaction between margins and tolerances should be recognized in order to avoid unrealistically tight tolerances which would be required if test margins were excessively reduced.

The maximum exposure time for acceptance random vibration tests, so as to not exceed the fatigue damage potential of the

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

qualification test, is strongly influenced by the relation between the qualification and acceptance test levels. If qualification test margins are relatively low, the number of allowable acceptance tests which can be repeated before the fatigue strength may be exceeded, as demonstrated by the qualification test, is seriously limited. This interaction should be considered in evaluating environmental qualification test margins. Further details of retest limits are provided in Section 6.3 of this handbook.

5.2 LIMIT AND ULTIMATE LOADS

5.2.1 Standa~d Criteria. Contents of Paragraphs 3.18 and 3.46 of MIL-STD-1540B (definitions of limit load and ultimate load) are as follows:

> *3.18 LIMIT LOAD*
>
> *The limit load is the maximum anticipated -load. or combination of loads, which a structure may be expected to experience during the performance of specified missions in specified environments. since the actual loads that are experienced in service are in part random in nature, statistical methods for predicting limit loads are employed wherever appropriate.*
>
> *3.46 ULTIMATE LOAD*
>
> *The ultimate load is the maximum static load to which a structure is designed. It is obtained by multiplying the limit load (see 3.18) by the ultimate factor of safety.*

5.2.2 Rationale for Definitions of Limit and Ultimate Loads. The prime objective of the structural design process is to have a structure capable of functioning satisfactorily at the most severe operational loads and environmental conditions. Prediction of the extreme worst case anticipated loads takes into account environmentally induced loads and pressures, their time and phase relationships, frequencies, durations, statistical characteristics, and the manner in which various load sources combine. The worst case anticipated service loads are termed limit loads and are intended to represent a low probability extreme event. In case of uncertainty in dynamic load sources and dynamic structural characteristics, an uncertainty factor is usually incorporated when defining limit load.

In order to ensure satisfactory operation at limit load, it is required that the structure be capable of withstanding limit load conditions without gross yielding or, for conditions more

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

severe than limit load, without catastrophic failure, such as rupture or collapse. Such extreme conditions may arise from inaccuracies in analysis and verification testing, structural discrepancies, and variations in material and structural properties. Compensation for these variations is provided by the factor of safety. The ultimate loads therefore equal the limit loads multiplied by the ultimate factor of safety.

5.2.3 Guidance for Application of Limit and Ultimate Loads. Limit load and ultimate load are the critical design and test load levels in the structural integrity verification process. Structural adequacy of design is demonstrated by qualification tests conducted on flight quality hardware as described in Paragraph 6.3.1 of MIL-STD-1540B for general (nonpressurized) structures and Paragraph 6.4.10 for pressure vessels. Success criteria for the qualification test include as primary requirements that the structure sustain limit loads without any gross yielding or detrimental deformation and ultimate loads without rupture or collapse. If limit and ultimate load tests are required for qualification Of pressurized structure, including main propellant tanks and solid rocket motors, they must be conducted at the most critical combinations of external loads and internal pressure. This requirement is detailed in MIL-STD-1522.

Frequently, in order to establish additional confidence in the design, yield load (limit load times the yield factor of safety) is used in place of limit load. In such cases, the structure is required to sustain yield loads without excessive yielding or detrimental deformation.

5.3 ACOUSTIC ENVIRONMENT

5.3.1 Standard Criteria. Contents of Paragraph 3.20 of MIL-STD-1540B (definition of maximum predicted acoustic environment) is as follows:

> *3.20 MAXIMUM PREDICTED ACOUSTIC ENVIRONMENT*
>
> *The maximum predicted acoustic environment is the extreme value of fluctuating pressure occurring on the external surface of the space vehicle which occurs during liftoff, powered flight, or reentry. The maximum predicted acoustic environment test spectrum is specified based on one-third octave bands over a frequency range of 32 to 10.000 Hertz (Hz). The duration of the maximum environment is the total period when the overall amplitude is within 6 dB of the maximum overall amplitude. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods . The data must be tested to show a satisfactory*

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. Where there are less than three data samples, a minimum margin of 3 dB is applied to the prediction to account for the variability of the environment.

5.3.2 Rationale for Definition of Maximum Predicted Acoustic Environment. The acoustic environment experienced by a space vehicle is the forcing function which produces most of the vehicle vibration response at frequencies greater than 50 Hz. The relative contributions of the forcing function producing these vibration responses are dependent on the launch vehicle, the space vehicle configuration, and the particular location of interest. Vibration requirements for components on space vehicles, therefore, are nearly always linked directly to the acoustic environment to which the vehicle is exposed. The vibration criteria for a payload would be based on the acoustic level within the payload compartment but external to the payload. For launch vehicles it would be based on the level external to the applicable launch vehicle equipment zone. The acoustic environment is generally near maximum levels for approximately 10 seconds at launch due to ground-reflected acoustic energy emanating from the exhaust flow. During transonic and maximum dynamic pressure regions of flight, acoustic levels comparable to launch levels can exist for up to 30 seconds. Acoustic environments during these time periods can have large spatial variations. Consequently, acoustic design criteria for space vehicles are sometimes defined by zones. More commonly, however, a single criterion is defined which represents the maximum environment in one-third octave bands to which any vehicle surface is expected to be exposed. The goal is to define the maximum level in statistical terms as discussed in Paragraph 3.20 of MIL-STD-1540B. Seldom, however, does sufficient data exist to allow performance of rigorous statistical analysis. Nevertheless, the maximum expected acoustic environment is usually developed considering variations such as different launch pads, different trajectories, spatial variations within the launch vehicle payload compartment, and in some cases different launch vehicles.

5.3.3 Guidance for Application of Maximum Predicted Acoustic Environment. Generally, as stated in Paragraph 5.3.2, the maximum predicted acoustic environment represents the highest environment to which any vehicle surface is expected to be exposed. A single maximum environment definition is the preferred approach for most programs. This results in lower cost test programs and simplifies definition of the design criteria. In some cases, however, this may require significant

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

overdesign of many elements of the vehicle. For example, currently in the Space Transportation System (STS) orbiter cargo bay, a high intensity local environment exists near the cargo bay vents. For cases such as this, it may be more cost-effective to design only the hardware located near the vents for the local high intensity environment. However, the testing necessary to simulate this could be more costly, since most acoustic test facilities are geared to provide uniform environments. Additional test equipment and special test procedures would be needed to produce a local more intense environment.

5.4 PYROTECHNIC SHOCK ENVIRONMENT

5.4.1 Standard Criteria. Contents of Paragraph 3.22 of MIL-STD-1540B (definition of maximum predicted pyrotechnic shock environment) is as follows:

3.22 MAXIMUM PREDICTED PYRO SHOCK ENVIRONMENT

The pyro shock environment Imposed on the space vehicle components is due to structural response when the space or launch vehicle electro-explosive devices are activated. Resultant structural response accelerations resemble the form of superimposed complex decaying sinusoids which decay to a few percent of their maximum acceleration in 5 to 15 milliseconds. The maximum predicted pyro shock environment is specified as a maximum absolute shock response spectrum determined by the response of a number of single-degree-of-freedom systems using Q = 10. The Q is the acceleration amplification factor at the resonant frequency for a lightly damped sys tern. This shock response spectrum is determined at frequency Intervals of one-sixth octave or less over a frequency range of 100 to 10,000 Hz. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods. The data must be tested to show a satisfactory fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. where there are less than three data samples. a minimum margin of 4.5 dB is applied to account for the variability of the environment.

5.4.2 Rationale for Definition of Maximum Predicted Pyrotechnic Shock Environment. Pyrotechnic shock environments have caused flight failures of equipment in space Vehicles. The pyrotechnic shock environments have large variations in amplitude over the range of equipment mounting locations in a typical spacecraft. Shock levels vary strongly as a function of

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

structural path length and structural joints between the device or event generating the shock and equipment locations. wpically, experimental data will 6h0w large attenuation of higher frequency components of the shock spectra. For complex spacecraft structures, it is not unusual for peak shock spectrum amplitudes at frequencies in the 2000 to 10,000 Hz range to be reduced by 20 dB from source levels in a distance of a few feet. In repeated tests of an identical test configuration on well-designed space vehicles, data from a given location will show test-to-test variations of plus and minus 6 dB. For these reasons, it iS very important to give CarefUl consideration to the determination of, and philosophy to be used in, establishing maximum expected shock environments.

5.4.3 Guidance for APPlication of Maximum Predicted Pyrotechnic Shock Environment. Maximum predicted pyrotechnic shock environments can be defined in several ways. As discussed in Paragraph 3.22 of MIL-STD-1540B, it is desired that the maximum predicted value represent a ninety-fifth percentile value. Given that sufficient data are available, this can be a value defined for a specific location or for a given zone of a vehicle. Zonal requirements are preferred and are likely to be more cost-effective on multiple vehicle programs, because this approach minimizes the amount of requalification when changes are made. On multiple programs, changes from one vehicle to another usually occur which alter the shock environment for individual components but which may not significantly change the zonal environment. These changes include such items as the relocation of components, modifications to ordnance devices, preloading in separation hardware, and structural redesign. A penalty of this approach is possible overdesign of those components which may be in a quiet area of the zone. On a single vehicle program, it may be more cost-effective to tailor the maximum predicted shock environment for individual components or small groups of components, based on their proximity to shock-generating devices.

5.5 RANDOM VIBRATION ENVIRONMENT

5.5.1 Standard Criteria; Contents of Paragraph 3.23 of MIL-STD-1540B (definitions of maximum predicted random vibration environments) are as follows:

> *3.23 MAXIMUM PREDICTED RANDOM VIBRATION ENVIRONMENT*
>
> *The random vibration environment imposed on the space vehicle components is due to the liftoff acoustic field, aerodynamic excitations. and transmitted structure-borne vibration. The maximum predicted random vibration environment is specified as a power spectral density,*

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

based on a frequency resolution of 1/6 octave (or narrower) bandwidth analysis, over a frequency range of 20 to 2000 Hz. A different spectrum may be required for different equipment zones or for different axes. The component vibration levels are based on vibration response measurements made at the component attachment points during ground acoustic tests or during flight. The duration of the maximum environment is the total period during flight when the overall amplitude is within 6 dB of the maximum overall amplitude. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods. The data must be tested to show a satisfactory fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. Where there are less than three data samples, a minimum margin of 3 dB is applied to account for the variability of the environment.

5.5.2 Rationale for Definition of Maximum Predicted Random Vibration Environment. The maximum predicted vibration environments are principally used as design and testing requirements for components and subsystems. Often, for procurement of long lead items, the vibration environments must be established well before the vehicle structural design has matured. Information available to establish predictions is usually very limited. The variability of the environment is great, considering the large number of parameters which influence levels for any given component location. Cost and schedule impacts incurred if levels are raised after release of procurement contracts may be Substantial. For these reasons, considerable care and foresight are needed in establishing maximum predicted vibration environments.

5.5.3 Guidance for Application of Maximum Predicted Vibration Environment. Vibration environments in space vehicle structures at frequencies above approximately 50 Hz are primarily the result of an acoustic forcing function. The vibration environment in a given vehicle will be proportional to the level of acoustic excitation. Vibration levels throughout a vehicle are highly variable and dependent upon factors such as orientation, local resonances, damping, structural mass loading. and degree of coupling with adjacent structures. In establishing a maximum predicted environment, one must decide whether this is to be the maximum environment for a specific axis, for a specific location, for a given zone, or possibly the maximum for the entire vehicle or family of vehicles. Selection of the correct maximum vibration environment for a particular

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

program situation will be dependent on considerations such as the number of vehicles in the program, the design maturity of the vehicle, available test data, and in some cases even the design and qualification status of available components and subsystems. It is recommended that a zonal approach be followed in establishing maximum predicted vibration levels. In general, the practice of establishing vibration levels for individual components for specific locations should be avoided. Experience on past programs has shown that it can lead to numerous specification changes late in the program and costly retests.

5.6 THERMAL UNCERTAINTY MARGIN

5.6.1 Standard Criteria. Contents of Paragraphs 3.25 and 3.45 of MIL-STD-1540B (definitions of maximum and minimum predicted component temperature and thermal uncertainty margin) are provided below. These definitions are presented in order to show the relationship between the maximum and minimum predicted component temperatures and the thermal uncertainties.

3.25 MAXIMUM AND MINIMUM PREDICTED COMPONENT TEMPERATURES

The maximum and minimum predicted component temperatures are the highest and lowest temperatures that can be expected to occur on each component of the space vehicle during all operational modes Including an uncertainty factor. The component temperatures are predicted by an analytical thermal model for all operational modes. This analytical model Includes the effects of worst case combinations of equipment operation, internal heating, space vehicle orientation, solar radiation, eclipse conditions, ascent heating, and degradation of thermal surfaces during the life of the mission. The analytical model used in this prediction is usually validated by a space vehicle thermal balance test under the worst case operational modes. An appropriate margin for uncertainties is applied to the extreme component temperatures predicted by the analytical model, even after validation by a thermal balance test, to obtain the maximum and minimum predicted temperatures. This margin accounts for uncertainties in parameters such as complicated view factors, surface properties. contamination. radiation environment, joint conduction. and inadequate ground simulation. Because of these uncertainties. an uncertainty margin (see 3.45) of at least 11 deg C is included in all cases in determining the maximum or minimum predicted temperatures for space vehicle components. This 11 deg C thermal margin is applied to the temperature predictions made after the qualification thermal balance test. This implies that even larger thermal margins are required at the beginning of a program to accommodate changes that typically evolve from preliminary design to the final product.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

3.45 THERMAL UNCERTAINTY MARGIN

A thermal uncertainty margin is Included in the thermal analysis of space vehicles to account for uncertainties In parameters such as complicated view factors, surface properties, contamination, radiation environments, joint conduction, and inadequate ground simulation. For components that have no thermal control, or have passive thermal control, the maximum predicted component temperatures should be at least 11 deg C above the maximum temperature estimated for each component based on measurements and analysis, and the minimum temperature should be at least 11 deg C below the minimum temperature estimated for each component based on measurements and analysis. The 11 deg C is the thermal uncertainty margin for the component. For active thermal control subsystems, a remaining control authority of at least 25 percent for either or both hot and cold limits is specified as the thermal uncertainty margin. It is used to provide a control margin equivalent to the 11 deg C uncertainty margin specified for passively controlled components. For example, if a 100 watt capacity proportional control heater is used, it should operate at 80 watts or less to maintain the component above the minimum predicted temperature. The duty cycle should be less than 80 percent for an on-off heater. A control authority margin In excess of 25 percent should be demonstrated in cases where an 11 deg C change in the analytically predicted component temperatures would cause the temperature of any part of the actively controlled component to exceed an acceptable temperature limit.

5.6.2 Rationale for Definition of Thermal Uncertainty Margin. Reasons for utilizing an uncertainty margin are discussed in Paragraphs 3.25 and 3.45 of MIL-STD-1540B. Comparison of temperature prediction with actual flight data for various spacecraft over the years shows that about 95 percent of flight temperature have been within ± 11 deg C of the values predicted by the analytical thermal model. Thus, the ± 11 deg C uncertainty margin has been shown by experience to be necessary in order to assure high confidence that flight temperatures will not exceed the maximum and minimum predicted component temperatures. For active thermally controlled components, a heater margin of 25 percent is specified in lieu of ± 11 deg C margin specified for passively controlled components. This margin is established on the basis of experience and is demonstrated in tests by monitoring the duty cycle of the heater. The Specified maximum duty cycle of 80 percent demonstrates that the heater system has the required margin.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

5.6.3 Guidance for Application of Thermal Uncertainty Margin. With respect to the 80 percent heater duty cycle, it should be recognized that when the thermostat set points are fixed at a level higher than the minimum design requirement, it may not be possible to demonstrate by test that the duty cycle is equal to or less than 80 percent. It would then be required to show by analyses of test data that the heater system meets the 80 percent requirement at the minimum design temperature. For example, a component heater might be selected with a controller set point 6 deg C higher than the minimum Specified temperature of 4 deg C for that item. Since it requires more heat to maintain the component at 10 deg C than would be required to maintain" it at the minimum design temperature of 4 deg C, the heater selected would have a higher duty cycle. In that case, a 92 percent duty cycle measured with the 10 deg C control set point might be shown by analytical means to have equal or greater capability than the 80 percent duty cycle design requirement for a set point of 4 deg C. As another example, a heater-protected component might never reach a cold enough temperature, because of other test constraints, to provide data regarding its duty cycle at the minimum heater control point. A component heater might be selected with a controller set point of 10 deg C, but test constraints limited testing to temperatures above 20 deg C. Since it requires less heat to maintain the component at 20 deg C than would be required to maintain it at 10 deg C, the heater selected would have a lower duty cycle. In that case, a 72 percent duty cycle measured at the minimum test temperature of 20 deg C might be shown by analytical means to have equal or greater capability than the 80 percent duty cycle design requirement at the 10 deg C control set point. The requirement for heater margins in excess of 25 percent (i.e., duty cycles of less than 80 percent) may apply where small capacity heaters are used or where an 11 deg C decrease in the minimum local environment may cause a heater with 25 percent margin to lose control authority. Typically, this may occur for an inboard located component which is exposed to small local temperature variations, or has a high conductance interface with the local environment, or both.

Guidelines recommended for the application of these margins to specific thermal control devices are presented below.

5.6.3.1 Self-Regulating Heaters. Self-regulating heaters using a fixed resistance element which exhibit a large variation in resistance with temperature (such as "auto trace" or positive coefficient thermistors) are to be treated as passive devices. In those cases, ± 11 deg C temperature margin should be used in determining the required system characteristics. Heater control systems utilizing variable resistance or other proportional controls should be treated as active control devices. These

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

should demonstrate or be analyzed to show that a 25 percent heater capacity margin exists at the minimum predicted temperature.

5.6.3.2 Heat Pipes. Thermal margins applicable to heat pipes should be demonstrated by tests which are to be conducted at both the component level (i.e., heat pipe only) and at the highest level of assembly practical (e.g., subsystem or space vehicle installation). The thermal margins are defined separately for constant conductance heat pipes (which are treated as passive control devices) and variable conductance heat pipes (which are treated as active control devices).

5.6.3.2.1 Constant Conductance Heat Pipes. The heat transport capacity demonstrated at the component level should be at least 125 percent of that required for the nominal predicted heat load at the maximum predicted temperature of the evaporator. The nominal heat load is defined as that predicted by the analytical model for the worst combination of operational modes, environments, and surface properties.

The thermal performance test, which is conducted at the highest assembly level practical, should demonstrate the ± 11 deg C margin as applicable to all passive devices and should also provide, if possible, the data to demonstrate that each pipe is functional at the system level acceptance test.

5.6.3.2.2 Variable Conductance Heat Pipes. The following guidelines apply to variable conductance heat pipes utilizing noncondensible gas reservoirs for temperature control. At the component level, the heat transport capacity should be the same as defined for constant conductance heat pipes in Paragraph 5.6.3.2.1. The reservoir and evaporator temperature may be adjusted as required to facilitate the simplest test procedure with the ambient environment available.

Thermal performance of the variable conductance heat pipe system should be demonstrated at the highest assembly level feasible. The applicable thermal margins are defined in the following three paragraphs.

5.6.3.2.2.1 Heat Rejection Margin. When 125 percent of the nominal predicted heat load is applied to the evaporator mounting plate, under the worst hot case simulated conditions, the plate temperature should be equal to or less than the maximum predicted temperature.

5.6.3.2.2.2 Variable Conductance Range. When 110 percent of the nominal predicted heat load is applied to the evaporator mounting plate, under the worst hot case simulated

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

conditions, the heat pipe should still possess variable conductance, as proven by the location of the gas or working fluid vapor interface within the condenser portion of the pipe.

5.6.3.2.2.3 Heat Pipe Turn-Off. For a heat pipe which has a reservoir with an active temperature control system, the heat pipe should be turned off, i.e., decoupled from the condenser by virtue of the gas (vapor) location, when the evaporator mounting plate temperature is at least 6 deg C or higher than the minimum predicted temperature. For a heat pipe with a passively controlled reservoir, the turn-off points should be at least 11 deg C higher than the minimum predicted temperature.

5.7 BURST PRESSURE, MAXIMUM PREDICTED OPERATING PRESSURE. AND PROOF PRESSURE

5.7.1 Standard Criteria. Contents of Paragraphs 3.4, 3.21, and 3.34 of MIL-STD-1540B (definitions of burst pressure, maximum predicted operating pressure, and proof pressure) are presented in order to show the interaction between these definitions.

3.4 BURST PRESSURE

The burst pressure is the maximum test pressure that pressurized components withstand without rupture to demonstrate the adequacy of the design In a qualification test. It Is equal to the product of the maximum expected operating pressure. burst pressure design factor, and a factor corresponding to the differences in material properties between test and design temperatures.

3.21 MAXIMUM PREDICTED OPERATING PRESSURE

The maximum predicted operating pressure is the working pressure applied to a component by the pressurizing system with the pressure regulators and relief valves at their upper operating limit Including the effects of temperature, transient peaks, and vehicle acceleration.

3.34 PROOF PRESSURE

The proof pressure is the test pressure that pressurized components can sustain without detrimental deformation. The proof pressure is used to give evidence of satisfactory workmanship and material quality, or to establish maximum possible flaw sizes. It is equal to the product of maximum expected operating pressure (see 3.21),

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

proof pressure design factor, and a factor accounting for the difference in material properties between test and design temperature.

5.7.2 Rationale for Pressure Definitions. A pressure vessel or pressurized structure should be tested to demonstrate that it has sufficient Strength, stiffness, and integrity to withstand the maximum pressure anticipated during its service life without any gross yielding, detrimental deformation, or leakage. This anticipated maximum pressure is called the maximum predicted operating pressure. It is synonymous with maximum expected operating pressure (MEOP) and is analogous to maximum anticipated load.

Safety and mission success dictate that a pressure vessel or pressurized structure be capable of withstanding, without rupture, a pressure exceeding its maximum predicted operating pressure by an amount determined by uncertainties associated with its design, materials used, and by the degree of hazard in the intended use. This pressure is called the burst pressure. It is also called the design burst pressure when it is determined by the ultimate strength of the vessel.

To ensure acceptable quality of workmanship and general flightworthiness, acceptance tests are conducted on every pressure vessel and pressurized structure, including main propellant tanks and solid rocket motor cases. The pressure level for this test is called the proof pressure.

5.7.3 Guidance for Application of Pressure Terms. Structural adequacy of the design of pressure vessels and pressurized structures is demonstrated by qualification tests conducted on full-scale flight-quality hardware. Qualification test requirement include a test of one article of each pressure vessel design to burst pressure level. Burst pressure test requirements are given in Paragraph 6.4.10 of MIL-STD-1540B. Requirement for proof pressure tests, required for acceptance of every pressure vessel and pressurized structure, also are provided in Paragraph 6.4.10.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 6

GENERAL REQUIREMENTS

6.1 REQUIREMENTS FOR REUSABLE FLIGHT HARDWARE TESTING

6.1.1 Standard Criteria. Contents of Paragraph 4.1.5.2 of MIL-STD-1540B (requirements for reusable space vehicle hardware) are as follows:

4.1.5.2 Reusable Flight Hardware Testing

Reusable space vehicle hardware consists of the space vehicles and components Intended for repeated space missions. Airborne support equipment and space vehicles which perform their missions while attached to a recoverable launch vehicle are candidates for reuse, particularly for multiple mission programs. The reusable equipment would be subjected to repeated exposure to test, launch. flight, and recovery environments throughout its service life (see 3.37). The accumulated exposure the of space vehicles retained in the recoverable launch vehicle and of airborne support equipment is a function of the planned number of missions Involving this equipment and the retest requirements between missions. Airborne support equipment environmental exposure time is further dependent on whether or not its use is required during the acceptance testing of each space vehicle. In any case, the service life of reusable hardware should include all planned reuses and all planned retesting between uses.

The testing requirements for reusable space hardware after the completion of a mission and prior to its reuse on a subsequent mission depends heavily upon the design of the reusable item and the allowable program risk. For those reasons. specific details are not presented in this standard. Similarly. orbiting space vehicles that have completed their useful life spans may be retrieved by means of a recoverable launch vehicle, refurbished, and reused. Until some Insight is provided by experience as to the extensiveness of required refurbishment, detailed test guidelines cannot be provided. Based on present approaches, it is expected that the retrieved space vehicle would be returned to the contractor's factory for disassembly. physical Inspection, and refurbishment. All originally specified acceptance tests should be conducted before reuse,

6.1.2 Rationale for Reusable Flight Hardware Requirements. The advent of the STS has brought with it the concept of reusable flight equipment, which is novel to many in the space

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

vehicle community. While this concept is certainly not foreign to the aircraft industry, space vehicle technology has fashioned its own techniques in this area. The STS shuttle vehicle faced this problem in the design and test of a reusable launch vehicle, Most elements of the shuttle vehicle were designed to fly 100 missions.

Most space vehicles launched by a recoverable launch vehicle utilize airborne support equipment (ASE), which remains with the launch vehicle after space vehicle deployment. Some payloads are not deployed but perform their mission within the recoverable launch vehicle, return to earth with that launch vehicle, and fly additional missions. Such multiple mission equipment requires that special attention be given to qualification and acceptance test requirements.

6.1.3 Guidance for Use of Reusable Flight Hardware Requirements. For reusable flight equipment, it is useful to distinguish between environmental tests that are influenced by mission exposure duration and those that are not. Acoustic and vibration tests fall into the former group, while the latter is exemplified by thermal vacuum, thermal cycling, acceleration, EMC, humidity, and leak tests. It is important to note that reentry, while not normally a mission phase for single-use flight equipment, may impose a set of environmental test conditions for reusable flight equipment. An example would be the inclusion of reentry deacceleration in an acceleration test of a payload intended for multiple missions. The qualification test requirements for reusable flight equipment can be derived by the logical extension of the methodology contained in MIL-STD-1540B.

6.1.3.1 Vibration Qualification Tests. The vibration qualification durations required by MIL-STD-1540 are the greater of: three times the expected flight exposure time or three times acceptance test time, but not less than three minutes per axis. The expected flight exposure to maximum vibration levels for a single use component flown on an expendable launch vehicle is usually less than one minute, resulting in a qualification test duration of three minutes. MIL-STD-1540B also requires a vibration qualification margin of 6 dB above the acceptance test level. The longer duration at the higher level for the qualification test allows a prediction of t_A, which is the time it would take a flight article exposed to the lower flight or acceptance test levels to reach an equivalent fatigue damage as the qualification specimen. The following formula for t_A has been adopted by a number of space contractors.

$$t_A = (t_Q)(2)^{(a/6)(M-S-K)}$$

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

where

t_A = acceptance test time plus flight level exposure duration resulting in fatigue damage equivalent to damage accumulated during qualification test duration

t_Q = vibration qualification test duration

a = inverse slope of stress versus number of cycles fatigue curve for the most fatigue-critical material in the test article

M = margin between qualification and acceptance vibration inputs in decibels

s = 2, if the qualification and acceptance test hardware were fabricated about the same time and 3, if the qualification and acceptance test hardware were fabricated several years apart (and therefore, might not be uniform or identical)

K = a number ranging between 0.6 and 2.0, in accordance with Table I

TABLE I. Value of "K"

Test Tolerance in dB	"K" if Vibration Fixture Used in Qualification and Acceptance Are The Same	Are Different
1.5	0.6	1.2
2.0	0.75	1.5
3.0 (nominal)	1.0	2.0

This equation for t_A is considered more Comprehensive of fatigue equivalence considerations than simpler expressions sometimes-used which do not include the parameters S and K. For preliminary guidance, values of S ranging from 2 to 3 and K from 0.6 to 2.0 are suggested. If other data are available to refine these parameters for specific fatigue evaluations, alternate values for S and K should be used.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

Using this formula for nominal MIL-STD-1540B conditions, it can be shown that three minutes at qualification test levels, with a margin of 6 dB above acceptance test levels, for an "a" value of 6, gives a t_A of approximately 24 minutes. That means that 24 minutes of flight article exposure to acceptance test levels would result in fatigue damage equivalent to that produced by the vibration qualification test; i.e., for:

t_Q = 3 minutes (MIL-STD-1540B duration)

a = about 6 for 2024-T3 Aluminum. (6 can be assumed as a likely average, but requires specific evaluation of each component)

M = 6 dB (MIL-STD-1540B standard margin)

S = 2 (assumes qualification and acceptance hardware manufactured at the same time)

K = 1 (assumes same vibration fixture for qualification and acceptance hardware)

One derives

$$t_A(\text{MIL-STD-1540B}) = (3)(2)^{(6/6)(6-2-1)} = 24 \text{ minutes}$$

Allowing 1 minute for flight and 1 minute for acceptance testing, the 24 minutes for t_A would seem to allow the remaining 22 minutes for reacceptance testing. Due to unit-to-unit variations between the qualification article and the flight hardware, it is considered unrealistic to allow the use of the full 24 minutes for acceptance testing, reacceptance testing, and flight of the hardware. An acceptance test exposure time of 1/2 t_A is considered relatively conservative. This would allow approximately 12 minutes of flight article exposure to acceptance test (flight) levels, or a total of 10 reacceptance tests for nominal MIL-STD-1540B conditions.

Note that a reduction in the qualification margin from 6 dB to 3 dB has a drastic effect on the equivalent damage time, t_A. With the other conditions the same as used in the previous case, t_A for a 3 dB margin would be as follows:

$$t_A(3 \text{ dB margin}) = (3)(2)^{(6/6)(3-2-1)} = 3 \text{ minutes}$$

For this situation, the qualifying note regarding the parameters S and K should be kept in mind.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

The vibration qualification test duration for reusable flight hardware should be based on a similar fatigue-related rationale that would cover the planned acceptance testing, reacceptance testing, and flight time. Examples are provided for two situations involving reusable flight hardware for a payload intended to be flown six times.

6.1.3.1.1 Example I: Vibration Qualification Test with No Reacceptance Testing Between Flights. For this example, it is assumed that the flight hardware acceptance test is planned to be conducted just once prior to the initial flight and that reacceptance testing will not be required prior to each of the five subsequent flights. The flight time exposure information is usually provided in the launch vehicle interface document. For this example, the exposure to the maximum predicted vibration levels is estimated as a 10-second exposure at liftoff of the launch vehicle, a 20-second exposure during transonic and high dynamic pressure ascent flight, plus a 20-second exposure during reentry aeronoise. Because the total exposure during each flight is only 50 seconds (less than one minute), the standard vibration acceptance test duration would be one minute. If it is assumed that circumstances, i.e., retest after repair, dictate the equivalent of five acceptance tests prior to the initial flight, the acceptance test time would total five minutes (300 seconds). The total exposure time of the flight hardware would then be the total acceptance test time plus total flight time, or 300 seconds plus 300 seconds (for six flights), or 10 minutes total. The qualification test used should, therefore, be based on a fatigue-related rationale that would provide a tA (Example I) of greater than 10 minutes.

The vibration qualification test required by MIL-STD-1540B that uses the 6-dB qualification margin and the 3-minute duration would seem to satisfy this required tA (Exampl~ I) of 10 minutes, since it was shown that 24 minutes of exposure to acceptance test levels are required to equal the fatigue exposure experienced by the qualification article. A 3-dB qualification margin would, however, not meet the criteria, since it would only allow a tA of three minutes for equivalent fatigue damage.

6.1.3.1.2 Example II: Vibration Qualification Test with Reacceptance Testing Between Flights. For this example, it is assumed that the payload must undergo significant refurbishment between flights, making reacceptance testing prior to each flight advisable. It is assumed that the flight hardware averages three vibration acceptance tests per flight for a total of 18 minutes for six flights. As in Example 1, the flight time exposure to the maximum predicted vibration levels is 5 minutes

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

for six flights, so the total exposure time to the maximum predicted vibration levels is 23 minutes. The qualification test used should, therefore, be based on a fatigue-related rationale that would provide a t_A (Example II) of greater than 23 minutes. While this duration is less than the 24 minutes of equivalent fatigue exposure as the qualification specimen, it is considered too marginal. As indicated previously, it is recommended that the fatigue exposure of the flight article be approximately one-half that of the qualification specimen. If the qualification test duration is increased to 5 minutes, the formula for equivalent fatigue damage due to acceptance test levels gives t_A(Example H) as follows:

$$t_A(\text{Example II}) = (t_Q)(2)^{(6/6)(M-S-K)}$$

$$= (5)(2^3)$$

$$= 40 \text{ minutes}$$

Since the planned exposure time of 23 minutes is approximately half this 40-minute equivalent fatigue damage time, a 5-minute qualification test duration would be satisfactory for Example II.

6.1.3.1.3 <u>General Vibration Qualification Test Requirements.</u> In summary, the required vibration qualification test duration t_Q for reusable flight hardware can be determined as follows:

Let

n_{AT} = number of vibration acceptance tests planned

t_{AT} = acceptance test duration

k_2 = acceptance test multiplication factor to account for repeated tests

n_F = number of flights

t_F = flight duration

k_1 = fatigue damage exposure margin relative to qualification article (typically 2)

Then the required t_A is given by

$$t_A = (k_1)(k_2)(n_{AT})(t_{AT}) + (k_1)(n_F)(t_F)$$

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

Or the required t_Q is derived from

$$t_A = (t_Q)(2)^{(a/6)(M-S-K)}$$

Therefore

$$t_Q = \frac{t_A}{(2)^{(a/6)(M-S-K)}}$$

By substitution, the required vibration qualification test duration t_Q is

$$t_Q = \frac{(k_1)(k_2)(n_{AT})(t_{AT}) + (k_1)(n_F)(t_F)}{(2)^{(a/6)(M-S-K)}}$$

The required vibration qualification test duration t_Q for reusable flight hardware is, therefore, dependent on a number of variables. Aside from requiring a knowledge of acceptance test and flight duration, it is necessary to make judgments on (a) how much margin to allow between the fatigue damage experienced by the qualification article and the flight articles and (b) how many unplanned acceptance tests (retests) might be required.

6.1.3.2 Shock Qualification Tests. MIL-STD-1540B shock test requirements are primarily based on pyrotechnic shock events . For qualification, a shock level which is 6 dB above the maximum predicted (acceptance level) environment for flight duration is required. The number of required shocks is three times in each of three axes, for a total of 18 shocks.

MIL-STD-1540B acceptance shock tests are conducted at the maximum predicted level. One shock is required in each direction of each of three axes, for a total of six shocks.

It is recommended that the required number of shock events to be used during the shock qualification tests to qualify reusable flight hardware, N_{QS} , be determined based on three times the number of shock events experienced by the flight hardware, as follows:

$$N_{QS} = 3(N_{ATS} + N_{FS})$$

where

N_{QS} = number of qualification shocks at 6 dB above maximum predicted level

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

N_{ATS} number of acceptance test shocks experienced by the flight hardware at maximum predicted level

N_{FS} = number of flight shocks experienced by the flight hardware

The factor of three times the number of flight plus test events experienced by the flight article is based on the general philosophy for qualifying flight hardware used in MIL-STD-1540, where a factor of three is used for qualification relative to flight level exposure (acceptance test level). The shock level for qualification should be 6 dB above the maximum predicted environment (6 dB above the acceptance test level). In summary, the qualification shock test for reusable flight hardware is based on the MIL-STD-1540 philosophy that the qualification test should demonstrate that the flight hardware can withstand three times the number of anticipated flight and test events with a 6 dB margin to account for the variability of the hardware.

6.2 TAILORING

6.2.1 Standard Criteria. Contents of Paragraph 4.2.3 of MIL-STD-1540B (requirements for test tailoring) are as follows:

4.2.3 Tailoring

This standard specifies test requirements that have been shown to assure high reliability in space missions. However. it is intended that these test baselines should be tailored to each space program considering design complexity, state of the art, mission criticality, cost, and acceptable risk. For some space programs this tailoring may relax the requirements in this standard, while for other programs the requirements may be made more stringent to reduce risk of on-orbit failures or to demonstrate with greater confidence that the space vehicle or components perform adequately when all parameters, environments, and related uncertainties are considered. For example, the optional tests shown in the test baseline tables should be added as required tests, where appropriate, as determined from considerations of design features, required lifetime, environmental exposure, and expected usage. The tailoring is a continuing process throughout the acquisition that should be Implemented by the wording used to state the testing requirements In the specifications of the space" system, space vehicles, and components or in other applicable contractual documents.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

6.2.2 Rationale for Tailoring Requirement. Individuals who are familiar with MIL-STD-810D should note that "tailoring" is used in MIL-STD-1540B, and in this handbook, in a narrower sense than it is used in MIL-STD-810D. In both documents, the intent of the tailoring requirement is to impose only the minimum design and test requirements needed to assure that the items produced will meet the range of environments that could be encountered during the actual life cycle of the items. For example, the definitions in MIL-STD-1540B require analysis and actions that are equivalent in many ways to the initial tailoring steps required when MIL-STD-810D is used.

For most military systems, the testing and maintenance costs represent major elements of the life cycle cost. Unlike aircraft programs where the testing and maintenance costs are primarily incurred during operational use, the testing costs for spacecraft are primarily incurred prior to operational deployment since on-orbit maintenance is seldom possible. Because testing represents such a large expense, good management requires tailoring of the test program to assure that a cost-effective program is achieved. On one hand, any excessive testing clearly represents a waste of money and time. On the other hand, an undetected deficiency or failure can result in an unsuccessful launch or shortened on-orbit life. Because a single failure can result in a loss of several 100 million dollars, not including the loss of scientific or operational data, a considerable budget for quality control, and for testing that will ensure spacecraft success, is usually cost-effective. Successful space vehicles have been launched even though their procurement documentation contains only sketchy or limited quality assurance provisions. Conversely, programs can be found where extensive inspections and tests at every step of the acquisition process still resulted in unsuccessful missions. However, the preponderance of evidence is, as expected, that the use of extensive testing and other quality assurance provisions that are based upon those used for previously successful programs is the only cost-effective approach. For high reliability spacecraft, the testing costs may represent as much as 40 percent of the life cycle cost.

MIL-STD-1540B, therefore, was prepared from a composite of the tests that had been used by contractor to achieve successful space missions. The test baselines presented include the need for assurance in the areas of performance, safety, and reliability. The standard includes requirements for development, qualification, acceptance, and prelaunch validation tests. The acceptance tests are intended to assure. to the maximum extent possible, that all space vehicle equipment will operate through various operational modes while exposed to the maximum predicted environments. The test duration at each

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

environment is to be equivalent to the expected time duration associated With the maximum environmental levels. The qualification tests typically require that the space vehicle equipment operate through various operational modes while exposed to design environments that include the environmental design margin. The environmental design margins provide a 6-dB or 10-deg C margin over the acceptance test levels. The qualification test duration at each environment is typically three times the acceptance test duration. To provide a basis for standardization of components, a minimum design range is specified. The design range is also the qualification test environment. For example, the minimum thermal range for component design, and hence for qualification tests, is from -34 deg C to +71 deg C. The minimum overall random vibration design level, and hence the minimum qualification test level, is 12 grins for 3 minutes. The minimum acoustic design level, and hence the minimum qualification test level, is 144 dB overall for 3 minutes. The minimum acceleration design level, and hence the minimum qualification test level, is 20 g. These minimum design levels, and hence qualification test levels, represent environmental levels commonly found on most spacecraft; however, they are not so severe as to cause design problems for most components. By designing and qualifying the components to these common requirements, a prudent minimum level of design ruggedness is provided. In addition, the components might be relocated on a spacecraft or might be used on other spacecraft without redesign or requalification. These minimum design ranges also assure an effective acceptance stress screening test for all components.

MIL-STD-1540B, therefore, establishes a uniform set of definitions and general ground testing requirements for space vehicles. It is intended that these baseline requirements will be tailored to fit each program’s requirements while recognizing the desire to meet the minimum standard requirements where practicable. MIL-STD-1540B provides a common framework from which program managers can identify and evaluate deviations in their testing and quality assurance plans. The extent of acceptable deviations is a tradeoff among program requirements, acceptable risk, and testing costs including schedule delays. Because the cost-effectiveness of these tradeoffs is difficult to evaluate statistically due to the small sample size for each program, an evaluation of the deviations from MIL-STD-1540B should be included in all program reviews.

6.2.3 Guidance for Use of Tailoring Requirement. Like many standardization documents, MIL-STD-1540B is structured to assist in the tailoring process. A few of the ways that MIL-STD-1540B and the requirements associated with testing can be tailored are discussed in the following paragraphs.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

6.2.3.1 By Limiting. For some applications, portions of the entire test program given in MIL-STD-1540B are not applicable, and exceptions can be included with the compliance statement. For example, the definitions, the component level tests, and the prelaunch validation tests may all be applicable to a launch vehicle program. In that case, the acceptance test requirement statement might read: "Except for the vehicle level testing, the acceptance tests shall be performed in accordance with MIL-STD-1540B." The details of launch vehicle acceptance test requirements would then be stated separately.

One of the provisions of MIL-STD-1540B that has provided some requests for deviation is the 6-dB qualification test margin. Of this 6-dB margin, 3 dB are provided to accommodate the maximum allowed testing tolerance. Provisions are incorporated in the definition of environmental design margin, Paragraph 3.12 of MIL-STD-1540B, to automatically reduce this 3-dB test tolerance portion in accordance with the actual testing tolerances used by the contractor. In other words, should the contractor choose to spend more time adjusting test levels to closer tolerances, then the 6 dB may be automatically reduced accordingly without a formal deviation being required. The definition of environmental design margin in MIL-STD-1540B also suggests circumstances where the remaining 3-dB margin might be changed. For example, if it is judged that a 2-dB allowance would provide an acceptable reliability risk, the component design requirement might be 5 dB (instead of 6 dB).

6.2.3.2 By Supplying Other Detailed Requirements. Many items that are pertinent to the test requirements are omitted from MIL-STD-1540B to make it directly applicable to a wide variety of space vehicle programs without deviation. These items include:

a. Maximum predicted environmental levels

b. Time duration for exposure to the maximum predicted environments

c. Transportation, handling, and storage environment

d. Required development tests

e. Applicable safety standards

f. Test sequence to be followed at each level of assembly (The test sequences given in MIL-STD-1540B are only suggested sequences.)

g. Tests identified as options in the standard that are required on the program

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

h. Component level tests that can be satisfied or accomplished by tests at higher levels of assembly

i. Prelaunch validation test flow

j. Requirements for development-test vehicles

k. Requirements for qualification test vehicles

l. Detailed test plans and procedures

m. Retest Requirements after modification, change, or repair to the hardware. computer programs. or test configuration

n. Acceptable basis of flight certification of all flight hardware (by qualification tests or by analysis of previous hardware usage)

By the inclusion of pertinent data on each of these items in the appropriate section of the specification or contract, the actual test requirements for the item are established or can be determined by analysis. For example, the location of components on a space vehicle, the specific orbit, the equipment duty cycle, and other design factors would be used in a thermal model to calculate the maximum and minimum predicted temperatures. If the maximum predicted temperature for a component were +71 deg C, then an acceptance test would be conducted using a maximum temperature of +71 deg C, and a qualification test would be conducted using a maximum temperature of +81 deg C. If the maximum predicted temperature for another component were +49 deg C, then the acceptance test would be conducted using a maximum temperature of +61 deg C, and the qualification test would be conducted using a maximum temperature of +71 deg C (see Paragraph 3.9 in MIL-STD-1540B). Therefore, it is clear that items not directly Stated in the specification or in MIL-STD-1540B influence the actual testing. Of course, the definitions of the maximum and minimum predicted temperatures are given in MIL-STD-1540B to ensure a uniform determination of its value, including an allowance for uncertainties (see Paragraph 3.25 in MIL-STD-1540B). In much the same way, the inclusion of related data in the specification for all the items listed above can modify the actual test requirements without modifying or deviating from the general requirements in MIL-STD-1540B.

6.2.3.3 By Supplementing Requirements, In some cases, it is clear to the government program office that the requirements stated in MIL-STD-1540B are inappropriate for a specific item on

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

that program. For example, a battery may not be able to withstand the -34 deg C to +71 deg C nominal design temperature range given in MIL-STD-1540B for components. For that program, it had been determined that active thermal controls would be incorporated to maintain the battery temperature between +10 deg C and +30 deg C. In that case, the specification requirement for the battery might read: “The battery shall be designed to operate over a temperature range of O deg C to +40 deg C.” This is a realistic thermal range for a battery, 60 the qualification test range would then be O deg C to +40 deg C. The companion acceptance test range would be +10 deg C to +30 deg C.

6.2.3.4 By Contractor’s Choice. Many items in MIL-STD-1540B are stated in ways that allow the contractor to select the most cost-effective approach. Some of these, such as the selection of test tolerances and test sequences, have been mentioned above. Another example is the method of flight qualification. Flight qualification of components can always be accomplished by qualification testing at the component level; however, for many items such as fluid lines, wiring harnesses, and structural components, testing at higher levels of assembly is usually cost-effective. In addition, previously used devices may sometimes be qualified by less costly approaches such as by equipment similarity or analysis. By allowing the contractor to select the appropriate approach for each item, repetitious or unnecessary qualification testing may be avoided. Of course, the contracting officer normally reserves the right to review and approve the adequacy of the flight qualification effort. This is usually accomplished by the required approval of test plans, test procedures, or data submitted as data items under the terms of the contract.

6.2.3.5 By Limiting Data Items. MIL-STD-1540B implies that a large number of associated documents will be prepared by the contractor. These include:

a. Detailed test plans and test sequences for items at the various levels of assembly

b. Detailed test procedures for all tests including the pass-fail test criteria for each test

CO Test records

d. A data bank to provide traceability of test data, the accumulation of trend data on critical parameters, a record of all test discrepancies, and a record of their disposition

e. Development test reports.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

f. Flight certification lists .

g. Qualification test reports

h. System safety plan

i. System failure mode and effects analysis to determine critical parameters

j. Operational time line to establish functional modes and requirements for the programmed orbit mission tests

k. Transportation and handling plan

Unless a report or data item is identified on the Contract Data Requirement List, such as DD Form 1423, AFSC Form 708, or AFSC Form 709, it will not be submitted to the contracting officer for review or approval. Of course, to save effort, only those reports or data items absolutely required to determine compliance with the program requirements should be requested (listed). The actual tests that will be conducted may be influenced greatly by whether the test procedures and other associated documents must be reviewed or approved by the contracting officer. The data item list can therefore modify to some degree the extent of the testing and the total contractor effort.

6.3 RETEST

6.3.1 Standard Criteria. Contents of Paragraph 4.3 of MIL-STD-1540B (requirements for retest) are as follows:

4.3 RETEST

If a test discrepancy (see 3.42) occurs, the test should be Interrupted and the discrepancy verified. If the discrepancy is dispositioned as due to the test setup. software. or to a failure in the test equipment. the test being conducted at the time of the failure may be continued after the repairs are completed. as long as the discrepancy did not result in an overstress test condition. If the discrepancy is dispositioned as a failure In the item under test, the preliminary failure analysis and appropriate corrective action shall normally be completed before testing is resumed. If the failure occurs during system testing, the test may be continued if the discrepant area is not affected by the continuation of testing.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

The conducting of a proper failure analysis plays an important part in the decision on the type of retest. It should Include the determination of whether a failure occurred, the cause of the failure, the physics of the failure, and isolatlon of the failure to the smallest replaceable item. The degree of retest shall be determined for each case based upon the nature of the failure. In the case of d Significant redesign of a component, all previous qualification tests shall be repeated. After significant component rework, all previous acceptance tests except burn-in shall be repeated. In the case of extensive component rework, repetition of the burn-in is also required. Where the redesign or rework of the component is very minor, it may be acceptable to only repeat functional testing and the test in which the failure occurred.

Where significant redesign or rework of components is required as the result of failure during system level testing, the system level test in which the failure occurred, as well as functional testing of the failed subsystem, shall be repeated. Repetition of system level environmental tests may be necessary if the redesign was extensive or the number of components changed out and connecters demated is so large as to reduce confidence in the space vehicle.

6.3.2 General Rationale for Retest Requirements. Retest is the repeat of previously conducted tests due to a test discrepancy or other factors related to the items previously tested.

Discrepancies may occur at any point in the qualification or acceptance test sequences of space vehicle systems or components . When a discrepancy occurs, a failure analysis is conducted to determine the cause of failure and to determine if there are any generic or lot-related problems that could affect other vehicles. If it is determined that the item being tested failed, it is important to try to determine why the failure occurred at that point in the test sequence. In other words, Are there deficiencies in the tests at lower levels of assembly that allowed the defect to go undetected? If the failed item is a qualification or flight article, a decision must be made as to whether repeats of previous tests (retests) or special tests are required after correction. If the space vehicle or components have been redesigned or reworked to correct the failure, tests conducted prior to the failure might require repetition to

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

verify the adequacy of the corrective action. Basically, retest requirements after failures of qualification or flight articles during testing depend on the nature of the failure, the point of occurrence in the test program, the degree of redesign and rework required for repair, criticality of equipment, and other factors. The Criteria specified in MIL-STD-1540B are subject to judgment. Their major purpose is to establish ground rules for such judgment and to aid in the preplanning of minimum retest criteria for specific space and launch vehicles. This is particularly applicable to system level retest requirements.

When a component is removed from a vehicle or a vehicle connector is broken. verification of vehicle flightworthiness is required subsequent to the replacement. While actual retest requirements are usually determined by Material Review Board disposition, the preplanning of retests can avoid unplanned emergency actions.

6.3.3 General Guidance for Use of Retest Requirements. The requirements of MIL-STD-1540B are illustrated in Figure 1 for specific actions immediately following a test discrepancy.

The test is interrupted and a determination is made as to whether the discrepancy is due to a failure of the item under test or a failure of the system performing the test (test setup, software, or equipment). Even if the item under test did not initially fail, it is possible that it could have been overstressed by a failure of the test equipment. After a determination is made that no overstress of the test item exists, the test may be continued after repairs of the system performing the tests are completed. If the test item has failed, either originally or due to overstress, test activities resume normally Only after a preliminary failure analysis which determines the cause and corrective action. Final failure analysis is shown as a continuing function to indicate that initial evaluations are sometimes inconclusive and that further action may be required, particularly if the failure represents a generic or lot-related problem. For long-term corrective action, one should determine if the failure could have been, and therefore should have been, detected at a lower level of assembly or in an earlier test. If that is the case, be sure to document all corrective actions that are appropriate at each level of assembly, including all changes in test procedures.

The results of failure analysis play an important part in the decision on the degree of retest. If the test item had to be redesigned or reworked extensively, repetition of many of the previously conducted tests might be necessary to restore confidence in the functional and environmental performance of

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

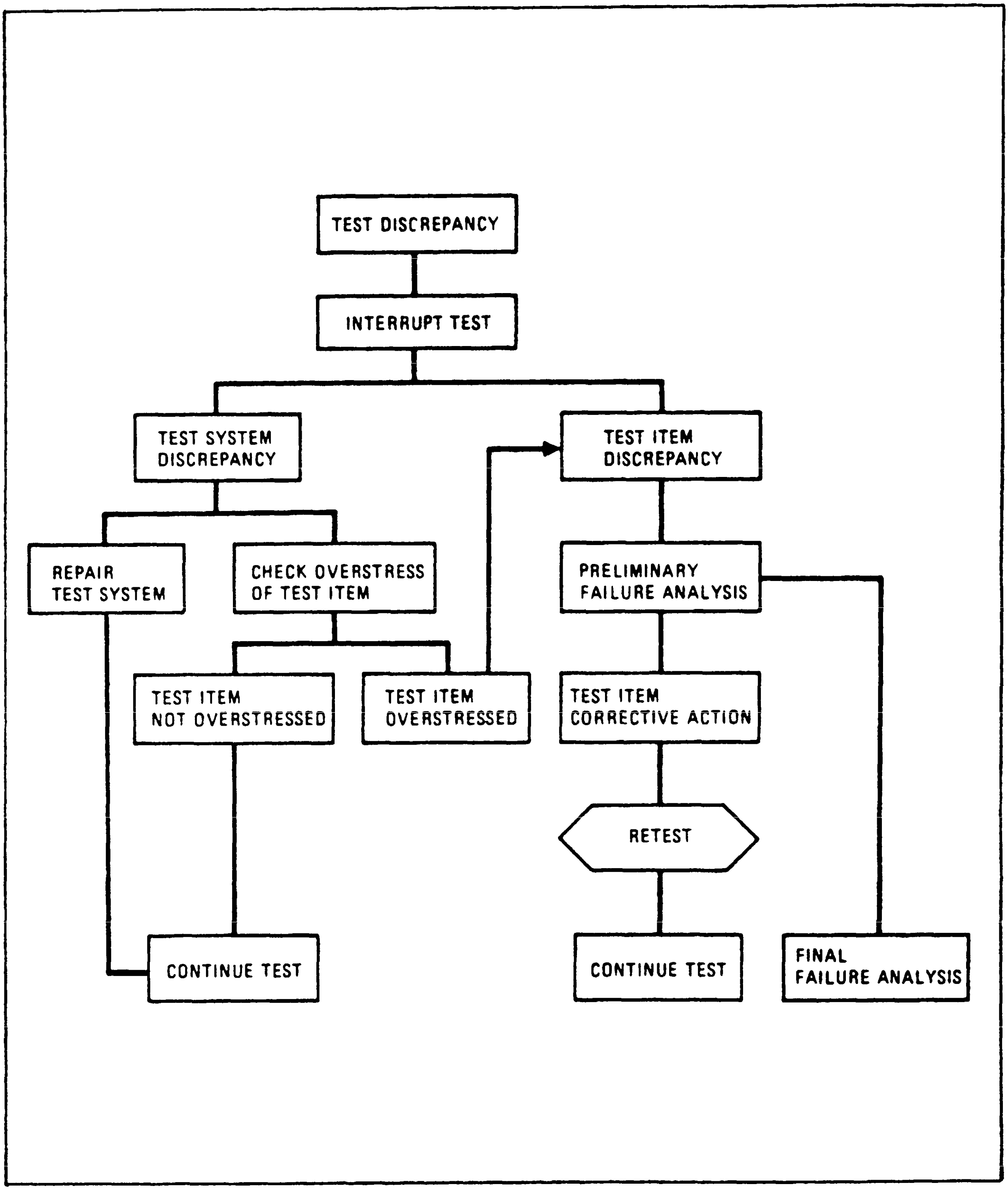

FIGURE 1. Activity After Test Discrepancy.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

the item. The following summarizes the retest requirements specified in MIL-STD-1540B:

- Component major redesign
  - Repeat all previous qualification tests.
- Component significant rework
  - Repeat all previous acceptance tests except for burn-in.
- Component extensive rework
  - Repeat all acceptance tests plus burn-in test.
- Component minor redesign or rework
  - It may be acceptable to only repeat functional testing and the test in which the failure occurred.
- Component failure during system level test
  - Retest components per previously stated ground rules.
  - Repeat system level test in which failure occurred plus functional test of failed subsystem.
- System major redesign and rework
  - Repeat system level environmental tests.

Maximum confidence in the integrity of a redesigned or repaired test article following corrective action exists if all previous tests are repeated. Since this is often costly, time-consuming, and impractical, compromises must be made on the degree of retesting. The degree of retest should be evaluated for each case considering the nature of the failure, the degree of redesign and rework required, and whether any previous tests could possibly have induced the failure or were invalidated by the corrective action. The decision therefore becomes a judgment on the amount of acceptable risk.

Different guidelines have been developed for component retests and for space vehicle (system) retests because of the nature of the design. In general, most components within a space vehicle are installed to be removable and replaceable.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

The retest of a space vehicle after component removal and replacement can therefore be preplanned, depending on the point in the test sequence when a removal and replacement occurred. The retest of a component if a failure occurs during a component test is more unpredictable, since the parts and hardware for the component often are not designed for ease of removal and replacement.

The standard criteria for retests have been categorized with respect to the corrective action. Note that the corrective action is either redesign or rework. The degree of redesign and rework plus the effect of the corrective action on previous tests become the major drivers in the retest decision.

An anomaly requiring redesign as a corrective action would typically occur during qualification article testing. This is based on the rationale that this testing precedes the acceptance testing of flight articles and, therefore, the majority of design problems will be discovered during this phase. A redesign may be classified as “major” or “minor.”

An anomaly causing rework as a corrective action may occur during any type of testing. The rework may be caused by implementation of a redesign or by a repair which does not change the design. The rework may be significant or relatively minor. A significant rework may invalidate a number of previously conducted tests. A minor rework may have relatively small effect on the validity of previous tests. It is the purpose of this discussion to provide some considerations leading to judgments on the significance of reworks.

6.3.3.1 Component Retests. For component test activities, Figure 2 depicts the sequence of events after a component discrepancy during a test has been verified. If at all possible, it is desirable to freeze the hardware and software in the discrepant mode to allow a determination of failure cause. It is recognized that complete failure analysis can be lengthy, and that often tests must be continued before failure analysis can be completed. A preliminary failure analysis can be conducted to determine whether test continuation is practical or whether the test must be aborted. Factrs entering into this decision are ease of isolation, ease of repair, and feasibility of continuing the test without repairing the discrepancy. Such a situation might exist where redundancy exists within a component and the test could be continued on the redundant leg. An additional reason for test continuation without repair would be the need to troubleshoot and isolate the failed hardware or parts by test. After the failed hardware is isolated, the component is redesigned or repaired. In either case, the degree of component rework governs the amount of

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

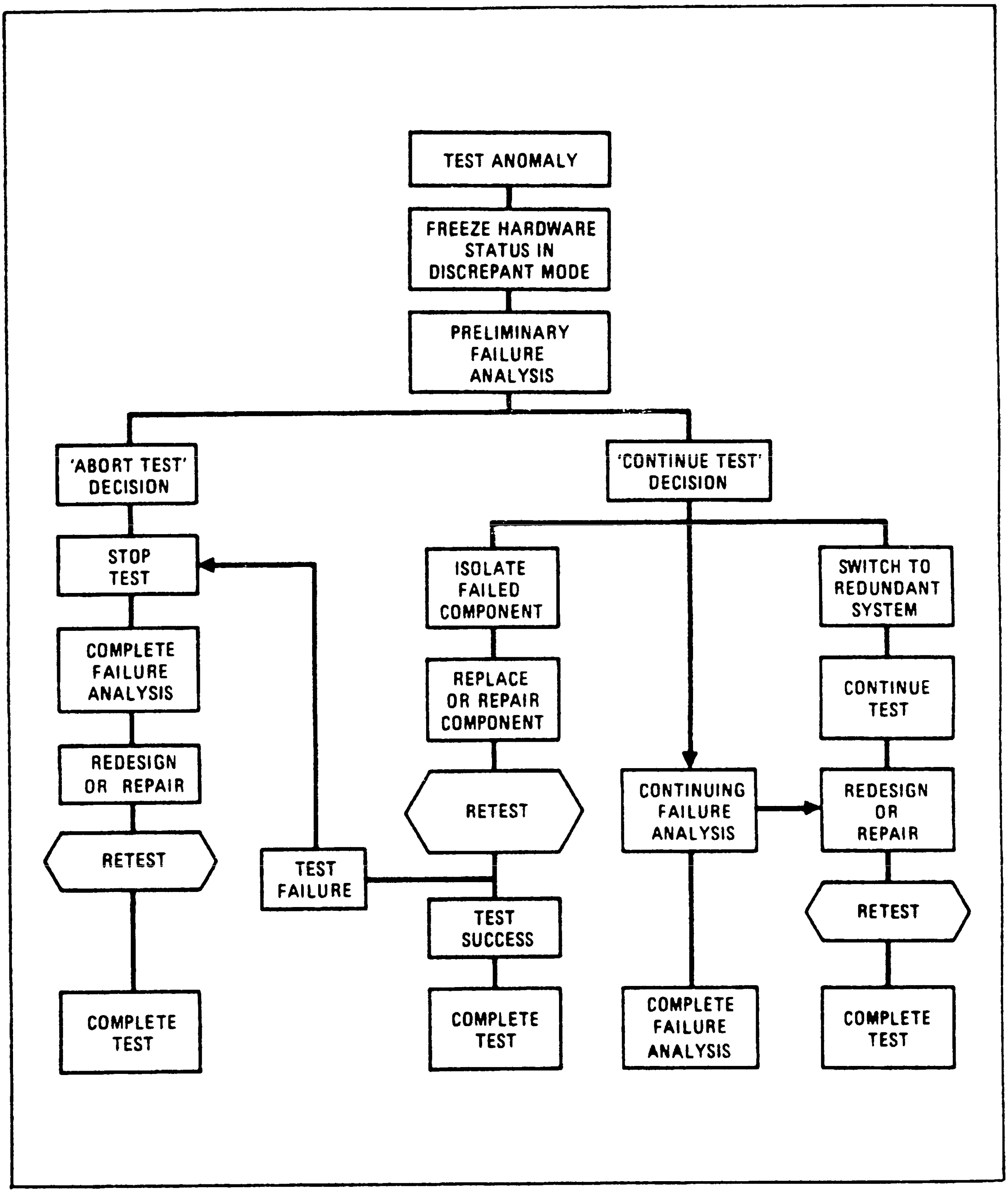

FIGURE 2. Component Test Activity After Discrepancy.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

retest necessary. If a defective part or subassembly can be replaced by simply disconnecting and reconnecting electrical connectors using plugs or pins, retests may be minimized. However, component rework generally results in considerable uncertainty regarding the validity of previous tests, and considerable retest is necessary to keep risks acceptable.

6.3.3.2 Space Vehicle Retests. Figure 3 depicts the activity after an anomaly has been discovered during a test of a space vehicle. As with component test discrepancies, it is desirable to freeze the hardware and software in the discrepant mode. After a preliminary failure analysis to determine the safety or hazard of continuing the test, a “continue” or “abort” test decision can be made.

For the “abort” test decision, the preliminary failure analysis may have revealed that the test results are too uncertain for continuation and that the system requires extensive redesign or rework. The test is therefore stopped and a more detailed failure analysis is completed to determine the exact cause and rework required. After completion of the rework, applicable retests are performed and the test is completed.

For the “continue” test decision, a failed item can readily be isolated and quickly replaced or repaired. If a retest shows that the replacement or repair is successful, the test may be completed. If unsuccessful, the activity will proceed along similar lines as an “abort” test decision. In other cases, a redundant system may be available, and testing may continue on the redundant leg with a parallel activity to perform a more complete failure analysis on the failed system. In all cases, the failure analysis is finally completed to assure that no generic problems exist.

6.3.4 Retest of Components with Major Redesign

6.3.4.1 Rationale for Retest of Components with Major Redesign. The corrective redesign of a component is defined as “major” if the test article after redesign violates one or more of the commonly used ground rules for qualification by similarity. This is based on the rationale that such redesigns or related parameters will have invalidated previous tests. The following rationale apply to the test article(s) and the qualification article(s) with major redesign:

a. Functional Input or Output Requirements. If electrical or mechanical performance requirements of the component previously tested are revised by the redesign, the previous functional tests are

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

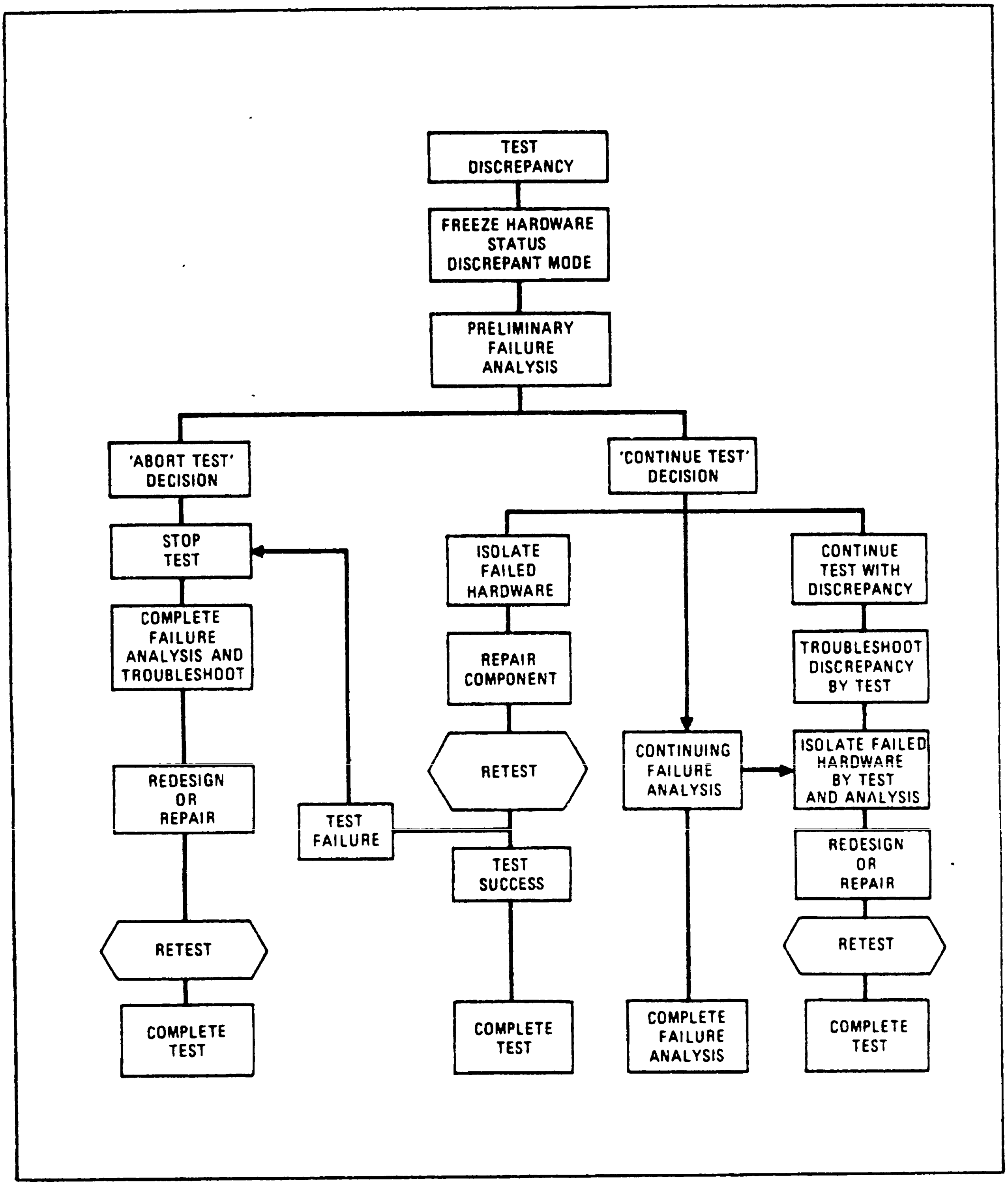

FIGURE 3. Space Vehicle Test Activity After Anomaly.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

no longer considered valid. Consideration must be given to the magnitude of the changes. Relatively small percentage changes may occur in performance requirements, such as revising tolerances on parameters, which do not change the basic characteristic of the component. An example might be an electronic component with the same configuration as before, where outputs are revised by minor tuning of adjustable devices within the component,

b. Environmental and Life Requirements. If the applicable operating or nonoperating environmental and life requirements such as thermal vacuum, thermal cycling, vibration, acoustic, pyrotechnic shock, acceleration, humidity, EMC, fatigue, or wearout were made more severe than the environments experienced by the previously tested component, the applicable environmental tests of the component prior to design are invalidated.

c. Thermal Effects. If analysis shows that the redesign has or could cause thermal effect different from the previous configuration, or if the redesign introduces elements which have not demonstrated a capability to survive the thermal environment, the previous thermal tests will be invalidated.

d. Dynamic Response. If analysis shows that the redesign has or could change dynamic responses different from the previous configuration, or if the redesign introduces elements which have not demonstrated a capability to survive the dynamic environment, the previous dynamic tests will be invalidated.

e. Materials and Manufacturing Processes. Changes in materials and manufacturing processes due to the redesign can invalidate previous tests due to different thermal effects, dynamic responses, and static responses of the redesign. Analysis Of changes in this area is required to determine their effect on the validity of previous tests. It is important to recognize that a detailed knowledge is required of the difference in the manufacturing process between the previously tested component and the redesigned component. Minor production changes in methods of manufacture can lead to an invalidation of most

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

previous tests due to uncertainty about the revised manufacturing method.

f. Weight, Size, Mechanical, and Electrical Configuration. Analysis is required to determine whether changes in these parameters have been of sufficient magnitude to significantly change thermal effects or static and dynamic responses.

6.3.4.2 Guidance for Retest of Component with Major Redesign. In this case, the test article after redesign violates one or more of the commonly used ground rules for qualification by similarity, 60 the previous tests will have been invalidated. Therefore, the following guidelines apply to the acceptance test article(s) and the qualification article(s):

o Repeat all previous tests on the redesigned test article.

Notes:

o Evaluate whether repeat of previous test(s) will degrade component and refurbish component hardware subject to degradation.

o Requalify redesigned components prior to flight article acceptance test continuation.

6.3.5 Retest of Components with Significant Rework

6.3.5.1 Rationale for Retest of Components with Significant Rework. The corrective rework of a component is defined as “significant” if the rework has caused a loss in confidence that tests prior to the rework are still valid. Since the rework corrective action is being treated separately from redesign, the major item of concern is the adequacy of the manufacturing and repair processes to perform the rework. The risk of the rework action may be divided into two categories: the risk of degrading the component by the repair operation and the risk of replacing a part with one that has not been screened by the previous component tests. The following rationale apply to the test article(s) and qualification article(s) with significant rework:

a. Amount of Disassembly and Reassembly. If a component requires considerable disassembly to obtain access to perform the repair and subsequent reassembly, the majority of previous tests are probably invalidated, even if the actual repairs are relatively simple.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

b. Quantity and Complexity of Disconnects and Reconnects. The number of disconnects to remove a failed part or failed hardware, the nature of the disconnects, and the complexity of performing the repair are important in evaluating the risk of. degrading the hardware. If a part can be simply unplugged, the risk of invalidating a previous test would appear less, since a functional test after the repair is completed could verify the adequacy of the repair, and possible damage to surrounding parts is low. A repair requiring Soldering or welding involves the risk of damage to surrounding hardware which could invalidate previous tests.

c. Access to Inspect. In-process inspection is an important part of manufacturing. As a component is manufactured, visual inspection with optical aids, local measurements using hand-held test equipment such as voltmeters, force-gauge measures of compression, tension, or torque, local temperature measurements, and other inspection devices are used to inspect the adequacy of the assembly as hardware is being installed. If a repair can be inspected locally in the same manner as it was inspected during original manufacture, considerable confidence in its adequacy can be obtained. In general, it is noted that a repair which does not allow the same degree of in-process inspection as was done during original manufacture has invalidated previous tests.

d. Repair Techniques. During original manufacture of a component, automated or manual production tooling may be used, depending on quantity. As an example, the soldering or welding of parts may be fully or partially automated and may be performed within the confines of a clean bench which protects the system from contamination. A repair may be performed under different conditions, using considerably different tooling and techniques than were used during original manufacture; it has invalidated the previous tests.

As a general observation, note that judgments relative to the risk of component degradation by rework are highly dependent on knowledge of the processes used during original manufacture.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

Consequently, a repair on a component preferably is coordinated with the original manufacturer. Regardless of how the repair is performed, a risk of not discovering some defect exists if all previous tests are not repeated. It is the degree of acceptable risk which determines whether previous tests should be repeated.

e. Lack of Replaced Part Screening. Although parts are usually screened prior to installation, there is no assurance that this is the case. If a part is replaced, it is necessary to know its previous test experience. If it has not been screened to
f. the same degree as the original part, the component tests conducted prior to the failure have been invalidated.

6.3.5.2 Guidance for Retest of Component with Significant Rework. In this case, the rework has caused a loss in confidence that tests prior to the rework are still valid. The major item of concern is the adequacy of the manufacturing and repair processes. The rework must avoid repair processes that degrade the component, and parts that are used for replacements should be adequately screened. The following guidelines apply to the acceptance test article(s) and qualification article(s):

- Repeat all previous tests after rework.

Note:
- Evaluate whether repeat of previous tests will degrade component and refurbish component hardware subject to degradation prior to repetition of previous tests.

6.3.6 Retest of Components with Minor Redesign or Rework

6.3.6.1 Rationale for Retest of Components with Minor Redesign or Rework. A minor redesign or rework is one that does not fit the definitions for major redesign or significant rework. A minor rework or redesign may have involved no parts replacement, such as tuning a system by adjustable devices, or may have involved replacement of an easily unplugged or detachable part.

6.3.6.2 Guidance for Retest of Components with Minor Redesign or Rework. For minor redesign or rework such as tuning, adjusting, or replacement of an easily detachable part, the following guidelines are provided:

- Evaluate whether replaced part(s) have been screened to the same degree or more severe than environments during component tests.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

- o If replaced parts screening is not adequate, repeat all previous component tests.

- o If replaced parts screening is adequate, evaluate whether previous tests induced failure or were invalidated by the repair.

- o If previous tests induced failure or were invalidated, repeat applicable previous test(s) and continue testing from point stopped.

- o If previous tests were not affected by rework, repeat the test(s) during which the failure occurred, and continue testing from point stopped.

6.3.7 Retest of Space Vehicle with Major Redesign

6.3.7.1 Rationale for Retest of Space Vehicle with Maior Redesign. The definition of "major redesign" follows basically the same ground rules as for component. However, some details are different. For purposes of retest guidelines, a space vehicle redesign is defined as "major" if the redesign has caused significant changes in parameters and has thereby invalidated a number of previous tests. Major redesign of a space vehicle is relatively rare, even during qualification testing. Nevertheless, it may occur. The following rationale apply to a space vehicle or qualification vehicle with major redesign:

a. Functional Input or Output Requirement. If electrical or mechanical performance requirements of any subsystem previously tested are reviewed by the redesign, the previous functional tests are no longer considered valid.

b. Environmental Requirement. If the applicable operating or nonoperating environmental requirements such as EMC, acoustic, pyrotechnic shock, vibration, thermal cycling, thermal balance, or thermal vacuum are made more severe than the environments experienced by the space vehicle prior to redesign, they will be invalidated.

c. Thermal Effects. If analysis shows that the redesign has or could cause thermal effects different from the previous configuration, the previous thermal tests will be invalidated.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

d. <u>Dynamic Response</u>. If analysis shows that the redesign has or could change dynamic responses from the previous configuration, the previous dynamic tests will be invalidated.

e. <u>Materials and Manufacturing Processes</u>. Although relatively rare at the space vehicle level, changes in materials and manufacturing processes due to the redesign can invalidate previous tests due to different thermal effects, dynamic responses, and static responses of the redesign. Analysis of changes in this area is required to determine their effect on the validity of previous tests.

f. <u>Weight, Size, Mechanical, and Electrical Configuration</u>. For components, a major redesign resulted in the guideline of repeating all previous tests, since it would be very difficult to determine portions of previous tests invalidated. Since a space vehicle is composed of a number of subsystems such as electrical power, attitude control, telemetry, instrumentation, command, structure, thermal control, and propulsion, it is possible that redesign of a specific subsystem has not affected other subsystems. Consequently, consideration can be given to repetition of only those previous tests invalidated by the redesign.

9. <u>Component Relocation</u>. If a component is relocated on a space vehicle, it can invalidate a number of the previously conducted tests related to the configuration and mass properties of the space vehicle. These may include EMC, acoustic, thermal balance, random vibration, and pyrotechnic shock test, plus the mass and center of gravity (e.g.) related operations including spin balancing.

6.3.7.2 <u>Guidance for Retest of Space Vehicle with Major Redesign</u>. In this case, the space vehicle redesign has caused significant changes in parameters and has thereby invalidated a number of previous tests. Therefore, the following guidelines apply to the acceptance test article(s) and the qualification article(s):

0 Evaluate which previous test(s) were invalidated by the redesign.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

- Perform functional tests to verify that all equipment (primary and redundant) meet performance requirements.

- Repeat all previous environmental tests on redesigned and reworked subsystem.

Notes:

- Evaluate whether repeat of previous test(s) will degrade components or interconnecting hardware and replace components or hardware subject to degradation.

- All replacement components must have passed acceptance test.

- Requalify redesigned components and subsystem prior to flight article acceptance test continuation.

6.3.8 Retest of Space Vehicle with Significant Rework

6.3.8.1 Rationale for Retest of Space Vehicle with Significant Rework. The definition of a significant space vehicle rework is the same as for components. In addition, the considerations related to space vehicle degradation by rework are similar but not identical. In general, the repair of components requires more severe disassembly and disconnect actions than the rework of a space vehicle. As an example, while parts or other hardware in components are often soldered or welded, the assembly of space vehicles is more modular with most electrical components connected by removable electrical connectors and mounted by removable mounting hardware. Mechanical components also are usually removable by nondestructive means and usually can be reinstalled without the use of special manufacturing processes. Consequently, the risk of space vehicle degradation by rework is somewhat lower than the risk of component degradation.

6.3.8.2 Guidance for Retest of Space Vehicle with Significant Rework. The definition of a significant space vehicle rework is the same as for components. In addition, the considerations related to space vehicle degradation by rework are similar but not identical.

As with components, the amount of disassembly and reassembly and the quantity plus complexity of disconnects and reconnects must be considered in order to reach a judgment on the significance of the rework and the degree by which previous

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

tests may have been invalidated. Inspection after reassembly plays an important part as with components. If accessibility and means of inspection are available to assure that the replaced part has been installed to the same standards as the original assembly, and if relatively few other parts are disturbed, the risk of invalidating previous tests is reduced. Again, however, regardless of how the rework is performed, an increased risk of not discovering some defects exists if all previous tests are not repeated.

The preplanning of space vehicle tests following component repair or replacement can be performed by establishing a retest matrix which denotes the system level retest(s) to be performed after repair or replacement of any component. The matrix should list the applicable tests which must be performed for retest of a specific component or assembly as illustrated by Table II. As an example, on a specific space vehicle, the matrix consists of approximately 200 tests and 27 components. The applicable retest following component replacement is marked with an “X” in the affected block. The tests are referred to by paragraph number and name of the test as designated in the test procedure document. This method of preplanning retests of replaced components on space vehicles has been implemented successfully on space programs, and has avoided the crunch of emergency and time-constrained decisions during testing. It is recognized that on small or one-of-a-kind programs, such a preplanned approach is not always possible, since early preparation and checkout of such preplanned procedures is necessary and budgets or schedules do not always allow such planning. Nevertheless, such an approach can be cost-effective, particularly if problems are anticipated.

For component tests, the screening of parts used for replacement is important. For space vehicles, the degree of component screening is a critical parameter. Although MIL-STD-1540B requires complete component testing prior to installation on the space vehicle, some components are not tested over their full range of performance requirements until they are assembled on the space vehicle. For those cases, consideration must be given to the previous tests missed by the replacement component, if the tests are not repeated.

In view of the above, the following retest guidelines are recommended for space vehicles with significant rework:

0 Evaluate whether previous tests induced the failure and which tests were invalidated by the rework.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE II. Retest Matrix

Procedure Paragraph Number	Test Requirements	Component Requiring Retest											
	Test Name	A	B	C	D	E	F	G	H	I	J	K	Etc.
1.0	TT&C Check												
1.1	Downlink or Uplink Control	X	X	X						X			
1.2	Primary Power Application	X	X							X			X
1.3	Transponder Mode Switching		X							X			X
1.4	Antenna Control		X									X	X
2.0	NAV System Ground Test												
2.1	Warmup	X					X						X
2.2	Standby	X					X						
2.3	On-pad Align or Calibrate	X					X						
2.4	Post-calibrate Gyro Mode	X					X						
Etc.													

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

- Determine the subsystem(s) test(s) affected by the rework.

- Assure that all replacement components have been component acceptance-tested.

- Perform an abbreviated functional test after rework to assure that all equipment is operational after rework.

- Repeat all environmental tests considered to have induced failure or were invalidated by the rework.
- Repeat the functional test during which the failure occurred.

- For higher degree of risk minimization, repeat the acoustic test regardless of its involvement with the failure.

- For highest degree of risk minimization, repeat all previous environmental tests.

6.3.9 Retest of Space Vehicle with Minor Redesign or Rework

6.3.9.1 Rationale for Retest of Space Vehicle with Minor Redesign or Rework. A minor redesign or rework is one that does not fit the definitions.for major redesign or significant rework. Examples of a minor space vehicle redesign or corrective rework are as follows:

a. An adjustment to "tune" a component

b. Replacement of an easily accessible electrical component with "plug-in" connectors whose continuity after replacement can be easily checked

c. Replacement of an easily accessible mechanical component with fittings whose torque and leakage can be easily checked

6.3.9.2 Guidance for Retest of Space Vehicle with Minor Redesign or Rework. Minor redesign or rework that does not fit the definitions for major redesign or significant rework. The following guidelines are recommended for a minor space vehicle rework:

- Evaluate whether previous tests induced the failure or were invalidated by the rework.

- **Assure** that all replacement components have been acceptance-tested.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

- Perform an abbreviated functional test to verify that the replaced component(s) are operational after the rework.
- Repeat all environmental tests considered to have induced the failure or were invalidated by the rework.
- Repeat the functional test during which the failure occurred.

6.3.10 Retest Limits. The accumulated test time on test articles must be considered when dynamic retests are planned, so that their fatigue life is not expended. For vibration tests, the characteristics of fatigue failures as related to test level and time can be used to determine how much time may be accumulated at acceptance test levels without exceeding the fatigue encountered by a similar qualification article at qualification test levels and durations. The following formula for t_A has been adopted by a number of space contractors.

$$t_A = (t_Q)(2)^{(a/6)(M-S-K)}$$

where

t_A = acceptance test time plus flight level exposure duration resulting in fatigue damage equivalent to damage accumulated during qualification test duration

t_Q = vibration qualification test duration

a = inverse slope of stress versus number of cycles fatigue curve for the most fatigue-critical material in the test article

M = margin between qualification and acceptance vibration inputs in decibels

S = 2, if the qualification and acceptance test hardware were fabricated about the same time and 3, if the qualification and acceptance test hardware were fabricated several years apart (and therefore, might not be uniform or identical)

K = a number ranging between 0.6 and 2.0, in accordance with Table I (Paragraph 6.1.3.1)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

In order to determine retest time available, consideration must be given” to normal vibration test exposures during initial component acceptance tests, vehicle level tests, and flight. The use of this equation is discussed in Paragraph 6.1.3.1 of this handbook.

6.4 TEST DATA ANALYSIS

6.4.1 Standard Criteria. Contents of Paragraph 4.4 of MIL-STD-1540B (requirement for test data analysis) are as follows:

4.4 TEST DATA ANALYSIS

A test data bank containing all pertinent system, space vehicle. subsystem. and component test data taken throughout the program shall be maintained, To permit as complete an evaluation as possible of component. subsystem. and space vehicle performance under the various specified test conditions, all relevant test measurements and the environmental conditions imposed on the units shall be recorded on magnetic tape or by other suitable means. These records are intended for post-test analysis to supplement the real-time monitoring and to facilitate replaceable item. The degree of retest shall be the mechanized accumulation of trend data for the critical test parameters. Test data shall be examined for out of tolerance values and for characteristic signatures. Transient responses and mode switching tests shall be examined for proper response. The test data shall also be compared across major test sequences for trends or evidence of anomalous behavior,

6.4.2 Rationale for Test Data Analsis Requirement. Test data analysis is conducted to ensure that all specification requirements are met and to eliminate any incipient failures. Also, analysis ensures that a data base exists from component to system level, and among all like items of hardware, from which nominal performance variability can be determined and degrading trends identified. The data bank is also necessary in evaluation of anomalies which occur in orbital use of the system.

6.4.3 Guidance for Use of Test Data Analysis Requirement. Test methodology and monitored parameters should be the same from component through system level to the maximum extent possible. Selected trends together with test data are recommended to be used as an integral element of hardware certification. Key parameter sheets should include all critical parameters, and any unusual or unexpected trends should be evaluated to determine the existence of any trends towards an

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

out-of-limit value or of an incipient failure within a component or system interface. Comparison should be made to previous like components to aid in determining whether the anomaly is peculiar to that component or is generic in nature.

The requirement is applicable to those selected components, subsystems , and systems whose operating characteristics are judged complex and whose nominal repeatability is dependent on the stability of its constituent elements. Implementation requires a test methodology which looks at the same or related critical parameters at each level of test, such that degradation' or failure detected at higher levels of assembly can be traced to the most probable cause at a lower level.

A matrix should be made showing evidence of test data review and data acceptance at each post-test review. Each matrix would then become part of the acceptance data package at the component, vehicle, and system levels.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

SECTION 7

DEVELOPMENT TESTING

7.1 DEVELOPMENT TESTS

7.1.1 Standard Criteria. Contents of Paragraph 5.1 of MIL-STD-1540B pertaining to development testing are as follows:

> *The objective of the development tests is to assure that testing of critical items at all levels of assembly is sufficient to validate the design approach. Requirements for development testing therefore depend upon the maturity of the subsystems and components used and upon the operational requirements of the specific program. Development tests are necessary to validate new design concepts and the application of proven concepts and techniques to a new configuration. Development tests are also conducted to verify design criteria for structures and components and to determine design margins and failure modes. Development tests may be conducted on breadboard equipment, prototype hardware, or the development test vehicle equipment and software. When development tests are proposed on qualification or flight hardware, the approval of the contracting officer is required.*
>
> *By its nature, development testing cannot be reduced to a standardized set of procedures. The development test requirements are necessarily unique to each new space vehicle. It is not the intent of this section to define the required development tests, but to provide guidelines for conducting appropriate tests when their need has been established.*

7.1.2 Rationale for Development Tests. Development tests are conducted on breadboard equipment, prototype hardware, or on prototype software to validate the design or manufacturing approach.

For hardware, particular concern is on packaging design, electrical and mechanical performance, and capability to withstand environmental stress. New designs should be characterized across worst case voltage, frequency, and temperature variations at the breadboard level. Functional testing in thermal and vibration environment is normally conducted. For electronic boxes, thermal mapping in a vacuum environment for known boundary conditions may be needed to

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

verify the internal component thermal analysis. The correlated thermal model is then used to demonstrate that critical piece part temperature limits, consistent with reliability requirements and performance, are not exceeded. Development tests involving mounting methods for parts, board sizes and thickness, number of layers, or installation method should be performed to evaluate new interconnect systems. Temperature cycling and random vibration development testing should be conducted to evaluate the entire package.

Tests of structural and thermal development models are often necessary to confirm dynamic and thermal environmental criteria for design of subsystems, to verify mechanical interfaces, and to assess functional performance of deployment mechanisms and thermal control systems. Space vehicle development testing also provides an opportunity to develop handling and operating procedures as well as to understand system interactions. A mechanical fit and operational interface test with the launch vehicle and handling facilities at the launch site is recommended.

7.1.3 Guidance for Development Tests. Specific development tests are conducted when their need has been identified by the contractor or when they are contractually required. It is not the intent of MIL-STD-1540B to limit development testing, but to encourage without restrictions appropriate development tests.

7.2 MODAL SURVEY TESTING

7.2.1 Standard Criteria. Contents of Paragraph 5.3 of MIL-STD-1540B pertaining to modal survey tests of space vehicles are as follows:

> *A modal survey is normally conducted to define or verify an analytically derived dynamic model of the space vehicle for use in launch vehicle flight loading event simulations and for use in examinations of post-boost configuration elastic effects upon control precision and stability. This test is conducted on a flight quality structural subsystem as augmented by mass simulated components. The data obtained should be adequate to define orthogonal mode shapes, mode frequencies. and mode damping ratios of all modes which occur within the frequency range of Interest. In most Instances, modes in the frequency range from zero to 50 Hz should be measured.*

7.2.2 Rationale for Modal Survey Tests. The modal survey test is an important element in the flight loads

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

environment definition, which is essential to the verification of the flightworthiness of the space vehicle structural design and to the satisfaction of flight safety requirements. Usually the critical loads experienced by a spacecraft structure in flight are highly dependent upon the dynamic characteristics of the spacecraft. For this reason, it is necessary that the accuracy of the spacecraft model be determined through the experimental measurement of the natural modes of the flight configuration.

7.2.3 Guidance for Modal Survey Tests. Modal survey tests are conducted to determine the natural mode frequencies and the mode damping ratios. They should accurately map the mode shape vectors of all modes in the frequency range of interest, which is usually taken to be from zero to 50 Hz. Orthogonality of the measured mode shapes is the most frequently applied criterion for the accuracy of the mode test measurements . Acceptable orthogonality is indicated when all the off-diagonal terms in the normalized modal mass matrix are less than 0.10. This is a technically demanding requirement and is likely to be achieved only when careful attention is given to planning and pretest preparations as well as to the proper execution of the test.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 8

SPACE VEHICLE AND SUBSYSTEM LEVEL TESTS

8.1 SPACE VEHICLE TEST BASELINES

8.1.1 Standard Criteria. Contents of Paragraphs 6.2 and 7.1 of MIL-STD-1540B (requirements for space vehicle qualification and acceptance tests baselines) are as follows:

6.2 SPACE VEHICLE QUALIFICATION TESTS

The space vehicle qualification test baseline consists of all the required tests specified in Table I. The test baseline shall be tailored for each program, giving consideration to both the required and optional tests; however, deviations from the baseline requirements for the required tests shall be approved by the contracting officer. Additional special tests such as alignments, instrument calibrations, antenna patterns, and mass properties that are conducted as acceptance tests for flight vehicles shall be conducted on the qualification flight vehicle unit. If the space vehicle is controlled by on-board data processing. the flight version of the computer software shall be resident in the space vehicle computer for these tests. The verification of the operational requirements shall be demonstrated to the extent practicable.

Table I. Space Vehicle Qualification Tests

Test	Reference Paragraph	Suggested Sequence	Required (R) OR Optional (O)
Functional	6.2.1	1(1)	R
EMC	6.2.2	2	R
Acoustic	6.2.3	5	R(2)
Vibration	6.2.4	5	O
Pyro Shock	6.2.5	4	R
Pressure	6.2.6	3, 6	R
Thermal Vacuum	6.2.7	9	R
Thermal Balance	6.2.8	8	R
Thermal Cycling	6.2.9	7	O(3)

Notes: (1) Electrical and mechanical functional tests shall be conducted prior to and following each environmental test.

(2) Conduct vibration in place of acoustic test for vehicles of compact shape and with weight less than 180 kilograms.

(3) Required if thermal cycling acceptance test 7.1.8 is conducted.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

7.1 SPACE VEHICLE ACCEPTANCE TESTS

The space vehicle acceptance test baseline consists of all the required tests specified in Table III. The test baseline shall be tailored for each program, glvlng consideration to both the required and optional tests; however. deviations from the baseline requirements for the required tests shall be approved by the contracting officer.

Table III. Space Vehicle Acceptance Tests

Test	Reference Paragraph	Suggested Sequence	Required (R) OR Optional (O)
Functional	7.1.1	1(1)	R
EMC	7.1.2	2	O
Acoustic	7.1.3	5	R(2)
Vibration	7.1.4	5	O
Pyro Shock	7.1.5	4	O
Pressure	7.1.6	3, 6	R
Thermal Vacuum	7.1.7	8	R(3)
Thermal Cycling	7.1.8	7	O
Storage Tests	7.1.9	-	O
Special Tests	7.1	-	O

Notes: (1) Electrical functional tests shall be conducted prior to and following each environmental test.

(2) Conduct vibration in place of acoustic test for vehicles of compact shape and with weight less than 180 kilogram.

(3) Requirements are modified if Thermal Cycling test 7.1.8 is conducted.

Additional special tests normally conducted by space vehicle programs include alignments, instrumentation calibrations, and measurements of mass properties, antenna patterns, and magnetic field. Since performance and accuracy requirements are generally program peculiar, and test methods are typically contractor peculiar, these tests are not included in this standard.

If the space vehicle is controlled by on-board data processing, the flight version of the computer software shall be resident in the space vehicle computer for these tests. The verification of the operational requirements shall be demonstrated in these tests to the extent practicable.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

8.1.2 Rationale for Space Vehicle Test Baseline Requirements. Environmental qualification tests are a formal demonstration that a production vehicle (or prototype) is adequate to successfully sustain specified environmental design levels. These tests are mainly performed to determine if there are factors that may have been overlooked during design, analysis, OK manufacturing. Additionally, the environments used during these tests are the design levels that are more severe than these predicted to occur during flight in order to account for variabilities in subsequent production articles and other uncertainties. Qualification test requirements, therefore, incorporate margins which are added to the range of environmental extremes and stresses expected to occur in service. Before qualification testing, the space vehicle should have been subjected to the same controls, inspections, alignments, and tests imposed on flight vehicles. This includes completion of the environmental acceptance tests.

The environmental tests required for space vehicle qualification are EMC, acoustics (vibration for certain configurations), pyrotechnic shock, thermal balance, thermal vacuum, and pressure test of fluid subsystems before and after the pyrotechnic shock and acoustic tests. Functional tests are required before and after each environmental test. Thermal cycling at ambient pressure is an optional test but becomes a required test if thermal cycling is imposed for space vehicle acceptance testing.

For certain configurations, random vibration may replace acoustic testing as one of the required tests. In general, these situations arise when the space vehicle is of small size and has a high density. For such a small compact vehicle, acoustic noise may not adequately excite vibratory responses, due to insufficient surface area over which the acoustic pressures may act, and due to a frequency mismatch between the excitation and the natural vibration frequencies related to the dimensions of the space vehicle. In such a case, vibration testing is used to generate a more realistic response in the test specimen.

Environmental acceptance tests are conducted on space vehicles to demonstrate flightworthiness and to disclose quality deficiencies in the flight article. Acceptance tests are intended to satisfy these goals by subjecting the space vehicle to the maximum environmental exposures expected in service. The test program is comprised of a series of tests; some are required tests, while others are optional. Required vehicle-level acceptance tests include thermal vacuum, acoustic (or vibration for certain configurations), pressure test of fluid subsystems, and functional tests before and after each

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

environmental test. Augmenting the required tests are those optional tests which are considered appropriate in accordance with the goals and characteristics of a given space vehicle program. Among the optional acceptance tests are EMC, pyrotechnic shock, and thermal cycling. If thermal cycling is performed, the thermal vacuum testing requirements for the space vehicle are reduced, and the number of thermal cycles specified for the thermal vacuum test may be reduced from four to one.

8.1.3 Guidance for Use of Space Vehicle Test Baseline Requirements. The Suggested sequence of environmental tests is based on two considerations: preserving the sequence or concurrent nature of the service environments, and assuring that potential failures will be detected as early in the test sequence as possible. Therefore, dynamic tests, which simulate the launch and ascent environment and are generally of short duration with limited performance testing, should precede thermal vacuum and thermal cycling tests, which simulate long duration orbital environments where greater opportunity is afforded for more extensive diagnostic testing. However, in recognition of program-peculiar requirements, such as the buildup sequence and logistic considerations, the order of testing in MIL-STD-1540B is only a suggested rather than a required sequence. However, the sequencing used should recognize that the thermal vacuum test is an orbital performance check that should be run towards the end of the sequence.

In order to minimize changes to test setups and instrumentation, the acceptance test exposures required for the qualification article may be integrated with the qualification test program by performing the acceptance level test just prior to the qualification level test. For example, in conducting the space vehicle acoustic qualification test, the acceptance level acoustic environment would be imposed for its prescribed duration before imposition of the full qualification acoustic environment. By conducting the acceptance test just before the applicable qualification test exposure, a secondary objective of validating the environmental acceptance test program is accomplished.

The thermal cycling test, which may be imposed at the space vehicle level, has proved to be extremely useful and cost-effective in disclosing latent defects. Thermal cycling tests are also useful for periodic testing of vehicles in storage to assure that they remain flight-ready.

The mechanical and electrical functional tests are extremely important elements in the test baselines. The functional tests are conducted prior to and after each of the environmental tests. They should be designed to verify that performance of the components and of the space vehicle meets the

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

specification requirements, that the components and the space vehicle are compatible with ground support equipment, and that all software used is validated, such as in computer-assisted commanding and data processing. In addition, the electrical functional tests should include negative logic testing to verify lockout, to assure that no function other than the intended function was performed, and to verify that the signal was not present other than when programmed. To the extent practicable, the functional tests should also be designed 60 that a data base of critical parameters can be established for trend analysis. This is accomplished by measuring the same critical parameters in all of the functional tests conducted before, during, and after each of the baseline environmental tests. During these tests, the maximum use of telemetry should be employed for data acquisition, problem identification, and problem isolation. This can assist in mechanizing the data base for trend analysis and provides training for on-orbit flight support.

The trend data and the final ambient functional test conducted prior to shipment of the space vehicle to the launch base provide the data to be used as success criteria during launch base testing. For this reason, the vehicle level functional tests should be designed so that they can be duplicated, as nearly as possible, at the launch base.

It is extremely important that functional tests be conducted before and after each environmental test. These functional tests provide the criteria for judging successful survival of the space vehicle in a given test environment. It is also important to perform functional tests of space vehicle subsystems while the environment is being imposed. This is especially important for the thermal balance or thermal vacuum tests, since the space vehicle is expected to be fully operational under these conditions. It is usually considered appropriate during acoustic or vibration tests to have the vehicle in an operating mode representative of launch and ascent. The launch and ascent time period usually involves a minimum level of functional performance, with many subsystems inoperative. However, ample evidence has been gathered to demonstrate that dynamic tests should be performed on fully functional space vehicles with their performance monitored for intermittents. Many defects such as improper mounting or intermittents, which otherwise escape detection by pre- and post-test functional checks, reveal themselves during environmental tests. For example, intermittent may be caused by foreign bodies, contaminants, inadequate clearances, cracks, debonds, and damaged connectors that might only be revealed during environmental tests. Therefore, regardless of the functional mode of the space vehicle during launch and ascent, the space vehicle should be functionally operated and monitored during dynamic as well as thermal tests to increase overall test

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

effectiveness. Practical limitations frequently restrict the extent of operation of space vehicle subsystems during the relatively brief acoustic test. In recognizing this problem, MIL-STD-1540B permits extended functional testing with subsystems operating and monitored, but conducted at a level 6 dB lower than the required test level, after the required environmental exposure has been satisfied.

For small compact spacecraft, acoustic testing will not provide adequate environmental simulation, and random vibration should supplant the acoustic test. MIL-STD-1540B directs that vibration testing be' considered for vehicles of compact shape and weight less than 180 kilograms (approximately 400 pounds). For a launch vehicle such as the STS, which produces considerable acoustic noise in the low frequency range below 100 Hz, the wavelengths of the dominant frequencies are longer than 10 feet. If a small heavy cylindric space vehicle, 4 feet in diameter and 3 feet long, were tested in a representative acoustic environment, the resulting vibration response of the vehicle might fall short of simulating actual conditions in the low frequency range. In such an instance, random vibration testing could become the preferred mode of testing. If there is insistence on an acoustic test mode, it may become necessary to include the interfacing structure with the space vehicle test specimen to achieve adequate simulation. This could include cradles which hold the space vehicle or associated upper stage, or even a portion of the launch vehicle. The proportions of the test article should correlate with those of the environmental frequency range of interest. Where either test may be appropriate, equivalent vibration and acoustic criteria should be derived by analysis or empirical observations to provide corresponding criteria. In addition to considering fidelity of simulation, a number of practical issues are involved in this matter. Random vibration equipment capabilities are limited in terms of displacement, force output, and frequency range. An acoustic chamber which simulates the ascent acoustic environment from 25 to 10,000 Hz can usually accommodate relatively large vehicles, regardless of their weight. However, a random vibration test facility imposes weight limitations based upon vehicle plus fixture weight because of its force limitations. In addition, mechanical vibration exciters have difficulty generating frequencies above 2000 Hz. Also, a very real danger exists of anomalous behavior of the vibration exciter such as sudden shutdowns, runaways, and line transients. When the space vehicle is intimately attached to a vibration exciter of significant force capability, much damage can be inflicted unless careful attention is devoted to safeguards. The decision to perform either acoustic or random vibration tests involves much engineering judgment. Situations may arise in which some combination of acoustic and vibration tests provides the best

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

solution. The low frequency portion of the environment may best be simulated by mechanical vibration, while the mid and high frequencies may be more suitably tested by acoustic methods. Familiarity with the capabilities of the two test methods and an understanding of the physical aspects of the environmental simulation aids in selecting the best combination of tests.

Tables III through VI summarize the important parameters of space vehicle environmental tests. They are useful as concise references to the major test requirements and for comparing qualification to acceptance test requirements.

TABLE III. Thermal Cycling Test--Space Vehicle Qualification and Acceptance Test Parameters

Thermal Cycling Test Parameters	Qualification - Para. 6.2.9	Acceptance - Para. 7.1.8
Temperature Range Differential	Max. possible within constraints, with minimum of 70°C	Max. possible within constraints, with minimum of 50°C
Temperature Extremes	Not specified in para. 6.2.9	Not specified in para. 7.1.8
Number of Cycles	No. of cycles = 125 percent of acceptance test = 50 minimum	40 minimum
Dwell	Duration not specified. On last cycle only, at each temp. extreme, for functional test.	Duration not specified. On last cycle only, at each temp. extreme, for functional test.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE IV. Thermal Vacuum Test--Space Vehicle Qualification and Acceptance Test Parameters.

Thermal Vacuum Test Parameters	Qualification - Para. 6.2.7	Acceptance - Para. 7.1.7
Temperature Range and Extremes	Min. predicted to max. predicted temp. environments plus environmental design margin of 10°C, for one component in each vehicle equipment area	Min. predicted to max. predicted temp. environments, for one component in each vehicle equipment area
Number of Cycles	Min. of 8 cycles	Min. of 4 cycles if thermal cycling not performed
Dwell	Min. of 8 hours soak at each temp. extreme of each cycle	Min. of 8 hours soak at each temp. extreme of each cycle
Pressure	10^{-4} Torr or less	10^{-4} Torr or less

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

TABLE V. Pyrotechnic Shock Test--Space Vehicle Qualification and Acceptance Test Parameters

Pyrotechnic Shock Test Parameters	Pyro Shock Qualification - Para. 6.2.5	Pyro Shock Acceptance - Para. 7.1.5
Shock Level	Max. predicted shock environment plus environmental design margin of 6 dB	Max. predicted shock environment
Number of Shocks (number of firings)	At least one firing of each pyrotechnic device. 3 firings for devices producing shocks within 6 dB of max. measured response from any device.	Required for re-furbishable devices only. One firing of each device causing significant shocks to Critical and shock-sensi-tive components.

TABLE VI. Acoustic Test--Space Vehicle Qualification and Acceptance Test Parameters

Acoustic Test Parameters	Qualification - Para. 6.2.3	Acceptance - Para. 7.1.3
Sound Pressure Level	Greater of: maximum predicted environ-ment plus environ-mental design margin of 6 dB, or 144 dB overall	Greater of: max. predicted environ-ment; or 138 dB overall
Test Duration	Greater of: 3 times expected flight exposure time, or 3 times acceptance test duration, or 3-minutes minimum	Greater of: max. expected flight exposure time, or l-minute minimum

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

8.2 SPACE VEHICLE ACOUSTIC TESTS

8.2.1 Acoustic Qualification Tests

8.2.1.1 Standard Criteria. Contents of Paragraph 6.2.3 of MIL-STD-1540B (requirements for space vehicle qualification acoustic test) are as follows:

6.2.3 Acoustic Test, Space Vehicle Qualification

6.2.3.1 Purpose. This test demonstrates the ability of the space vehicle to withstand or, if appropriate, to operate in the design level acoustic environment which is the maximum level Imposed in flight plus a design margin. This test also verifies the adequacy of component vibration qualification criteria,

6.2.3.2 Test Description. The space vehicle shall be installed in a reverberant acoustic cell capable of generating desired sound pressure levels. It shall be mounted on a flight-type support structure or reasonable simulation thereof. The mechanical configuration of the space vehicle shall be as it is during ascent (for example, solar arrays and antennas stowed). Where possible, ground handling equipment and test equipment shall be removed. Adequate dynamic instrumentation shall be installed to measure vibration responses at attachment points of critical and representative components.

6.2.3.3 Test Levels and Duration. The acoustic test spectrum shall be the design environment (see 3.8) which is the maximum predicted flight environment (see 3.20) plus the design margin (6 dB : see 3.12). However, the overall sound pressure level of the qualification test shall not be less than 144 dB. Exposure test time shall be at least three times the expected flight exposure time to the maximum flight environment, or three times the acceptance test duration if that is greater, but not less than 3 minutes. Operating time should be divided approximately equally between redundant circuits. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

6.2.3.4 Supplementary Requirements. During the test all electrical and electronic components, even if not operating during launch, -shall be electrically energized and sequenced through operational modes to the maximum extent possible with the exception of components that may sustain damage if energized. Continuous monitoring of several perceptive

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

parameters shall be provided to detect intermittent failures. Functional tests are required before and after the environmental exposure.

8.2.1.2 Rationale for Qualification Acoustic Tests. Acoustic qualification tests are a formal demonstration that a production space vehicle can successfully sustain the specified acoustic design levels. The space vehicle acoustic qualification test also serves as a source for accurate vibration data which may be used to compare with component qualification test requirements, as well as forming a reference for evaluating vibration levels encountered during acoustic acceptance testing of subsequent vehicles.

8.2.1.3 Guidance for Qualification Acoustic Test. A critical element in the space vehicle acoustic qualification test is the instrumentation used to measure the acoustic levels and the vibration response of the equipment subjected to the acoustic inputs. The quantity of instrumentation required may vary widely from program to program due mainly to the size and complexity of the test vehicle; however, sufficient vibration data should be obtained such that every component may be evaluated. For large vehicles, it would not be unusual to have in excess of 100 accelerometer measurements. Where large numbers of measurements are not feasible and when each component cannot be instrumented, emphasis should be placed on those components which have exhibited poor component level qualification history or which are known to have less than 6 dB qualification margins. It may be feasible to choose locations which are representative of several component mountings. In general, measurements should be made on primary or secondary structure at component attachment points. Measurement on the component attachment flanges or lugs is acceptable only when there is no room on the adjacent structure.

In general, triaxial measurements should be taken; however, a single axis may be taken when it is known to be the higher response axis or is the axis of maximum component sensitivity. The data acquisition system should have the capability of acquiring accurate data from 20 to at least 2000 Hz.

8.2.2 Acoustic Acceptance Tests

8.2.2.1 Standard Criteria. Contents of Paragraph 7.1.3 of MIL-STD-1540B (requirements for space vehicle acceptance acoustic test) are as follows:

7.1.3 Acoustic Test, Space Vehicle Acceptance

7.1.3.1 Purpose. This test simulates the acoustic and vibration environment imposed on a space vehicle in flight

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

in order to detect material and workmanship defects that might not be detected in a static test condition.

7.1.3.2 Test Description. Same as 6.2.3.2.

7.1.3.3 Test Levels and Duration. The acoustic spectrum shall represent the maximum predicted flight environment as defined In 3.20. The overall sound pressure level for acceptance testing shall not be less than 138 dB. The exposure the at full acceptance test level shall equal or exceed the maximum expected flight exposure time, but the test time shall not be less than 1 minute. Operating time should be divided approximately equally between redundant circuits. Where insufficient the is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

7.1.3.4 Supplementary Requirements. Same as 6.2.3.4.

8.2.2.2 Rationale for Acceptance Acoustic Tests. Acoustic acceptance tests are conducted on space vehicles to demonstrate flightworthiness and to disclose quality deficiencies by subjecting each flight article to the maximum acoustic exposure expected in service. The space vehicle acoustic acceptance test also serves as a source for vibration data which may be used to compare with component expected flight levels, component acceptance test levels, space vehicle qualification levels, and as a diagnostic aid in the event of component malfunction or failure.

8.2.2.3 Guidance for Acceptance Acoustic Tests. An important element in the space vehicle acoustic acceptance test is the instrumentation used to measure the acoustic levels and the vibration response of the equipment subjected to the acoustic inputs. The quantity of instrumentation is governed by the size and complexity of the test vehicle. Particular attention should be given to those components critical to the flight mission, and whose qualification test margin is less than 6 dB or which have a poor vibration test history. Single-axis measurements may be made in lieu of triaxial, when that axis has been shown to be the higher response axis or is the axis of maximum component sensitivity. A total of 12 measurements is considered nominal. In some instances, the accelerometer and some of its wiring may be left in place for flight, if its removal would require partial disassembly and thus cause additional testing. In general, accelerometer locations should duplicate those used in the qualification testing.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

8.3 SPACE VEHICLE VIBRATION TESTS

Vibration tests, per Paragraphs 6.2.4 and 7.1.4 of MIL-STD-1540B, are conducted in place of acoustic tests for vehicles of compact shape and with weight less than 180 kilograms. The rationale and guidance for space vehicle qualification and acceptance vibration tests are the same as for acoustic tests (see Paragraphs 8.2.1 and 8.2.2 above).

8.4 SPACE VEHICLE PYROTECHNIC SHOCK TESTS

8.4.1 Pyrotechnic Shock Qualification Test

8.4.1.1 Standard Criteria. Contents of Paragraph 6.2.5 of MIL-STD-1540B (requirements for space vehicle qualification pyrotechnic shock test) are as follows:

6.2.5 Pyro Shock Test, Space Vehicle Qualification,

6.2.5.1 Purpose. This test demonstrates the capability of the space vehicle to withstand or, if appropriate, to operate in the design level pyro shock environments which are the levels predicted for flight plus a design margin. This test also verifies the adequacy of component pyro shock criteria.

6.2.5.2 Test Description. In this test or series of test segments, all pyrotechnically operated devices and other equipment capable of imparting a significant shock impulse to the space vehicle shall be operated. Separation subsystem shocks are often more severe than those from other pyrotechnic devices, and operation of the separation subsystems is therefore particularly significant. For these tests, the space vehicle shall be suspended or otherwise supported so as to preclude the possibility of recontact between separated portions thereof. When significant shock levels are predicted from subsystems not on board the space vehicle under test. such as the launch vehicle separation shock, the adapter subsystem or suitable simulation shall be attached and appropriate pyrotechnics or other means used to simulate the shock imposed. Adequate dynamic instrumentation shall be installed to measure pyro shock responses in 3 axes at attachment points of critical and representative components.

Support of the space vehicle varies with the configuration and may vary during the course of this test series. To permit optimum positioning and prevent damage to such items as deployment booms, paddles, and ejectable, a series of individual test setups or deployment restraints may be required. The test setup shall permit. as nearly as possible, flightlike dynamic response of the space vehicle structure.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

6.2.5.3 Test Levels and Duration. All pyrotechnic devices (e.g., explosive bolt, nut, pin puller, marmon clamp, etc.) shall be fired at least one time. Those pyrotechnic devices producing shock levels within 6 dB of the maximum shock response measured from any of the devices shall be fired two additional times to provide the expected variability in the shock environment. Firing of both primary and redundant pyres shall be in the same sequence as they are designed to fire in flight.

6.2.5.4 Supplementary Requirements. Electrical and electronic components shall be operating and monitored to the maximum extent possible. Functional tests are required before and after environmental exposure.

8.4.1.2 Rationale for Pyrotechnic Shock Qualification. The pyrotechnic shock qualification tests are a formal demonstration that a production space vehicle can successfully sustain the specified pyrotechnic shock design levels. The pyrotechnic shock qualification test also serves as a source for accurate shock data, which may be used for comparison with component qualification test requirements, and for forming a data base for evaluation of shock levels measured during acceptance shock testing of subsequent vehicles.

8.4.1.3 Guidance for Pyrotechnic Shock Qualification Test. A critical element in the space vehicle pyrotechnic shock qualification test is the instrumentation used to measure the pyrotechnic shock response levels of the equipment subjected to the pyrotechnic shock inputs. The quantity of instrumentation required may vary widely from program to program due mainly to the size and complexity of the test vehicle; however, sufficient data should be obtained such that every component may be evaluated. For large vehicles, it would not be unusual to have in excess of 100 accelerometer measurements. Where large numbers of measurements are not feasible and when each component cannot be instrumented, emphasis should be placed on those components which have exhibited poor component level qualification history or which are known to have less than 6 dB qualification margins. It may be feasible to choose locations which are representative of several component mountings. In general, measurement should be made on primary or secondary structure at component attachment points. Measurement on the component attachment flanges or lugs is acceptable only when there is no room on the adjacent structure. Shocks from all potential shock-generating events should be measured.

In general, triaxial measurements should be taken; however, a single axis may be taken when it is known to be the higher

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

response axis or is the axis of maximum component sensitivity. The data acquisition system should have the capability of acquiring accurate data from 100 to at least 10,000 Hz at frequency intervals of one-sixth octave or less.

In addition, if no design verification or development shock testing was conducted, it is highly desirable to obtain data to aid in characterization of the source shock. Measurements should be made within 6 inches of the source with as few intervening mechanical transitions as possible.

8.4.2 Pyrotechnic Shock-Acceptance Test

8.4.2.1 Standard Criteria. Contents of Paragraph 7.1.5 of MIL-STD-1540B (requirements for space vehicle acceptance pyrotechnic shock test) are as follows:

7.1.5 Pyro Shock Test, Space Vehicle Acceptance

7.1.5.1 Purpose. This test simulates the dynamic shock environment Imposed on a space vehicle in flight in order to detect material and workmanship defects.

7.1.5.2 Test Description. Same as 6.2.5.2.

7.1.5.3 Test Levels and Duration. . Pyrotechnic shock acceptance testing of space vehicles shall be required in those instances where the shock-producing mechanism can be readily refurbished for flight, as is often the case for explosive nuts, bolts, pinpullers, and clamps. one firing of those pyrotechnic devices causing significant shocks to critical and shock sensitive components shall be conducted. Firing of both primary and redundant pyros is required in the same relationship as they will be used in flight. However, where the pyrotechnic mechanism explosively severs structure by detonation of detonating fuse or shaped charge, such testing shall not be included or required. To aid in fault detection, the pyro shock test shall be conducted with subsystems operating and monitored to the maximum extent practical.

8.4.2.2 Rationale for Pyrotechnic Shock Acceptance Test. Pyrotechnic shock acceptance tests are conducted on space vehicles to demonstrate flightworthiness and to disclose quality deficiencies by subjecting each flight article to the maximum pyrotechnic shock exposure expected in service. The space vehicle pyrotechnic shock acceptance test also serves as a source for data which may be used to compare with component expected flight levels, component acceptance test levels, space vehicle qualification levels, and as a diagnostic aid in the event of component malfunction or failure.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

8.4.2.3 Guidance for Pyrotechnic Shock Acceptance Test. An important element in the space vehicle pyrotechnic shock acceptance test is the instrumentation used to measure the pyrotechnic shock levels and the vibration response of the equipment- subjected to the pyrotechnic shock inputs. The quantity of instrumentation is governed by the size and complexity of the test vehicle. Instrumentation may be restricted. to those components which are critical to-the flight mission, and whose qualification test margin is less than 6 dB or which have a poor vibration test history. Single-axis measurements may be made in-lieu of triaxial, when that axis has been shown to be the higher response axis or is the axids of maximum component sensitivity. A total of 12 mea6urement6 is considered nominal. In some instances, the accelerometer and some of its wiring may be left in place for flight, if its removal would require partial disassembly and thus cause additional testing. In general, accelerometer locations should duplicate those used in the qualification testing.

8.5 SPACE VEHICLE PRESSURE TESTS

8.5.1 Standard Criteria. Contents of Paragraphs 6.2.6 and 7.1.6 of MIL-STD-1540B (requirements for space vehicle qualification and acceptance pressure tests) are as follows:

> *6.2.6 Pressure Test, Space Vehicle Qualification*
>
> *6.2.6.1 Purpose. This test demonstrates the capability of fluid subsystems to met the flow, pressure, and leakage rate requirements specified.*
>
> *6.2.6.2 Test Description. The space vehicle shall be placed in a facility that provides the services and safety conditions required to protect personnel and equipment during the testing of high pressure subsystems and in the handling of dangerous fluids. Tests shall be performed to verify compatibility with the test setup and to ensure that proper control of the equipment and test functions is provided. The requirements of the subsystem including flow. leakage, and regulation shall be measured while operating applicable valves, pumps. and motors. The flow checks shall verify that the plumbing configurations are adequate. Checks for subsystem cleanliness, moisture levels, and pH shall also be made. Where pressurized subsystems are assembled with other than brazed or welded connections. the specified torque values for these connections shall be verified prior to leak checks.*
>
> *In addition to the high pressure test, propellant tanks and thruster valves shall be tested for leakage under propellant servicing conditions. The system shall be*

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

evacuated to the internal pressure normally used for propellant loading and the systems pressure monitored for any indication of leakage.

6.2.6.3 Test Levels and Duration. The subsystem shall be pressurized to proof pressure (see 3.34) and held for 5 minutes, then the pressure shall be reduced to the maximum predicted operating pressure (see 3.21). Unless specified otherwise, the proof pressure equals 1.5 times the maximum operating pressure. This sequence shall be conducted three times. Inspection for leakage after these cycles shall be at the maximum operating pressure. The duration of the evacuated propulsion system leak test shall not exceed the time that this condition is normally experienced during propellant loading.

6.2.6.4 Supplementary Requirements. Applicable safety standards shall be followed in conducting all tests. Specially formulated bubbleforming solutions are suitable for detecting external leakage at such locations as joints, fittings, plugs, and lines, where the allowable limits are from 0.00001 to 0.01 cubic centimeters per second (cubic cm per see). Solutions that are used for detecting leaks shall be compatible with the media being leak tested or with the media which could contact any residues. Liquid displacement methods may be used for detecting leakage through poppet seats and internal seals for measurement requirements of 0.1 to 30 cubic cm per sec. Hellum or radioactive tracer gas leak detectors may be used for leakage rates from 0.0000001 to 0.0001 cubic cm per sec. The use of halogen gas detectors for liquid propulsion subsystems shall be avoided. Leak tests shall be conducted only after satisfactory proof pressure tests have been completed. Leak detection and measurement procedures may require vacuum chambers. bagging of the entire space vehicle or localized areas. or other special techniques to achieve the required accuracies.

7.1.6 Pressure Test, Space Vehicle Acceptance

7.1.6.1 Purpose. This test demonstrates the capability of fluid subsystems to meet the flow. pressure, and leakage requirements specified in the space vehicle specification.

7.1.6.2 Test Description. Same as 6.2.6.2.

7.1.6.3 Test Levels and Duration. The leak checks shall be performed by pressurizing the subsystem to maximum

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

operating pressure and holding at this pressure for a period commensurate with the leakage method being employed.

7.1.6.4 Supplementary Requirements. Same as 6.2.6.4.

8.5.2 Rationale for Pressure Test Requirements. The pressure tests defined in the standard criteria above are conducted after assembly of a fluid subsystem. It is assumed that each component has previously been pressure-qualified and acceptance-tested. Consequently, the main emphasis of the subsystem level pressure tests is the pressure and leakage integrity of interconnects. Since components might have degraded during storage, transport, handling, and assembly operations, subsystem proof pressure tests are required in addition to inspection for leakage. Tables I (Qualification) and III (Acceptance) of MIL-STD-1540B require two vehicle or subsystem level pressure tests: one before pyrotechnic shock tests and one after the acoustic test.

For qualification tests, three proof pressure tests are required each time a subsystem pressure test is conducted. For the two subsystem pressure tests required by MIL-STD-1540, this requires a total of six proof pressure tests. The three proof pressure cycles required for qualification are based on the general concept of providing a qualification margin above the acceptance test values. Since acceptance and qualification proof pressures are required to be the same, the greater number of qualification proof pressure cycles (three times the acceptance cycles) are considered to provide this margin.

For acceptance pressure tests, a single proof pressure test is required each time a subsystem pressure test is conducted. For the two subsystem acceptance pressure tests per Table IV of MIL-STD-1540B, this requires a total of two proof pressure tests . For these tests, the system is raised to proof pressure and held for five minutes at this pressure. The purpose of the five-minute hold is to allow time for potential yield of the materials or for potential crack growth to occur. The magnitude of the proof pressure is as required in the subsystem specification. MIL-STD-1522 provides proof pressure requirements for components. The component with the lowest proof pressure requirement within the subsystem governs the subsystem proof pressure magnitude.

Leakage of subsystems is usually determined at interconnects and at exits for gases such as at thrusters and fill or drain fittings. The maximum allowable leakage governs the leakage testing method.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

8.5.3 Guidance for Use of Pressure Tests. The test description of Paragraph 6.2.6.2 of MIL-STD-1540B provides a synopsis for guidance. Further guidance for proof pressure tests is provided in MIL-STD-1522. Guidance for leakage tests is provided by the leakage testing handbook, NASA S-69-1117.

8.6 SPACE VEHICLE THERMAL VACUUM TESTS

8.6.1 Thermal Vacuum Qualification Tests

8.6.1.1 Standard Criteria. Contents of Paragraph 6.2.7 of MIL-STD-1540B (the requirements for space vehicle thermal vacuum qualification tests) are as follows:

6.2.7 Thermal Vacuum Test, Space Vehicle Qualification

6.2.7.1 Purpose. This test demonstrates the ability of the space vehicle to meet design requirements under vacuum conditions and temperature extremes which simulate those predicted for flight plus a design margin.

6.2.7.2 Test Description. The space vehicle shall be placed in a thermal vacuum chamber and a functional test performed to assure readiness for chamber closure. The vehicle shall be zoned Into separate equipment areas based on the location of critical components within each area. Components that operate during ascent shall be monitored for corona, and multipacting (see 3.27) as applicable, as the pressure is reduced to the lowest specified level. Equipment that does not operate during launch shall have electrical power applied after the test pressure level has been reached, A temperature cycle begins with the space vehicle at ambient temperature. The temperature is reduced to the specified low level and stabilized. Component temperature stabilization has been achieved when the rate of temperature change is no more than 3 deg C per hour. Following the cold soak. the temperature shall be raised to the highest specified level and stabilized. Following the high temperature soak, the space vehicle shall be returned to ambient temperatures to complete one temperature cycle. Functional tests shall be conducted during the first and last temperature cycle at both the high and low temperature limits with functional operation and monitoring of perceptive parameters during all other cycles. In addition to the temperature cycles, the chamber shall be programmed through various orbital operations. Operational sequences shall be coordinated with expected orbital environments, and a complete cycling of all equipment shall be performed including the operating and monitoring of redundant equipment and

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

paths. System electrical equipment shall be operating and monitored throughout the test. Strategically placed temperature monitors shall assure attainment of temperature limits. Strategically placed witness plates and quartz crystal microbalances or other instrumentation shall be installed in the test chamber to assure that outgassing from the space vehicle and test equipment does not degrade system performance beyond specified limits.

6.2.7.3 <u>Test Levels and Duration</u>. Temperatures in various equipment areas shall be controlled by the external test environment and internal heating resulting from equipment operation so that during the hot cycle the temperature on at least one component in each equipment area at its design high temperature and one component during the cold cycle is at its design low temperature. The temperature extremes shall be established by a survey of predicted temperatures in various equipment areas and may have to be adjusted to the performance of the most sensitive components in a particular area. Temperatures on the components shall not be allowed to exceed the design levels for the components. The pressure shall be maintained at 0.0133 pascals (0.0001 Torr) or less. All orbital operational conditions and all equipment functional modes including redundancy shall be tested. The qualification test shall include at least eight complete hot-cold cycles at the maximum predicted orbital rate of temperature change and with at least an 8-hour soak at each temperature extreme. Operating time should be divided approximately equally between redundant circuits.

6.2.7.4 <u>Supplementary Requirements</u>. Since the purpose of the more severe temperature extreme is to demonstrate an adequate design margin, it may be necessary to force temperature extremes at certain locations by altering thermal boundary conditions locally or by altering the operational sequence to provide additional heating or cooling. Adjacent equipments may be turned on or off; however, any special conditioning within the space vehicle shall generally be avoided. External baffling, shadowing. or heating shall be utilized to the extent feasible.

8.6.1.2 <u>Rationale for Thermal Vacuum Qualification Tests</u>. The objective of the vehicle level qualification thermal vacuum test is to verify satisfactory functional performance of the vehicle when it is exposed to vacuum conditions and design level temperature extremes. During this test, temperatures of individual components must not be allowed

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

to exceed their component qualification levels. If component failures or anomalies occur during this vehicle level test, thermal data are needed to aid failure analysis and to determine whether performance and material degradation due to environment exposure are within acceptable limits.

8.6.1.3 Guidance for Thermal Vacuum Qualification Tests. The vehicle is divided into separate equipment areas or zones for the thermal vacuum tests. The equipment areas are defined by the number of critical or sensitive components selected as drivers for the test. For example, an entire equipment compartment may be defined as an equipment area, or a compartment could be subdivided into critical components within the compartment. A space vehicle may be divided into as many equipment areas as necessary to test critical subsystems and components over the thermal range. Note that some subsystems may be located within more than a single equipment area.

The quantity of instrumentation required for the thermal vacuum tests may vary widely from program to program depending on the size, complexity, and thermal sensitivity of vehicle equipment. Sufficient thermal data should be obtained such that every component may be evaluated. It is recommended that consideration be given to instrumenting components such as the following:

- Those components whose function is sensitive to variation in thermal conditions, such as gyroscopes, should be instrumented with several thermocouples in order to detect thermal gradients which may exist across the component.

- All flight-critical components should be instrumented with thermocouple in order to verify the component qualification requirements, with respect to temperature extremes.

- Those components which have a surface facing heat sources or cold walls should be instrumented with a thermocouple on that surface in order to prevent that surface from being exposed to temperatures beyond its qualification limit. These thermocouples should not be used for test control of an equipment zone. Thermocouples should be placed on at least one other surface (surfaces not facing the heat source or cold wall), in order to detect temperature gradients across the unit and to determine the control temperature of the unit.

- Components which are not flight-critical should also be instrumented with thermocouples if they

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

are temperature-sensitive, or if the temperatures they will see in flight cannot be predicted through thermal analysis with sufficient accuracy.

- Thermocouple should be strategically located on components with heat sinks or thermal shunts so that an assessment of these systems can be made.

- Instrumentation, such as quartz microbalances and liquid nitrogen-cooled cold fingers, should be strategically located to monitor the rate and quantity of outgassing and to collect contamination data.

- All test instrumentation should have current calibration and alignment dates prior to installation on the test vehicle.

- The power consumption of pertinent components should be recorded prior to test initiation at several voltage levels.

- Equipment that is operational during launch should be operational during the chamber pressure pump-down. Components whose design is semivented should also be operational during the chamber pressure pump-down. These components should be monitored for corona and multipacting during this time.

- All flight thermocouples should be operational throughout the thermal vacuum test. All thermocouples, both flight and test, should record temperature data in real time. Hard copies of the temperature data should be obtained periodically and before, during, and after significant events.

- In the event of a power outage or failure of the real-time data acquisition system, precautions should be preplanned to prevent the space vehicle and components from being exposed to environments beyond their qualification limits.

- Photographs of the test article orientation within the thermal vacuum chamber, and of the locations of all thermocouples and contamination monitors, should be taken prior to closeout of the thermal vacuum chamber.

- Those components and hydraulic lines which contain fluids should be closely inspected before and

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

after the test and, if at all possible, they should be monitored for leaks during the thermal vacuum test.

0 If at all possible, periodic visual checks of the space vehicle should be conducted during this test (i.e., through portholes).

8.6.2 Thermal Vacuum Acceptance Tests

8.6.2.1 Standard Criteria. Contents of Paragraph 7.1.7 of MIL-STD-1540B (the requirements for space vehicle thermal vacuum acceptance tests) are as follows:

7.1.7 Thermal Vacuum Test, Space Vehicle Acceptance

7.1.7.1 Purpose. This test detects material, process, and workmanship defects that would respond to thermal vacuum and thermal stress conditions and verifies thermal control.

7.1.7.2 Test Description. Same as 6.2.7.2.

7.1.7.3 Test Levels and Duration. Temperatures in various equipment areas shall be controlled by the external test environment and internal heating resulting from equipment operation so that the hot (or cold) temperature on at least one component in each equipment area equals the maximum (or minimum) predicted temperature as defined in 3.25. The temperature extremes shall be established by a survey of predicted temperatures in various equipment areas and may have to be adjusted to performance of the most sensitive components in a particular area. The pressure shall be maintained at 0.0133 pascals (0.0001 Torr) or less. Duration shall be sufficient to test all orbital operational conditions and all equipment functional modes Including redundancy. Operating time should be divided approximately equally between redundant circults. If the thermal cycling test (7.1.8) is not conducted, the thermal vacuum acceptance test shall include at least four complete hot-cold cycles at the maximum predicted orbital rate of temperature change and have at least an 8-hour soak at each temperature extreme of each cycle.

During one temperature cycle, thermal equilibrium shall be achieved at both hot and cold extremes to allow verification of performance of the thermostats, louvers, heat pipes, electric heaters, and the control authority of active thermal systems. Thermal equilibrium has been achieved when equipment temperature change is not more than 3 deg C per hour.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

7.1,7.4 Supplementary Requirements. It may be necessary to force temperature extremes at certain locations by altering thermal boundary conditions locally or by altering the operational sequence to provide additional heating or cooling. Any special conditioning within the space vehicle shall generally be avoided. External baffling, shadowing. or heating shall be utilized to the extent possible.

8.6.2.2 Rationale for Thermal Vacuum Acceptance Tests. Thermal vacuum acceptance tests are conducted on space vehicles to demonstrate flightworthiness and to disclose quality deficiencies by subjecting each flight article to vacuum conditions and design level temperature extremes expected in service. The space vehicle thermal vacuum acceptance test also serves as a source for data which may be used to compare with component expected flight levels, component acceptance test levels, space vehicle qualification levels, and as a diagnostic aid in the event of component malfunction or failure. The thermal vacuum acceptance test serves as a source for thermal data which may be used to compare with component design temperature limits, component acceptance test levels, system qualification levels, and as a diagnostic aid in the event of component failure during or after the test. During the system test, temperatures of individual components should not be allowed to exceed their component acceptance test levels.

8.6.2.3 Guidance for Thermal Vacuum Acceptance Tests. The quantity of instrumentation required may vary widely from program to program depending on the complexity and thermal sensitivity of vehicle equipment. Sufficient thermal data should be obtained such that all flight-critical, thermally sensitive components may be evaluated. During the test, temperatures of individual components must not be allowed to exceed their component acceptance levels.

8.7 SPACE VEHICLE THERMAL BALANCE TEST

8.7.1 Standard Criteria. Contents of Paragraph 6.2.8 of MIL-STD-1540B (requirements for space vehicle thermal balance qualification test) are as follows:

6.2.8 Thermal Balance Test, Space Vehicle Qualification

6.2.8.1 Purpose. This test verifies the analytical thermal model and demonstrates the ability of the space vehicle thermal control subsystem to maintain components, subsystems, and the entire space vehicle within the specified operational temperature limits. This test also verifies the adequacy of component thermal design criteria.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

6.2.8.2 Test Description, The qualification space vehicle shall be tested to simulate the thermal environment seen by the space vehicle during the transfer orbit and orbital mission phases. Tests shall be conducted over the full mission range of seasons, equipment duty cycles, solar angles, and eclipse combinations so as to include the worst case high and low temperature extremes for all space vehicle components. Special emphasis shall be placed on defining the test conditions expected to produce the maximum and minimum battery temperatures. Sufficient measurements shall be made on the space vehicle Internal and external components to effect verification of the space vehicle thermal design and analyses. The power requirements of all thermostatically controlled heaters shall be verified during the test. The test chamber, with the test item installed, shall provide a pressure of 0.0133 pascals (0.0001 Torr). or less. Where appropriate, provisions shall be made to prevent the test item from "seeing" warm chamber walls by using black-coated cryogenic shrouds of sufficient area and shape that are capable of approximating liquid nitrogen temperatures. The space vehicle thermal environment may be supplied by one of the following three methods:

a. *Method I. Absorbed Flux. The absorbed solar, albedo, and planetary irradiation is simulated using heater panels or IR spectrum adjusted for the external thermal coating properties and projected by IR lamps or heater panels.*

b. *Method II. Incident Flux. The Intensity, spectral content, and angular distribution of the Incident solar, albedo, and planetary Irradiation is simulated.*

c. *Method III. Combination. Thermal environment is supplied by a combination of incident and absorbed irradiation.*

The selection of the method and fidelity of the simulation depends upon details of the space vehicle thermal design such as vehicle geometry, the size of internally produced heat loads compared with those supplied by the external environment, and the thermal characteristics of the external surfaces. Instrumentation shall be incorporated down to the component level to evaluate total space vehicle performance within operational limits as well as to identify component problems. The space vehicle shall be operated and monitored throughout the test. Dynamic

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

orbital simulation of the space vehicle thermal environment shall be provided unless the external space vehicle temperature does not vary significantly with time. For example, static simulation is usually adequate for spinning space vehicles.

6.2.8.3 Test Levels and Duration. Test conditions and durations for this test are dependent upon the space vehicle configuration design, and mission details. Normally boundary conditions for evaluating thermal design shall include: (a) maximum external absorbed flux plus maximum internal dissipation, and (b) minimum external absorbed f1ux plus minimum internal power dissipation. The thermal time constant of the subsystems and orbital maneuvering bath influence the time required for the space vehicle to achieve thermal equilibrium and hence the test duration. Thermal equilibrium has been achieved when the equipment temperature change is no more than 3 deg C per hour. The tests should simulate the full range of seasons, equipment duty cycles, solar angles. and eclipse combinations so as to produce the worst case high and low temperature extremes for all space vehicle components.

6.2.8.4 Supplementary Requirements. This test augments and validates the detailed thermal analysis. Pass criteria depend not only on survival and operation of each equipment within specified temperature limits, but also on correlation of the test with theoretical thermal models. As a goal, correlation of test results to the thermal model predictions should be within ± 3 deg C. Lack of correlation with the theoretical models may indicate either a deficiency in the model, test setup, or space vehicle hardware. The thermal balance test can be combined-with the thermal vacuum test. The correlated thermal math model is then used to make the finals temperature predictions for the various mission phases. including prelaunch, ascent, and on-orbit. The thermal margins are then based on these final temperature predictions.

8.7.2 Rationale for Thermal Balance Test Requirememts. The main purpose of the thermal balance vacuum test is to provide thermal data to verify the adequacy of the thermal model of the space vehicle being tested. This test should be conducted for one-of-a-kind spacecraft, the lead vehicle of a series of spacecraft, a block change in a series of vehicles, upper stages, and sortie pallets designed to fly with the shuttle. Since the test is designed to provide thermal data to

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

verify the space vehicle thermal model, ample thermocouples or thermistors consistent with MIL-STD-1540B instrumentation accuracy should be used to obtain the appropriate information. Typically, two orbital environments are simulated: one hot and the other cold. These environments, however, may not be the hottest or coldest for the space vehicle. Test or subsystem restrictions may prevent running the hottest and coldest environments . Again, the test is to verify the thermal model, not to test the spacecraft at its extremes.

8.7.3 Guidance for Use of Thermal Balance Test. After the test is completed, the temperature predictions made before the thermal model for the test environments are compared to the corresponding test data. Those differences that fall outside the correlation goal of ±3 deg C require either a good explanation or a model adjustment, depending on how large the differences deviate from ±3 deg C. The correlated math model is then used to make the final temperature predictions for the various mission phases, including prelaunch, ascent, and on-orbit.

The thermal margins are then based on these final temperature predictions. If these passive margins are less than 11 deg C or its equivalent for active systems, then either a design change or a waiver to MIL-STD-1540B is required. As noted in Paragraph 3.25 of MIL-STD-1540B, the 11 deg C passive thermal margin or its equivalent for an active system, is applied to the final orbital temperature predictions made by the correlated model. This implies, as stated in MIL-STD-1540B (Paragraph 3.25), that even larger thermal margins (passive or active) are required at the beginning of a program in order to account for design changes that almost inevitably occur during the evolution of a program. This is a cost-effective means of avoiding costly design changes late in the program.

8.8 SPACE VEHICLE THERMAL CYCLING TESTS

8.8.1 Thermal Cycling Qualification Test

8.8.1.1 Standard Criteria. Contents of Paragraph 6.2.9 of MIL-STD-1540B (requirements for space vehicle thermal cycling qualification tests) are as follows:

> *6.2.9 Thermal Cycling Test, Space Vehicle Qualification*
>
> *6.2.9.1 Purpose. This test demonstrates the ability of the space vehicle to withstand the thermal stressing environment of the space vehicle thermal cycling acceptance test (7.1.8) plus a design margin.*
>
> *6.2.9.2 Test Description. The space vehicle shall be placed in a thermal chamber at ambient pressure, and a*

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

functional test shall be performed to assure readiness for the test. The space vehicle shall be operated and monitored during the entire test, except that space vehicle power may be turned off if necessary to reach stabilization at the cold temperature. Space vehicle operation shall be asynchronous with the temperature cycling, and redundant circuits shall be operated with approximately equal the on each redundant circuit. Unfavorable combinations of temperature and humidity shall be avoided so there is no moisture deposition either on the exterior surfaces of the space vehicle or inside spaces where the humidity is slow to diffuse, e.g., multilayer insulation and enclosed electronic equipment, When the relative humidity of the inside spaces of the space vehicle is below the value at which the cold test temperature would cause condensation. the temperature cycling shall begin. One complete temperature cycle is a period beginning at ambient temperature then cycling to one temperature extreme and stabilizing, then to the other temperature extreme and stabilizing, and then returning to ambient temperature. Strategically placed temperature monitors installed on components shall assure attainment and stabilization of the temperature extremes at several components. Auxiliary heating and cooling may be employed for selected temperature-sensitive components, e.g., batteries. If it is necessary to achieve the temperature rate of change, parts of the space vehicle such as solar panels and passive thermal equipment may be removed for the test. The last temperature cycle shall be a soak cycle during which the space vehicle shall remain at each temperature extreme while a functional test, including testing of redundant circuits, is conducted.

6.2.9.3 Test Levels and Duration. The space vehicle temperature-range from hot to cold shall be the maximum possible within the constraints of the component design temperatures. The minimum space vehicle temperature range should be 70 deg C. Auxiliary heating and cooling may be used to protect selected temperature sensitive components. The average rate of change of temperature from one extreme to the other shall be as rapid as possible. The test shall include 25 percent more thermal cycles than the thermal cycling acceptance test (7.1.8).

8.8.1.2 Rationale for Thermal Cycling Qualification Test. The objective of the vehicle level qualification thermal cycling test is to verify satisfactory functional performance of the vehicle when it is exposed to design level temperature extremes. An examination of failures found during space vehicle thermal vacuum testing indicates that the vacuum-related

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

failures and verification of the thermal control system occur early, and the later failures are probably due to temperature cycling. There is also a large set of data from thermal cycling tests at lower assembly levels, which suggests that a space vehicle thermal cycling test is a very effective test for surfacing latent defects. Thermal cycling tests are much less costly than thermal vacuum tests and are believed to be more revealing of thermal problems in most components than thermal vacuum tests. Thermal cycling tests may therefore be used to reduce the number of thermal vacuum testing cycles required during acceptance and thereby achieve a total test program that is more revealing and may be less costly. During thermal cycling tests of the vehicle, temperatures of individual components must not be allowed to exceed their component qualification levels. If component failures or anomalies occur during this vehicle level test, thermal data are needed to aid failure analysis and to determine whether performance and material degradation due to environment exposure is within acceptable limits.

8.8.1.3 Guidance for Use of Thermal Cycling Qualification Test. The vehicle level qualification thermal cycling test is required if a vehicle level acceptance thermal cycling test is required. The qualification thermal cycling test adds 25 percent more thermal cycles and a 10 deg C margin to the thermal cycling acceptance test for a total of 50 cycles over a 70 deg C range. Full qualification level thermal vacuum testing is still required. The retention of full qualification level thermal vacuum tests is necessary because the reduction of acceptance thermal vacuum testing cycles depends on the confidence obtained from the qualification thermal vacuum test. The acceptance test cycle reduction is based on the premise that the vacuum-related failures will all surface during the first temperature cycle of the acceptance thermal vacuum test, and that the temperature-related failures will all have been identified in the preceding thermal cycling test. The space vehicle qualification test program is intended to verify these premises. Also, the space vehicle qualification thermal vacuum test demonstrates the ability of the space vehicle to meet design requirements in the thermal vacuum environment. Thus, there is no reduction of the temperature cycles during the space vehicle qualification thermal vacuum test.

The quantity of instrumentation required for the space vehicle thermal cycling tests may vary widely from program to program depending on the size, complexity, and thermal sensitivity of vehicle equipment. Sufficient thermal data should be obtained such that every component may be evaluated. In general, the thermal instrumentation required is the same as for a thermal vacuum test (see Paragraph 8.6).

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

8.8.2 Thermal cycling Acceptance Test

8.8.2.1 Standard Criteria. Contents of Paragraph 7.1.8 of MIL-STD-1540B (requirements for space vehicle thermal cycling acceptance tests) are as follows:

7.1.8 Thermal Cycling Test, Space Vehicle Acceptance

7.1.8.1 Purpose. This test detects material, process, and workmanship defects by subjecting the space vehicle to a thermal cycling environment.

7.1.8.2 Test Description. Same as 6.2.9.2.

7.1.8.3 Test Levels and Duration. The space vehicle temperature range from hot to cold shall be the maximum possible within the constraints of the components acceptance temperatures. The minimum space vehicle temperature range shall be 50 deg C. Auxiliary heating and cooling may be used to protect selected temperature sensitive components. The average rate of change of temperature from one extreme to the other shall be as rapid as possible. Operating time should be divided approximately equally between redundant circuits. The minimum number of thermal cycles shall normally be 40.

7.1,8.4 Supplementary Requirements. If this test is implemented, only one thermal cycle is required in the thermal vacuum acceptance test specified in 7.1.7. Consideration should be given to conducting this test where considerable disassembly for rework of components has occurred or if maximum confidence in the system is required.

8.8.2.2 Rationale for Thermal Cycling Acceptance Test. All available data point to the high effectiveness of this test to surface defects. An examination of failures found during space vehicle thermal vacuum testing indicates that the vacuum-related failures and verification of the thermal control system occur early, and the later failures are probably due to temperature cycling. Thus, the space vehicle thermal vacuum test may be reduced to one temperature cycle if the space vehicle thermal cycling test option is also selected. This combination of two tests is believed to be more effective than only a thermal vacuum test for four temperature cycles. There also is a large set of data from thermal cycling tests at lower assembly levels which suggests that a space vehicle thermal cycling test is a very effective test for surfacing defects.

The stress test aspects of the acceptance thermal cycling tests have been found to be an important contribution to successful orbiting vehicles. Past programs have shown a correlation between more ground testing and less failures on

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

orbit. However, the reduction of acceptance thermal vacuum testing cycles depends on the confidence obtained from the qualification thermal vacuum test. The acceptance test cycle reduction is based on the premise that the vacuum-related failures will all surface during the first temperature cycle of the acceptance thermal vacuum test, and that the temperature- related failures will all have been identified in the preceding thermal cycling test. The space vehicle qualification test program is intended to verify these premises.

8.8.2.3 Guidance for Use of Thermal Cycling Acceptance Test. The data available from thermal cycling space vehicles indicated that the test effectiveness is relatively insensitive to the temperature rate of change, at least for the range of values that might be achievable for a space vehicle. In the interest of minimizing testing time and cost, the temperature change should be as fast as practical. Experience has shown that a complete temperature cycle can be achieved in less than eight hours. Analysis of the test results also indicates that the effectiveness of the thermal cycling test is a function of both the number of cycles and the range of temperature, and that the number of cycles is the more important parameter. Because of the limited data from space vehicle tests, it is not appropriate to specify the number of cycles for different temperature ranges. Instead, a single temperature range of 50 deg C was specified, which appears to be a representative value for many space vehicles. At this temperature range, the calculated test-effectiveness curve begins to flatten at about 40 cycles. For vehicles that can be tested at different temperature ranges, the number of cycles can be tailored.

It is believed that the stress which precipitate defects into failures during the thermal cycling test is mainly mechanical motion resulting from differential expansion and contraction of materials. This is supported by the types of failures which are identified during thermal cycling tests. These include broken wires, cold or broken solder joints, changes of adjustment, and SO forth. Some failures may only be manifest at the temperature extremes and not at other points in the temperature cycle. As an example, a broken solder joint may be making contact at ambient temperatures and may open at a temperature extreme. In order to detect such failures, the last temperature cycle contains temperature soak periods, with a functional test conducted at each temperature Soak extreme.

The quantity of instrumentation required for the space vehicle thermal cycling tests may vary widely from program to program depending on the size, complexity, and thermal sensitivity of vehicle equipment. Sufficient thermal data should be obtained such that every component may be evaluated. In general, the thermal instrumentation required is the same as for a thermal vacuum test (see Paragraph 8.6).

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

The space vehicle thermal cycling test is relatively new. Therefore, more data should become available on the relationship among test effectiveness, number of cycles, and temperature range as more programs elect this option. The tradeoff between cycles and temperature range may be clarified by further experience.

8.9 STRUCTURAL LOAD TEST

8.9.1 Standard Criteria. Contents of Paragraph 6.3.1 of MIL-STD-1540B (requirements for structural static load test, subsystem qualification) are as follows:

6.3.1 Structural Static Load Test, Subsystem Qualification

6.3.1.1 Purpose. This test demonstrates the adequacy of the structure to meet requirements of strength or stiffness, or both, with the desired design margin when subjected to simulated critical environments, such as temperature and loads, predicted to occur during its service life.

6.3.1.2 Test Description. The structural configuration, materials, and manufacturing processes employed in the qualification test specimens shall be identical to those of flight articles. When structural items are rebuilt or reinforced to meet specific strength or rigidity requirements, all modifications shall be structurally identical to the changes incorporated in flight articles, The support and load application fixture shall consist of an adequate replication of the adjacent structural section to provide boundary conditions which simulate those existing in the flight article. Static loads representing the design limit load and the design ultimate load (see 3.46) shall be applied to the structure, and measurements of the strain and deformation shall be recorded. Strain and deformation shall be measured before loading, after removal of the limit loads, and at several intermediate levels up to limit load for post-test diagnostic purposes. The test conditions shall include the combined effects of acceleration, pressure, preloads. and temperature. These effects can be simulated in the test conditions as long as the failure modes and design margins are enveloped by the simulations. For example, temperature effects, such as material degradation and additive thermal stresses, can often be accounted for by increasing mechanical loads. Analysis of flight profiles shall be used to determine the proper sequencing or simultaneity for application of thermal stresses. When prior loading histories affect the structural adequacy of the test article, these shall be included in the test requirements. If more than one ultimate load condition is to be applied to the same test specimen, a method of sequential load application shall be developed by which each condition may,

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

in turn, be tested to progressively higher load levels. The final test may be taken to failure to substantiate the capability to accommodate internal load redistribution, to provide data for any subsequent design modification effort. and to provide data for use in any weight reduction programs. Failures at limit load shall include material yielding or deflection which degrade mission performance and at ultimate load shall include rupture or collapse.

6.3.1.3 Test Levels and Duration

a. Static Loads. The loads. other than internal pressure in pressure vessels, shall be increased until failure occurs or until the specified test loads are reached.

b. Temperature. Critical flight temperature-load combinations shall be used to determine the expected worst case stress anticipated in flight.

c. Duration of Loading. Loads shall be applied as closely as possible to actual flight loading times, with a minimum dwell time sufficient to record test data such as stress, strain, deformation. and temperature.

6.3.1.4 Supplementary Requirements. Pretest analysis shall be conducted to identify the locations of minimum design margins and associated failure modes which correspond to the selected critical test load conditions. This analysis shall be used to locate instrumentation, to determine the sequence of loading conditions, and to afford early indications of anomalous occurrences during the test. This analysis shall also form the basis for judging the adequacy of the test loads. Internal loads resulting from the limit test conditions shall envelop all critical internal loads expected in flight; however, excessive internal loads peculiar to the test shall be avoided. In cases where a load or other environment has a relieving effect, the minimum. rather than the maximum, expected value shall be used in defining limit test loading conditions. In some instances, where only a small number of flight vehicles have been included in the program, the cost of a dedicated test article may represent an unacceptably high percentage of the program cost. In such cases, the failure test would not be conducted, and it would be necessary to subject flight hardware to test loads prior to flight. In this event, special precautions shall be taken to ensure that the structure can still withstand its predicted flight environment after it has been subjected to the test loads. Such precautions shall include at least the

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

special design requirement that no permanent deformation detrimental to mission performance shall occur and the inspection requirement that sufficient nondestructive testing be conducted after the test to ensure the integrity of the structure. Alternatively, each flight vehicle shall be proof-tested; proof levels may be less than ultimate levels but shall exceed limit levels. In this case, the vehicle shall be designed to withstand the proof levels without permanent deformation detrimental to mission performance, and a thorough post-test inspection of each flight vehicle shall be conducted to ensure the integrity of the structure.

8.9.2 Rationale for Requirements. These test requirements are intended to demonstrate the adequacy of the structural strength and stiffness of the space vehicle.

8.9.3 Guidance for use Of Requirements. Expanded guidance is provided for the situation in which dedicated test articles are not provided, and flight hardware is subjected to test loads l Table VII shows successful past examples of methods used to obtain static load qualification of flight structures.

TABLE VII. Flight Use of Static Load Qualification Test Equipment.

Program	Details of Static Load Qualification Test
A	Components from the development test model were subsequently used as flight hardware. Decision was made after post-test examination revealed that hardware had been tested well below yield strength.
B	After proof loading, vehicle was put through detailed refurbishment program and retested. Some minor rework was necessary to bring it up to flight configuration. Test article was used successfully as second flight article. Practice is to be continued in this program.
C	Refurbished test article (centerbody) is intended for use as third flight article and has been declared flightworthy. Article was tested to ultimate with no detectable yielding.
D	Support structure was dedicated qualification test article (not flown). Some reduced level qualification test experiment modules were successfully flown.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

Selection of specific options should be made on the basis of program-unique needs. Increased design levels may be used to reduce program risks for either the flight-test or ground-test phase of the program, with attendant weight penalties.

If an option permitting flight use of qualification hardware is selected, it is imperative that it be understood and accepted at program start. Understanding and early planning are essential to the successful flight use of a prototype satellite.

The factors given in Table VIII are minimum factors of safety to be used in conjunction with sound design practices and thorough analytical and test verifications of the design. These verifications include fully coupled dynamic load analysis by means of structural-dynamics modeling and modal test surveys; detailed stress analyses to show positive margins of safety; use of proven materials with well-characterized allowable; and adequate development and qualification test programs. Table VIII also shows the design and test options that are recommended for use with structural subsystems. In addition, these factors are to be used in conjunction with the following:

a. Industry standard manufacturing and inspection procedures that satisfy prevailing military standards and specifications

b. Additional factors to account for uncertainty in dynamically induced loads

c. Thorough monitoring of design, development, analysis, and testing

Option 2 in Table VIII qualifies a small fleet by means of a static test to 125 percent of limit load, with the condition that no detrimental deformation occurs during the test. This condition may require additional test instrumentation, at carefully chosen locations on critical structural elements, and careful post-test inspection. No demonstration of ultimate load-carrying capability is provided. However, the combination of a test to 125 percent of limit load, coupled with an ultimate design factor of safety equal to 1.4 (minimum), provides assurance of structural integrity equivalent to those of the other options.

The factors of safety given in Table VIII are for general structure and do not include factors of safety for pressure conditions (e.g., for pressure vessels or for hydraulic and pneumatic systems), for thermal load conditions, nor for special structures such as bearings, journals, or glass windows.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE VIII. Structural Design and Test Options a/

TYPE OF STRUCTURE	Test Level Factor b/	Design Factors of Safety (F.S.)			Test Success Criteria	Typical Application
		Yield	Ultimate			
			Unmanned Flight	Manned Flight		
NEW STRUCTURES c/						
Option 1. Test dedicated test article to ultimate (baseline)	1.4	1.0		1.4	No Failures d/ (for F.S. = 1.4)	Fleet
	1.25	1.0	1.25		No detrimental deformation e/ (for F.S. = 1.0)	
Option 2. Reduced level test on single flight article	1.25	1.25	1.4	1.4	No detrimental deformation d/,f/	Small fleet
Option 3. Proof test each flight article	1.1	1.1	1.25	1.25	No detrimental deformation d/	Few flight articles
EXISTING STRUCTURES g/						

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

Table VIII. Structural Design and Test Options. (Continued)

NOTES :

a/ Factors of safety shown here are minimum values for general structure. They apply to limit internal loads, stresses, or strains, resulting from mechanically induced loads (except pressure) which occur during various mission phases. Yield factors of safety larger than shown herein may be used to reduce risk of detrimental deformations during test. Ultimate factors of safety for manned or unmanned flights should be selected individually for each loading condition. Factors of safety for pressure and thermal loading conditions also apply when pertinent.

b/ Test level factor = factor multiplying limit load. The limit load is discussed in Paragraph 5.2 of this document.

c/ Option 1 is used for programs having a fleet of weight-critical flight articles and is the conventionally accepted practice. Option 2 may be used for programs having a small fleet with costly but not weight-critical structural subsystems. Option 3 is applicable to programs having one or at most a few weight-critical flight articles.

d/ A failure is any rupture, collapse, seizure, excessive wear, excessive deformation, or any other phenomenon which prevents any portion of the vehicle structure from sustaining the specified test loads and temperatures.

e/ Detrimental Deformation = Either elastic or inelastic deformation resulting from the application of test loads and temperature which prevents any portion of the vehicle structure from performing its intended function, or which prevents the unloaded structure from keeping its original dimensions and alignment within Specified manufacturing and assembling tolerances.

f/ A minimum margin of safety equal to 0.15 should be used for instability failure modes when Option 2 is used.

g/ For existing structures to be used in new missions, the design and test verification of either existing designs or off-the-shelf structures should conform to one of the design and test options given above for the new-mission loads. Reinforcing and partial or local redesign of existing structures are acceptable to upgrade the load-carrying capability of the original design.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

Utilization of qualification test hardware for flight generally leads to overdesign. Therefore, consideration should be given to using a dedicated structural qualification subsystem only, with smaller payload items being qualified by Options 2 or 3, Table VIII. Program D (see Table VII) was a case of such a combination of qualification strategies. In that instance, a large support structure was a dedicated qualification test item (not flown). The smaller experimental packages (which were qualifications and flight articles) were qualified by Options 2 and 3 and, in one particular case, by a combination of these two options.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

SECTION 9

COMPONENT LEVEL TESTS

9.1 COMPONENT TEST BASELINES

9.1.1 Standard Criteria. Contents of Paragraphs 6.4 and 7.3 of MIL-STD-1540B (requirement for component qualification and acceptance tests) are as follows:

6.4 COMPONENT QUALIFICATION TESTS

The space vehicle component qualification test baseline consists of all the required tests specified in Table II. The test baseline shall be tailored for each program, giving consideration to both the required and optional tests; however. deviations from the baseline of required tests shall be approved by the contracting officer. Each component that is acceptance tested as a component shall undergo comparable qualification tests as a component. Component qualification tests shall normally be accomplished entirely at the component level. However, in certain circumstances, required component qualification tests may be conducted partially or entirely at the subsystem or space vehicle levels of assembly. Tests of components such as interconnect tubing. radio frequency circuits. and wiring harnesses are examples where at least some of the tests can usually be accomplished at higher levels of assembly.

Where components fall into two or more categories of Table II, the required tests specified for each category shall be applied, For example, a star sensor may be considered to fit both "Electronic Equipment" and "Optical Equipment" categories. In this example, a thermal cycling test would be conducted since it is required for electronic equipment. even though there is no requirement for thermal cycling optics. Similarly. an electric motor-driven actuator fits both 'Electrical Equipment" and "Moving Mechanical Assembly" categories. The former makes thermal cycling a required test, even though this test is optional for the moving mechanical assembly category.

7.3 COMPONENT ACCEPTANCE TESTS

The space vehicle component acceptance test baseline consists of all the required tests specified in Table IV. The test baseline shall be tailored for each program,

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

MIL-STD-1540B TABLE II. Component Qualification Tests

Test	Ref. Para-graph	Sugg. Seq.	Electronic or Electri-cal Eqpmt.	Antennas	Moving Mechan. Assemb.	Solar Panel	Batteries	Valves	Fluid or Propul. Equipmt.	Pres-sure Vessels	Thrusters	Thermal Equip-ment	Optical Equip-ment
Functional	6.4.1	1 (1)	R	R	R	R	R	R	R	R	R	R	R
Thermal Vacuum	6.4.2	9	R	R	R	R	R	O	R	O	R	R	R
Thermal Cycling	6.4.3	8	R	O	O	O	O	O	O	-	-	-	-
Sinusoidal Vibration	6.4.4	5	O	O	O	-	O	O	O	O	O	O	O
Random Vibration	6.4.5	4	R	R (3)	R	-	R	R	R	R	R	R	R
Acoustic	6.4.6	4	-	R (3)	-	R	-	-	-	-	-	-	-
Pyro. Shock	6.4.7	3	R	O	O	O	O	O	O	-	O	O	O
Acceleration	6.4.8	7	O	R	O	O	O	-	-	O	-	-	R
Humidity	6.4.9	10	O	O	O	O	O	O	O	O	O	O	O
Pressure	6.4.10	11	-	-	R	-	R (2)	R	R	R	R	-	-
Leak	6.4.11	2,6,12	R (2)	-	R	-	R (2)	R	R	R	O	O	-
EMC	6.4.12	13	R	O	O	-	-	-	-	-	-	-	-
Life	6.4.13	14	O	O	O	O	O	O	-	O	O	O	O

Legend:
R = Required
O = Optional test
- = No requirement

Notes:
(1) Functional tests shall be conducted prior to and following environmental test.
(2) Required only on sealed equipment or pressurized equipment.
(3) Either random vibration or acoustic test required with the other optional

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

MIL-STD-1540B TABLE IV. Component Acceptance Tests

Test	Ref. Para-graph	Sugg. Seq.	Electronic or Electri-cal Eqpmt	Antennas	Moving Mechan. Assemb.	Solar Panel	Batteries	Valves	Fluid or Propul. Equipmt.	Pres-sure Vessels	Thrusters	Thermal Equip-ment	Optical Equip-ment
Functional	7.3.1	1(1)	R	R	R	R	R	R	R	R	R	R	R
Thermal Vacuum	7.3.2	7	R(2)	O	R	O	R	R	R	O	R	R	R
Thermal Cycling	7.3.3	6	R	O	O	O	O	R	R	-	-	-	-
Random Vibration	7.3.4	4	R	R(4)	R	-	O	R	R	O	R	R	R
Acoustic	7.3.5	4	O	R(4)	-	O	-	-	-	-	-	-	-
Pyro Shock	7.3.6	3	O	-	O	-	-	-	-	-	-	-	O
Pressure	7.3.7	9	-	-	O(3)	-	R(3)	R	R	R	O	-	-
Leak	7.3.8	2,5,10	R(3)	-	R(3)	-	R(3)	R	R	R	O	-	-
Burn-in	7.3.9	8	R	-	O	-	-	R	-	-	R	-	-

Legend: R = Required
O = Optional test
\- = No requirement

Notes:
(1) Functional tests shall be conducted prior to and following environmental test.
(2) Required only on unsealed units and on high power RF equipment.
(3) Required only on sealed or pressurized equipment.
(4) Either random vibration or acoustic test required with the other optional.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

giving consideration to both the required and optional tests; however, deviations from the baseline of required tests shall he approved by the contacting officer. Component acceptance tests shall normally be accomplished entirely at the component level. However, in certain circumstances, the required component acceptance tests may be conducted partially or entirely at the subsystem or space vehicle levels of assembly. Acceptance tests of components such as interconnect tubing, radio frequency circuits, and wiring harnesses are examples where at least some of the tests can usually be accomplished at higher levels of assembly.

Where components fall into two or more categories of Table IV, the required tests specified for each category shall be applied. For example, a star sensor may be considered to fit both 'Electronic Equipment" and "Optical Equipment" categories. In this example, a thermal cycling test would be conducted since it is required for electronic equipment, even though there is no requirement for thermal cycling optics. Similarly, an electric motor-driven actuator fits both "Electrical Equipment" and "Moving Mechanical Assembly" categories. The former makes thermal cycling a required test, even though this test is optional for the moving mechanical assembly category.

9.1.2 Rationale for Component Test Baseline Requirement. Environmental qualification tests are a formal demonstration that a production component (or prototype) is adequate to successfully sustain specified environmental design levels. These tests are mainly performed to determine if there are factors that may have been overlooked during design, analysis, or manufacturing. Additionally, the environments used during these tests are the design levels that are more severe than those predicted to occur during flight in order to account for variabilities in subsequent production articles and other uncertainties. Qualification test requirements, therefore, incorporate margins which are added to the range of environmental extremes and stresses expected to occur in service. These design environmental levels are typically based upon the maximum and minimum predicted environmental levels for an item during its operational life plus the appropriate environmental design margin. The maximum expected extremes of the operational environments are defined in Paragraphs 3.8 and 3.9 of MIL-STD-1540B. For example, the standard operating thermal range for components of -24 deg C to +61 deg C is usually used when the maximum predicted operating range is less severe. The environmental design margins specified are primarily intended to incorporate the allowable test condition tolerances and to accommodate any differences among production

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

units. The environmental design margins are also intended to assure qualification test levels that are more severe than the maximum operating ranges that can occur in flight and help assure against performance degradation and fatigue failures due to repeated acceptance testing and operational use. For example, the 10 deg C environmental design margins specified in MIL-STD-1540B make the standard thermal design range for components from -34 deg C to +71 deg C. This standard design range for space components is similar to that used for aircraft subsystems and therefore should not impose unusual design problems in most cases. In addition, this standard design range encourages the development of standard modules, provides a very revealing test screen for defective components, allows components to be moved to other locations on a spacecraft without affecting qualification, and may allow the use of a qualified component on other spacecraft without requalification.

Before qualification testing, the space components should have been subjected to the same controls, inspections, alignments , and tests imposed on flight component. This includes completion of the environmental acceptance tests.

Environmental acceptance tests are conducted on space components to demonstrate flightworthiness and to disclose quality deficiencies in the flight article. Acceptance tests are intended to satisfy these goals by subjecting the space component to the maximum environmental exposures expected in service. The test program is comprised of a series of tests; some are required tests, while others are optional.

The suggested test sequences require functional tests before and after each environmental test. Additionally, certain functional tests are required to be performed during some of the environmental tests. The sequencing is based on a combination of the order in which the environments are encountered during flight and the desire to perceive defects as early in the test sequence as possible. The categorization of tests into "required" and “optional)’ was guided by the sensitivity of the type of component to the specific environment and by the probability of encountering the environment. As an example, leak tests are required only on sealed or pressurized equipment, since such equipment is sensitive to loss of pressure, vacuum, or purge mechanism.

9.1.3 Guidance for Use of Component Test Baseline Requirements. The sequencing and categorization of the tests should be tailored to each specific component for each program. This tailoring should consider both increasing and decreasing the severity of the tests. For example, while random vibration tests for electronic components are normally more revealing than acceleration tests, some electronic components may require both types of tests.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

The humidity qualification test is designated as optional for all components; however, if components are not fully environmentally protected on the ground, such tests should become mandatory. This is also the case for such tests as fungus, sand. dust, salt spray, explosion-proofing, and radiation which are not specified in MIL-STD-1540B, but each should become mandatory when there are requirements.

Component qualification life tests are optional. These tests should be applied to selected components where an evaluation of Component reliability has determined that such tests are necessary to convey confidence that components have the capability to withstand the maximum duration or cycles of operation without fatigue or wearout failures.

The mechanical and electrical functional tests are extremely important elements in the test baselines. The functional tests are conducted prior to and after each of the environmental tests. They should be designed to verify that performance of the components meets the specification requirements, that the components are compatible with ground support equipment, and that all software used is validated. The electrical functional tests should apply electrical inputs of interfaces including redundant circuits and measure the component performance. The mechanical functional tests should apply mechanical inputs including torques, loads, and motions, and should measure performance. The electrical and mechanical inputs should be varied through their specification ranges to verify the component performance throughout the range. In addition, the electrical functional tests should include negative logic testing to verify lockout, to assure that no function other than the intended function was performed, and to verify that the signal was not present other than when programmed. To the extent practicable, the functional tests should also be designed so that a data base of critical parameters can be established for trend analysis. This is accomplished by measuring the same critical parameters in all of the functional tests conducted before, during, and after each of the baseline environmental tests.

It is extremely important that functional tests be conducted before and after each environmental test. These functional tests provide the criteria for judging successful survival of the space component in a given test environment. It is also important to perform functional tests of the component while the environment is being imposed, if the component is expected to be fully operational under that environment. Many defects, which otherwise escape detection by pre- and post-test functional checks, reveal themselves during environmental tests. For example, intermittent may be caused by foreign bodies, contaminants, inadequate clearances, cracks, debonds.,

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

and damaged connectors that might only be revealed during environmental tests. Therefore, regardless of the functional mode of the component during launch and ascent, the component should be functionally operated and monitored during dynamic as well as thermal tests to increase overall test effectiveness. Practical limitations frequently restrict the extent of operation of the component during the relatively brief acoustic or vibration tests. In recognizing this problem, MIL-STD-1540B permits extended functional testing with the component operating and monitored, but conducted at a level 6 dB lower than the required test level, after the required environmental exposure has been satisfied.

Tables IX through XIV summarize important parameters of component environmental baseline tests. They are useful as a concise reference to major test requirements and for comparing qualification to acceptance test requirements.

TABLE IX. Thermal Vacuum Test--Component Qualification and Acceptance Test Parameters.

Thermal Vacuum Test Parameters	Qualification - Para. 6.4.2	Acceptance - Para. 7.3.2
Temperature Range (Differential)	105°C	85°C
Temperature Extremes	Min. predicted with -10°C environmental design margin, to maximum predicted with the +10°C environmental design margin, or at least -34°C to +71°C	Min. predicted to max. predicted, or at least -24°C to +61°C
Number of Cycles	3 cycles minimum	1 cycle minimum
Dwell	12-hour minimum at temp. extremes	12-hour minimum at temp. extremes
Pressure	10^{-4} Torr or less	10^{-4} Torr or less

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE X. Thermal Cycling Test--Component Qualification and Acceptance Test Parameter.

Thermal Cycling Test Parameters	Qualification - Para. 6.4.3	Acceptance - Para. 7.3.3
Temperature Range (differential)	105°C min.	85°C min.
Temperature Extremes	Min. predicted with -10°C environmental design margin, to maximum predicted plus the +10°C environmental design margin, or at least -34°C to +71°C	Min. predicted to max. predicted, or at least -24°C to +61°C
Number of Cycles	3X acceptance (24 cycles min.)	8 cycles minimum
Dwell	1-hour minimum at temp. extremes (each cycle)	1-hour minimum at temp. extremes (each cycle)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XI. Pyrotechnic Shock Test--Component Qualification and Acceptance Test Parameters.

Pyrotechnic Shock Test Parameters	Pyro shock qualification - Para. 6.4.7	Pyro shock screening acceptance - Para. 7.3.6
Shock Level	Minimum level equal maximum predicted environment plus 6 dB environmental design margin	Maximum predicted environment
Number of Shocks	Number required in each direction of each of 3 axes to meet amplitude criteria 3 times (18 shocks)	One shock in each direction of each of 3 axes (6 shocks)
Shock Duration	Greater of 20 msec or flight shock duration	Not specified in Para. 7.3.6
Vibration Level	Not applicable - no vibration in Para. 6.4.7	Min. of 4.5 grins or 3 dB below acceptance vibration test level
Vibration Duration	Not applicable - no vibration in Para. 6.4.7	5 minutes dwell plus 10-second bursts (minimum of 20 bursts) for each of 3 axes

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XII. Random Vibration Test--Component Qualification and Acceptance Test Parameters.

Random Vibration Test Parameters	Qualification - Para. 6.4.5	Acceptance - Para. 7.3.4
Vibration Level	Spectrum for maximum predicted environment plus environmental design margin of 6 dB, and minimum of 12 grins overall for weight of 50 lb max.	Min. of spectrum for max. predicted environment, and minimum of 6 grins overall for weight of 50 lb max.
Test Duration	Greater of 3 times ex-petted flight exposure time per axis or 3 X accept. test duration per axis, and min. of 3-minutes per axis	Minimum of expected flight exposure time, and minimum of one minute per axis
Tolerances	± 1.5 dB overall ± 3.0 dB for 500-2000 Hz	± 1.5 dB overall ± 3.0 dB for 500-2000 Hz

Table XIII. Acoustic Test--Component Qualification and Acceptance Test Parameters.

Acoustic Test Parameters	Qualification - Para. 6.4.6	Acceptance - Para. 7.3.5
Sound Pressure Level	Greater of: maximum predicted environment + environmental design margin of 6 dB, or 144 dB overall	Greater of: maximum predicted environment, or 138 dB overall
Test Duration	Greater of: 3 times expected flight exposure time, or 3X acceptance test duration. or 3-minutes minimum	Greater of: expected flight exposure time, or l-minute minimum

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

TABLE XIV. Burn-in Test--Component Qualification and Acceptance Test Parameters.

Burn-in Test Parameters	Qualification	Acceptance - Para. 7.3.9
Temperature Range (differential)	No Qual. Test Specified by MIL-STD-1540B	85°C
Temperature Extremes		Min. predicted to max. predicted or at least -24°C to +61°C
Number of Temperature Cycles		18 cycles min. including thermal cycling test cycles
Total Operating Time	Items being qualified shall have completed the acceptance tests including applicable burn-in	300 hour minimum including thermal cycling time (or 100 cycles min. for cycle-sensitive components)
Dwell		1 hour minimum at temperature extremes

9.2 COMPONENT PRESSURE TESTS

9.2.1 Standard Criteria. Contents of Paragraphs 6.4.10 and 7.3.7 of MIL-STD-1540B (requirements for component qualification and acceptance pressure tests) are as follows:

6.4.10 PRESSURE TEST. COMPONENT QUALIFICATION

6.4.10.1 PURPOSE. This test demonstrates that the design and fabrication of such Items as pressure vessels, pressure lines, fittings, and valves provide an adequate

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

margin such that structural failure or excessive deformation does not occur at the maximum expected operating pressure.

6.4.10.2 TEST DESCRIPTION

a. *Proof Pressure.* For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical use temperature and subjected to a minimum of one cycle of proof pressure. A proof pressure cycle shall consist of raising the Internal pressure (hydrostatically or pneumatically, as applicable) to the proof pressure, maintaining it for 5 minutes, and then decreasing the pressure to zero. Evidence of permanent set or distortion that exceeds 0.2 percent or failure of any kind shall indicate failure to pass the test.

b. *Proof Pressure for Valves.* With the valve in the open and closed positions (if applicable), the proof pressure shall be applied for a minimum of three cycles to the inlet port for 5 minutes (hydrostatically or pneumatically, as applicable). Following the 5-minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The exterior of the unit shall be visually examined. Evidence of deformation that exceeds 0.2 percent or any failure shall indicate failure to pass the test. The test may be conducted at room ambient temperature.

c. *Burst Pressure (see 3.4).* For such items as pressure vessels, pressure lines, and fittings. the temperature of the component shall be consistent with the critical use temperature, and the component shall be pressurized (hydrostatically or pneumatically, as applicable and safe) to design burst pressure or greater. The internal pressure shall be applied at a uniform rate such that stresses are not imposed due to shock loading.

d. *Burst Pressure for Valves.* With the valve in the open or closed position, as applicable, the design burst pressure shall be applied to the inlet port for 5 minutes (hydrostatically or pneumatically, as applicable). Following the 5-minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The exterior of the unit shall

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

be visually examined for indications of deformation or failure. The test may be conducted at room ambient temperature.

6.4.10.3 TEST LEVELS

a. *Temperature. As specified in the test description. As an alternative, tests may be conducted at ambient room temperatures if the test pressures are suitably adjusted to account for temperature effects on strength and fracture toughness.*

b. *Proof Pressure. Unless otherwise specified, the proof pressure equals 1.5 times the maximum operating pressure.*

c. *Burst Pressure. Unless otherwise specified. the burst pressure equals two times the maximum operating pressure.*

6.4.10.4 SUPPLEMENTARY REQUIREMENTS. The component shall withstand proof pressure without leakage or detrimental deformation. Applicable safety standards shall be followed in conducting all tests.

7,3.7 Pressure Test, Component Acceptance

7,3.7.1 Purpose. This test detects material and workmanship defects which could result in failure of the pressure vessel or valves in usage.

7.3.7.2 Test Description. This test is the same as described in 6.4.10.2a and b, except that only one cycle shall be required, and test at elevated temperature is optional.

7.3.7.3 Test Levels. Same as 6.4.10.3.

7.3.7.4 Supplementary Requirements. Applicable safety standards shall be followed in conducting all tests.

9.2.2 Rationale for Pressure Test Requirement. The proof and burst pressure tests described in Paragraph 6.4.10 of MIL-STD-1540B are parts of the structural integrity verification program for all pressure vessels required by MIL-STD-1522. All pressure vessels, other than pressure vessels designed, fabricated, inspected, and tested in accordance with the ASME Boiler and Pressure Vessel Code, are classified as

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

fracture-critical components and, therefore, come under fracture control procedures. The design of such pressure vessels must satisfy minimum technical requirements for a fracture control program. These requirements include a comprehensive stress analysis, failure mode prediction based on results of the stress analysis, demonstration of safe-life and fail-safe design, and implementation of required quality assurance procedures. Satisfaction of these requirements, and their integration into a program of structural design, analysis, and test, assures the structural integrity of fracture-critical hardware.

9.2.3 Guidance for Use of Pressure Test Requirements. Note that the requirements discussed in this section apply to metallic pressure vessels and structures. Nonmetallic vessels and structures must have requirements established on a case-by-case basis.

Prior to test program planning, a detailed stress analysis of the structure is conducted under the assumption of no crack-like flaws in the structure. The analysis determines stresses and critical combinations of stresses resulting from loads, pressures, and temperatures associated with the expected operating environments. The results of the stress analysis determine potential failure modes of the structure. These are either ductile fracture or brittle fracture modes. Required test levels depend upon potential failure modes.

Pressure vessels and pressurized structures expected to fail in a ductile fracture mode may be conventionally designed. Such design uses design factors of safety and test factors selected on the basis of successful past expedience or specified by codes, specification, and standards (e.g., MIL-STD-1522). Typical design and test factors applied to these pressurized components are given in Table XV.

Pressure vessels and pressurized structures expected to fail in a brittle fracture mode are designed by a safe-life design method based on linear elastic fracture mechanics. This method establishes the appropriate design factor of safety and the associated proof factor. The proof pressure is calculated as the product of the limit pressure, proof factor, and a factor corresponding to the difference in material strength and fracture properties between test and design environments.

Pressure vessels and pressurized structures are qualified by a combination of a proof pressure test (a proof pressure combined with limit loads test if necessary), a burst pressure test (burst pressure combined with ultimate loads test if necessary), and, as appropriate, a safe-life test and fail-safe test. Environmental tests are performed with the proof and burst pressure tests to expose test units to the most severe

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XV. Design and Test Factors.

Component	Factors		
	Ultimate	Proof	Burst
Pressure vessels (tanks other than main propellant tanks and solid rocket motor cases)	--	1.50	2.00
Main propellant tanks and solid motor cases			
- Manned application	1.40	1.25	1.40
- Unmanned application	1.25	1.10	1.25
Pressurized structures			
- Flight loads: Manned	1.40	--	--
- Flight loads: Unmanned	1.25	--	--
- Internal pressure	2.00	1.50	2.00
Pressurized lines, fittings, and hoses			
- less than 1.5-inch diameter	--	1.50	4.00
- 1.5-inch diameter and larger	--	1.50	2.50
Accumulators, actuating cylinders, pumps, regulators, and valves	--	1.50	2.50

combination of environments, pressures, and loads. Test requirements are detailed in MIL-STD-1522.

Structural similarity of the flight hardware and the qualification test hardware ensures structural integrity of the flight hardware. In structuring the qualification test program, the highest practical level of assembly should be used. The

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

test fixtures, support structures, and test environments must not introduce erroneous test conditions. Qualification instrumentation and instrument locations should be based on the results of stress analysis. Instrumentation must provide sufficient data to ensure proper test conclusions. For qualification, one test article of each pressure vessel design is proof pressure tested in accordance with Paragraph 6.4.10 of MIL-STD-1540B without leak or detrimental deformation. It is then tested to the burst pressure level as described in Paragraph 6.4.10 of MIL-STD-1540B. The pressure vessel must sustain design burst pressure without rupture. The design burst pressure is calculated as the product of the limit pressure, burst pressure factor, and a factor corresponding to the differences in material strength and fracture properties between test and design temperatures.

Each pressurized component intended for flight must pass the one-cycle proof pressure test as described in Paragraph 7.3.7 of MIL-STD-1540B before flight.

9.3 COMPONENT LEAKAGE TESTS

9.3.1 Standard Criteria. Contents of Paragraphs 6.4.11 and 7.3.8 of MIL-STD-1540B (requirements for component qualification and acceptance leakage tests) are as follows:

6.4.11 Leakage Test, Component Qualification

6.4.11,1 Purpose. This test demonstrates the capability of pressurized components to meet the design leakage rate constraints specified in the component specifications.

6.4.11.2 Test Description and Alternatives. Component leak checks shall be made prior to initiation of, and following the completion of, component qualification thermal and vibration tests. Proof pressure tests per 6.4.10 shall be successfully completed before conducting leakage tests. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate. One of the following recommended methods shall be used:

a. Method I (gross leak test). The component shall be completely immersed in a liquid so that the upper most part of the test item is 5 ± 2.5 cm (2 ± 1 Inches) below the surface of the liquid. The critical side or side of interest of the component shall be In a horizontal plane facing up. The liquid, pressurizing gas, and the test item shall be 23 ± 10 deg c (73 ± 18 deg F). The gas used for pressurizing shall be clean and dry with a dewpoint

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

of at least -32 deg C (-25 deg F). Any observed leakage during Immersion as evidenced by a continuous stream of bubbles emanating from the component indicates a failure of seals.

b. Method II (fine leak test). The component shall be purged with nitrogen and then charged with helium to the required pressure (as specified in the component detail specification) before being sealed. The component shall then be placed in a suitable vacuum chamber and tested for helium leakage with a helium leak detector. The leakage rate shall be used to determine seal Integrity and shall not exceed the amount specified in the detailed component specification. This method is applicable to tape recorders and similar components.

c. Method III (for battery cases or pressurized components). The component shall be pressurized with dry nitrogen or other appropriate gas to the specified value. The pressure shall be monitored by a gage (or pressure transducer) for the required time. The drop in pressure shall not exceed the permitted amount as specified under the component specification.

d. Method IV (for hermetically sealed alkaline storage batteries). The battery shall be cleaned with alcohol while in the discharged state. A suitable indicator (e.g., dilute solution of phenolphthalein or other suitable color change Indicator) shall be applied to all seams, terminals. and pinch tubes subject to leakage of electrolyte. A change in the color of the indicator shall be an Indication of a leak. After testing, the test solution shall be removed (e.g., with distilled water).

e. Method V (for comporients of pressurized fluid systems). The components shall be pressurized to their maximum working pressure in each of the functional modes. Leakage shall be detected using an appropriate method (6.2.6.4). Propulsion system tanks and thrusters shall also be evacuated to the Internal pressure normally used for propellant loading and the Internal pressure monitored for Indications of leaking.

6.4.11.3 Test Levels and Duration. The leak tests shall be performed with the component pressurized at the maximum operating pressure and then at the minimum operating pressure if the seals are dependent upon pressure for proper sealing.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

The test duration shall be sufficient to detect any significant leakage. The test levels and duration for the typical methods of 6.4.11.2 are:

a. *Method I. The duration of Immersion shall be 60 minutes at each pressure,*

b. *Method II. The external test pressure shall be 0.133 pascals (0.001 Torr) or less and the duration of the test shall be 4 hours (for equipment that is operational in orbit for more than one day).*

c. *Method III. The test pressure is usually less than 343 kilopascals (50 psi). The pressure drop shall not exceed the specified amount (typically about 6.9 kilopascals (1 psi) in a 6-hour period at room temperature).*

d. *Method IV. The test results are visable within seconds.*

e. *Method V. The duration of the evacuated propulsion system component leak test shall not exceed the time that this condition is normally experienced during propellant loading.*

6.4.11.4 Supplementary Requirements. Component leak tests are considered adjunctive to the component qualification environmental tests in that their results are part of the success criteria for these tests.

7.3.8 Leakage Test, Component Acceptance

7.3.8.1 Purpose. This test demonstrates the capability of pressurized components to meet the leakage rate requirements specified in the component specifications.

7.3.8.2 Test Description and Alternatives. The component leak checks shall be made before and after exposure to each environmental acceptance test. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate. One of the methods given in 6.4.11.2 shall be used.

7.3.8.3 Test Levels and Duration. Same as 6.4.11.3.

9.3.2 Rationale for Leakage Test Requirements. The leakage tests are intended to demonstrate the capability of pressurized components to meet their design leakage rate constraints.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

9.3.3 Guidance for Use of Leakage *Test* Requirements. The NASA leakage testing handbook, NASA S-69-1117, provides detailed guidance for leakage testing Methods II, III, IV, and V plus a number of other specialized methods, including the use of radioactive tracers. The leakage test method should be selected to suit the design and performance requirements of the hardware item. It should prove that the item can function in its operational environment within specifications and without damaging leakage. Each test method listed in Paragraph 6.4.11.2 of MIL-STD-1540B also lists typical hardware to which the method can be applied.

Method I (gross leak test) describes an immersion leakage test which is a potentially destructive test. It is sometimes used on small parts where a gross leak in the item might be missed, due to the small cavity size, if a fine leak test were the only leak test conducted. Because it is a potentially destructive test, Method I is not recommended for space vehicle components. This method might have applicability for specialized development tests, qualification tests of some items, or tests of nonflight hardware.

9.4 COMPONENT LIFE TEST

9.4.1 Standard Criteria. Contents of Paragraph 6.4.13 of MIL-STD-1540B (requirements for life test, component qualification) are as follows:

> *6.4.13 Life Test, Component Qualification*
>
> *6,4.13.1 Purpose. This test demonstrates the reliability of the component and increases confidence that components which may have a wearout, drift, or fatigue-type failure mode have the capability to withstand the maximum duration or cycles of operation to which they are expected to operate during repeated ground testing and in flight without degradation of their function outside of allowable limits.*
>
> *6.4.13.2 Test Description. One or more components shall be set up to operate in conditions that simulate the flight conditions to which they would be subjected. These environmental conditions shall be selected for consistency with end use requirements and the significant life characteristics of the particular component. Typical environments are ambient, thermal, thermal vacuum, and various combinations of these. The test sample shall be selected at random from production units or shall be a qualification unit. The test shall be designed to demonstrate the ability of the component to withstand the maximum operating the and the maximum number of operational*

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

cycles predicted during its service life with a suitable margin. For components having a relatively low percentage duty cycle, it shall be acceptable to compress the operational duty cycle into a tolerable total test duration. For components which operate continuously in orbit, or at very high percentage duty cycles, accelerated test techniques may be employed if such an approach can be shown to be valid.

6.4.13.3 Test Levels and Duration

a. Pressure. Ambient pressure shall be used except for unsealed units where degradation due to a vacuum environment may be anticipated. In those cases, a pressure of 0.0133 pascals (0.0001 Torr) or less shall be used.

b. Environmental Levels. The maximum predicted environmental levels shall be used. For accelerated life tests, environmental levels may be selected that are more severe than flight levels. provided the higher stresses can be correlated with life at the predicted use stresses and do not introduce additional failure mechanisms.

c. Duration. The total operating time or number of operational cycles for a component life test shall be twice that predicted during the service life, including ground testing. in order to demonstrate an adequate margin.

d. Functional Duty Cycle. Complete functional tests shall be conducted before the test begins, after each 168 hours of operation and during the last 2 hours of the test. An abbreviated functional test shall be conducted periodically to ascertain that the component is functioning within specification limits.

6.4.13.4 Supplementary Requirements. For statistical type life tests, the duration is dependent upon the number of samples, confidence, and reliability to be demonstrated.

9.4.2 Rationale for Life Test Retirements. This test is intended to demonstrate a component's capability to perform for its mission duration. It is anticipated that it will be used when wearout, fatigue, or drift characteristics are unknown, and when premature failure will compromise mission goals. When these characteristics for an item have been determined to be adequate, a qualification life test is not required. This test does not demonstrate a quantitative reliability, such as

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

obtained from a Reliability Demonstration Test. The primary concern is hardware having moving parts, e.g. , moving mechanical assemblies or electromechanical assemblies. Other items such as batteries and pressure vessels may also be of concern.

9.4.3 Guidance for Use of Life Test Requirements. It is necessary to plan the life test specifically for each hardware item. The test should closely simulate actual usage conditions in terms of function, cycling, environment, and stress. Ideally, the test should continue to failure, and it should employ statistical samples which determine the mean wear out and variance, with the low end of the variance being in excess of the life requirements. However, this approach is usually not cost-effective or practical. Test unit availability is usually limited, with only one item often specified for qualification. Because two times the design life is specified as a duration for the test, the testing time can be excessive. Also, the life capability may be far in excess of the requirements, so testing to failure could take a long time.

Some classes of components rarely need life testing. Most of these components are electronic hardware. The life of electronic hardware which use solid state technology, proven packaging, and proven interconnection techniques has been adequate for normal space vehicle life requirements. If a component uses unproven interconnection or packaging techniques, then failure or fatigue due to incompatible coefficients of expansion should be considered. Such a component should be subjected to temperature cycling tests with many cycles of extreme range. Assurance of adequate life of electronic components can often be demonstrated at the part level. Parts should receive qualification tests, usually including life testing. Part life qualification tests, especially tests of electromechanical parts, should be reviewed for compatibility with mission needs.

9.5 COMPONENT BURN-IN TEST

9.5.1 Standard Criteria. Contents of Paragraph 7.3.9 of MIL-STD-1540B (requirements for burn-in test, component acceptance) are as follows:

7.3.9 Burn-in Test. Component Acceptance

7.3.9.1 Purpose. The purpose of the burn-in test shall be to detect material and workmanship defects which occur early in the component life.

7.3.9.2 Test Description. A modified thermal cycling test shall be used to accumulate the additional operational

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

hours required for the burn-in test of electronic and electrical components. While the component is operating (power on) and while perceptive parameters are being monitored, the temperature of the unit shall be reduced to the specified low temperature level. The unit shall be operated at the low temperature level for one hour or longer. The unit temperature shall then be increased to the specified high temperature level and operated for 1 hour or longer. The temperature shall then be reduced to ambient to complete one cycle of the burn-in test. The transitions between low and high temperatures shall be at an average sate greater than 1 deg C per minute.

For valves, thrusters, and other items where the number of cycles of operation rather than hours of operation is a better method to ensure detecting infant mortality failures, functional cycling shall be conducted at ambient temperature. For thrusters, a cycle is a hot firing which includes a start. steady state operation, and shutdown. For hot firings of thrusters utilizing hydrazine propellants, action shall be taken to assure that the flight valves are thoroughly cleaned of all traces of hydrazine propellant following the test firings. Devices that have extremely limited life cycles such as positive expulsion tanks are excluded from burn-in test requirements.

7.3.9.3 Test Levels and Duration

a. Pressure. Ambient pressure should normally be used.

b. Temperature. For cycling of electronic and electrical components. the extreme temperatures specified in 7.3.3.3.b shall be used.

c. Duration. The total operating the for electronic and electrical component burn-in shall be 300 hours including the operating time during thermal cycling per 7.3.3. The minimum number of temperature cycles shall be 18 including those conducted during the thermal cycling acceptance test. Additional test time beyond that required for thermal cycling shall be conducted at either maximum or minimum temperature. The last 100 hours of the component burn-in test shall be free of failures. For valves, thrusters, and other components where functional cyclic testing is a better burn-in method, a minimum of 100 cycles shall be conducted.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

d. *Functional Duty Cycle.* Functional tests shall be conducted at the start of this test to provide a baseline reference for determining if performance degradation occurs. The functional test shall be repeated after 150 hours of operation and during the last 2 hours of the thermal cycling test. Perceptive parameters for all circuits. Including all redundancy, shall be monitored to the maximum extent possible during the entire test sequence. On-off cycling of the electronics component shall be conducted during the test to simulate operational usage.

7.3.9.4 *Supplementary Requirements.* The reduction of system level failures by burn-in at the component level has a favorable impact on costs and schedules by stabilizing the failure rate at or near its minimum and ensuring the highest probability of mission success.

9.5.2 Rationale for Component Acceptance Burn-in Test Requirements. These tests are conducted at the component acceptance level as a screen for workmanship and material defects, or for some mechanical components, to wear-in moving surfaces. The objective is to eliminate infant mortality, "debug" the hardware, and enhance long-term reliability. Useful screens to enhance these objectives are temperature cycling, constant temperature soak, continuous power application, power cycling, vibration, and various combinations of these tests. Random vibration is usually a separate test, but temperature cycling, temperature soak, and power on-off cycling are all part of the typical burn-in test. Temperature cycling with power cycled on and off is usually considered the most effective screening test for electronic hardware. Ambient temperature power-on screening may be effective for wear-in of moving surfaces. The length of the operating cycle (duty cycle) can have an effect on reliability. Frequent on-off cycling in service might introduce failure modes which should be "debugged" during burn-in by simulating the expected usage conditions. The temperature soak might provide a screen for failure modes that could occur during in-service temperature cycling.

Most satellite programs perform within comparatively benign temperature conditions. If the temperature cycle range were based only on these benign conditions, little thermal stress would be produced. Therefore, the conditions established in MIL-STD-1540B, providing for an 85 deg C range, 300 hours duration (including other tests), and 18 cycles minimum, were designed to produce thermal stress which will screen out latent defects.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

9.5.3 Guidance for Component Acceptance Burn-in Test Requirements. The most important burn-in test is temperature cycling. The three key variables are the number of cycles, the rate of change in temperature, and the temperature range. Usually, the faster the rate of change, the larger the range, and the more cycles, the more effective the test.

Rates of change from -17.2 deg C per minute to 4.4 deg C per minute have been used, with the faster rates providing the best screening. The rates in MIL-STD-1540B were established to be consistent with the capabilities of equipment available to most contractors.

It is important that intermittent discrepancy conditions be discovered. Therefore, the test should be performed in a monitored power-on mode, including the temperature transition periods.

The hardware maturity, hardware design, and conditions of manufacture and quality control are variables which can affect the needed burn-in period. MIL-STD-1540B has standardized on 300 hours as the needed period, based on successful program practices. This 300 hours is the cumulative power-on testing during the entire component acceptance test. Due to the potential variables which can affect the burn-in period, the duration could be considered a tailoring parameter. If data are available which show (for a given manufacturer and design) that longer or shorter times are needed to reach the end of the infant mortality period, then tailoring should be considered. The 100 hours failure-free requirement should be maintained to provide a little confidence that the infant mortality period has been passed. Of course, for items that are intended to last for years without failure, the 100 hours of failure-free operation is more reassuring that statistically significant.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

SECTION 10

SUBASSEMBLY LEVEL QUALIFICATION AND ACCEPTANCE TESTS

10.1 STANDARD CRITERIA

Contents of Paragraphs 6.5 and 7.4 of MIL-STD-1540B (requirements for subassembly level qualification and acceptance tests) are as follows:

6.5 SUBASSEMBLY LEVEL QUALIFICATION TESTS

subassembly level qualification tests shall be conducted on those subassemblies that are subjected to environmental acceptance tests at the subassembly level. For other subassemblies, qualification tests are to be considered as optional unless specified otherwise in the contract. Functional or environmental qualification tests may be conducted at the subassembly level to detect material and workmanship defects, or to measure critical parameters, that cannot be accomplished satisfactorily at higher levels of assembly. When subassembly level qualification tests are planned, the subassemblies may be tested to similar requirements as components, or if more stringent requirements are used for acceptance test stress screening. then the more stringent levels shall be the basis for the qualification tests. In general, all parts shall be qualified to maximum and minimum environmental levels well in excess of the levels predicted for their specific application in the space vehicle.

7.4 SUBASSEMBLY LEVEL ACCEPTANCE TESTS

These tests are to be considered as optional unless specified otherwise in the contract. However, subassembly level acceptance tests are often cost-effective measures for reducing or avoiding failures in higher level tests and possibly in orbital operations. Acceptance test should be conducted at the subassembly level where this level provides a more perceptive test than would be possible at either the part or the component level. Functional or environmental acceptance tests are usually conducted at the subassembly level to detect material and workmanship defects, or to measure critical parameters, that cannot be accomplished satisfactorily at higher levels of assembly. When these acceptance tests are planned on subassemblies, they may be tested to similar requirements as components, or more stringent requirements for stress screening may be used.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

10.2 RATIONALE FOR REQUIREMENTS

The general rule is that it is almost always cheaper to find a problem at the lowest level of assembly possible. This means that subassembly testing should always be considered. The fact that MIL-STD-1540B may not specifically require subassembly testing does not in any way mean that it should not be done. Proper design requires that items at each level of assembly should have broader parameter tolerances and narrower environmental ranges than the subtier items that are used in its fabrication. In that way, manufacturing defects can be screened out at the lowest level of assembly possible, and items that pass subtier screening tests should not be expected to fail subsequent tests. Also, critical parameters that cannot be accurately measured at higher levels of assembly must be evaluated at lower levels of assembly. This usually means that some form of stress-screening tests are cost-effective at the subassembly level.

The extremely high cost of an on-orbit space vehicle failure means that all parts, materials, subassemblies, and components must be designed and fabricated to assure high reliability. Testing of the space vehicle itself and its components provides necessary screening checks, but they are insufficient to assure the reliability of the space vehicle. The other words, in-process screening tests, including stress screening, must be used at the-subassembly level, and at all subtier levels where appropriate, to assure a reliable space vehicle.

10.3 GUIDANCE FOR USE OF REQUIREMENTS

10.3.1 Applicability of Subassembly Testing to Mechanical, Electromechanical, and Electronic Subassemblies. Subassembly testing is almost always applicable to electronic equipment Such as printed circuit or wiring boards. Mechanical and electromechanical subassembly tests are generally performed on equipment containing moving parts if the parts can be practically tested when removed from the case of the assembly or component. For example, they are applicable to a solenoid coil or to actuator subassemblies for space vehicle shrouds, solar panels, and antennas. In a component level test the component case is the fixture for the subassembly, but in a subassembly test a special fixture is used to hold the subassembly. Difficulty in designing and using such test fixtures may result in a decision to eliminate environmental stress from the subassembly tests.

Electronic subassembly test fixtures are often simpler to design and use than mechanical subassembly fixtures. However, fixture design to simulate mounting within the component

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

relative to dynamic and thermal responses can also be complex. The subassembly tests are essentially environmental and functional tests of individual circuit boards. The testing concentrates on the electronic functioning of the circuit when it is exposed to environments such as vibration and temperature extremes. At the electronic component level, there is often difficulty in disassembly or failure isolation, and proper subassembly repair or replacement can be dificult due to test point availability and access problems. Subassembly level tests with or without stress screening can be used to alleviate these problems.

Parts screening usually is conducted using the maximum range of design or qualification conditions in the part specifications. Assuming proper applications of the parts, those conditions would always be more severe than the conditions specified for subassembly or component screening. Since the subassembly or component tests do not duplicate the stringent conditions of part level testing, they should never be viewed as a substitute for part level screening.

10.3.2 Test Procedures for Subassembly Tests. The following are major tests performed as common industry practices on space system subassemblies, as applicable to the individual unit under test:

- Electrical tests--continuity and short test, dielectric withstanding voltage, and insulation resistance
- Functional test
- Burn-in and wear in tests
- Thermal cycling test
- Random vibration test
- Particle screening test
- Over-stress screening

The nature of some subassemblies imposes restrictions on some of these stress-screening subassembly tests. For instance, temperature limitations may be imposed if certain oscillators are present on a circuit board, because they will not withstand more than a limited temperature range. Some boards may have inherent limitations for exposure to vibration, such as with board6 of a “foamed” component which are to be tested before foaming. In that case, the vibration test spectrum must be

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

tailored to avoid damaging the mounted but unfoamed parts. Another example is electro-optical subassemblies, which can be damaged by more than a limited rate of temperature change during thermal cycling tests. All tests must be designed with the applicable restrictive factors in mind.

Over-stress screening is a special testing technique where the test levels used exceed the design levels for the item. Over-stress screening is usually invented after an item has been fabricated to uncover particular types of latent defects or incipient failures that were just discovered in the items, and that cannot be uncovered by other means. For example, nicks may be discovered in the insulation of some of the wiring used in a wiring harness designed for a 28-volt circuit. It is known that when insulated wiring is exposed to about 1500 volts, an examination of leakage current will indicate the presence of nicks and other defects in the insulation. By exposing the 28-volt wiring to 1500 volts as required to reveal the defects in the insulation, an over-stress screening of the wiring can be used to identify the defective wire. Similarly, higher than the design levels of shock, vibration, temperature, pressure, radiation, or combinations of these or other parameters may be used to uncover certain types of defects in a particular device or subassembly. Obviously, extreme care must be used in selecting any form of over-stress screening in order to avoid test conditions that may damage the item being tested. For this reason, an over-stress screening test that is appropriate for one particular type of item may be inappropriate for another type of item, even though the type of defect being screened is the same.

To illustrate the kind of problems that need to be avoided, suppose the 1500 volts used for indicating the presence of nicks and other defects in the wiring harness over-stress screening damage a connector or other part attached to the wiring harness. Clearly, that would not be a good over-stress screening test. In another case, a vibration over-stress screening test intended to identify any loose electrical connections could produce a condition which causes small cracks (a potential or latent failure) in one of the-items. The cracks may cause an actual failure in a subsequent component or system test. That in turn might result in long delay and expense to correct the failure. Or, an undiscovered crack might cause a failure in orbit, resulting in partial loss of the mission. This illustrates that there is some financial and technical risk inherent in subassembly over-stress screening. However, careful analysis and prudent choice of the over-stress can greatly reduce the risk. Of course, the over-stress damage potential could be avoided by always including all test environments in the required design environment and thereby avoid after-the-fact over-stress screening decisions.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

Technical necessity and cost-effectiveness of subassembly testing are both dependent on the specific program being considered. Design factors such as board design complexity, number of subassemblies, parts reliability, parts testing program, toughness of subassembly environments, and amount of subassembly testing time are dependent *on* the specific program under consideration. Cost factors such as the cost of test equipment and the testing costs are also dependent on the specific item under consideration. If possible, a risk versus cost-effectiveness analysis for subassembly testing should be performed for each item.

10.3.3 Significant Data from Subassembly Thermal Cycling Stress Screening. Data which indicate the test-effectiveness and cost-effectiveness of subassembly thermal cycling stress screening have been reported by two large corporation.

Corporation “A” Investigation

Corporation “A” performed an extensive experimental investigation of effectiveness of thermal cycling stress screening of circuit boards.

There were 1,248 missile system circuit boards stress-screened in a thermal cycling environment between -40 deg C and +75 deg C for up to 48 cycles, while prior normal circuit board ambient temperature acceptance tests were retained. The rate of change in temperature was 10 deg C and 20 deg C per minute.

The failure histories of components containing these stress-screened boards and identical components with unstress-screened boards were monitored from ambient temperature subassembly tests and environmental component tests through customer ambient temperature component tests. The components containing stress-screened circuit boards proved to have lower failure rates, as their failure rate (in the customer component tests) was only one-fourth of the failure rate of components with unstress-screened boards.

It was concluded that circuit board thermal cycling stress screening is clearly effective for reducing component failure rates. The available data were not sufficient to show the effect of circuit board stress screening on system level failure rates, but the potential exists for reducing system level failures and improving system level reliability.

Corporation “B” Operations

Corporation “B” started thermal cycling stress screening of circuit boards while still retaining ambient temperature subassembly testing. There were 55 repetitions of a 2-hour

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

cycle between -55 deg C and +80 deg C. For 68 computer memory circuit boards, the failure rate in the subassembly level tests performed before and after stress screening was reduced from 27.4 to 11.8 percent, a reduction of 57 percent in failure rate. The failure rate at the component level was reduced from 20.8 to 13.2 percent, a reduction of 37 percent in failure rate. This is illustrated in Table XVI. A cost-effectiveness computation showed a net yearly savings of 12,573 person-hours for electronics test operations due to the institution of stress screening.

TABLE XVI. Reduction in Failure Rate Due to Stress Screening.

Tests on 68 Circuit Boards	No Stress Screening	With Stress Screening	Percent Reduction in Failure Rate
Subassembly tests	Failure rate = 27.4%	Failure rate = 11.8%	57%
Component tests	Failure rate = 20.8%	Failure rate = 13.2%	37%

10.3.4 <u>Significant Data from Subassembly Testing for A Large Space Vehicle Program</u>. Consultations with engineers closely involved in testing of a large military satellite program revealed the data shown in Table XVII concerning these space vehicle tests. These data are from normal testing operations (not stress screening).

The number of failures was largest at the subassembly level. Failures at the higher levels were dramatically less than subassembly test failures. These data tend to indicate the test-effectiveness of subassembly tests. The data do not prove that subassembly testing improves space vehicle reliability; but the potential for improved reliability exists, since it is not clear that testing only at the higher levels of assembly would have revealed all the failures found by subassembly tests. Of course, it would have been more costly to correct the failures had they been discovered during testing at the higher levels of assembly.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

TABLE XVII. Data from Subassembly Testing for a Large Space Vehicle Program.

Test	Total Hardware	Number of Failures
Circuit boards and slice tests for 4 satellites (subassembly tests)	3700 boards, 1256 slices	192
Component acceptance tests for 4 satellites	388 components	69
Satellite integration operations (subsystem tests)	3 satellites	18
Satellite acceptance tests (space vehicle tests)	3 satellites	12

10.3.5 Guidance Summary. Subassembly testing decisions are component and program dependent, and should be made by program management with guidance from design and test engineering. Subassembly tests may be optional; however, if critical parameters cannot be adequately verified by tests at higher or lower assembly levels, they should be verified by subassembly tests. Testing and replacement of defective units at the subassembly level usually involve relatively fewer engineers, technicians, and pieces of test equipment than at the component or space vehicle level. A test failure at the component, subsystem, or system level can involve many engineers and technicians and large-scale test setups, and can cause extensive delay to an entire project. The expense of an extensive subassembly test program can often be justified based only on avoiding the potentially higher expense of what otherwise could be failures during component, subsystem, or system tests. However, a risk versus cost-effectiveness analysis should be performed to evaluate the effectiveness of subassembly tests for each specific test program. Unfortunately, the technical necessity of specific subassembly tests, and the related historical cost data, are generally unavailable. Also, insufficient data exist at this time to prove whether on-orbit reliability of spacecraft have been increased by performing subassembly tests. However, it is clear that there is both a potential for cost savings and a potential for improved reliability.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

It is generally cost-effective to require space system subassemblies to be subjected to electrical continuity, functional, burn-in, thermal cycling, and random vibration tests. The technical parameters of these subassembly tests should be specified individually for each item by the system program office with guidance from design, test, and reliability engineers. The testing should be performed either to the same stress-screening environmental limits as the higher tier component tests or to tougher stress-screening limits applicable to each subassembly.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 11

FLIGHT USE OF QUALIFICATION EQUIPMENT

11.1 STANDARD CRITERIA

Contents of Paragraphs 8.0 through 8.4 of MIL-STD-1540B (flight use of qualification equipment) are as follows:

8.0 FLIGHT USE OF QUALIFICATION EQUIPMENT

Qualification tests are conducted to demonstrate that the design, manufacturing. and assembly have resulted in hardware conforming to specification requirements, The qualification tests required by this document incorporate the environmental design margin into the test levels to assure that flight units will meet the operational requirements for their service life. The vibration tests. acoustic tests, and thermal tests produce cyclic stresses that can encroach on the fatigue margins of interconnect wiring, solder joints, structural members, and similar items In the qualification test units. If equipment that has been subjected to qualification testing is planned for subsequent flight use, It is possible that the remaining fatigue margins are so low as to present a high risk of failure during flight. This is primarily due to the use of high test levels and long test durations during the baseline qualification tests. Therefore, the actual vehicle used for the 6.2 vehicle qualification tests or the components used for the 6.4 component qualification tests may not be suitable for subsequent flight.

Nevertheless. Initial program costs and schedule constraints may force the consideration of ways to make units used for qualification testing acceptable for flight. It should be recognized that the use of qualification items for flight always presents a higher risk than the use of standard acceptance-tested items for flight. This risk may be reduced by various strategies such as reducing qualification test levels and durations to reduce the encroachment on fatigue and wearout margins. The strategy used should be based upon specific program considerations. One method has been to replace all components on the qualification vehicle with "new" components that have passed component acceptance tests (see 8.3). Another way was to lower the space vehicle qualification test levels and test duration to avoid excessive encroachment on margins (see 8.2). on some programs, one or more qualification components have been

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

used as flight components (see 8.1). In such cases where program considerations are overriding. the contract may direct, or the contracting officer may approve, the use of qualification units for flight. Some of the strategies that have been used are presented in the following examples.

8.1 USE OF THE QUALIFICATION COMPONENTS FOR FLIGHT

When the qualification components are planned for flight use, the component qualification test program shall be modified from that specified in Section 6 to reduce cyclic stress levels. In addition, the component qualification testing shall be conducted on flight spares so that flight use is delayed or possibly never required. The flight space vehicle in which these qualification components are installed shall be acceptance-tested in accordance with the requirements of 7.1. This space vehicle qualification would be based on the requirements of 6.2.

8.1.1 Component Qualification Tests. When the component qualification tests are conducted on a component intended for subsequent flight. the component acceptance tests required by this standard are waived, except for the burn-in acceptance test of 7.3.9, and only the qualification test baseline specified in 6.4 is required with the following exceptions:

a. For the component thermal vacuum test (6.4.2). the temperature extremes shall be 5 deg C beyond the minimum and maximum predicted temperatures.

b. For the component thermal cycling test (6.4.3), the temperature cycles shall be conducted at 5 deg C beyond the acceptance temperature extremes (7.3.3.3 b).

c. For the component vibration qualification test (6.4.5). the test level shall be 3 dB greater than the maximum predicted level but not less than 9 grms.

d. For the component acoustic qualification test (6.4.6), the test level shall be 3 dB greater than the maximum predicted level but not less than 141 dB overall.

e. For the component pyrotechnic shock test (6.4.7), the shock spectrum shall be 3 dB greater than the maximum predicted level.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

f. For the component pressure test (6.4.10), only proof pressure tests per 6.4.10.3 a and b shall be conducted.

8.1.2 Component Certification for Flight. Upon completion of the modified qualification test program. the component test history shall be reviewed for excessive test the and potential fatigue type failures to determine if the unit is acceptable for flight or If refurbishment is required. Mission and safety critical qualification components should not be used for flight in systems where a redundant component is not provided.

8.2 USE OF THE FLIGHT VEHICLE FOR SPACE VEHICLE LEVEL QUALIFICATION

When the flight vehicle is also used for the vehicle level qualification tests, the space vehicle qualification test levels and durations shall be reduced as defined in 8.2.1. The components installed in this flight vehicle shall be acceptance-tested in accordance with the requirements of 7.3. The component qualifications would be based on the requirements of 6.4.

8.2.1 Space Vehicle Qualification Tests. If the space vehicle qualification tests are to be combined with the flight vehicle acceptance tests, the space vehicle level acceptance tests required by this standard are waived, and only the qualification test baseline in 6.2 is required with the following exceptions:

a. For the space vehicle acoustic qualification test (6.2.3), the test level shall be 3 dB greater than the maximum predicted level but not less than 141 dB overall. The duration of the test shall be the same as for the space vehicle acoustic acceptance test (7.1.3.3).

b. For the space vehicle vibration qualification test (6.2.4), the test levels shall produce vibration responses in the equipment which are 3 dB greater than the maximum predicted level. The duration of the test shall be the same as for the space vehicle vibration acceptance test (7.1.4.3).

c. For the space vehicle thermal vacuum qualification test (6.2.7), the number of hot-cold cycles shall be four and the temperature extremes shall be 5 deg C beyond the minimum and maximum predicted temperatures.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

d. *If the optional space vehicle thermal cycling test (6.2.9) is adopted as baseline, the minimum space vehicle temperature range shall be 60 deg C. The test should include 15 percent more thermal cycles than the space vehicle thermal cycling acceptance test (7.1.8.3).*

8.2.2 Space Vehicle Certification for Flight. Upon completion of the modified space vehicle qualification test program. the vehicle test history shall be reviewed for excessive test the and potential fatigue-type failure to determine if the vehicle is acceptable for flight or if refurbishment is required. If significant modifications are incorporated or numerous components are refurbished or replaced with new components subsequent to qualification testing, the space vehicle acceptance baseline specified in 7.1 shall be required prior to launch certification.

8.3 USE OF THE QUALIFICATION VEHICLE FOR FLIGHT

When the space vehicle used for vehicle level qualification testing of 6.2 is planned for subsequent flight use. all components shall be replaced with "new" components that have passed the component acceptance tests. The space vehicle is certified for flight when it satisfactorily completes the vehicle level acceptance tests of Section 7.

8.4 OTHER STRATEGIES

Various combinations of strategy may be considered. depending on specific program considerations and the degree of risk deemed acceptable. For example, method 8.1.1 may be combined with a vehicle qualified at reduced levels per 8.2.1 or with the qualification vehicle per 8.3. In such cases, the provisions of both methods apply, and the resultant risk would be increased appropriately.

11.2 RATIONALE FOR FLIGHT USE OF QUALIFICATION EQUIPMENT REQUIREMENTS

Past test practices (per MIL-STD-1540A) implicitly prohibited flight use of components or space vehicles subjected to qualification test levels. MIL-STD-1540B recognizes that program considerations may dictate the flight use of qualification test articles. The increased cost and complexity of space vehicles in combination with mounting pressure for cost reductions are the usual driving forces leading to this growing practice of flying qualification test articles. Both components and space vehicles which have undergone environmental test at qualification levels have been committed to an operational

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

role. The conventional practice of retiring qualification specimens from further service is still valid from a technical point of view, since the reliable life remaining in an article after the rigors of qualification testing cannot be established with certainty. It is safer, in a technical sense, to test flight components and vehicles to only acceptance test levels of environment. These acceptance test levels represent the maximum expected in service, but lie below qualification test levels by the design margins. Driven by cost considerations, however, some space vehicle programs have decided to fly their qualification test articles. The operational performance of such equipment in Air Force and NASA space vehicles has generally proven satisfactory. In most cases, however, there were modifications to the usual qualification test program to reduce the risk of flying “worn out” test articles. It is likely that the practice of flying qualification test articles will continue and expand. Guidance is therefore needed to formulate appropriate qualification test programs for equipment which will subsequently be used in service.

11.3 <u>GUIDANCE FOR FLIGHT USE OF QUALIFICATION EQUIPMENT</u>

MIL-STD-1540B recognizes that there are no standard criteria for flight use of qualification equipment. It is noted that the use of qualification equipment for flight presents a higher risk than the use of standard acceptance-tested items for flight. For items tested to their nominal design level (full qualification), this higher risk is primarily due to the uncertainties regarding fatigue margins and the uncertainties regarding the remaining life in the test articles following the qualification test. For items qualified to less than their full design levels (i.e., reduced environmental margins for qualification) , the higher risk in using the test articles is primarily due to uncertainties regarding differences between the test environments and the actual flight environments as well as uncertainties regarding the fatigue and wearout limits of the test units. If a reduced level qualification test is used, then production variabilities may also increase the risk in using other units that may only be acceptance-tested, i.e., tested to the maximum predicted flight environmental range.

In order to discuss the possible flight use of qualification test articles, it is necessary to define the various possible categories of the flight items. In this discussion, it is assumed that the design environments for the items are in accordance with the definitions in MIL-STD-1540B, Sections 3.8, 3.9, and 3.10 (see Section 5.1 in this handbook). Full qualification testing means that the items are tested to those design levels, and normal acceptance testing would typically be to the maximum predicted environmental range during flight. In this context, there are four possible categories of flight items as defined in Table XVIII.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XVIII. Possible Categories of Flight Items.

Category	Description
Category I	Items that have passed all normal acceptance tests and whose qualification is based upon the full qualification testing of another production unit
Category II	Items that have passed all normal acceptance tests and whose qualification is based upon reduced qualification testing of another production unit
Category 111	A test item that has passed reduced level qualification testing and is then used as a flight unit
Category IV	A test item that has passed the full qualification testing and is then used as a flight unit

Although the components installed in a flight space vehicle, particularly a production vehicle, are normally Category I items, it is clear that any of the components could instead be Category II, Category 111, or Category IV items. Similarly, the flight space vehicle into which the components are installed would normally be a Category I item; however, it is also clear that it could instead become a Category II, Category III, or Category IV space vehicle, depending upon both the vehicle level testing that is conducted on that flight vehicle and the qualification testing conducted on a separate space vehicle.

The baseline test program outlined in MIL-STD-1540B assumes that Category I components are used in a Category I flight vehicle. That may mean higher testing costs than other alternatives, since it requires a full qualification test program on another set of components and a full qualification test program on a separate space vehicle. This particular subsection addresses ways to reduce the program costs without increasing the risks beyond what is acceptable. Therefore, this subsection will not address the baseline category, i.e., Category I components in a Category I space vehicle.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

Example A illustrates an extreme case. One could postulate a program where only a single set of components is built and installed in a single space vehicle. One could further postulate that the components and the space vehicle are only to be tested over the maximum predicted flight environmental range (i.e., at normal acceptance test levels). The rationale in this example would be that if there is only one set of hardware produced, and if it is tested to the maximum predicted flight environments, that should be enough to prove it is flightworthy. In this example, the components are Category III and the space vehicle would be Category III. Certainly, the one set of equipment used in this case is less costly than the two sets required for Category I; and certainly, the one set of tests at acceptance test levels is less expensive and will have fewer failures than the two sets of tests required for Category I. The problem with this case is that the reductions in test levels have completely eliminated any provisions for the environmental design margin. The environmental design margin assures against environments that may not adequately simulate flight environments, allows retest without the risk of fatigue failure, and provides for test equipment tolerances. For example, the design margin accommodates the fact that the test environments are applied one at a time, while in flight they are combined. In addition, acceptance test levels are typically the maximum predicted flight levels that may only be the 95th percentiles. This means that more extreme flight levels might be expected on some components at least some of the time, and that fact increases the risk of failure during flight. It would therefore seem unlikely that a space vehicle program would be willing to accept the increased risks implied by Example A in the hope that the total equipment and testing costs would be reduced.

Example B illustrates another case. It is assumed that the qualification test articles that were tested over their full design environmental range (full qualification) were installed on a flight space vehicle that was then given a full qualification test. In this case, the components would be, Category IV and the space vehicle would also be Category IV. Here the reduction in hardware costs and testing costs is essentially the same as in Example A; however, since the test levels are higher in Example B, there are added risks that the items may not pass the tests. If the tests are satisfactorily completed, the risk of having a flight environment exceed the component or space vehicle test environments is greatly reduced, and the probability of mission success increases. On the other hand, the potential for a component fatigue failure or wearout has been increased due to the more severe testing in this example. It would seem unlikely that a space vehicle program would accept the increased risks implied by this example in the hope that the total equipment and testing costs would be reduced.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

The examples given in Section 8 of MIL-STD-1540B for the flight use of qualification test articles are intended to present a more reasonable balance between increasing risks and reducing costs than either Example A or Example B above. It is important to recognize that the reduced qualification tests discussed in Paragraphs 8.1.1 and 8.2.1 of MIL-STD-1540B are suggestions and are not to be construed as recommendations or as specific requirements. In order to provide visibility regarding these reduced qualification test examples, Tables XIX and XX compare the reduced qualification tests with standard qualification test levels and durations.

TABLE XIX. Flight Use of Qualification Component; Modified Test Program--Reduced Level Qualification Test.

MIL-STD-1540B Component Tests	Components for Standard Qualification Test (Nonflight Use)	Qualification Components for Flight Use: Reduced Level Qualification Test
Thermal Vacuum	Test level has 10°C margin beyond max. & min. predicted. Minimum of 3 cycles.	Test level has 5°C margin beyond max. & min. predicted. Minimum of 3 cycles.
Thermal Cycling	Test level has 10°C margin beyond max. & min. predicted. Minimum of 24 cycles.	Test level has 5°C margin beyond max. & min. predicted. Minimum of 24 cycles.
Random Vibration	Test level has 6 dB margin beyond max. predicted. Min. of 12 grins overall. Min. of 3 min per axis.	Test level has 3 dB margin beyond max. predicted. Min. of 9 grins overall. Min. of 3 min per axis.
Acoustic	Test level has 6 dB margin beyond max. predicted. Min. of 144 dB overall. Min. of 3 min.	Test level has 3 dB margin beyond max. predicted. Min. of 141 dB overall. Min. of 3 min.
Pyrotechnic Shock	Test level has 6 dB margin beyond max. predicted. 3 shocks per direction per axis; 18 shocks total.	Test level has 3 dB margin beyond max. predicted. 3 shocks per direction per axis; 18 shocks total,

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XX. Flight Use of Qualification Space Vehicles; Modified Test Program--Reduced Level Qualification Test

MIL-STD-1540B Space Vehicles Tests	Space Vehicle for Standard Qualification Test (nonflight use)	Qualification Space Vehicle for Flight Use: Reduced Level Qualification Test
Acoustic	Test level has 6 dB margin beyond max. predicted. Min. of 144 dB; min. of 3 min.	Test level has 3 dB margin beyond max. predicted. Min. of 141 dB; min. of 1 min.
Random Vibration	Test level has 6 dB margin beyond max. predicted. Minimum of 3 min per axis.	Test level has 3 dB margin beyond max. predicted. Minimum of 1 min per axis.
Thermal Vacuum	Test level has 10°C margin beyond max. & min. predicted. Min. of 8 cycles.	Test level has 5°C margin beyond max. & min. predicted. Min. of 4 cycles.

Table XIX shows examples of reduced qualification test levels for qualification components intended for flight and a comparison of these reduced test levels with standard qualification test levels. Similar examples are shown in Table XX for reduced qualification level space vehicle tests compared with standard tests.

Note that the reduced qualification level component tests are at one-half the design margins for full qualification durations. The reduced qualification level space vehicle tests are at one-half the design margins, and the test durations are the same as the nominal vehicle acceptance test durations. This strategy still provides a funnel effect to maximize test rigor at the lower level of assembly. Other variations in strategy are possible and should be considered. depending on specific program considerations.

11.3.1 <u>Guidance for Qualification Test Margins on Qualification Equipment Used for Flight</u>. Qualification test requirements are established at levels which exceed environments and stresses expected in operational service and in normal

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

acceptance testing. This is accomplished by application of environmental design margins to the maximum anticipated extremes of the service environment. In general, the maximum expected service environmental condition is augmented in terms of amplitude and duration for dynamic and load-inducing environments and in terms of temperature range for thermal environments. Numerous considerations are involved in determining appropriate margins. Only those which are affected by a decision to fly qualification articles are treated herein.

11.3.1.1 Amplitude versus Time. There is an uncertain risk in using equipment with a relatively small remaining fatigue life for flight. However, the risk of fatigue failure can be reduced significantly by reducing qualification test margins. In regard to the dynamic environments, e.g., vibration and acoustics, the fatigue life is more strongly affected by amplitude than by exposure duration. Assuming that stress levels in a dynamic qualification test are above the endurance limit (where fatigue damage is accumulated), a reduction in test amplitude by a factor of two is expected to extend the fatigue life by more than an order of magnitude. This expectation is based upon study of fatigue characteristic of typical materials used in space Vehicles. Test duration provides an important contribution to test effectiveness by allowing sufficient time to monitor operational performance over an extended time. If concern for possible fatigue damage during qualification tests motivates a reduction in test requirements, an amplitude decrease is therefore better than a test duration decrease.

11.3.1.2 Fatigue Damage Concerns. When the decision is made to fly qualification equipment, concern is often expressed regarding the probable fatigue damage suffered by these articles during qualification tests. In response to this concern, concessions are sometimes granted in terms of decreased test levels and exposure duration in order to reduce the possibility of fatigue failure during flight. Such reduction should be cautious, since the rigorousness of the qualification test is thereby diminished. Although it is anticipated that the space vehicle and components will not encounter flight conditions exceeding the maximum predicted environments, some uncertainty is inherent in the predictions. The design test margins were established to compensate for these and other uncertainties. A reduction of the design test margins means a reduction in the qualification test levels which increases the risk that unexpected events may exceed the equipment design capability. It is sometimes argued that rather than increasing the risk of fatigue failure in flight, the qualification test specimens are less risky, because they benefit from substantial test margins relative to the comparatively benign flight environment. Essentially it is argued that the higher the test margins, the

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

lower the risk of flight damage. Also, later flight articles tested at acceptance levels benefit from high qualification test margins, because more acceptance test repetitions can occur before the capability demonstrated by qualification is exceeded. However, regardless of the margin applied in setting qualification test levels, the life remaining in the qualification test article is still undefined due to uncertainties in estimating fatigue life.

11.3.1.3 Thermal Extremes. Thermal testing is generally believed to be a less serious fatigue damage threat than dynamic testing. The number of test temperature cycles and the test exposure time are usually considerably less than the flight environment. The number of thermal stress cycles is not sufficient to provide the large number of stress reversals required for fatigue failure. Thermal margins, which are fixed increases in the maximum expected temperature ranges, do not appear to influence fatigue life as strongly as the ratio-type margins of dynamic and load environment. The hazard in testing flight equipment to qualification temperature extremes lies in the risk of exceeding the temperature design limits of the hardware, but this risk is generally small. In most cases, the qualification temperatures may be used for flight hardware with a relatively low risk.

11.3.1.4 Vehicle Design. Another ingredient involved in qualification test margins is the space vehicle design philosophy. Most space vehicles incorporate a high degree of redundancy in system design by using redundant strings of components . This redundancy is enhanced by cross-strapping of individual components from one string to another.

With redundant component. it is assumed that only one of the components has been qualification-tested and that the other has been acceptance-tested. For this case, if qualification-tested equipment is used in one string and standard acceptance-tested equipment is in the other string, a decision must be made as to which string should be active initially, and which string should remain dormant until required. Consideration should be given to the somewhat higher risk of fatigue failure for the qualification-tested equipment in one string, and the decision should be made accordingly.

For mission-critical nonredundant application, it appears safest to fly standard acceptance-tested components that were qualified using other nonflight qualification units.

11.3.1.5 Number of Flight Articles. Full qualification testing is performed to show that the generic equipment can operate at environmental levels that are more extreme than

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

predicted. This provides confidence that variabilities among flight articles and uncertainties in the testing and in the prediction of environments are accounted for. At the same time, a higher qualification test level adds some risk of fatigue or other damage to the test article.

If the number of flight articles is large, no reduction of qualification test levels may be appropriate. Although higher qualification test levels may add some risk of fatigue damage to the qualification test article, they add much confidence in the adequacy of the generic design. Successful completion of qualification tests with larger margins also allow an increased number of repetitions of acceptance tests of subsequent copies without fear of fatigue damage. The articles following the qualification article may be acceptance-tested only.

If just one or two flight articles are involved, the qualification equipment used for flight represents a large percentage of the total build. In order to minimize the risk of fatigue damage to this equipment, a reduction in qualification tests may be appropriate as shown in Tables XIX and XX of this document. If relatively small variability exists between production articles, the second and other articles following the first article that was given a reduced qualification test may be acceptance-tested only. If considerable variability exists, those generic follow-on articles are recommended to be qualification-tested with the same qualification margins as the first flight article.

11.3.1.6 Consistency with Level of Assembly. The advisability of maintaining consistent qualification margins for testing at different levels of assembly is sometimes questioned. However, the relationship of test rigor and level of assembly is purposeful. It is intended to aid in early identification of environmental susceptibility. This “funnel effect” is best preserved by maintaining consistent margins in both component and space vehicle testing. If test requirements are consistent and properly established, the system level test will approach, but not exceed, the stress level of component test margins.

11.3.2 Guidance for Application Strategy on Flight Use of Qualification Equipment. Beyond the consideration of test margins, a number of flight- and test-related questions may arise. Should components which have been tested to qualification levels be used to assemble the qualification test vehicle, or should the vehicle be built from components tested at acceptance levels? If the latter, how should qualification-tested components be utilized? Should they serve-as flight spares? Should qualification test components be refurbished and

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

retested at acceptance levels prior to flight? What constitutes refurbishment? In what order should the qualification test vehicle be flown relative to other flight vehicles? Some views on these questions are in the following text. The treatment is not intended to be prescriptive but to provide points for consideration.

11.3.2.1 Use of Qualification Components in Qualification Vehicle. In general, it appears reasonable to use the components subjected to qualification testing to build the qualification test vehicle. Testing at lower levels of assembly is normally more rigorous environmentally than at higher levels of assembly, to aid in early problem identification.

Therefore, during the vehicle qualification test, the qualification components will accumulate fatigue damage at a relatively lower rate than they did during component qualification. The idea of containment of the fatigue risk within a single flight vehicle is also worthy of consideration. In any event, space vehicles with adequate redundancy can provide backup for the qualification components with redundant components which have not been exposed to component qualification test levels. The risk of Catastrophic failure causing loss of the mission is thereby diminished. This benefit is lost, however, if internally redundant component designs are used.

11.3.2.2 Use of Qualification Components as Spares. Some believe that the qualification test components are best utilized as spares to be used in the factory as needed and flown only if needed, and that the qualification test vehicle should be fabricated with acceptance-tested component. This strategy appears prudent if the program makes no other provisions for spares. In that case, it should be understood that these qualification test components may be flown as needed without reservation. This does not eliminate concerns with fatigue damage . Spare units are subject to rework due to engineering design changes or part replacement while on the Shelf. Such modification usually requires additional environmental acceptance testing to validate the rework. In this manner, qualification components used as spares may accumulate much more test time than the normal acceptance test units. This should be considered in estimating the risk.

11.3.2.3 Refurbishment. The question of refurbishment of qualification equipment is ambiguous. It infers that equipment exposed to qualification test levels requires inspection or perhaps upgrade or repair. It may be prudent to replace delicate mechanisms (such as precision bearings) which could suffer life-limiting damage that is difficult to

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

immediately detect. However, the advisability of partial disassembly, inspection, and rework of suspect areas in qualification components on a routine basis should be questioned. Such rework usually imposes a requirement that environmental acceptance tests be conducted after reassembly.

Discovery of incipient fatigue failures (which would require rework) is highly uncertain. In fact, component construction with foam sealants or potting compounds, and use of many extremely small parts, makes visual inspection difficult. Aside from rework necessitated by mandatory engineering change, replacement of suspect parts, and mechanisms whose life is diminished by test exposure, refurbishment does not appear fruitful and tends to add further test exposure to the equipment.

11.3.2.4 Flight Strategy The decision concerning when to fly the qualification test vehicle must be based on limited experience. In the normal situation, where the first vehicle built is qualification-tested, it seems logical to fly it first. Past successes indicate that the risk of latent fatigue or over-stress failure is low. Furthermore, the longer the qualification test vehicle remains in the manufacturing facility, the more likely it is to be reworked and retested. Due to rapidly advancing technology, it is also subject to obsolescence due to redesigned components. As long as the qualification test components have been built to full flight-quality standards, it appears logical to fly the vehicle as soon as possible. This often would make it the first flight article. Production or launch schedules may dictate that the second production unit be completed prior to completion of all qualification tests on the first unit. This would usually result in designating the qualification unit as a launch spare and then as the second flight article.

11.3.3 Summary of Guidance for Qualification Test Margins and for Flight Use of Qualification Equipment. The following summary is a guide for structuring an environmental qualification test program for test articles to be used as flight hardware:

a. For dynamic tests, reducing the qualification margin in terms of amplitude rather than time causes less fatigue damage and permits more thorough performance testing.

b. Regardless of test margins, the fatigue damage sustained by qualification specimens is undefined.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

c. From a possible fatigue damage viewpoint, there is less necessity for reducing test margins for thermal qualification tests than for dynamic tests.

d. Redundant system design reduces the mission risk associated with flying qualification-tested equipment.

e. A program planning to build a significant number of the same space vehicles may benefit by maintaining larger design (qualification test) margins.

f. Consistent test margins at different levels of assembly would appear appropriate in most cases.

g. Building the qualification test vehicle with qualification-tested components represents a reasonable approach.

h. For some components, general refurbishment of the qualification test article may not be desirable and may even be harmful.

i. If the qualification test vehicle is to be flown, there appears to be little value in delaying the flight until later in the program.

j. The use of qualification-tested components as initial factory checkout units, or as flight spares to be used only if needed, has been a successful policy on many space programs.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 12

PRELAUNCH VALIDATION TESTS

12.1 PRELAUNCH VALIDATION TEST DESCRIPTION

12.1.1 Standard Criteria. Contents of Paragraphs 9.4, 9.4.1, 9.4.3, and 9.4.4 of MIL-STD-1540B (description of prelaunch validation tests) are as follows:

> *9.4 PRELAUNCH VALIDATION TEST DESCRIPTIONS. The prelaunch validation tests shall exercise and demonstrate satisfactory operation of the space vehicle through all of Its mission phases. to the maximum extent practical. Test data shall be compared to corresponding data obtained in factory tests to identify trends which indicate performance degradation within specification limits. Each test procedure used shall include test limits and success criteria sufficient to permit a rapid determination as to whether or not processing and integration of the vehicle should continue. However, the final acceptance or rejection decision, in most tests, depends upon the results of post-test data analysis.*
>
> *9.4.1 Functional Test. Electrical functional tests shall be conducted that duplicate. as nearly as possible. the factory functional tests of 7.1.1.2. Mechanical tests for leakage, valve and mechanism operability, and fairing clearance shall be conducted.*
>
> *9.4.1.1 Simulators. Simulation devices shall be carefully controlled and shall be permitted only when there is no feasible alternative for conducting the test. When it is necessary to employ simulators in the conduct of prelaunch validation tests of the space vehicle, the interfaces disconnected in the subsequent replacement of the simulators with flight hardware shall be revalidated. Simulators shall be used for the validation of ground support equipment prior to connecting it to flight hardware.*
>
> *9.4.2.2 Explosive Circuits. When explosive circuits are involved, approved simulation devices shall be used where appropriate. Before connection of pyrotechnic devices to their respective circuits, line continuity checks shall be made for the presence of the "Fire" signal at the squib connection when commanded. A line continuity stray voltage check shall be made immediately prior to the connection of any pyrotechnic device, and this check shall be repeated whenever that comection is opened and prior to reconnection.*

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

9.4.3 Integrated System Tests. Total launch system readiness shall be demonstrated through an integrated, fully assembled launch systems test prior to flight. This test shall include an evaluation of radio frequency (rf) interference between system elements, electrical power interfaces, and the command and control subsystems. On a new space vehicle design or a significant design change to the telemetry. tracking, or receiving subsystem of an existing space vehicle, a test shall be run on the first vehicle to ensure nominal operation and that pyrotechnics (simulators) do not fire when the vehicle is subjected to the worst case range electromagnetic Interference environment.

9.4.4 Compatibility Test, On-orbit System

9.4.4.1 Purpose. This test validates the compatibility of the space vehicle and the on-orbit command and control network. For the purpose of establishing this testing baseline, it is assumed that the on-orbit command and control network is (or operationally interfaces with) the Air Force Satellite Control Facility (AFSCF). This test demonstrates the ability of the space vehicle, when in orbit, to properly respond to the AFSCF hardware, software, and operations team as specified in the AFSCF Orbital support Plan. For programs which have a dedicated ground station, compatibility tests shall also be performed with the dedicated ground station.

9.4.4.2 Test Description. Facilities to perform on-orbit system compatibility tests exist at the Western Test Range (WTR) and the Eastern Test Range (ETR). At both locations, the AFSCF can command the space vehicle and process telemetry from the space vehicle as well as perform tracking and ranging, thus verifying the rf compatibility, the command software, and the telemetry modes. The tests include the following:

a. Verification of rf, analog. and digital compatibility of command, telemetry, and tracking links.

b. Verification of AFSCF capability to control the space vehicle using single, block, unsecure, and secure commands as required for on-orbit support.

c. Verification of AFSCP capability to process, display, and record space vehicle telemetry link or links as required for on-orbit support.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

d. Verification of AFSCF capability to track the space vehicle using angle, doppler, and range tracking as required for on-orbit support.

9.4.4.3 Supplementary Requirements. This test should be run as soon as feasible after the space vehicle arrives at the launch base. The test is made with every space vehicle to verify system interface compatibility. The test shall be run using the software model versions that are integrated into the operational on-orbit software of the space vehicle under test. A preliminary compatibility test may be run prior to the arrival of the space vehicle at the launch base by the use of prototype subsystems, components, or simulators as required to prove the interface. Preliminary compatibility tests may be run using preliminary software. Normally. a preliminary compatibility test is run once for each series of space vehicles to check design compatibility, and is conducted well in advance of the first launch to permit orderly correction of hardware. software, and procedures as required. Changes in the interface from those tested in the preliminary test shall be checked by the compatibility tests conducted just prior to launch.

12.1.2 Rationale for Prelaunch Validation Tests. The purpose of the prelaunch validation tests is to verify by end-to-end tests that each critical path in the launch system, in the on-orbit system, and in the reentry system is satisfactory; i.e., there are no out-of-tolerance conditions or anomalous behavior. Duplication of the factory functional tests is also intended to provide data for trend analysis that might provide evidence of a problem, even though all measurement were within tolerances. Whether electrical, mechanical, or both, all critical paths or circuits shall be verified from the application of the initiating signal through completion of each event. This testing is intended to verify that an event command or signal was properly generated and sent on time, that it arrived at its correct destination, that no other function was performed, and that the signal was not present other than when programmed. Once successfully accomplished, that particular critical path or circuit is considered validated. Not all end-to-end tests can be performed with only flight hardware, as in the case where an explosive event is involved. In cases where end-to-end testing cannot be performed with the flight hardware, appropriate simulation devices should be used to exercise the flight hardware to the maximum extent possible. Simulation devices should be carefully controlled and should be permitted only when there is no feasible alternative for conducting the test. All of the events that occur during the mission profile should be tested in the flight sequence to the

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

extent that is practical. The space vehicle should be operated through the ascent sequence, separation and engine ignition phase, orbital injection, on orbit, and if applicable, recovery phase. Redundant components and subsystems should also be validated in the same manner.

12.1.3 <u>Guidance for Use of Prelaunch Validation Tests</u>. Because signals or commands can be communicated to the space vehicle in a variety of ways, no single end-to-end test configuration can be defined. Consequently, the term "end-to-end test" was not used in MIL-STD-1540B, but the prelaunch validation tests described include the classical functional tests, end-to-end tests, and sequential tests defined in other documents. The end-to-end tests should include negative logic tests to verify lockout, to assure that no other function than the intended function was performed, and that the signal was not present other than when programmed.

For the space shuttle cargoes that have a link through the orbiter, the end-to-end test includes verification of orbiter to cargo interfaces through an orbital functional simulator prior to cargo installation in the orbiter.

The compatibility of the space vehicle and the on-orbit command and control network is a further part of the system end-to-end testing.

12.2 PROPULSION SYSTEM LEAK AND FUNCTIONAL TEST

12.2.1 <u>Standard Criteria</u>. Contents of Paragraph 9.4.2 of MIL-STD-1540B (requirements for propulsion system leak and functional test) are as follows:

> *9.4.2 <u>Propulsion System Leak and Functional Test</u>. A functional test of the space vehicle propulsion subsystem shall be conducted to verify, to the maximum practical extent. the proper operation of all components. Propulsion system leakage rates shall be verified to be within allowable limits.*

12.2.2 <u>Rationale for Propulsion System Leak and Functional Test Requirements</u>. Functional testing of the propulsion subsystem is conducted to verify that all components are operating properly. Leakage testing of the propulsion subsystem is performed to verify that space vehicle transport and handling has not degraded the previously factory-tested system.

12.2.3 <u>Guidance for Use of Propulsion System Leak and Functional Test Requirements</u>. Prior to leakage testing, a pressure test at maximum expected operating pressure (MEOP) is

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

recommended. Leakage rates are recommended to be verified at the MEOP unless specified otherwise. However, testing should be conducted at the minimum pressure if the valves or fittings have a greater tendency to leak at minimum operating pressures than at maximum.

If the structural integrity of the system has been violated since the time that the last proof pressure test was conducted, a proof pressure test prior to leakage test is recommended. All pressure tests at the launch site should be performed within the requirements imposed by the existing range safety requirements,

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

THIS PAGE INTENTIONALLY LEFT BLANK

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

SECTION 13

INDEX

13.1 SUBJECT INDEX

Page

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

13.1 SUBJECT INDEX (continued) Page

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

13.1 SUBJECT INDEX (continued) Page

13.2 MIL-STD-1540B PARAGRAPH CROSS REFERENCE INDEX

The cross reference in the following Table XXI is provided to indicate which paragraphs of MIL-STD-1540B are discussed in this handbook, and to identify the paragraph numbers and page numbers of the corresponding discussion in the handbook. Paragraphs of MIL-STD-1540B that are not listed in this cross reference are not specifically addressed in the handbook.

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XXI. Cross Reference Index to MIL-STD-1540B Paragraphs

MIL-STD-1540B PARAGRAPH NUMBER	MIL-STD-1540B PARAGRAPH TITLE	THIS HANDBOOK'S CORRESPONDING PARAGRAPH	PAGE
1.2	Application	4.1	5
3.	DEFINITIONS		
3.4	Burst Pressure	5.7	24
3.8	Design Environments, Space Vehicles	5.1	9
3.9	Design Environments, Space Vehicle Components	5.1	9
3.12	Environmental Design Margins	5.1	10
3.18	Limit Load	5.2	14
3.20	Max. Pred. Acoustic Environment	5.3	15
3.21	Max. Pred. Operating Pressure	5.7	24
3.22	Max. Pred. Pyrotechnic Shock Environment	5.4	17
3.23	Max. Pred. Random Vibration Environment	5.5	18
3.25	Max. and Min. Pred. Component Temperatures	5.6	20
3.34	Proof Pressure	5.7	24
3.45	Thermal Uncertainty Margin	5.6	20
3.46	Ultimate Load	5.2	14
4.	GENERAL REQUIREMENTS		
4.1	Testing Philosophy	Foreword	iii
4.1.5.2	Reusable Flight Hardware Testing	6.1	27
4.2.2	Test Condition Tolerances	5.1	11
4.2.3	Tailoring	6.2	34
4.3	Retest	6.3	40-60
4.4	Test Data Analysis	6.4	60
5.	DEVELOPMENT TESTING	7.1, 7.2	63-65
6.	QUALIFICATION TESTING		
6.2	Space Vehicle Qualification Tests	8.1	67-95
6.2.3	Acoustic Test, Space Veh. Qual	8.2.1	76
6.2.4	Vibration Test, Space Veh. Qual	8.3	79
6.2.5	Pyro. Shock Test, Space Veh. Qual.	8.4.1	79
6.2.6	Pressure Test, Space Veh. Qual	8.5	82
6.2.7	Thermal Vacuum Test, Space Veh. Qual	8.6.1	85
6.2.8	Thermal Balance Test, Space Veh. Qual	8.7	90
6.2.9	Thermal Cycling Test, Space Veh. Qual	8.8.1	93

(Continued on next page)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE TECHNICAL LIBRARY ABBOTTAEROSPACE.COM

TABLE XXI. Cross Reference Index to MIL-STD-1540B Paragraphs

(Continued)

MIL-STD-1540B PARAGRAPH NUMBER	MIL-STD-1540B PARAGRAPH TITLE	THIS HANDBOOK'S CORRESPONDING PARAGRAPH	PAGE
6.3.1	Structural Static Load Test, Subsystem Qualification..........	8.9	98-104
6.4	Component Qualification Tests..........	9.1	105-115
6.4.2	Thermal Vacuum Test, Component Qual.	Table X	111
6.4.3	Thermal Cycling Test, Component Qual.	Table IX	112
6.4.5	Random Vibration Test, Component Qual.	Table XII	114
6.4.6	Acoustic Test, Component Qual.......	Table XIII	114
6.4.7	Pyro. Shock Test, Component Qual....	Table XI	113
6.4.10	Pressure Test, Component Qual.......	9.2	115
6.4.11	Leakage Test, Component Qual........	9.3	120
6.4.13	Life Test, Component Qual...........	9.4	123
6.5	Subassembly Level Qualification Tests..	10	129-136
7.	ACCEPTANCE TESTS		
7.1	Space Veh. Acceptance Tests............	8.1	68-75
7.1.3	Acoustic Test, Space Veh. Accept....	8.2.2	75
7.1.4	Vibration Test, Space Veh. Accept...	8.3	80
7.1.5	Pyro. Shock Test, Space Veh. Accept.	8.4.2	81
7.1.6	Pressure Test, Space Veh. Accept....	8.5	83
7.1.7	Thermal Vacuum Test, Space Veh. Acceptance......................	8.6.2	89
7.1.8	Thermal Cycling Test, Space Veh. Acceptance......................	8.8.2	96
7.3	Component Acceptance Tests.............	9.1	107-115
7.3.2	Thermal Vacuum Test, Comp. Accept...	Table X	111
7.3.3	Thermal Cycling Test, Comp. Accept..	Table IX	112
7.3.4	Random Vibration Test, Comp. Accept.	Table XII	114
7.3.5	Acoustic Test, Comp. Acceptance.....	Table XIII	114
7.3.6	Pyrotechnic Shock Screening Test, Component Acceptance.............	Table XI	113
7.3.9	Burn-in Test, Component Acceptance................	9.5 & Table XIV	115,125
7.4	Subassembly Level Acceptance Tests.....	10	129-136

(Continued on next page)

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

TABLE XXI. Cross Reference Index to MIL-STD-1540B Paragraphs

(Concluded)

MIL-STD-1540B PARAGRAPH NUMBER	MIL-STD-1540B PARAGRAPH TITLE	THIS HANDBOOK'S CORRESPONDING PARAGRAPH	PAGE
8.	FLIGHT USE OF QUALIFICATION EQUIPMENT.....	11	137-156
8.1	Use of the Qualification Components for Flight.....	11, Table VII, & Table XIX	138-151
8.2	Use of the Flight Vehicle for Space Vehicle Level Qual......	11, Table VII, & Table XX	138-151
9.	PRELAUNCH VALIDATION TESTS		
9.4	Prelaunch Validation Test Descriptions		
9.4.1	Functional Test......................	12.1	153
9.4.2	Propulsion System Leak and Functional Test..................	12.2	156
9.4.3	Integrated Systems Test.............	12.1	153
9.4.4	Compatibility Test, On-orbit Systems	12.1	154

Custodians
Air Force - 19

Preparing Activity
Air Force - 19

(Project No. 1810-F031)

Document 0126e/Arch 0202b
Document 0127e/Arch 0202b
Document 0129e/Arch 0203b
Document 0130e/Arch 0203b

U. S. GOVERNMENT PRINTING OFFICE: 1985 - 505-038/23242

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

RUCTIONS: In a continuing effort to make our standardization documents better, the DoD provides this form for use in itting comments and suggestions for improvements. All users of military standardization documents are invited to provide stions. This form may be detached, folded along the lines indicated, taped along the loose edge *(DO NOT STAPLE)*, and d. In block 5, be as specific as possible about particular problem areas such as wording which required interpretation, was igid, restrictive, loose, ambiguous, or was incompatible, and give proposed wording changes which would alleviate the ems. Enter in block 6 any remarks not related to a specific paragraph of the document. If block 7 is filled out, an owledgement will be mailed to you within 30 days to let you know that your comments were received and are being dered.

E. This form may not be used to request copies of documents, nor to request waivers, deviations, or clarification of ication requirements on current contracts. Comments submitted on this form do not constitute or imply authorization ive any portion of the referenced document*(s)* or to amend contractual requirements.

☆ U.S. Government Printing Office: 1982—360-679/3804

(Fold along this line)

(Fold along this line)

AL BUSINESS
TY FOR PRIVATE USE $300

NO POSTAGE
NECESSARY
IF MAILED
IN THE
UNITED STATES

BUSINESS REPLY MAIL

FIRST CLASS PERMIT NO 73236 WASH DC

POSTAGE WILL BE PAID BY HQ U.S. AIR FORCE

SD/ALM
P.O. Box 92960
Worldway Postal Center
Los Angeles, CA 90009-2960

THIS DOCUMENT PROVIDED BY THE ABBOTT AEROSPACE
TECHNICAL LIBRARY
ABBOTTAEROSPACE.COM

(TO DETACH THIS FORM, CUT ALONG THIS LINE.)

STANDARDIZATION DOCUMENT IMPROVEMENT PROPOSAL

(See Instructions – Reverse Side)

1. DOCUMENT NUMBER	2. DOCUMENT TITLE
MIL-HDBK-340(USAF)	Application Guidelines for MIL-STD-1540B; Test Requirements for Space Veh

3a. NAME OF SUBMITTING ORGANIZATION

b. ADDRESS *(Street, City, State, ZIP Code)*

4. TYPE OF ORGANIZATION *(Mark c*

☐ VENDOR

☐ USER

☐ MANUFACTURER

☐ OTHER *(Specify)*

5. PROBLEM AREAS

a. Paragraph Number and Wording:

b. Recommended Wording:

c. Reason/Rationale for Recommendation:

6. REMARKS

7a. NAME OF SUBMITTER *(Last, First, MI)* – Optional	b. WORK TELEPHONE NUMBER *(I Code)* – Optional
c. MAILING ADDRESS *(Street, City, State, ZIP Code)* – Optional	8. DATE OF SUBMISSION *(YYMMD*

DD FORM 1426

PREVIOUS EDITION IS OBSOLETE.