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Testing

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Foreword

This Standard is one of the series of ECSS Standards intended to be applied together for the management, engineering and product assurance in space projects and applications. ECSS is a cooperative effort of the European Space Agency, national space agencies and European industry associations for the purpose of developing and maintaining common standards.

Requirements in this Standard are defined in terms of what shall be accomplished, rather than in terms of how to organize and perform the necessary work. This allows existing organizational structures and methods to be applied where they are effective, and for the structures and methods to evolve as necessary without rewriting the standards.

The formulation of this Standard takes into account the existing ISO 9000 family of documents.

This Standard has been prepared by the ECSS Testing Working Group, reviewed by the ECSS Technical Panel and approved by the ECSS Steering Board.

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Scope

1.1 General

a. This Standard,

1. provides standard environmental and performance test requirements for a space system and its constituents.
2. defines the test requirements for products and systems that are generally applicable to all projects. It also defines the documentation associated with testing activities.
3. applies to all types and combinations of project, organization and product.
4. is applicable to space systems and its constituents as defined in ECSS-E-00 “Space engineering - Policy and principles”.
5. covers each stage of verification by testing, as defined by ECSS-E-10-02, for a space system from development to post-landing.

b. This Standard does not:

1. define acceptance criteria, specifications or procedures for any particular project or class of projects.

   NOTE These and other detailed test requirements applicable to a particular project are defined in the applicable technical specifications and statement of work.

2. cover the following:
   - sounding rockets,
   - launch facilities,
   - test facilities,
   - training facilities and ground refurbishment,
   - logistic facilities, and
   - engine testing.
3. apply to software testing, (which is covered by ECSS-E-40) and to hardware below equipment levels (such as components).

1.2 Tailoring

The requirements defined in this Standard should be tailored to match the requirements of the particular profile and circumstances of a project.

NOTE Tailoring is a process by which individual requirements or specifications, standards and related documents are evaluated and made applicable to a specific project by selection, and in some exceptional cases, modification of existing or addition of new requirements.

This Standard should be used in combination with the ECSS-E-10-02. In particular, when tailored, it can become the "Test Requirement Specification" applicable to the specific project (see ECSS-E-10-02 for details).

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Normative references

The following normative documents contain provisions which, through reference in this text, constitute provisions of this ECSS Standard. For dated references, subsequent amendments to, or revisions of any of these publications do not apply. However, parties to agreements based on this ECSS Standard are encouraged to investigate the possibility of applying the most recent editions of the normative documents indicated below. For undated references the latest edition of the publication referred to applies.

ECSS-P-001	Glossary of terms
ECSS-E-10-02	Space engineering — Verification
ECSS-E-70 Part 1	Space engineering — Ground systems and operations — Principles and requirements
ECSS-E-70 Part 2	Space engineering — Ground systems and operations — Document requirements definitions (DRDs)
ANSI/ASME PTC19.1	Test uncertainty, instruments and apparatus: Supplement to ASME performance test codes
ISO 3740:1980	Acoustics — determination of sound power levels of noise sources — Guidelines for the use of basic standards and for the preparation of noise test codes
ISO 3744:1994	Acoustics — Determination of sound power levels of noise sources using sound pressure — Engineering method in an essentially free field over a reflecting plane

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Terms, definitions and abbreviated terms

3.1 Terms and definitions

The following terms and definitions are specific to this Standard in the sense that they are complementary or additional to those contained in ECSS-P-001.

3.1.1
acceptance stage

verification stage with the objective of demonstrating that the product is free of workmanship defects and integration errors and ready for its intended use

[ECSS-E-10-02A]

3.1.2
airborne support equipment (ASE)

equipment installed in a recoverable launch vehicle to provide support functions and interfaces for the spacecraft during launch and orbital operations of the recoverable launch vehicle

NOTE ASE includes the hardware and software that provides the structural, electrical, electronic and mechanical interfaces with the launch vehicle. ASE is recovered with the launch vehicle.

3.1.3
burst pressure

maximum test pressure that pressurized equipment withstands without rupture to demonstrate the adequacy of the design in a qualification test

NOTE 1 Burst pressure is equal to the product of the maximum expected operating pressure, a burst pressure design factor, and a factor corresponding to the differences in material properties between test and design temperatures.

NOTE 2 An item subjected to a burst pressure test is not used for other purposes.

3.1.4
design environments

composite of the various environmental loads, to which the hardware is designed

NOTE Each of the design environments is based upon:

- the maximum and minimum predicted environments during the operational life of the item;
- the qualification margin that increases the environmental range to provide an acceptable level of confidence that a failure does not occur during the service life of the item;
- uncertainties and tolerances related to the analytical prediction.

3.1.5
environmental design margin
increase of the environmental extremes for the purpose of design and qualification above the levels expected during the life cycle

NOTE Environmental design margin includes levels such as mechanical, thermal, radiation as well as the time of exposure of them.

3.1.6
environmental test
simulation of the various constraints (together or separately) to which an item is subjected during its operational life cycle

NOTE Environmental tests cover natural and induced environments.

3.1.7
fundamental resonance (for structural modes)
first major significant resonances as observed during one-axis vibration test for each of the three test axes

NOTE 1 The term fundamental resonance is used in conjunction with notching of sinusoidal vibration input spectrum for item qualification.

NOTE 2 Significant resonances are modes that have an effective mass greater than 10 % of the total mass of the item.

3.1.8
in-orbit stage
verification stage valid for projects whose characteristics (e.g. mission and in-orbit operations) require in-orbit verification

[ECSS-E-10-02]

3.1.9
integrated system test
test that has the scope to verify that the performance of the element meets the specification requirements, in terms of correct operation in all operational modes, including back-up modes and all foreseen transients

NOTE Integrated system test is also known as "system functional test".

3.1.10
integrated system check
sub-set of the integrated system test, able to involve all major functions, at the maximum extent automatically performed and with the scope to provide the criteria for judging successful survival of the element in a given test environment, with a high degree of confidence, in a relatively short time

NOTE Integrated system check is also known as "abbreviated functional test".

3.1.11
limit load

maximum anticipated load, or combination of loads, which a structure is expected to experience during the performance of specified missions in specified environments

NOTE Since the actual loads that are experienced in service are in part random in nature, statistical methods for predicting limit loads are generally employed.

3.1.12
low level sinusoidal vibration

exposing an item to a frequency sweep of low level sinusoidal vibrations to show possible deficiencies in workmanship, as a consequence of another environment

NOTE Low level sinusoidal vibration test is also known as "signature test".

3.1.13
maximum predicted acceleration

acceleration value determined from the combined effects of the quasi steady acceleration and the transient response of the vehicle to engine ignition, engine burnout and stage separation

NOTE Where the natural frequency of the equipment mount or mounting structure can couple with engine initiated transients, the maximum predicted acceleration level accounts for the possible dynamic amplification.

3.1.14
maximum predicted acoustic environment

maximum value of the time average r.m.s. SPL (sound pressure level) in each frequency band occurring below payload fairing or within STS orbiter cargo bay, which occurs during lift-off, powered flight or re-entry

NOTE The maximum predicted acoustic environment test spectrum is specified in octave or 1/3 octave bands over a frequency range of 31,5 Hz to 10 000 Hz. The duration of the maximum environment is the total period when the overall amplitude is within 6 dB of the maximum overall amplitude.

3.1.15
maximum predicted operating pressure

working pressure applied to equipment 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

3.1.16
maximum predicted pyro shock environment

maximum absolute shock response spectrum determined by the response of a number of single degree of freedom systems using an acceleration amplification factor at the resonant frequency of lightly damped system (Q = 10)

NOTE 1 The shock response spectrum is determined at frequency intervals of one-sixth octave or less over a frequency range of 100 Hz to 4 000 Hz or more.

NOTE 2 The pyro shock environment imposed on the spacecraft equipment is due to structural response when the space or launch vehicle electro-explosive devices are activated. Resultant structural response accelerations have the form of superimposed complex decaying sinusoids that decay to a few percent of their maximum acceleration in 5 ms to 15 ms.

3.1.17
maximum predicted random vibration environment

random vibration environment imposed on the spacecraft, subsystems and equipment due to the lift-off acoustic field, aerodynamic excitations, and transmitted structure-borne vibration.

NOTE 1 A different spectrum can exist for different equipment zones or for different axis. The equipment vibration levels are based on vibration response measurements made at the equipment attachment points during ground acoustic tests or during flight. The duration of the maximum environment is the total period during flight when the overall level is within 6 dB of the maximum overall level.

NOTE 2 The Power spectral density is based on a frequency resolution of 1/6 octave (or narrower) bandwidth analysis, over a frequency range of 20 Hz to 2 000 Hz

3.1.18
maximum predicted sinusoidal vibration environment

predicted environment imposed on the spacecraft, subsystems and equipment due to sinusoidal and narrow band random forcing functions within the launch vehicle or spacecraft during flight or from ground transportation and handling

NOTE 1 In flight, sinusoidal excitations are caused by unstable combustion, by coupling of structural resonant frequencies (POGO), or by imbalances in rotating equipment in the launch vehicle or spacecraft. Sinusoidal excitations occur also during ground transportation and handling due to resonant responses of tires and suspension systems of the transporter.

NOTE 2 The maximum predicted sinusoidal vibration environment is specified over a frequency range of 5 Hz to 100 Hz for flight excitation.

3.1.19
maximum and minimum predicted equipment temperatures

highest and lowest temperatures that are expected to occur in flight on each equipment of the spacecraft during all operational and non-operational modes which include uncertainties

3.1.20
model philosophy

definition of the optimum number and characteristics of physical models to achieve a high confidence in the product verification with the shortest planning and a suitable weighing of costs and risks

[ECSS-E-10-02A]

3.1.21
moving mechanical assemblies

mechanical or electromechanical devices that control the movement of one mechanical part of a spacecraft relative to another part

NOTE Moving mechanical assemblies include: deployment mechanisms, pointing mechanisms, drive mechanisms, design mechanisms and the actuators, motors, linkages, latches, clutches, springs, cams, dampers, booms, gimbals, gears, bearings and instrumentation that are an integral part of these mechanical assemblies (e.g. recorders).

3.1.22
multipacting
resonant back and forth flow of secondary electrons in a vacuum between two surfaces separated by a distance such that the electron transit time is an odd integral multiple of one half the period of the alternating voltage impressed on the surface

NOTE 1 Multipacting does not occur unless an electron impacts one surface to initiate the action, and a secondary emission of one or more electrons at each surface to sustain the action takes place.

NOTE 2 Multipacting is an unstable self-extinguishing action which occurs at pressures less than $6{,}65 \times 10^{-2}$ hPa, however, it becomes stable at a pressure less than $1{,}33 \times 10^{-3}$ hPa.

NOTE 3 The pitting action resulting from the secondary emission of electrons degrades the impacted surfaces. The secondary electron emission can also increase the pressure in the vicinity of the surfaces causing ionization (corona) breakdown to occur.

NOTE 4 These effects can cause degradation of performance or permanent failure of the radio frequency cavities, waveguides or other devices involved.

3.1.23
notching of sinusoidal vibration input spectrum
notching of the shaker input spectrum to limit structural responses at resonant frequencies according to qualification or acceptance loads

NOTE Notching of sinusoidal vibration input spectrum is a general accepted practise in vibration testing.

3.1.24
operational modes
combination of operational configurations or conditions that can occur during the service life for equipment or spacecraft

EXAMPLE Power-on or power-off, command modes, readout modes, attitude control modes, antenna stowed or deployed, and spinning or de-spun.

3.1.25
post-landing stage
verification stage valid for projects where characteristics for post-landing verification is performed (e.g. multimission projects)

[ECSS-E-10-02A]

3.1.26
pre-launch stage
verification stage with the objective to verify that the flight article is properly configured for launch and capable to function as planned for launch

[ECSS-E-10-02A]

3.1.27
proof pressure
test pressure for pressurized equipment to sustain without detrimental deformation

NOTE 1 The proof pressure is used to give evidence of satisfactory workmanship and material quality, or to establish maximum possible flaw size.

NOTE 2 The proof pressure is equal to the product of maximum expected operating pressure (see 3.1.15), proof pressure design factor, and a factor accounting for the difference in material properties between test and design temperature.

3.1.28
qualification stage
verification stage with the objective to demonstrate that the design conforms to the requirements including margins

[ECSS-E-10-02A]

3.1.29
service life
total life expectancy of an item, equipment or space vehicle

NOTE The service life starts at the completion of assembly of the item and continues through all acceptance testing, handling, storage, transportation, launch operations, orbital operations, refurbishment, retesting, re-entry or recovery from orbit, and reuse if applicable.

3.1.30
space element
product or set of products intended to be operated in outer space

NOTE 1 In order to avoid repetition in the level of decomposition of the space product, the term element is used to define "systems within the system". The term element is used to identify any system within the space system.

NOTE 2 Elements that operate entirely in space or on the ground are referred to as "Space segment" and "Ground Segment" respectively.

3.1.31
space vehicle
integrated set of subsystems and equipment capable of supporting an operational role in space

NOTE A space vehicle can be an orbiting vehicle, a major portion of an orbiting vehicle, or a payload that performs its mission while attached to a launch or upper-stage vehicle. The ground support equipment is considered to be a part of the space vehicle.

3.1.32
stabilized test temperature
specified temperature for equipment and subsystem tests that has been achieved and has not changed by more than 1 °C during the previous one-hour period

NOTE During system level tests, performance verification testing may be started when the rate of change is below 1 °C within a time period equal or near the time constant of the spacecraft.

3.1.33
temperature reference, reference point
physical point located on the equipment providing a simplified representation of the equipment thermal status

NOTE 1 Depending upon the equipment dimensions, more than one temperature reference may be defined.

NOTE 2 The temperature of the reference point is measured by temperature sensors during test. The temperature distribution within the equipment and hot spots on the external casing due to point heat sources are not used as reference points.

3.1.34
ultimate load
maximum static load to which a structure is designed

NOTE 1 It is obtained by multiplying the limit load by the ultimate factor of safety.

NOTE 2 See annex A.3

3.2 Abbreviated terms

The following abbreviated terms are defined and used within this Standard.

Abbreviation	**Meaning**
ABCL	as-built configuration list
ABM	apogee boost motor
AFT	abbreviated functional test
AOCS	attitude and orbit control system
APTC	ambient pressure temperature cycling
ASE	airborne support equipment
ATC	accelerate thermal cycling
AVM	anti vibration mounts
AVT	acceptance vibration test
BAPTA	bearing and power transfer assembly
CCB	configuration control board
CIDL	configuration item data list
CoG	centre of gravity
DRD	document requirements definition
ECLS	environmental control and life support
EGSE	electronic ground support equipment
EMC	electromagnetic compatibility
EQM	engineering qualification model
ESD	electrostatic discharge
ESSEH	environmental stress screening of electronic hardware
FM	flight model
FOP	flight operations plan
GSE	ground support equipment
HFE	human factors engineering
hi-rel	high reliability

ISC	integrated system check
ISST	integrated subsystem test
IST	integrated system test
LCDA	launcher coupled dynamic analysis
LEO	low Earth orbit
LEOP	launch and early orbit phase
LV	launch vehicle
M	mass
MEC	microgravity environmental compatibility
MIL (spec)	specification of the US Department of Defense
MLI	multi-layer insulation
MoI	moment of inertia
NCR	nonconformance report
NDIU	network data interface unit
NRB	nonconformance review board
PFM	protoflight model
PSD	power spectral density
PTC	passive thermal control
POGO	propulsion generated oscillations
QAVT	qualification vibration test
QM	qualification model
r.m.s.	root-mean-square
RSS	root-sum-square
RF	radio frequency
SADM	solar array drive mechanism
SFT	system functional test
SPL	sound pressure level
STM	structural-thermal model
STS	space transportation system
TC	telecommand
TCL	test configuration list
TCS	thermal control subsystem
TM	telemetry
TMM	thermal mathematical model
TT&C	telemetry, tracking and command
TWT	travelling wave tube
TWTA	travelling wave tube assembly
VTC	vacuum temperature cycling

4

General requirements

4.1 Testing philosophy

4.1.1

Test programmes are an essential part of the overall verification to ensure that the product achieves all the design, performance and quality requirements.

4.1.2

a. Test planning, test requirements, and test criteria shall be derived from the design requirements.

NOTE The design requirements indicate the expected performance of the test item as well as the conditions under which it operates.

b. To facilitate this process, all system, element, subsystem, and equipment product specifications shall indicate their intended verification method (test, analysis, review of design, or inspection) and level for each stated requirement (verification matrix), as defined in the ECSS-E-10-02.

c. Starting from the verification matrix, a dedicated test matrix shall be prepared defining the detailed types of tests to be implemented.

d. Based on the test requirements as above documented, a test plan shall be established as part of the overall verification plan.

e. The test plan shall:

1. describe and sequence all the tests,
2. state the objectives and the scope of the tests,
3. specify the test facilities, and
4. govern the execution of all tests.

EXAMPLE By setting test documentation and approval requirements, cleanliness and product assurance requirements, handling of discrepancies, and personnel safety requirements.

f. The determination of test sequences should be based on two main considerations:

1. Preserve the order in which environments are encountered during the operational life, and

2. Detect potential failures and defects as early in the test sequence as possible.

g. When defining a test baseline, the specific item characteristics (e.g. design maturity and margins, qualification status, and model philosophy) and the programmatic characteristics (e.g. cost and acceptable risks) shall be considered for each project.

h. Testing is the preferred verification method with the lowest risk, but, because it represents a large expense, the tailoring of the test programme should also assure that a cost effective programme is achieved (see ECSS-E-10-02).

4.1.3

The overall test programme shall encompass different stages at different project levels and in different phases in accordance with the verification process definition (see ECSS-E-10-02).

EXAMPLE Development, qualification, acceptance testing, both on-ground and in-orbit.

4.1.4

The test programme should satisfy the specific objectives of each stage prior to the accomplishment of next stage, using a sequential bottom-up product approach.

4.1.5

a. Design suitability should be demonstrated in the development phase prior to the start of the formal qualification testing.

b. Qualification testing should be completed and consequential design improvements incorporated, prior to the initiation of the flight item manufacturing.

4.1.6

a. The environmental tests should be imposed sequentially, rather than in combination.

NOTE Features of the hardware design or of the service environments can justify the necessity of combined environments in some tests.

b. Where it is not feasible to test a product as a single entity, major assemblies, modules or stages of the product can be tested, with the support of analyses, simulations or simulators.

EXAMPLE The service and payload module in the case of a space vehicle and first~~,~~ or second stage in the case of a launcher, which can be integrated and tested separately.

NOTE Launch vehicle testing is very intensive during development and qualification stages and more limited for production models. Apart from functional tests, very few tests are done on complete launch vehicles. Most of the tests are conducted at equipment and subsystem level. Element level testing takes place only at the launch range or in qualification flights.

4.2 Model philosophy

4.2.1

The test baseline definition shall be based on the project model philosophy.

NOTE Referring to the model philosophies, and related activity identification, see annex B.1 of ECSS-E-10-02A, it can be seen how the qualification, acceptance and protoflight test activities are shared among the different models. This sharing depends on model philosophy, project characteristics and model representativeness.

4.2.2

The following project model philosophy situations can exist:

a. Prototype approach

1. The qualification testing can be conducted on one or more qualification model (QM), according to the project requirements and objectives, always with qualification levels and duration.
2. For tests on more than one QM, the tests shall be performed on the different models according to their representativeness (e.g. functional qualification is performed on EQM) and the test sequences for each model shall be adapted accordingly.
3. The FM shall be subjected to complete acceptance testing.

b. Protoflight approach

1. All the qualification tests shall be performed on the same model to be flown, normally with qualification levels and reduced duration.
2. The protoflight model (PFM) should be subjected to a test programme defined on a case-by-case basis.

   NOTE The test programme combines both qualification and acceptance tests to satisfy the qualification and the acceptance objectives, see subclauses 4.4 and 4.5.

c. Hybrid approach

1. A combination of the prototype and protoflight rules shall be applied.
2. Specific qualification testing in the critical areas can be conducted on dedicated models (e.g. STM, QM, EQM or others).
3. In critical areas acceptance testing shall be performed only on the PFM.

4.3 Development testing

4.3.1

a. The objective of development testing is to support the design feasibility and to assist in the evolution of the design.

b. Development tests are used to validate new design concepts and the application of proven concepts and techniques to a new configuration.

4.3.2

a. Development tests shall be used to confirm

1. performance margins,
2. manufacturability,
3. testability,
4. maintainability,
5. reliability,

6. life expectancy,
7. failure modes and
8. compatibility with safety requirements.

NOTE 1 Development tests need not be subjected to the same levels and controls associated with the qualification, acceptance, pre-launch, in-orbit and post-landing test campaigns.

NOTE 2 Development testing cannot be reduced to a standardized set of procedures.

NOTE 3 Development testing requirements are dependent upon the maturity of design and upon the operational requirements of the specific project.

b. Where practicable, development tests should be conducted over a range of operating conditions that exceed the design limits to identify marginal design features.

c. Development tests can be conducted on mock-ups, development (breadboards) and integration models.

d. Adequate records of test configuration, test results and other pertinent data shall be maintained so that this information is available to supplement other portions of the verification programme as required, or when failure occurs during the functioning of qualification and flight models.

4.4 Qualification testing

4.4.1

The objective of qualification testing is the formal demonstration that the design implementation and manufacturing methods have resulted in hardware and software conforming to the specification requirements.

4.4.2

The purpose of qualification testing shall be to demonstrate that the items perform satisfactorily in the intended environments with sufficient margins.

4.4.3

The qualification test levels shall exceed the maximum predicted levels by a factor of safety which assures that, even with the worst combination of test tolerances, the flight levels shall not exceed the qualification test levels.

4.4.4

The qualification testing shall be conducted on dedicated qualification models (QM) that are produced from the same drawings, using the same materials, tooling and methods as the flight item.

NOTE The standard of the QM may be slightly relaxed with respect to flight standard, e.g. by using lower quality parts instead of the space qualified hi-rel ones, without affecting the test validity (EQM).

4.4.5

Minor changes with respect to the flight models can be made to the QM used for qualification testing, providing they do not affect the qualification objectives

EXAMPLE Adding items such as thermocouples, strain gauges and monitoring leads, necessary to conduct the test.

4.4.6

a. The complete flight element (i.e. flight vehicle) should be used during environmental testing for qualification.

NOTE Environmental testing of a complete element is not always feasible (e.g. due to its size it can exceed the capacity of a test facility). In this case, qualification of an element can be achieved by a combination of tests on various assembly levels, supported by analysis and similarity assessment, review of design, and inspection.

b. The requirements for tests on levels below the element shall be derived from the system qualification requirements.

c. The effects of constituents, which are interacting on the element level, but which are not present during these tests, shall be taken into consideration.

4.4.7

The environments used during qualification tests shall be the design levels which are usually more severe than those predicted to occur during flight in order to account for variability in subsequent production articles and other uncertainties (see subclause 4.8.1).

4.5 Acceptance testing

4.5.1

The purpose of acceptance testing is to demonstrate conformance to specification and to act as quality control screens to detect manufacturing defects, workmanship errors, the start of failures and other performance anomalies, which are not readily detectable by normal inspection techniques.

4.5.2

The acceptance tests shall be formal tests conducted to demonstrate the adequacy and readiness of an item for delivery and subsequent usage.

4.5.3

Acceptance testing shall not create conditions that exceed safety margins or cause unrealistic modes of failure.

4.5.4

Acceptance tests shall be conducted on flight models under environmental conditions no more severe than those expected during the mission.

4.5.5

The acceptance tests shall be conducted on all the flight products (including spares) according to the model philosophy of subclause 4.2, and ground elements.

4.5.6

a. Systems and elements that cannot be flight acceptance tested as a whole due to size limitations of available test facilities shall undergo a controlled acceptance process corresponding to the qualification process described in subclause 4.1.

b. All tests below the system level shall be traceable to the system acceptance requirements.

c. Effects of constituents, which are interacting on system level, but which are not present during these tests, shall be taken into account.

4.6 Protoflight testing

4.6.1

a. In a minimum risk programme, as in the prototype approach (see ECSS-E-10-02), the products that were subjected to qualification tests are not eligible for flight, since there was no demonstration of remaining life of the product.

b. The protoflight approach (see ECSS-E-10-02) can present a higher risk than the prototype approach in which design margins are demonstrated by testing of a dedicated qualification product.

c. A strategy to minimize the risk can be applied by enhancing development testing, by increasing the design factors of safety and by implementing an adequate spare policy.

d. The protoflight approach can be applied at each level of decomposition of space system.

e. For detailed considerations on the model philosophy see ECSS-E-10-02A, annex B.1.

4.6.2

The protoflight test levels and durations shall be as follows:

a. protoflight test levels: as qualification margins (see subclause 4.8.1.2);

b. protoflight test durations: as acceptance durations (see subclause 4.8.1.3).

4.7 Retesting

4.7.1 General

a. Since the scope and the nature of retesting differs so much, standard test requirements or test guidelines shall be decided on a case-by-case basis.

b. There are five situations, which can result in a retest described in 4.7.2 to 4.7.6 below.

4.7.2 Failure or anomaly during qualification, acceptance or protoflight testing

a. The failure or anomalies occurred during the testing activities, shall be recorded on a nonconformance report (NCR) and the nonconformance review board (NRB) shall convene to decide on the course of actions to take (see ECSS-Q-20-09).

b. After the implementation of the NRB disposition and the elimination of the failure cause, a retest should be performed.

c. The NRB shall decide on the necessity and the extent of the retest in order to demonstrate the correctness of the disposition made.

4.7.3 Implementation of a design modification after completion of qualification testing

The configuration control board (CCB) shall evaluate and decide the extent of the qualification test sequence to be repeated if:

a. a design modification is implemented after the completion of qualification testing, or

b. a modification (after being qualified) is retrofitted to an already accepted piece of hardware (or software), in which case the complete or part of the acceptance test sequence (as defined in clause 6) shall be repeated to demonstrate the absence of poor quality or bad workmanship.

4.7.4 Long duration storage of flight hardware after acceptance testing

4.7.4.1

a. If flight hardware is stored for an extended time (six months or longer) between acceptance and shipment to the launch site, periodic checks on its health status shall be performed.

b. Periodic maintenance shall be performed on items such as travelling wave tubes (TWTs), tape recorders, batteries (conditioning) and special lubricated mechanisms and motors.

c. Age sensitive items shall be assessed for their life expectancy and replaced if they do not achieve the operating requirements prior to shipment.

4.7.4.2

Upon removal of flight hardware from storage for shipment it shall be determined, if a full or partial set of mechanical and functional tests (as defined in clause 6) shall be repeated.

NOTE The determination depends on storage time, the complexity of the subject hardware and whether any items were temporarily removed (e.g. batteries, sensors and propellant) or disconnected.

4.7.5 Hardware to be re-flown

a. Hardware that is re-flown shall be thoroughly inspected for any hidden damage.

NOTE The type and extent of retests (acceptance) depends on the following:

- the time between the first acceptance test and the intended second usage;
- whether the hardware is used in the same application;
- the residual fatigue life.

b. If the conditions and the configurations are different from the previous usage, then a delta qualification should be performed.

4.7.6 Flight use of qualification hardware

4.7.6.1 Introduction

Often former qualification hardware is refurbished to be used as

- flight hardware (typically when there is more than one item of the same hardware needed), or
- flight spare.

This approach usually comes from programme costs and schedule constraints.

4.7.6.2 Assessment

A detailed assessment shall be established by the design engineer and the quality engineer to determine the refurbishment required to make the hardware acceptable, e.g. replacement of items, which are over-stressed or have the potential to be over-stressed due to qualification testing.

4.7.6.3 Acceptance tests

After the refurbishment the hardware should be subjected to a full or partial acceptance test depending on the extent of the refurbishment and disintegration.

4.8 Test conditions and tolerances

4.8.1 Test levels and duration

4.8.1.1 General

a. Margins to both test levels and duration should be applied to the environmental tests.

NOTE Functional and performance test requirements do not present margins.

b. Each test environment should be based on actual flight data, often scaled for differences in parameters, or if more reliable, by analytical prediction or a combination of analysis and flight data.

c. A margin can include an increase in level or range, an increase in duration or cycles of exposure, as well as any other appropriate increase in severity.

4.8.1.2 Qualification test levels and duration

a. The qualification environmental conditions shall stress the hardware to more severe conditions than those expected to occur during the service life. Therefore, the qualification test levels and duration shall be the predicted flight maximum increased by a qualification margin.

NOTE The qualification margin is intended to:

- avoid qualification test levels and duration that are less severe than those expected in flight;
- provide test levels which cover minor differences among qualification and flight units due to variations in parts, materials, processes, manufacturing and degradation during usage;
- provide appropriate test duration against fatigue failures due to repeated testing and operational use (e.g. acceptance plus flight(s) and re-testing);
- meet requirements under extreme conditions of flight, which when expressed statistically are the P99/90 estimates (i.e. an environment not expected to be exceeded on 99 % of flights, estimated with 90 % confidence).

b. Qualification testing shall not create conditions that exceed applicable design safety margins or cause unrealistic modes of failure.

c. The qualification test margin for levels and duration specified in Table 1, with the indication, of the applicability to equipment or space element levels, shall be applied.

NOTE The table does not include tests for some ambient conditions such as humidity and toxic-off gassing because they are performed exposing the hardware to the environment without margin.

Table 1: Qualification test levels and durations

Test	Levels		Duration	
	Equipment	Space element	Equipment	Space element
Shock	+6 dB [a]	N/A	3 shocks in both directions of 3 axes	3 activations of explosive firing
Acoustic	+4 dB [a]	+3 dB	2 min [b]	2 min [b]
Vibration	Random/Sine: +4 dB	Random/Sine: +3 dB	Random: 2,5 min per axis [b] Sine: 2 octave/min 1 sweep up and down (5 Hz-100 Hz)	Random: 2 min per axis [b] Sine: 2 octave/min (5 Hz-100 Hz) (notching, if necessary)
Thermal cycling	10 °C extension of maximum and minimum predicted temperatures [a]	10 °C extension of maximum and minimum predicted temperatures [c]	8 cycles	8 cycles
Thermal vacuum	10 °C extension of maximum and minimum predicted temperatures [a]	10 °C extension of maximum and minimum predicted temperatures [c]	8 cycles if combined with thermal cycling, 1 cycle if thermal cycling is performed	8 cycles if combined with thermal cycling, 1 cycle if thermal cycling is performed
EMC	+6 dB EMC safety margin		depending on operating modes	
Static/ acceleration	1,25	1,25	100 s + 50 s per mission	Sufficient to record test data
Pressure	1,5 (proof), 2 (burst)	1,5	5 min (3 cycles for valves) only for proof	5 min (3 cycles)
Life	N/A (margins only for accelerated tests)		4 times operating life	

[a] If the equipment qualification is carried out for multi-project utilization standard spectra or temperature limits can be used.

[b] Duration is dependent on the number of missions.

[c] A suitable distribution of the flight delta temperature for equipment is used, i.e. temperature limit is reached as soon as one unit in a selected area is at the hot and cold temperature reached during the unit qualification thermal testing.

4.8.1.3 Acceptance test levels and duration

a. The acceptance test levels shall cover the maximum flight levels encountered during the service life (except for equipment thermal tests where a margin shall be used to cover uncertainties of mathematical model) according to Table 2.

b. The test duration shall not degrade the item and be taken into account in the lifetime of the item itself.

c. The acceptance test levels and durations to be applied are summarized in Table 2, with the indication, where necessary, of the applicability to equipment or space element levels.

NOTE Table 2 does not include tests for some ambient conditions such as humidity and toxic-off gassing because they are performed exposing the hardware to the environment without margin.

d. For vibration, the random acceptance test spectrum shall not be lower than the minimum spectrum (see Table 2).

Table 2: Acceptance test levels and durations

Test	Levels		Duration	
	Equipment	Space element	Equipment	Space element
Shock	Maximum expected shock spectrum	N/A	1 shock in both directions of 3 axes + dwell and burst tests	1 activation of explosive firing
Acoustic	Envelope of maximum expected acoustic spectrum		2 min	1 min
Vibration	Random: Envelope of maximum and minimum expected spectrum [a]	Random: Envelope of maximum and minimum expected spectrum [a]	Random: 2 min per axis	Random: 1 min per axis
Thermal cycling	5 °C extension of maximum and minimum predicted temperatures [b]	Flight temperature [c]	4 cycles	4 cycles
Thermal vacuum	5 °C extension of maximum and minimum predicted temperatures [b]	Flight temperature [c]	4 cycles if combined with thermal cycling, 1 cycle if thermal cycling is performed	4 cycles if combined with thermal cycling, 1 cycle if thermal cycling is performed
Pressure	1,5	1	5 min (only one cycle)	Sufficient to establish leakage

[a] For random acceptance test spectrum see 4.8.1.3 d.
For the equipment, the minimum spectrum is the acceptance vibration test (AVT) spectrum having no relation with the expected mission and derived from experience on several projects.

[b] If the equipment acceptance is carried out for multi-project utilization standard spectra or temperature limits can be used.

[c] A suitable distribution of the flight delta temperature for equipment is used, i.e. temperature limit is reached as soon as one unit in a selected area is at the hot and cold temperature reached during the unit acceptance thermal testing.

4.8.1.4 Protoflight test levels and duration

a. The protoflight approach as defined in subclause 4.6 combines the qualification and acceptance tests into protoflight tests with modified test levels and durations.

b. The protoflight qualification test levels and durations shall be as follows:

1. protoflight test levels: as qualification levels (see Table 1);
2. protoflight test durations: as acceptance durations (see Table 2).

4.8.2 Test measurements

4.8.2.1 General

Measurement uncertainties should be in accordance with ANSI/ASME PTC 19.1.

4.8.2.2 Test tolerances

The test level tolerances specified in Table 3 shall be applied to the nominal test value specified.

NOTE The tolerances specified in Table 3 are the maximum range within which the test level may vary. They are exclusive of instrument accuracy from test parameter NO TAG to 12. From test parameter 13. to 16. they are coincident with the maximum measurement instrument accuracy.

4.8.2.3 Accuracy of test instrumentation

a. The accuracy of instruments used to control or monitor the test parameters shall be verified periodically and shall be compatible with the test objectives.

b. The accuracy of the instruments shall be consistent with the tolerance for the variable to be measured, as specified in Table 3, and should be at least one third of the tolerance itself.

c. All instrumentation to be used for testing shall be subjected to approved calibration procedures and shall be within the normal calibration period at the time of the test.

Table 3: Maximum allowable test tolerances

Test parameters	Tolerances	
a Temperature (°C)	High	Low
from -50 °C to +100 °C	$T_{Max\,-0}^{\,+3}$	$T_{Min\,-3}^{\,+0}$
below -50 ° C or above +100 °C	$T_{Max\,-0}^{\,+4}$	$T_{Min\,-4}^{\,+0}$
3. Relative humidity	±5 %	
4. Pressure (in vacuum chamber)		
> 1.3 hPa	±15 %	
1.3 10^{-3} hPa to 1.3 hPa	±30 %	
< 1.3 10^{-3} hPa	±80 %	
5. Acceleration	0 / +10 %	
6. Sinusoidal vibration		
Frequency	±2 % from 10 Hz to 2 000 Hz	
Amplitude	±10 %	
Sweep rate	±5 %	
7. Random vibration		
Frequency	±5 % (or 1 Hz whichever is greater)	
PSD from 20 Hz to 500 Hz (filter bandwidth 25 Hz or less) PSD from 500 Hz to 2 000 Hz (filter bandwidth 50 Hz or less)	-1/ +3 dB for Qualification and -3/+1.5 dB for Acceptance	
Random overall g r.m.s.	±10 %	
8. Acoustic noise		
Sound pressure level		
1/3 octave band (centre frequency)	±3,0 dB	
Overall	±1,5 dB	
9. Test time	0/+10 %	

Table 3: Maximum allowable test tolerances *(continued)*

Test parameters	Tolerances
10. Shock	
Response spectrum amplitude (1/6 octave centre frequency)	±6 dB (with 30 % of the response spectrum centre frequency amplitudes greater than nominal test specification)
Shock duration	
≤ 20 ms	0/+20 %
> 20 ms	0/+10 %
Shock level	0/+20 %
11. Solar flux	
Mean value	±3 % on reference plane(s)
Test volume	±1 %
12. Infrared flux	
Mean value	±3 % on reference plane(s)
13. Mass	
Equipment	$\pm 2 \times 10^{-2}$ kg or ± 0,1 % (whichever is greater)
Space vehicle, element or module	±0,3 kg
14. Centre of gravity (CoG)	
Equipment	Within a 1 mm radius sphere
Space vehicle, element or module	±2,5 mm along Z-axis
15. Moment of inertia (MoI)	
Equipment	±3 %
Space vehicle, element or module	±1 % each axis
16. Leak rate	$\pm 10^{-5}$ Pa m^3 s^{-1} of Helium at 1013 hPa pressure differential
17. Microgravity	
Quasi-static force or torque	±10 %
Dynamic forces	±20 %
Sound-power:	
1/3 octave band centre frequencies	
32,5 Hz - 160 Hz	±3 dB
160 Hz - 500 Hz	±2 dB
18. Audible noise	
32,5 Hz - 160 Hz	±3 dB
160 Hz - 15 kHz	±2 dB
19. Flow rate	±5 %
20. Strain	±10 %

4.9 Operations validation testing

4.9.1 Introduction

4.9.1.1

This subclause addresses the specific test philosophy applicable to the ground segment implementation and mission operations testing. In accordance with commonly accepted definitions (see ECSS-E-70), the "ground segment" or "operational entity" consists of:

a. organizational elements operating the space mission and consisting of staff, manuals, plans, mission rules, and procedures;

b. functional elements which form the ground segment (software or hardware) facilities used in the operations of the mission.

4.9.1.2

The objective of the ground segment and operations validation testing covers both of the element categories distinguished above and can be formulated as follows:

> To obtain assurance prior to launch that the integrated hardware and software facilities perform as intended and that the ground segment can accomplish all operational mission requirements within the constraints imposed by the operational environment.

4.9.1.3

a. The testing philosophy as outlined under subclause 4.1 should be applied to the ground segment.

b. Due to the specific characteristics of the ground segment rigid applicability of the space vehicle test principles should not be imposed. In particular, the ground segment equipment need not to be subjected to precisely the same qualification and test requirements as "flight" hardware since post-launch maintenance and replacement is possible. Furthermore, many essential space vehicle test requirements are not relevant for ground hardware (e.g. model philosophy and environmental testing).

4.9.1.4

Specific acceptance and validation testing requirements for ground segment elements shall be performed for the in-orbit elements when adequate availability and performance of the ground segment constitutes a critical condition for the fulfilment of the mission objectives.

4.9.2 Overview of ground segment facilities

The following list provides a high-level overview of the different kinds of ground segment support facilities to consider (see ECSS-E-70).

a. Mission control system, consisting of:

- Operations control system,
- Payload control system,
- Mission exploitation system (often under responsibility of a "user" organization);

  NOTE The mission control system comprises all monitoring, control and data processing software and hardware, the mission operations rules, plans and procedures, as well as the control centre control facilities and the relevant operations and maintenance support staff.

b. Ground station system;

NOTE The ground station system includes telemetry and telecommand subsystems as well as antenna, ranging, timing, and station monitor and control subsystems.

c. Communications network system.

NOTE The communications network system covers the data and voice communications lines and their connections to the ground stations and mission control system as well as data communications facilities with the users.

4.9.3 Ground segment subsystem acceptance tests

4.9.3.1
The objective of acceptance tests on ground segment subsystems is to confirm or to establish confidence in the performance of a particular individual subsystem (with its specific hardware or software function) at the conclusion of its development stage. This refers specifically to the "integrity" of the equipment with respect to its conformance to the applicable functional, performance, and interface requirements.

4.9.3.2
a. Test objectives and test procedures shall be documented and approved before the start of the tests.

b. Test results shall be documented in corresponding test reports immediately following the completion of the test.

4.9.3.3
a. Through acceptance testing of a subsystem, it shall be demonstrated and documented that the subsystem fulfils all specified functional and performance requirements.

b. Acceptance tests should include the confirmation of specific user requirements relating to the performance of the equipment within the intended operational environment.

4.9.3.4
Specific dedicated test facility (hardware or software) should be developed or procured along with the subsystem itself for verifying functional performances and interface requirements.

4.9.3.5
a. All new or modified hardware and software subsystems shall be subjected to rigorous acceptance testing before delivery.

b. Software functions associated with the hardware should be considered as an integral part of its overall functional capabilities.

4.9.3.6
Software acceptance tests shall be conducted by running and evaluating a number of specific software tasks as specified before and in cooperation with the software user.

4.9.4 Ground segment integration tests

4.9.4.1
The objective of ground segment integration test is to establish confidence that two or more assembled subsystems are capable of functioning together, i.e. satisfying all relevant interface requirements.

4.9.4.2
Integration testing shall be performed in a progressive step-by-step manner, building up the complete ground system out of elements which by themselves were previously integrated and validated in terms of their functional and interface performances.

4.9.4.3
The relevant interface characteristics with other parts of the ground segment should be tested with the help of software simulators before exercising the actual hardware interfaces.

4.9.5 Operations validation tests

4.9.5.1 Overview and background

4.9.5.1.1
Operations validation tests refer to the system-level ground segment validation.

The objective of operations validation tests is to establish the correct functioning of a substantial portion of the (partly or wholly) integrated ground segment and is summarized as follows:

a. To validate the correctness of the operations control and monitoring software in relation to the space vehicle design and performance requirements;
b. To validate the interface integrity and robustness of the integrated hardware and software subsystems within realistic nominal and back-up operations scenarios;
c. To validate the procedural contents of the flight operation plan (FOP) at system level. Therefore, it is important that these tests are conducted as far as practical in accordance with the control procedures and timelines outlined in the FOP;
d. To confirm the functional and performance integrity of the relevant ground stations within the relevant operational context;
e. To validate the integrity of the data communications interfaces between the control centre and the ground stations (i.e. by means of data flow tests coupled with functional and performance tests).

NOTE Table 4 provides examples of ground segment and operations validation test activities.

4.9.5.1.2
Operations validation tests shall be conducted within a realistic operational environment in order to demonstrate the ability to meet specific operational conditions, for instance imposed by time critical operations sequences.

NOTE When new subsystem facilities are involved, these tests can often require a number of iterations or “restarts”.

4.9.5.1.3
To achieve the system-level test objectives the categories of tests distinguished in 4.9.3 and 4.9.4 shall be conducted in a sequential order.

4.9.5.1.4
Subsystem acceptance tests shall be completed satisfactorily before initiating their successive integration; similarly, operations validation tests shall be started only after successful integration testing of significant parts of the ground segment.

Table 4: Examples of ground segment validation tests

Mission control system	Monitoring and control system validation
	Validation of deliverable data products and services
	Validation of operations plans and procedures
	Validation of timeline and mission rules
	Validation of flight dynamics capabilities
Ground station facilities	RF compatibility tests
	TM and TC data interface tests
	Station computer functional tests
	Ranging and antenna pointing data interface tests
Communications network	TM and TC data flow tests
	Ground station system operations tests
	Ranging and antenna pointing operations tests
NOTE: Additional tests are given in subclauses 4.9.5.2 to 4.9.5.4.	

4.9.5.2 RF compatibility tests

4.9.5.2.1
The objective of the RF compatibility test is to establish confidence that the uplink and downlink characteristics of the space vehicle and of the ground stations, to be used during the launch and early orbit phase (LEOP) and operational phase of the mission, are compatible in terms of their RF characteristics.

4.9.5.2.2
A series of standard tests should be performed using the ground station (or a representative model) and a satellite “suitcase” model consisting of the space vehicle telecommunications and data handling subsystems.

NOTE Sometimes, the actual space vehicle is used.

4.9.5.3 Space vehicle to ground segment compatibility tests

4.9.5.3.1
The objective of the space vehicle to ground segment compatibility tests is to validate the monitor and control system with regard to its TM and TC interactions with the space vehicle.

4.9.5.3.2
a. Space vehicle to ground segment compatibility tests shall be performed directly with the actual satellite linked to the control centre via a data communications network for TM, TC and voice.

b. A network data interface unit (NDIU) shall be attached to the space vehicle for performing the ground station TM and TC data processing functions.

NOTE “Space vehicle compatibility tests” (which aim at validating the “compatibility” between the actual satellite and the mission control system) is sometimes referred to as “system validation tests”. This term is discouraged since these tests cover only one specific (although extremely important) interface verification and not a complete ground segment system-level validation.

4.9.5.3.3

a. Prior to the compatibility tests with the space vehicle, the control system should be validated with the aid of a dedicated software simulator.

b. Actual telemetry tapes generated during satellite check-out tests can be used.

4.9.5.3.4

Two compatibility tests should be performed with the satellite typically at about 6 and 3 months before launch and each with a duration of one to two weeks as follows:

a. During these tests, all space vehicle operating modes and their reconfigurations should be exercised in realistic operations sequences.

b. The first compatibility test shall focus on software and database validation activities that includes all monitor and control software facilities.

c. The second compatibility test shall

1. (re)validate outstanding software capabilities, and

2. exercise and validate critical FOP (flight operation plan) operations sequences with the actual space vehicle.

4.9.5.4 Simulations programme

4.9.5.4.1

The objectives of simulation activities is to ultimately validate the mission rules, the timelines, and operations procedures contained in the flight operation plan.

4.9.5.4.2

a. Simulations shall be performed within the actual operations environment containing the integrated mission control system, the ground stations and communications network as well as the relevant control and support teams.

b. A software simulator shall be used for generating the telemetry data and for handling telecommand.

4.9.5.4.3

a. Simulations shall be designed to familiarize the flight control teams and support personnel in the control centre and at the ground stations with the relevant mission operations activities as defined in the operations plan.

b. The monitor and control activities for the relevant flight sequences shall be exercised and validated under realistic conditions as imposed by the actual timeline and actual organizational characteristics.

c. Both nominal and contingency operations procedures shall be exercised in the simulations programme.

4.9.5.4.4

a. The flight dynamics team and the space vehicle engineering support team should be considered as an integral part of the mission control team.

NOTE The number of teams involved in simulations gradually increases as the simulation programme progresses.

b. The specific interfaces between the various teams should be exercised in realistic conditions during a number of individual simulations sessions in order that problems are minimized during the actual operations.

4.9.6 Ground segment mission readiness test

4.9.6.1

The objective of the ground segment mission readiness test is to confirm the operational readiness status of the ground segment, in particular the control system and the communications network in the relevant nominal and back-up modes.

4.9.6.2

Ground segment mission readiness should cover extended data flow tests with additional processing, display requirements or specific on-station performance checks.

4.10 Test data

a. For an evaluation of the product performance under the various specified test conditions, test measurements and the environmental conditions on the products shall be recorded.

b. These records shall be used for post-test analysis to supplement the real-time monitoring and to facilitate the accumulation of trend data for the critical test parameters.

c. The test data should be compared across major test sequences for trends or evidence of anomalous behaviour.

d. Test data shall be examined for possible anomalies. For details of the nonconformance control procedure see ECSS-Q-20-09.

e. Test data can be used as inputs to analysis (to mathematical models) for verification closure.

4.11 Test documentation

4.11.1 Introduction

a. Test standard documentation shall consist of the following DRDs:

1. Test specification (see ECSS-E-10-02A, annex F),
2. Test procedure (see ECSS-E-10-02A, annex G),
3. Test report (see ECSS-E-10-02A, annex H).

b. The documentation should be generated at all product levels as part of the verification documentation.

4.11.2 Test specification

4.11.2.1

The objectives of the test specification is to define the test requirements to be implement to demonstrate the item performance. Test specification may be combined with AIV plan and test procedure (see ECSS-E-10-02A, annex F.1).

4.11.2.2

a. The test specification shall be prepared for each major test activity described in the test plan activity sheets with the objective to detail test requirements.

b. The test specifications shall contain as minimum, the

1. activity objectives,
2. selected approach,

3. article configuration,
4. set-up description,
5. GSE to use,
6. equipment and instrumentation,
7. conditions for the activity,
8. facilities to use,
9. sequence of activities with the detailed requirements,
10. pass-fail criteria,
11. organization and responsibilities,
12. involved documentation,
13. relationship with product assurance activities, and
14. schedule.

NOTE Parts of the test specification can be contained in separate documents (e.g. test configuration and measurement plan).

4.11.3 Test procedure

The test procedure shall be written in conformance with the test specification and specify the following information, as a minimum:

a. Objective of the test including identification of the requirements to be verified.

b. Identification and configuration of the test article, including:
   1. applicable documentation (e.g. CIDL, ABCL and TCL),
   2. deviation from specified requirement baseline, and
   3. equipment reference axes.

c. Test set-up identification, including:
   1. test facility and data handling,
   2. ground support equipment and manuals,
   3. instrumentation (including accuracy and calibration data), and
   4. adapters and interfaces.

d. Test conditions (reference to test specification), including:
   1. test levels and duration,
   2. tolerances, and
   3. data acquisition and reduction.

e. Step-by- step instructions for operation, including:
   1. test preparation,
   2. test performance (step-by-step procedure),
   3. pass-fail criteria, and
   4. past test activities (relationships).

f. Safety and security instructions.

g. Personnel required and responsibilities.

h. Criteria and rules for deviations.

4.11.4 Test report

The test reports shall describe the execution and the results of the test activity and contain:

a. The as-run test procedure including any deviations and failures (NCR).

b. Facility data including environmental data and conditions.

c. Test article data including measurements made on the item under test.

d. Evaluation of the test data.

e. Summary of the test evaluation and conclusion.

5

Qualification testing

5.1 Equipment test requirements

5.1.1 Introduction

The requirements provided in this clause are of two types:

a. General

The general qualification philosophy provides requirements for multi-project utilization (i.e. different launchers and different missions).

With this philosophy the equipment is qualified independently from the specific project, assuring the compatibility with the requirements of a specified set of launchers (ARIANE 4 and 5, and STS) and with a wide range of mission types.

The general philosophy makes use of standard spectra for structural tests and of standard temperature limits for each category of equipment.

b. Specific

The specific qualification philosophy provides requirements tailored for each specific project.

With this approach the equipment qualification is limited to the requirements of a particular project (e.g. launcher mission).

The philosophy specifies the criteria for determination of test levels in terms of margins to be applied. In principle, the test durations shall be the same as specified in the general approach.

5.1.2 Equipment classification

5.1.2.1

Equipment shall be classified according to the following categories:

a. Category A:
Off-the-shelf equipment with no modifications and subjected to a qualification test programme for space applications at least as severe as that imposed by the actual project specifications.

Further qualification testing need not be performed.

b. Category B:
Off-the-shelf equipment with no modifications that was already tested and

qualified but subjected to a different qualification programme or to a different environment.

Delta qualification test programme shall be decided and performed on a case-by-case basis.

c. Category C:
Off-the-shelf equipment with minor design modifications.

Delta or full qualification test programme shall be decided case-by-case depending on the impact of the modification.

d. Category D:
Completely new equipment designed and developed or existing equipment with major redesign.

A full qualification test programme shall be performed.

5.1.2.2

a. The type and the extent of the test programmes to be performed for each category depend also on the project model philosophy. The hardware matrix shall identify, for each equipment, the related qualification status and models.

b. The equipment qualification test baseline shall consist of the tests specified in Table 5.

NOTE The categorization of tests into "required", "optional" and "not required" is guided by the sensitivity of the type of equipment to the specific environment and by the applicability of the environment.

c. Where equipment falls into two or more types of Table 5 the required tests specified for each type shall be applied.

EXAMPLE A star sensor can be considered to fit both "electronic equipment" and "optical equipment" types, therefore, an EMC test is conducted since it is applicable for electronic equipment, even though there is no requirement for optical equipment.

5.1.3 Rules for test programme

5.1.3.1

The following rules shall be taken into account in establishing the test programme:

a. Equipment qualification tests should be accomplished entirely at the equipment level.

NOTE In some circumstances equipment qualification may be conducted partially or entirely at a higher level (e.g. interconnecting tubes, radio frequency circuits and wiring harness)

b. Leak tests shall be performed only on sealed or pressurized equipment, where equipment is sensitive to loss of pressure or vacuum.

c. If the equipment is not fully environmentally protected on the ground and in flight, then a humidity qualification test shall be performed.

NOTE For equipment which is environmentally protected , the humidity qualification test is optional.

d. A humidity qualification test should be performed for tests such as fungus, sand, dust, salt spray, and radiation.

e. Equipment qualification life tests shall be optional unless a reliability analysis has shown that such tests are essential to demonstrate that the equipment can withstand the maximum duration or cycles of operation without fatigue or wear-out failures.

f. Functional and performance tests shall be performed prior and following environmental tests.

g. Equipment functional and performance tests shall be performed at the beginning, the end and at specific times during the test sequence to check that equipment has survived a certain test environment.

h. Equipment functional tests shall be performed while the environment is being imposed, if the equipment is expected to be (fully) operational under that environment.

NOTE Many defects, which otherwise escape detection by pre- and post test functional checks, reveal themselves during environmental tests.

i. The performance and functional tests should be designed so that a database of critical parameters can be established for trend analysis.

j. The database of critical parameters shall be established by measuring the same critical parameters in all the functional tests conducted before, during and after each of the baseline environmental tests.

k. Any unusual or unexpected trends shall be evaluated to determine the existence of any trends towards an out of limit value or of an incipient failure.

l. The EGSE or other support systems of the test article shall:

1. not influence the results of EMC tests,
2. be designed for low emissions, and
3. be hardened against susceptibility testing.

5.1.3.2

a. The qualification test sequence is given in Figure 1 should be followed.

NOTE The sequencing in Figure 1 is based on a combination of the order in which the environments are encountered during flight and the purpose to perceive defects as early in the test sequence as possible.

b. The sinusoidal and random vibration should be performed at the same time per axis.

Table 5: Equipment qualification test baseline

Test	Reference subclause	Recommended sequence	Category/type of equipment											
			a	b	c	d	e	f	g	h	i	j	k	l
Physical properties	5.1.4	1	R	R	R	R	R	R	R	R	R	R	R	R
Functional and performance	5.1.5	2[1]	R	R	R	R	R	R	R	R	R	-	R	R
Humidity	5.1.6	3	O	O	O	O	O	O	O	O	O	O	O	O
Leak	5.1.7	4,6,11,14[7]	R[2]	-	R[2]	R	R	R	O	O	-	-	-	-
Pressure	5.1.8	5[7]	R[2]	-	R[2]	R	R	R	R	-	-	-	-	-
Acceleration	5.1.9	7	O	R[10]	O	O	-	O	-	-	R[10]	-	-	O
Sinusoidal vibration	5.1.10	8	R	R	R	R	R	R	R	R	R	R	R	-
Random vibration	5.1.11	9	R	R[3]	R	R	R	R	R	R	R	R	R	-
Acoustic	5.1.12	9	-	R[3]	-	O	-	-	-	-	-	O	O	R
Shock	5.1.13	10	R[10]	O	O	O	O	-	O	O	O	-	-	O
Corona and arcing	5.1.14	12	R[5]	R[5]	O	O	O	-	-	-	-	-	-	-
Thermal vacuum[9]	5.1.15	13[6]	R	R	R	R	R	O	R	R	R	O	R	R
Thermal cycling[11]	5.1.16	13[6]	R	R	R	R	R	O	R	R	R	O	R	O
EMC/ESD	5.1.17	15	R	O	-	R	O	-	O	-	-	-	-	-
Life	5.1.18	16	O	O	O	O	O	O	O	O	O	O	O	O
Microgravity[4]	5.1.19	17	R	R	-	R	R	-	R	-	R	R	R	-
Audible noise[8]	5.1.20	18	R	R	-	R	R	-	R	-	-	R	R	-

Categories / Types of Equipment

a = Electronic and electrical equipment

b = Antennas

c = Batteries

d = Valves

e = Fluid or propulsion equipment

f = Pressure vessels

g = Thrusters

h = thermal equipment

i = Optical equipment

j = Mechanical equipment

k = Mechanical moving assemblies

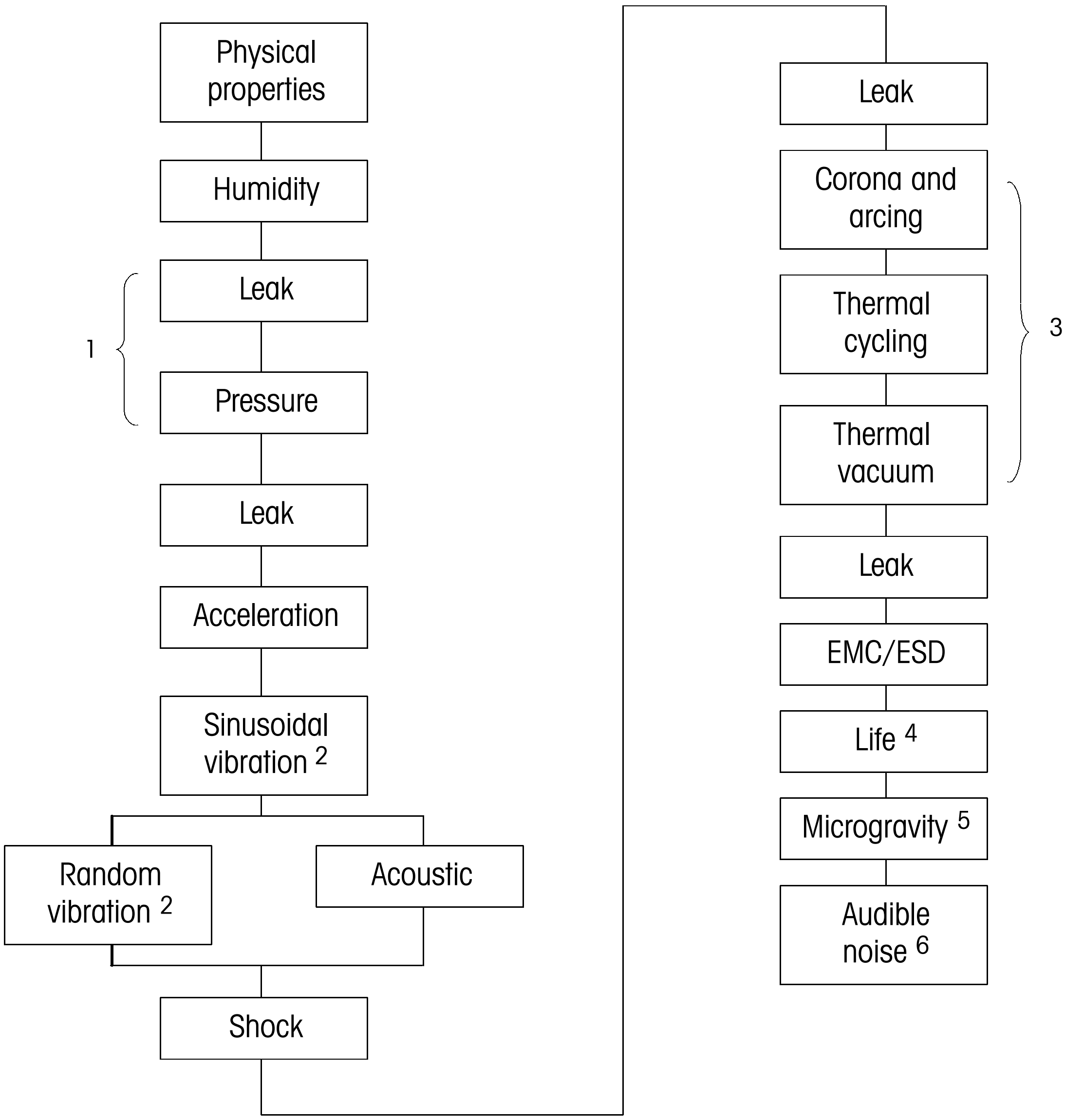
l = Solar arrays

Key

R = Required O = Optional - = Not required

Notes

1. See 5.1.2.2 f.
2. Required only on sealed or pressurized equipment.
3. Either random vibration or acoustic test required with other optional.
4. Mission dedicated for in-orbit active mechanical assemblies.
5. To be performed for equipment which is switched on during ascent phase only.
6. Can be combined in thermal vacuum cycling test.
7. Can be combined.
8. Required for equipment for manned space vehicle.
9. Required for equipment exposed to vacuum and sensitive to it.
10. Required for equipment sensitive to the environment and located in zones where the environment is critical.
11. Including temperature test.



Notes

1 These tests may be combined.

2 See 5.1.3.2 b.

3 These tests may be combined.

4 Performed for completeness. It can be performed on separate equipment.

5 Equipment for microgravity utilization space vehicle.

6 To be performed for manned space vehicle.

Figure 1: Equipment qualification test sequence

5.1.4 Physical properties measurements, equipment qualification

5.1.4.1 Purpose

The purpose of physical properties measurements is to determine the equipment physical characteristics, i.e. dimensions, mass, centre of gravity and momentum of inertia.

5.1.4.2 Test description

a. Dimensions

1. The characteristic dimensions of the equipment (height, width, and length) shall be measured.
2. The interface dimensions shall be measured.

b. Mass

The mass of the equipment shall be measured.

c. Centre of gravity

1. The position of the centre of gravity shall be determined with respect to a given co-ordinate system for three mutually perpendicular axes.
2. This measurement should be performed by means of standard laboratory instrumentation.

d. Momentum of inertia

1. The momentum of inertia shall be measured with respect to the given co-ordinate system.
2. A torsional pendulum and standard laboratory instrumentation should be used.

5.1.4.3 Supplementary requirements

a. Alternative methods can be used, provided that they assure the required accuracy for the measurements.

b. The test article shall be in flight configuration. Any non-flight items, which cannot be removed, shall be subtracted by calculation.

c. For equipment with simple shapes, the centre of gravity location and momenta of inertia can be determined by calculation.

5.1.5 Functional and performance test, equipment qualification

5.1.5.1 Purpose

The purpose of the functional and performance test is to verify that the electrical and mechanical functions and the performance of the equipment conform to the equipment specification.

5.1.5.2 Functional and performance test basic requirements

a. Electrical tests shall include application of expected voltages, impedance, frequencies, pulses, and wave forms at the electrical interface of the equipment, including all redundant circuits.

b. Mechanical tests

1. Mechanical tests shall include application of torque, load and motion as appropriate.
2. Test parameters shall be varied throughout their specification ranges and the sequences expected in flight operation.

3. The equipment output shall be measured to verify the equipment performance to specification requirements.

c. Functional performance shall include electrical continuity, stability, response time, alignment, pressure, leakage or other special functional tests related to a particular equipment configuration.

d. Functional and performance tests shall be executed during test exposures if the equipment is operational during its mission.

5.1.5.3 Functional and performance test supplementary requirements

a. Complete functional and performance tests shall be performed at the beginning and at the end of the test sequence.

b. Reduced functional and performance tests shall be conducted before and after each environmental exposure as follows:.

1. The test shall be designed in order to provide reliable criteria for judging successful survival of the equipment to the test environment.
2. A database of critical parameters shall be established for trend analysis.
3. Unusual or unexpected trends should be evaluated to determine the existence of any drift towards an out of limit value or the start of a failure.

c. Functional tests shall be performed while the environment is being imposed, if the equipment is expected to be fully operating under that environment (exceptions solar panels and antenna feeds during thermal vacuum tests).

d. The benefit to equipment (functionally operated and monitored during the environmental tests, regardless of the functional mode of the equipment during launch and ascent), shall be evaluated to increase the overall test effectiveness, without jeopardizing equipment integrity.

NOTE Many defects, which otherwise escape detection by pre- and post-test functional checks, reveal themselves during environmental tests e.g. intermittence can be caused by inadequate clearances, cracks, debonds and damaged connectors that can only be revealed during environmental tests (e.g. relay change over during vibration).

5.1.6 Humidity test, equipment qualification

5.1.6.1 Purpose

The purpose of this test is to demonstrate the ability of the equipment to withstand the humid environment that can be imposed upon the equipment during its life on ground.

5.1.6.2 General

a. In cases where exposure is controlled during the life cycle to standard ambient conditions, verification by test shall not be performed.

b. Since some items can experience higher levels of humidity in manned flights, the humidity test shall be performed if the equipment is sensitive to humidity.

5.1.6.3 Humidity test description and levels

a. Pre-test conditions

1. The equipment shall be installed in the humidity chamber.
2. Chamber temperature shall be at room-ambient conditions.

b. Cycle 1

1. The temperature shall be increased to +35 °C over a 1-hour period;
2. The relative humidity shall be increased to not less than 95 % over 1-hour period with the temperature maintained at +35 °C.

3. The conditions specified above shall be maintained for 2 hours.
4. The temperature shall be reduced to 2 °C over a 2-hour period with the relative humidity stabilized at not less than 95 %.
5. The conditions specified above shall be maintained for 2 hours.

c. Cycle 2
   1. Cycle 1 shall be repeated with the exception that the temperature shall be increased from +2 °C to +35 °C over a 2-hour period
   2. Moisture shall be not added to the chamber until +35 °C is reached.

d. Cycle 3
   1. The chamber temperature shall be increased to +35 °C over a 2-hour period without adding any moisture to the chamber.
   2. The test equipment shall be dried with air at room temperature and 50 % maximum relative humidity by blowing air through the chamber for 6 hours.
   3. The volume of air used per minute shall be between one and three times the volume of the test chamber.
   4. A suitable container can be used in place of the test chamber for drying the test equipment.

e. Cycle 4
   1. The equipment shall be placed in the test chamber and the temperature increased to +35 °C and the relative humidity increased to 90 % over a 1-hour period.
   2. The conditions above shall be maintained for at least 1 hour.
   3. The temperature shall be reduced to +2 °C over a 1-hour period with the relative humidity stabilized at 90 %
   4. The conditions above shall be maintained for at least 1 hour.
   5. A drying cycle should follow (see Cycle 3).

5.1.6.4 Humidity test supplementary requirements

a. The equipment shall be checked prior to the test and at the end of Cycle 3 (within 2 hours after the drying) and visually inspected for deterioration or damage.

b. The equipment shall be tested during the Cycle 4 period of stability (i.e. following the 1-hour period after reaching +35 °C and 90 % relative humidity conditions).

c. The equipment shall be visually inspected for deterioration or damage after removal from the chamber.

5.1.7 Leakage test, equipment qualification

5.1.7.1 Purpose

The purpose of leakage test is to demonstrate the ability of pressurized equipment to meet the design leakage rate constraints specified in the equipment specification.

5.1.7.2 Leakage test description

5.1.7.2.1 General

a. Equipment leak checks shall be made prior to initiation of, and following the completion of equipment qualification thermal and structural tests.

b. Proof pressure tests defined in subclause 5.1.8.2 shall be successfully completed before conducting leakage tests.

c. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

5.1.7.2.2 Method

One of the following methods shall be used:

a. Method I (gross leak test)

1. The equipment shall be completely immersed in a friendly and non-corrosive liquid, so that the uppermost part of the test item is 5 cm ±2,5 cm below the surface of the liquid.

   NOTE Any observed leakage during immersion as evidenced by a continuous stream of bubbles emanating from the equipment indicates a failure of seals.

2. The critical side or side of interest of the component shall be in a horizontal plane facing up.

3. The liquid, pressurizing gas, and the test items shall be 23 °C ±10 °C.

4. The gas used for pressurizing shall be clean and dry with a dew-point of at least -32 °C.

b. Method II (fine leak test)

1. This method shall be applied to fluid loop components and similar equipment.

2. The equipment shall be purged with nitrogen and then charged with helium to the pressure specified in the equipment specification before being sealed.

3. The equipment shall be placed in a suitable vacuum chamber or bag or container and tested for helium leakage with a helium leak detector.

4. The leakage rate shall be used to determine seal integrity and shall not exceed the amount specified in the equipment specification.

c. Method III (for battery cases or pressurized equipment)

1. The equipment shall be pressurized with dry nitrogen or other appropriate gas to the specified value.

2. The pressure shall be monitored by a gauge (or pressure transducer) for the required time.

3. The drop in pressure shall not exceed the permitted amount as specified in the equipment specification.

d. Method IV (for hermetically sealed alkaline storage batteries)

1. The battery shall be cleaned with alcohol while in the discharged state.

2. A suitable indicator (e.g. dilute solution of phenolphthalein or other suitable colour change indicator) shall be applied to all seals, terminals and pinch tubes subject to leakage of electrolyte

   NOTE A change in the colour of the indicator is an indication of a leak.

3. After testing, the test solution shall be removed (e.g. with distiled water).

e. Method V (for equipment of pressurized fluid systems)

1. The equipment shall be pressurized to their maximum design pressure in each of the functional modes.

2. Leakage shall be detected using the standard method as described in Method II.

3. All pressurized fluid equipment shall be evacuated to the internal pressure normally used for propellant loading and the internal pressure monitored for indications of leaking.

f. Method VI (for pressurized modules or compartments)

1. Full scale pressurized modules or compartments shall be subjected to pressure decay tests for verification of total leak rate requirements under limit pressure.
2. This leak test shall verify the integrity of the pressure shell and all seals and measure the total leak rate of the assembled module over a sufficient time to confirm the long duration "keep pressure" capability.

5.1.7.3 Leakage test levels and duration

5.1.7.3.1 General

a. The leak tests shall be performed with the equipment pressurized at the maximum operating pressure and then at the minimum operating pressure if the seals are dependent upon pressure for proper sealing.

b. The test duration shall be sufficient to detect any significant leakage.

5.1.7.3.2 Test levels and durations for specific methods

The test levels and duration for the typical methods of subclause 5.1.7.2 are as follows.

a. Method I

The duration of immersion shall be 60 minutes at each pressure.

b. Method II

The external test pressure shall be 10^{-3} hPa or less and the duration of the test shall be at least 2 hours (for equipment that is operational in orbit for more than one day).

c. Method III

1. The test pressure should be less than 3 430 hPa.
2. The pressure drop shall not exceed the specified amount (typically about 69 hPa in a 6 hour period at room temperature).

d. Method IV

The test results should be visible within seconds.

e. Method V

The duration of the evacuated propulsion system equipment leak test shall not exceed the time that this condition is normally experienced during propellant loading.

f. Method VI

The duration shall be 10 hours minimum.

5.1.8 Pressure test, equipment qualification

5.1.8.1 Purpose

The purpose of the pressure test is to demonstrate that the design and structural integrity of such items as pressure vessels, pressure lines, fittings and valves provide an adequate margin such that structural failure or excessive deformations do not occur at the maximum expected operating pressure.

5.1.8.2 Pressure test description

The following test shall be performed as part of a fracture control programme (see ECSS-E-30-01).

a. Proof pressure

1. For pressure vessels, pressure lines and fittings, the temperature of the equipment shall be consistent with the operational temperature and subjected to a minimum of one cycle of proof pressure.
2. 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 initial condition.

NOTE Any evidence of permanent deformation, distortion or failure of any kind indicates a failure to pass the test.

b. Proof pressure for valves

1. With the valve in open and closed position, the proof pressure shall be applied for a minimum of three cycles to the inlet port for 5 minutes (hydrostatically or pneumatically).
2. Following the 5 minutes pressurization period, the inlet pressure shall be reduced to ambient conditions.
3. The exterior of the unit shall be visually examined.

NOTE 1 Evidence of deformation or any failure indicates a failure to pass the test.

NOTE 2 The test may be conducted at room ambient temperature.

c. Burst pressure for pressure vessels, pressure lines and fittings

1. The temperature of the equipment shall be consistent with the operational temperature.
2. The equipment shall be pressurized (hydrostatically or pneumatically) to design burst pressure.
3. The internal pressure shall be applied at a uniform rate such that stresses are not imposed due to shock loading.

NOTE Any evidence of rupture when tested at design burst pressure indicates a failure to pass the test.

d. Burst pressure for valves

1. With the valve in the open and closed position, the design burst pressure shall be applied to the inlet port for 5 minutes (hydrostatically or pneumatically).
2. Following the 5 minutes pressurization period, the inlet pressure shall be reduced to ambient conditions.
3. The exterior of the unit shall be visually examined for indications of deformation or failure.

NOTE 1 Any evidence of rupture of the equipment indicates a failure to pass the test.

NOTE 2 The test may be conducted at room ambient temperature.

e. Collapse pressure

1. Collapse pressure test should be performed for equipment that experience internal vacuum in ambient pressure environment (e.g. all fluid loops and components during filling).
2. The equipment shall be installed in a hermetic sealed chamber.

3. After evacuation, the chamber shall be pressurized by helium.
4. Test endurance (>120 s) and leakage rate shall be recorded.

5.1.8.3 Pressure test levels

a. Temperature

The temperature of the pressure test is specified in the pressure 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 resistance.

b. Proof pressure

The proof pressure should equal 1,5 times the maximum operating pressure for vessels and for other equipment.

c. Burst pressure

The burst pressure should be equal to twice the maximum operating pressure for vessels and for other equipment.

5.1.8.4 Pressure test supplementary requirements

a. The equipment shall withstand proof pressure without leakage or detrimental deformation.

b. After burst pressure, no equipment or any of its parts shall be used for further qualification activities or as flight hardware.

c. Applicable safety standards shall be followed in conducting all tests.

5.1.9 Constant acceleration test, equipment qualification

5.1.9.1 Purpose

The purpose of the constant acceleration test is to determine the capability of equipment to avoid structural damage, and to operate correctly when it is subjected to constant accelerations generated during the various phases of the vehicle flight.

NOTE The constant acceleration test is intended for equipment sensitive to static acceleration.

5.1.9.2 General

A test method is described in 5.1.9.3, however, other test methods can be used, providing that they satisfy the conditions specified in subclause 5.1.9.4.

5.1.9.3 Constant acceleration test description

a. The equipment should be mounted to a test fixture through its normal mounting points, protecting the test article by a shroud if necessary.

b. The equipment should be tested in each of three mutually perpendicular axes.

NOTE The specified accelerations apply to the centre of gravity of the test item.

c. When a centrifuge is used, the length of the arm (measured to the geometric centre of the test item) should be at least five times the dimension of the test item measured along the arm.

NOTE Inertial equipment, such as gyros and platforms, can use counter rotating fixtures on the centrifuge arm.

d. When the dimensions of the equipment are too large with regard to the arm length, the test programme should specify which sensitive points are subjected to the specified acceleration within ±10 %.

e. During each test, the revolving arm is started progressively and its rotation accelerated uniformly so as to reach, within a period of time (which should not be less than 15 s), the speed corresponding to the proof acceleration for the tested equipment.

f. This acceleration should be maintained during the application time shown in subclause 5.1.9.4, and then the speed returned progressively to zero within a period of time which should also not be less than 15 s.

5.1.9.4 Constant acceleration test levels and duration

a. General

The test levels along each axis in both directions shall be:

Acceleration: 7,5 g Duration: 3 min

b. Specific

The test level shall be those expected in flight increased by the qualification margin.

5.1.9.5 Constant acceleration supplementary requirements

a. A functional test (see subclause 5.1.5) should be conducted before and after the acceleration test.

b. Detailed visual checks on should be carried out where functional tests are not performed (e.g. reflectors and solar panels).

c. Electrical equipment, if operated during ascent and descent, shall be powered during the test and parameters monitored to detect intermittent or persistent failures during the test, for each axis.

5.1.10 Sinusoidal vibration test, equipment qualification

5.1.10.1 Purpose

The purpose of sinusoidal vibration testing is to demonstrate the ability of the equipment to withstand low frequency excitations of the launcher increased by a qualification factor.

5.1.10.2 Sinusoidal vibration test description

a. The equipment shall be hard-mounted to a fixture through its normal mounting points.

b. The equipment shall be tested in each of three mutually perpendicular axes.

c. Any significant resonant frequencies (with amplification factor $Q > 3$) shall be noted and recorded.

d. The induced cross axis accelerations at the attachment points should be limited to the maximum test levels specified for the cross axis.

e. For induced cross axis acceleration above the specification, levels shall be agreed with the contracting authority.

f. Any pressurized equipment to be tested (e.g. if pressurized during ascent) shall be pressurized at the maximum operating pressure and monitored for internal pressure decay.

5.1.10.3 Sinusoidal vibration test levels and duration

5.1.10.3.1 Resonance search

a. Low level sinusoidal vibration test shall be performed to determine resonance frequencies to evaluate the item integrity.

b. Resonance search shall be carried out before and after vibration test sequence of each axis as specified in Table 6.

Table 6: Resonance search test levels

frequency	level	sweep rate
(5 – 2 000) Hz	0,5 g	2 octave per minute

5.1.10.3.2 Sinusoidal qualification

a. Sinusoidal tests shall be conducted for all axes, 1 sweep-up (and down if required), 2 octaves per minute.

b. The levels are hereafter defined according to the qualification methodology:

1. General

The test level shall be the envelope of the maximum expected spectra (for the different launchers: ARIANE 4 and 5, and STS) increased by the qualification margin (see Table 1).

In particular:

(a) For equipment with first frequency > 100 Hz and mass ≤ 50 kg the test levels are specified in Table 7 (see justification and complementary information in annex A.2).

Table 7: Sinusoidal qualification test levels for equipment with first frequency > 100 Hz and mass ≤ 50 kg

Frequency	Level	Remark
(5 – 21) Hz	11 mm (0 – peak)	no notching
(21 – 60) Hz	20 g (0 – peak)	
(60 – 100) Hz	6 g (0 – peak)	

(b) For equipment with first frequency > 100 Hz and mass > 50 kg the test levels are specified in Table 8.

Table 8: Sinusoidal qualification test levels for equipment with first frequency > 100 Hz and mass > 50 kg

Frequency	Level	Remark
(5 – 15) Hz	11 mm (0 – peak)	
(15 – 50) Hz	10 g (0 – peak)	no notching
(50 – 80) Hz	6 g (0 – peak)	
(80 – 100) Hz	3,5 g (0 – peak)	

(c) For equipment with first frequency ≤100 Hz and mass ≤50 kg.

* The same levels and duration as (a) above with notching, and
* response at critical locations and near the centre of gravity (CoG) of equipment shall be limited to acceleration level definition, see subclause 5.1.9.4.

(d) Equipment with first frequency ≤100 Hz and mass >50 kg.

* Same levels and duration as (b) above, with notching, and
* response at critical locations and near the centre of gravity (CoG) of equipment shall be limited to acceleration level definition, see subclause 5.1.9.4.

2. Specific

The test level shall be the expected flight level increased by the qualification margin.

5.1.10.4 Sinusoidal vibration supplementary requirements

a. A functional test should be conducted before the sinusoidal vibration and after its completion (or after random vibration, if performed).

b. Detailed visual checks shall be carried out where functional tests are not performed (e.g. reflectors and solar panels).

c. Equipment powered during launch shall be powered during the test. Several perceptive parameters shall be monitored to detect intermittent or persistent failures during the test, for each axis.

d. If equipment is mounted on dynamic isolators in the space vehicle, the equipment shall be mounted on these isolators during the qualification tests.

e. An intermediate level should be performed for fragile equipment before applying the qualification levels.

f. The test item shall be attached to the vibrator exciter table using a rigid test fixture capable of transmitting the vibration conditions specified.

g. The test fixture shall be designed to minimize fixture response at resonance within the test frequency range.

h. The variation of transmissibility between test item mounting points shall not exceed a factor of 3 dB between 5 Hz and 100 Hz.

i. A pre-test of the empty fixture shall be performed to verify the correct dynamic behaviour of the fixture and the proper function of the control loop with sufficient levels and durations to provide diagnostic capability.

j. If the fixture resonance characteristics prevent a single point from being representative of the input, a multi-point control system shall be performed and the average or the r.m.s. of all control points used.

NOTE In special cases (e.g. sensitive equipment and high dispersion of control values), the control may be performed on the peak value.

5.1.11 Random vibration test, equipment qualification

5.1.11.1 Purpose

The purpose of random vibration testing is to demonstrate the ability of the equipment to withstand the random excitation produced by the launcher, increased by a qualification margin, and to the transmitted acoustic noise excitation.

5.1.11.2 Random vibration test description

a. The equipment shall be mounted to a rigid fixture through its normal mounting points and tested in each of three mutually perpendicular axes.

b. Any pressurized equipment to be tested (e.g. if pressurized during ascent) shall be pressurized at the maximum operating pressure and monitored for internal pressure decay.

c. Resonance search shall be carried out before and after the vibration test for each axis as specified in table 9:

Table 9: Random vibration test levels

Frequency	Level	Sweep rate
(5 – 2 000) Hz	0,5 g	2 octave per minute

5.1.11.3 Random vibration test levels and duration

The levels for random vibration testing are defined according to the qualification philosophies:

a. General

The test level shall be the envelope of the maximum expected spectra (for the different launchers: ARIANE 4 and 5, and STS) increased by the qualification margin (see Table 1).

In particular:

1. Equipment with mass M ≤50 kg as specified in table 10 (see additional information in annex A.2). All axis without notching and any deviations shall be agreed by the contracting authority.

Table 10: Random vibration test levels and duration for equipment with mass M ≤ 50kg

Location	Duration	Levels	
Equipment located on "external panel [a] or with unknown location	Vertical [b] 2,5 min/axis	(20 – 100) Hz	+3 dB/octave
		(100 – 300) Hz	PSD(M) [c] = 0,12 g^2/Hz × (M + 20 kg)/(M + 1 kg)
		(300 – 2 000) Hz	-5 dB/octave
	Lateral [b] 2,5 min/axis	(20 – 100) Hz	+3 dB/octave
		(100 – 300) Hz	PSD(M) [c] = 0,05 g^2/Hz × (M + 20 kg)/(M + 1 kg)
		(300 – 2 000) Hz	-5 dB/octave
Equipment not located on "external" panel [a]	All axes 2,5 min/axis	(20 – 100) Hz	+3 dB/octave
		(100 – 300) Hz	PSD(M) [c] = 0,05 g^2/Hz × (M + 20 kg)/(M + 1 kg)
		(300 – 2 000) Hz	-5 dB/octave

[a] Panel directly excited by payload acoustic environment.

[b] Equipment vertical axis = perpendicular to fixation plane.
Equipment lateral axis = parallel to fixation plane.

[c] M = equipment mass in kg, PSD = Power Spectral Density in g^2/Hz.

2. Apogee motors, tanks, batteries and equipment of mass > 50 kg as specified in table 11. Notching is possible for tanks and motors, if design loads can be exceeded at centre of gravity (CoG) in fundamental resonance. Notching for batteries is to be agreed by the contracting authority.

Table 11: Random vibration test levels and duration for equipment with mass M >50kg

Frequency	Level	Remark
(20 – 110) Hz	+3 dB/octave	
(110 – 700) Hz	0,09 g^2/Hz	11,12 g r.m.s.
(700 – 2 000) Hz	3 dB/octave	
Duration: all axes – 2,5 min/axis.		

b. Specific

The test level shall be the expected flight level increased by the qualification margin.

5.1.11.4 Random vibration test supplementary requirements

a. A functional test should be conducted before and after the random vibration test (or sinusoidal test, if performed).

b. Detailed visual checks shall be carried out where functional tests are not performed. (e.g. reflectors and solar panels)

c. Equipment powered during launch, including all redundant circuits, shall be energized and functionally sequenced through various operational modes during the test.

d. Perceptive parameters shall be monitored to detect intermittent or persistent failures during the test, for each axis.

e. If the equipment is mounted on dynamic isolators in the space vehicle, the equipment shall:

   1. Be mounted on these isolators during the qualification test;
   2. Control vibration test levels at the input to the isolators;
   3. Perform an additional random vibration test with the equipment hard-mounted to the test fixture in each axis using the test level and duration as specified in table 12.

f. Additional random vibration test shall be performed as specified in Table 12 for equipment which use anti-vibration mounts (AVM) to be applied on the equipment hard-mounted to the test fixture (QAVT):

Table 12: Additional random vibration for AVM

Frequency	Level	Remark
(20 – 80) Hz	+3 dB/octave	
(80 – 350) Hz	0,067g^2/Hz	7,87 g r.m.s.
(350 – 2 000) Hz	3 dB/octave	
Duration: all axes – 2,5 min/axis – no notching		

g. One or more control points can be used. In case of more than one control point, the average shall be used. In special cases (e.g. sensitive equipment and high dispersion of control values), the control can be performed on the peak value.

h. For equipment which is to be re-flown, the test duration per axis shall be 100 s plus 50 s per mission.

5.1.12 Acoustic test, equipment qualification

5.1.12.1 Purpose

The purpose of the acoustics test is to demonstrate the ability of the equipment to withstand the qualification level acoustic environment of the launch.

NOTE Acoustic tests are conducted on equipment with large surfaces which are likely to be susceptible to acoustic noise excitations; for this type of equipment random vibration testing is not performed.

5.1.12.2 Acoustic test description

a. The equipment shall be installed in a reverberant acoustic cell which is capable of generating the specified sound pressure levels.

b. A uniform sound energy density throughout the chamber shall be established.

c. The configuration of the equipment shall be the same as it is subjected to during flight, such as deployed or stowed.

d. The test should be performed with the equipment mounted on a flight type support structure and with ground handling equipment removed.

e. Equipment shall be operating if it is operating during launch.

5.1.12.3 Acoustic test levels and duration

The test levels are hereafter defined according to the qualification philosophies:

a. General

The test level shall be the envelope of the maximum expected spectra (for the different launchers: ARIANE 4 and 5, and STS) increased by the qualification margin (see Table 1).

NOTE The sound pressure levels for qualification are compiled in Table 13.

b. Specific

The test level shall be the expected flight level increased by the qualification margin.

Table 13: Acoustic qualification test level and duration

Octaves		1/3 Octaves	
Centre frequency (Hz)	**Level (dB) ref. 2×10^{-5} N/m^2**	**Centre frequency (Hz)**	**Level (dB) ref. 2×10^{-5} N/m^2**
31,5	130	31,5	126
		40	128
		50	129,5
63	135,5	63	131
		80	132
		100	132,5
125	139	125	134
		160	136
		200	138
250	143	250	139
		315	137,5
		400	135
500	138	500	133
		630	131
		800	129
1 000	132	1 000	127
		1 250	125,5
		1 600	124,5
2 000	128	2 000	123
		2 500	121,5
		3 150	120,5
4 000	124	4 000	119
		5 000	117,5
		6 000	116,5
8 000	120	8 000	115
		10 000	113,5
Test overall level: 147 dB Test duration: 2 min Spectrum adjustment requirements: the duration shall not exceed 30 s at full levels			

5.1.12.4 Acoustic test supplementary requirements

a. A functional test should be performed before and following the acoustic test.

b. Detailed visual checks shall be carried out where functional tests are not performed. (e.g. reflectors and solar panels)

c. During the test, electrical and electronic equipment, including all redundant circuits, powered during launch shall be energized and functionally sequenced through various operational modes, and perceptive parameters monitored to detect intermittent or persistent failures during the test.

d. Calibration runs, with an empty chamber, shall be performed prior to test in order to obtain the specified spectrum.

e. The levels for calibration runs shall be the same as those planned for the equipment, i.e. low, intermediate and qualification levels.

f. The low level run shall be repeated after the qualification level run.

g. For large equipment (testing volume > 1 m^3), a minimum of three microphones shall be positioned around the test article at a distance away from any surface to measure the field sound pressure level without being affected by surface absorption or re-radiation.

h. For equipment which is to be re-flown, the test duration shall be 120 s plus 50 s per mission.

5.1.13 Shock test, equipment qualification

5.1.13.1 Purpose

The purpose of shock testing is to demonstrate the ability of the equipment to withstand the shocks induced by the separation of the payload from the launcher, the booster burn out, the various pyrotechnic events in the payload (e.g. apogee boost motor ignition and solar array deployment) and shocks from nominal and emergency landing of reusable vehicles.

5.1.13.2 Shock test description

a. The equipment shall be mounted to a fixture using its normal mounting points.

   NOTE Mounting of the equipment on the actual or dynamically similar structure provides a more realistic test than mounting on a rigid structure such as a shaker armature or slip table.

b. The selected test method shall achieve the required shock spectrum with a transient that has a duration comparable to the duration of the expected in-flight shock.

   NOTE 1 Numerous test techniques are currently applied to perform shock tests.

   NOTE 2 Methods such as shaped pulses, complex decaying sinusoids, impact devices, and pyrotechnically excited fixture can be used. Electrodynamic shakers can reproduce shock transients with a prescribed spectrum.

c. Trial tests using a test configuration shall be conducted to validate the proposed test method before testing qualification hardware.

d. Adequate number of transducers shall be mounted at the input points for verification of successful input.

e. The test method shall be acceptable, provided that:

   1. a transient with the prescribed shock spectrum is generated within specified tolerances, and

2. the applied shock transient provides a simultaneous application of the frequency components as opposed to a serial application of shock frequency components.

5.1.13.3 Test levels and duration

a. General

The shock spectrum in each direction of the three orthogonal axes shall be equivalent to a half sinusoidal pulse of 0,5 ms duration and 200 g (0-peak) amplitude, see Figure 2.

b. Specific

The test levels shall be the expected flight levels increased by the qualification margin.

c. Shock test performance

1. At least three shocks shall be imposed to meet the amplitude criteria in both directions on each of the three orthogonal axes.
2. If a suitable test environment can be generated to satisfy the amplitude requirement in all six axial directions by a single application, that test environment shall be imposed three times.
3. If an imposed shock meets the amplitude requirements in only one direction of a single axis, the shock test shall be conducted a total of 18 times in order to get three valid test amplitudes in both directions of each axis.

5.1.13.4 Shock test supplementary requirements

a. A visual inspection should be made before and after the shock test.

b. A functional test shall be performed before and after all shock tests.

c. Detailed visual checks shall be carried out where functional tests are not performed (e.g. reflectors and solar panels).

d. Equipment, including redundant circuits, powered during launch shall be energized and monitored.

e. If the equipment is mounted on dynamic isolators in the space vehicle, the equipment shall be mounted on these isolators during the qualification test.

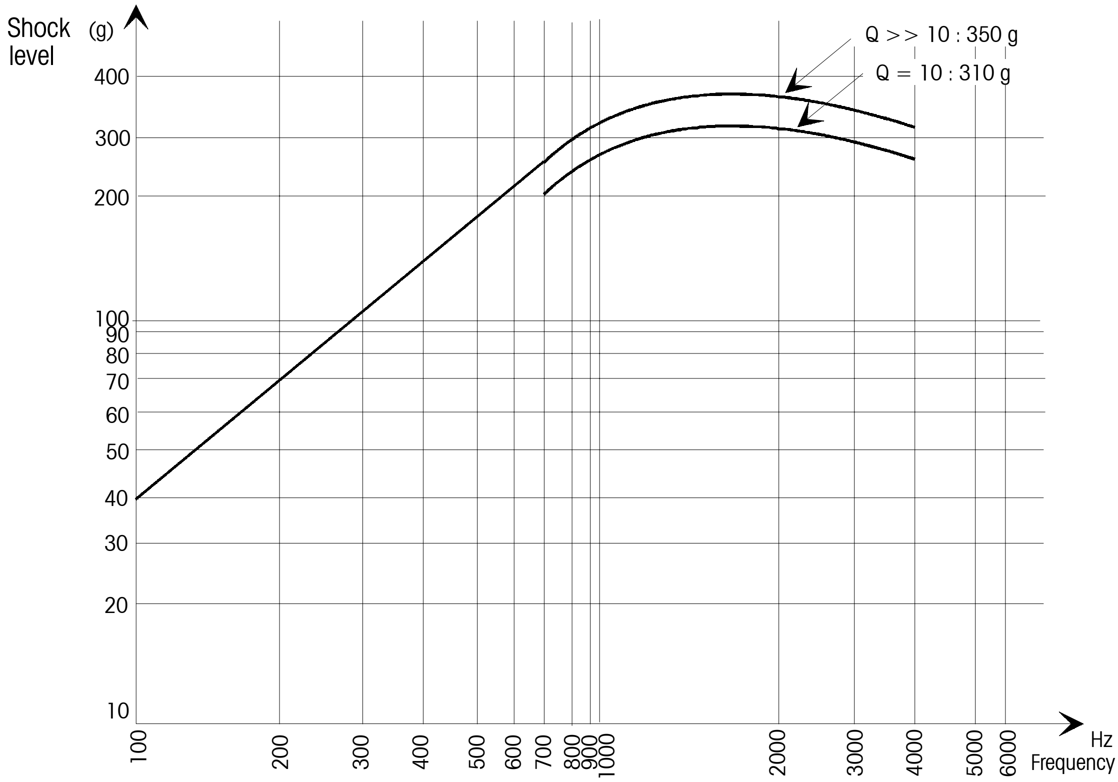


Figure 2: Shock spectrum equipment qualification

5.1.14 Corona arcing detection, equipment qualification

a. Equipment that is energized during ascent or descent and which is exposed to the critical low pressure atmosphere shall be monitored for corona and multipacting detection, as the pressure is reduced from ambient to 0,1 Pa within 10 min (minimum) simulating the launch profile.

b. Equipment operating at a voltage >500 V, shall be tested for corona arcing.

5.1.15 Thermal vacuum test, equipment qualification

5.1.15.1 Purpose

The purpose of the thermal vacuum test is to demonstrate the ability of the equipment to perform in a thermal vacuum environment that simulates the worst conditions in-orbit, including an adequate margin.

5.1.15.2 General

All equipment temperatures shall refer to reference point temperatures (see 3.1.33).

5.1.15.3 Thermal vacuum test description

a. The equipment shall be mounted in a vacuum chamber, in a thermally controlled environment (see Figure 3).

b. Temperatures shall be controlled, measured and selected such that it can be guaranteed that the test item experiences actual temperatures equal to or beyond the minimum and maximum qualification temperatures in the test environment.

c. The equipment shall be qualified using the type of fixations and mounting as designed in the equipment specification.

d. The equipment shall be fixed and mounted in accordance with one of the test methods specified in subclause 5.1.15.4.

NOTE For equipment not covered in subclause 5.1.15.4 refer to the contracting authority.

5.1.15.4 Thermal vacuum test methods

5.1.15.4.1 Non-special equipment, internally mounted

The equipment shall be arranged as follows (see Figure 3):

a. The equipment shall be bolted to a mounting panel as specified in the equipment interface specification.

b. With the exception of the mounting contact area, the mounting panel shall be black-painted.

c. The mounting panel should have the following characteristics:

1. thickness representing standard platforms and sidewalls (if thickness is not known, a value of 20 mm should be used);
2. length and width approximately equal at least to twice the nominal base dimensions of the equipment;
3. flatness and stiffness to meet equipment surface requirements.

d. The mounting panel shall be temperature controlled.

e. During the test, the shroud or the panel temperatures shall be controlled to provide the space vehicle internal environment to give the qualification temperature level on the equipment itself.

f. The reference point temperature shall be selected so as to conform to criteria below:

NOTE The reference point is defined in the equipment interface specification.

1. The number of reference points should be kept to a minimum, one point whenever possible.

   NOTE Complex equipment use more than one reference point.

2. The reference point should be located at the outer surface of the equipment on its base plate or near to its mounting feet.
3. The reference point shall be representative of the mean temperature level of the equipment.
4. A temperature sensor shall be located at the reference point, as an integral part of the equipment.

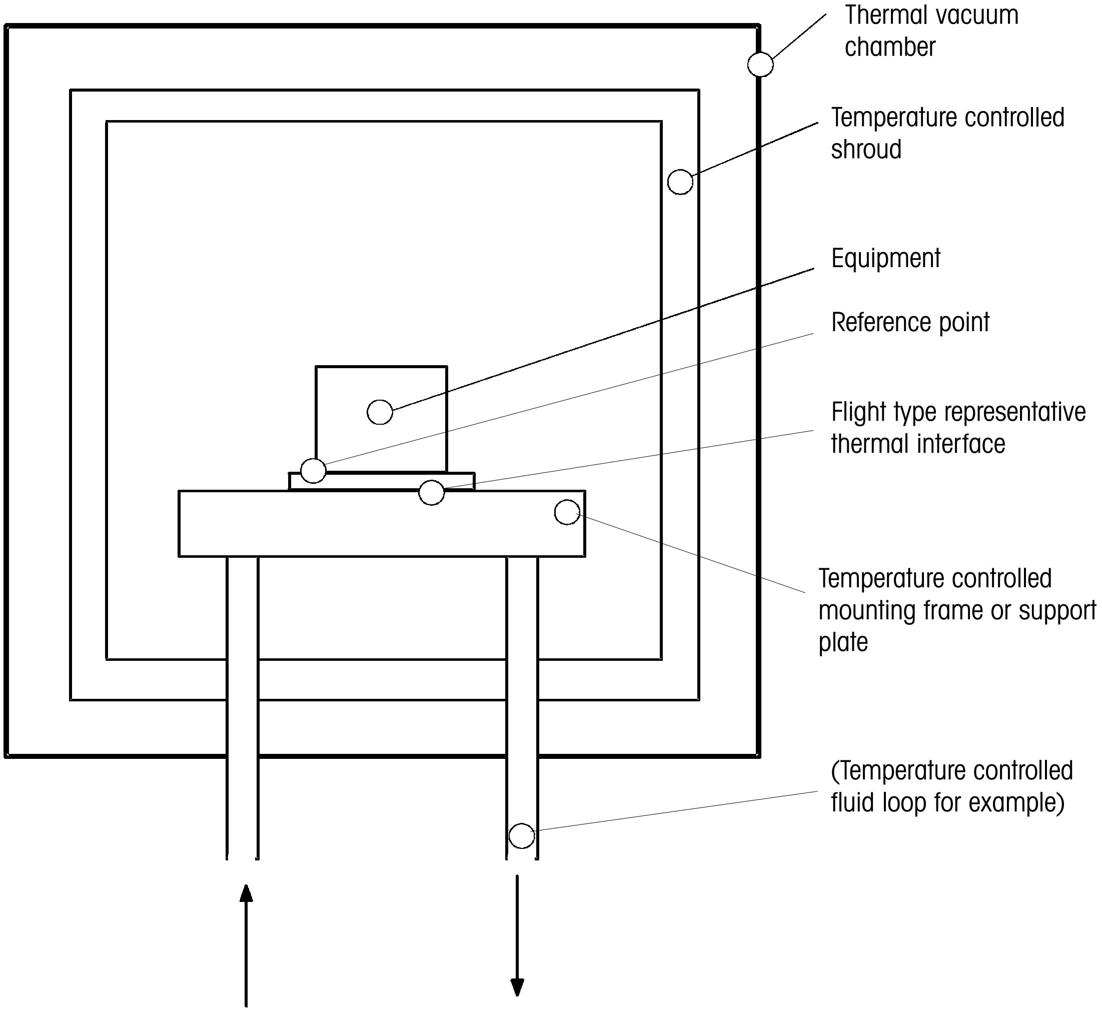


Figure 3: Equipment thermal vacuum test arrangement

5.1.15.4.2 Special equipment, internally mounted

a. Certain internally mounted equipment shall be subject to special test provisions.

EXAMPLE

* Solar array drive mechanism,
* sensors having viewing apertures seeing space,
* attitude thruster assemblies,
* high power TWTs having direct radiating collectors, and
* high dissipating, direct radiator cooled equipment.

b. For special equipment as described above, the test method given for internally mounted equipment (see subclause 5.1.15.4.1) should be modified, to give a reasonable representative test environment.

5.1.15.4.3 Equipment, externally mounted

a. Any equipment mounted outside the main space vehicle body (e.g. antennas, deployment mechanisms, pointing mechanisms, solar arrays and external power dumps) shall have special test requirements.

b. The test arrangement shall be designed to give the specified qualification temperatures on the equipment, which is representative of heat flows to and from the environment.

5.1.15.4.4 Thermal radiative equipment

An equipment which is mostly radiative cooled can be suspended inside the test chamber, in this case no conductive path shall exist between the equipment under test and the fixation.

5.1.15.5 Thermal vacuum test temperatures

The equipment temperatures shall be defined for the following conditions:

- minimum and maximum operating qualification,
- minimum and maximum non-operating qualification, and
- minimum and maximum start-up qualification.

a. General

The temperature range for an effective qualification should be ±10 °C with respect the value defined in subclause 6.1.10.4.

NOTE Standard qualification temperature limits (increased or decreased by test tolerances, as indicated in Table 3) are provided in annex B for various type of equipment.

b. Specific

1. The qualification temperatures shall be defined with a 10 °C margin with respect to the predicted flight temperatures

   NOTE For worst cases, refer to annex B.

2. The values above shall be increased or decreased by the test tolerances, as indicated in Table 3.

5.1.15.6 Thermal vacuum test conditions

a. Equipment shall be tested in a thermal vacuum environment, having a pressure of 10^{-5} hPa or less.

b. The thermal vacuum test shall start with a pump-down.

c. The adjustment of the first temperature exposure can commence after the pressure is less than 10^{-4} hPa.

NOTE For corona and arcing detection equipment qualification see subclause 5.1.14.

5.1.15.7 Thermal vacuum test cycle and duration

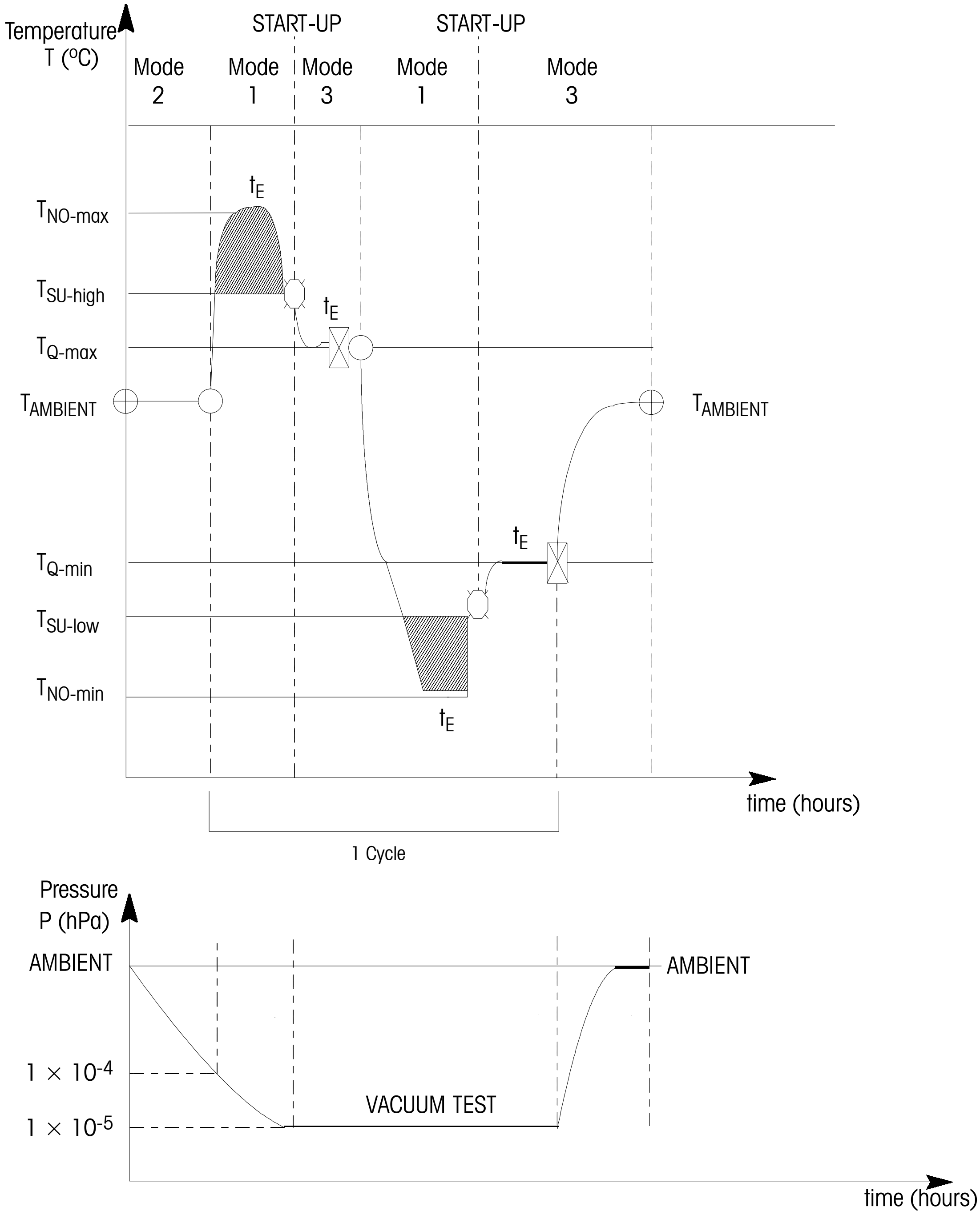
The equipment shall be tested in the thermal vacuum test sequence performing one thermal cycle as described below and shown in Figure 4.

a. The temperature cycle shall begin with the initial functional and performance test and the chamber at ambient temperature ($T_{AMBIENT}$).

b. The equipment shall be switched-off and the pressure decreased.

c. At a pressure of 10^{-4} hPa, the temperature shall be increased up to the maximum non-operating level ($T_{NO\text{-}max}$).

NOTE The temperature is increased to provide for better outgassing.



For explanation of symbols see Table 14

Figure 4: Equipment thermal vacuum test sequence

d. After a dwell time t_E, the temperature shall be decreased to the maximum (hot) start-up level ($T_{SU-high}$) and then the temperature stabilized at the high operating temperature (T_{Q-max}).

e. After the time t_E, the functional and performance test shall be performed.

f. The equipment shall be switched off and the temperature decreased and maintained at the minimum non-operating temperature (T_{NO-min}) during a time t_E.

g. The temperature shall be increased to the minimum (cold) start-up temperature (T_{SU-low}) and the equipment switched on.

h. When stabilized at the low operating level ($T_{Q\text{-}min}$), and after the time t_E, the functional and performance test shall be performed.

i. The temperature and pressure shall be raised to ambient conditions ($T_{AMBIENT}$) and the final functional test performed.

NOTE Characteristic parameters of thermal vacuum test are specified in table 15.

j. The temperature rate of change < 20 °C/min shall apply only to equipment within the space vehicle. For equipment outside the space vehicle, higher gradients are specified in the appropriate equipment specifications.

Table 14: Legend and symbols for Figures 4, 9 and 10

Symbol	Description
T	Test item temperature
$T_{AMBIENT}$	Ambient temperature
$T_{NO\text{-}max}$	Maximum non-operating temperature (highest design temperature for the equipment to survive not powered)
$T_{NO\text{-}min}$	Minimum non-operating temperature (lowest design temperature for the equipment to survive not powered)
$T_{SU\text{-}high}$	Maximum start-up temperature (highest design temperature of the equipment, at which the equipment can be switched on)
$T_{SU\text{-}low}$	Minimum start-up temperature (lowest design temperature of the equipment, at which the equipment can be switched on)
$T_{Q\text{-}max}$	Maximum qualification temperature (highest design temperature at which the equipment demonstrates full design ability)
$T_{Q\text{-}min}$	Minimum qualification temperature (the lowest design temperature at which the equipment demonstrates full design ability)
P	Pressure
MODE 1	Functionally inert (test item not energized). Normally applicable to the non-operating condition.
MODE 2	Partially functioning. Conditions as detailed in applicable design specifications, but normally applicable to conditions during launch.
MODE 3	Fully functioning (test item fully energized and fully stimulated). Normally applicable to conditions during orbit.
⊕	Initial and final "functional and performance test"
⊠	Intermediate reduced functional and performance test
t_E	Dwell time
	Switch-on (Start-up)
○	Switch-off

Table 15: Thermal vacuum test parameters (qualification)

Parameter	Condition / Value
Start cycle	Hot (Hot start for outgassing)
n (number of cycles)	1
t_E (dwell time at T_{hot}/T_{cold}) [a, b]	2 hours
dT/dt (temperature rate of change)	< 20 °C/min
Stabilization criterion	1 °C/1 hour

[a] $T_{hot} = T_{Q\text{-}max}$ or $T_{NO\text{-}max}$.
[b] $T_{cold} = T_{Q\text{-}min}$ or $T_{NO\text{-}min}$.

5.1.15.8 Thermal vacuum test supplementary requirements

a. The equipment shall be subjected to functional and performance test before and after the thermal vacuum test at ambient conditions.

b. Functional and performance tests shall be performed at the minimum and maximum operating temperature levels.

c. During the remainder of the test, the equipment shall be monitored to detect intermittent or persistent failures.

d. Monitoring of the RF output for corona shall be conducted during chamber pressure reduction for those RF units that are energized during launch.

e. RF equipment shall be operated at maximum power and at design frequency.

f. For mechanisms operating in vacuum, a complete verification at the identified worst cases. of the functional capability shall be included.

g. If thermal gradients play an important role in the functional behaviour of the mechanisms, they shall be implemented in the test set-up.

h. The force or torque design margin shall be measured at identified worst cases on moving mechanical assemblies.

i. For equipment with mechanism, the vacuum should be below 10^{-6} hPa with a target value of 10^{-8} hPa pressure and a diffusion pump shall not be used.

j. The partial pressure of oxygen shall be less than 10^{-7} hPa with a target value of 10^{-9} hPa pressure.

k. The decision to perform tests on items such as antennas, solar arrays and any item made of carbon composites and honeycomb, with relaxed pressure requirements or to make an additional effort (e.g. bake-out – to reduce particle or gas release) shall be made on a case-by-case basis.

NOTE Pressure requirements can be difficult to reach when testing antennas, solar arrays and any item made of carbon composites and honeycomb.

5.1.16 Thermal cycling test, equipment qualification

5.1.16.1 Purpose

The purpose of the thermal cycling test is to demonstrate the ability of the equipment under test to fulfil all functional and performance requirements over the qualification temperature range at ambient pressure.

5.1.16.2 Thermal cycling test description

a. The equipment shall be mounted in a temperature chamber.

b. Temperatures shall be controlled, measured and selected such that it can be guaranteed that the test item experiences actual temperatures equal to or beyond the minimum and maximum qualification temperatures.

c. This above shall be achieved by adopting one of the following methods in subclause 5.1.16.3.

5.1.16.3 Test methods

5.1.16.3.1 First test method

a. The equipment shall be bolted to a mounting panel as specified in the equipment interface specification, and as shown in Figure 5 and described in subclause 5.1.15.4.

b. The assembly shall be suspended in a temperature facility and the chamber flooded with dry air or nitrogen to preclude condensation on and within the equipment at low temperature.

c. The chamber shall be driven to a fixed temperature.

d. Instrumentation shall be provided to allow accurate temperature control of the average equipment temperature for comparison with the specified test levels.

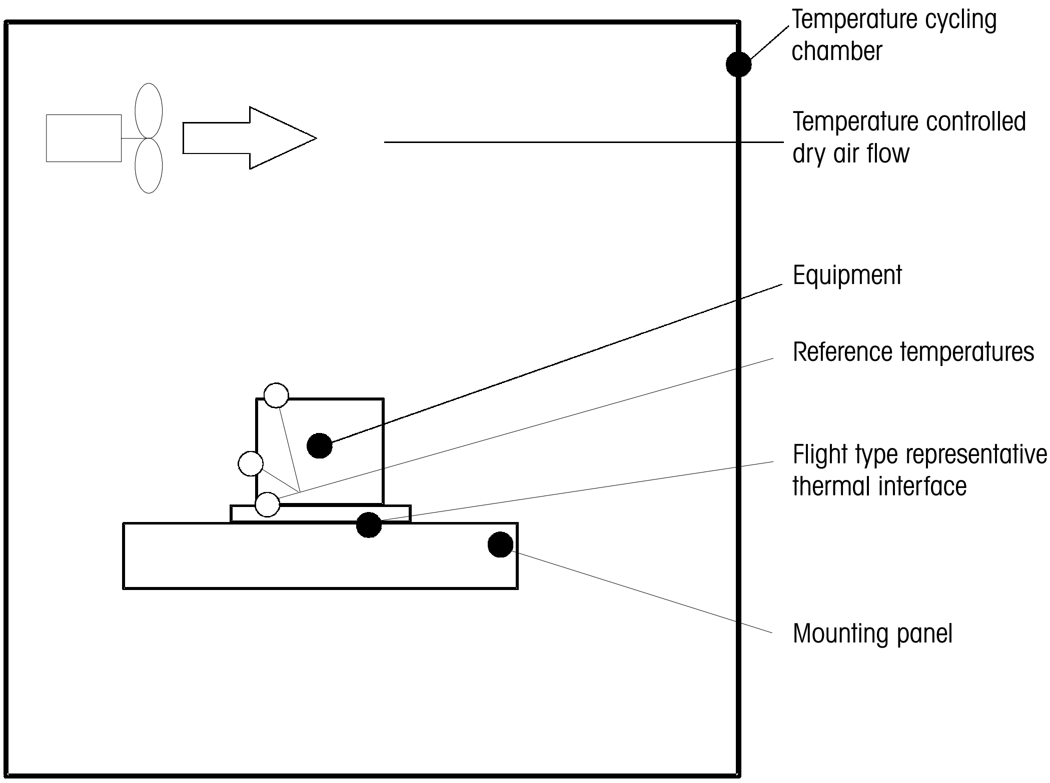


Figure 5: Equipment thermal test arrangement with surface temperature control

5.1.16.3.2 Second test method

The test method shall be as in Figure 5 with the following modifications:

a. The mounting panel shall be replaced by a thermally controlled conductive heat sink (e.g. cold-plate).

b. The equipment shall be bolted directly to this heat sink (see Figure 6).

c. The chamber air temperature shall be driven to a fixed temperature to represent the space vehicle internal environment.

d. The heat sink temperature shall be controlled to give the qualification temperature level on the equipment.

e. The temperature adjustments or averaging shall be referred to the reference point temperature.

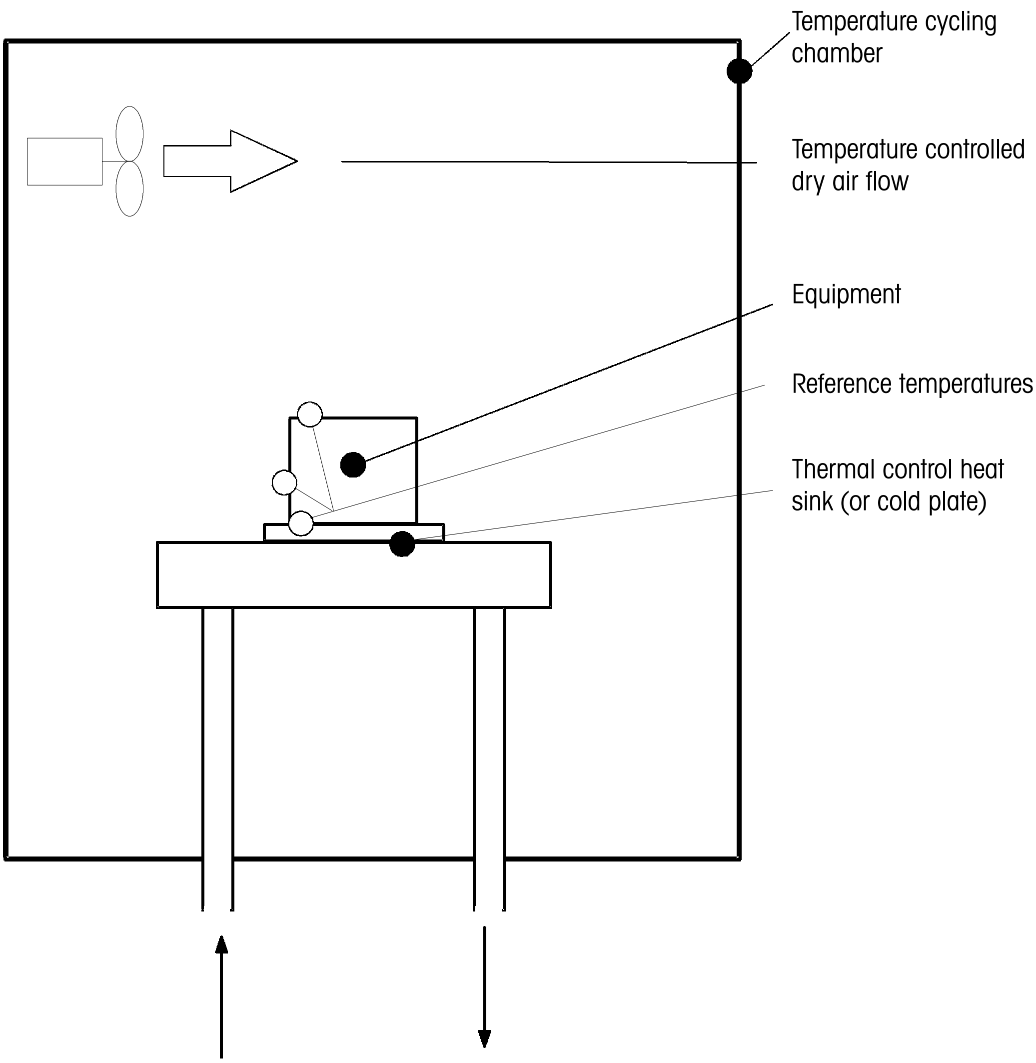


Figure 6: Equipment thermal test arrangement with direct temperature control

5.1.16.3.3 Third test method

a. This method is project specific,and detailed configuration shall be discussed with the customer's technical counterpart.

b. The test item shall be installed in a specially designed test chamber that simulates the following conditions.

1. Blow air through the test item for forced air cooling.
2. Blow air around the test item housing for surface cooling.
3. Simulate adjacent units to the test item.
4. Adjustable air flow and air temperature through chamber.

NOTE The principle test set-up is shown in Figure 7.

c. The airflow rate shall be adjusted as specified in the interface specification.

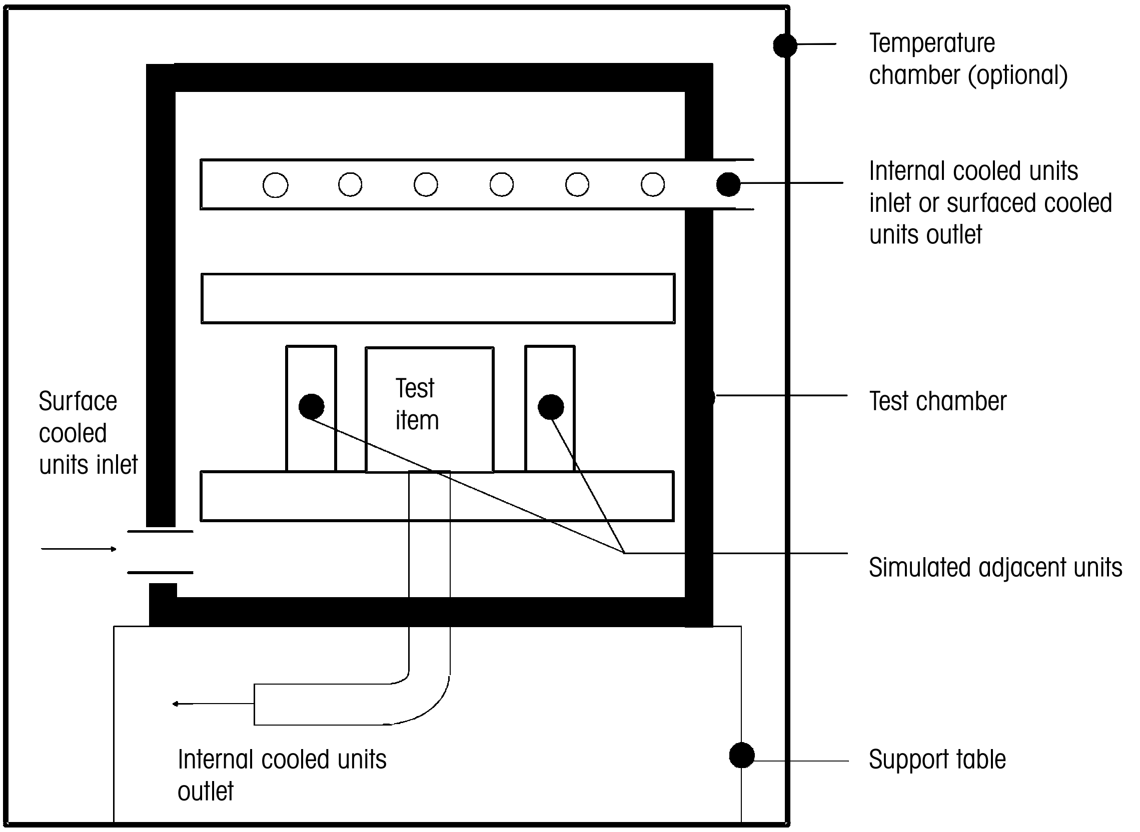


Figure 7: Equipment thermal test arrangement for forced air cooling (internal or surface cooled unit method)

5.1.16.3.4 Fourth test method

a. Accelerated thermal cycling (ATC) shall be used to simulate several thousands of thermal cycles, for instance for low Earth orbit (LEO) applications, in a time compatible with project schedule.

NOTE The principle of ATC operation is shown in Figure 8.

b. The test item shall be installed in a specially designed test chamber in which fans blow alternatively cold and warm dry nitrogen.

c. The chamber temperature control and the measurement of test items temperature shall be independent.

d. The use of this method shall be discussed with the contracting authority for application to solar cell samples.

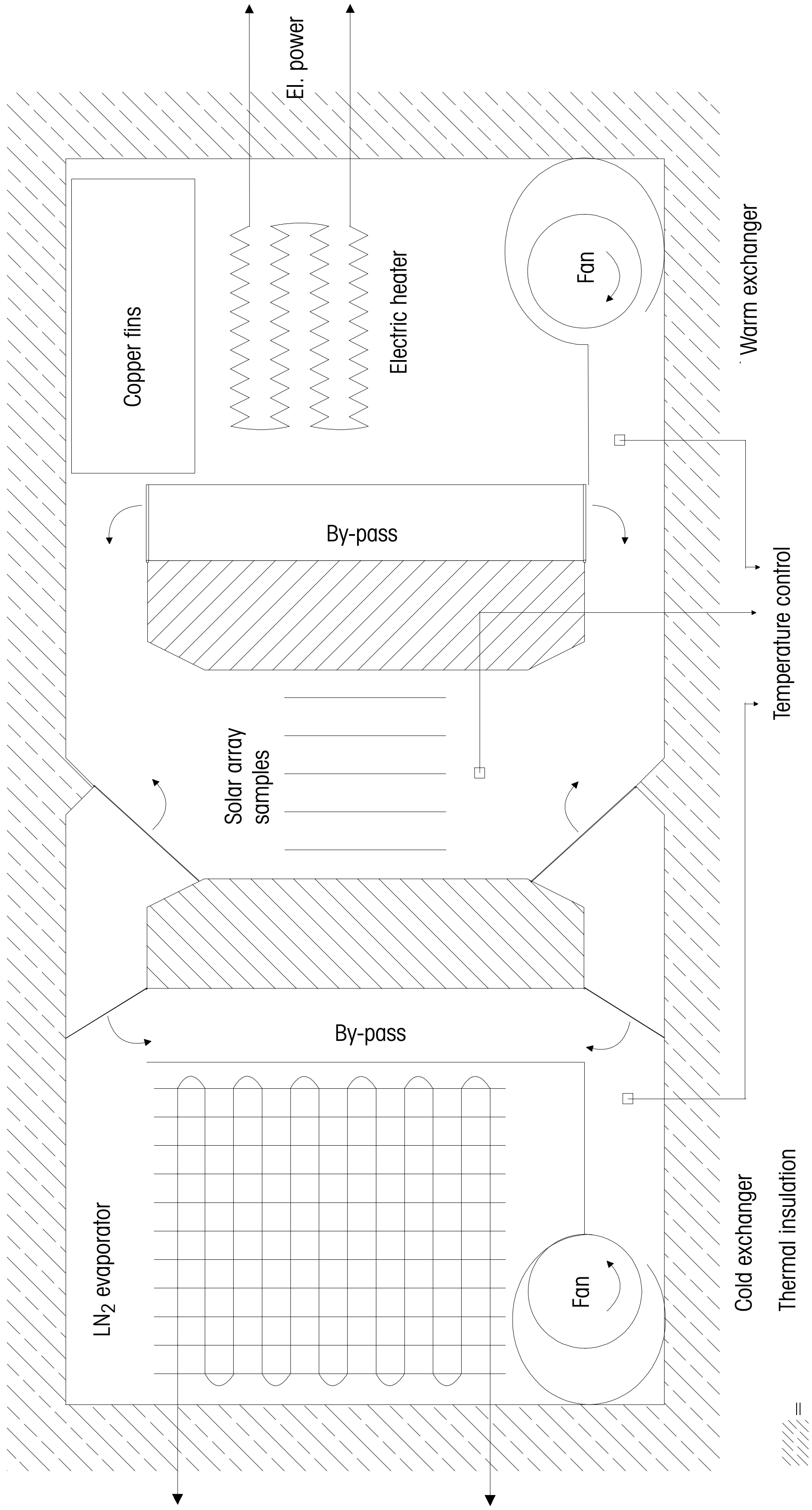


Figure 8: Accelerated thermal cycling test set-up

5.1.16.4 Thermal cycling test temperatures

The qualification temperature limits shall be in accordance with subclause 5.1.15.5.

5.1.16.5 Thermal cycling test cycles and duration

The equipment shall be tested in the thermal cycling test sequence performing 8 thermal cycles, as shown in Figure 9.

NOTE Characteristic parameters of the thermal cycling test are given in Table 16.

a. The first thermal cycle begins after the equipment is functionally tested at ambient temperature.

b. The equipment shall be switched off and the temperature increased up to the high non-operating level (T_{NO-max}).

c. After a dwell time t_E, the temperature shall be decreased to the maximum (hot) start-up level ($T_{SU-high}$) and then the temperature shall be stabilized at the high operating temperature (T_{Q-max}).

d. After the time t_E, the functional and performance test shall be performed.

e. The equipment shall be switched off and the temperature shall be decreased and maintained at the minimum non-operating temperature (T_{NO-min}) during a time t_E.

f. The temperature shall be increased to the minimum (cold) start-up temperature and the equipment switched on.

g. When stabilized at the low operating level (T_{Q-min}), and after the time t_E, the functional test shall be performed.

h. The equipment shall be cycled between T_{Q-max} and T_{Q-min} until the number of cycles specified in Table 16 is achieved.

i. During the last cycle, the equipment shall be functionally tested at T_{Q-max} and T_{Q-min}.

j. At the end of last cycle, the temperature shall be raised to ambient conditions and the final functional and performance test performed.

k. The temperature rate of change < 20 °C/min shall apply only to equipment within the space vehicle. For equipment outside the space vehicle, higher gradients are specified in the appropriate equipment specifications.

Table 16: Thermal cycling test parameters (qualification)

Parameter	Condition / Value
Start cycle	Hot
n (number of cycles)	8
t_E (dwell time at T_{hot}/T_{cold}) [a, b]	2 h
dT/dt (temperature rate of change)	< 20 °C/min
Stabilization criterion	1 °C/h

[a] $T_{hot} = T_{Q-max}$ or T_{NO-max}.
[b] $T_{cold} = T_{Q-min}$ or T_{NO-min}.

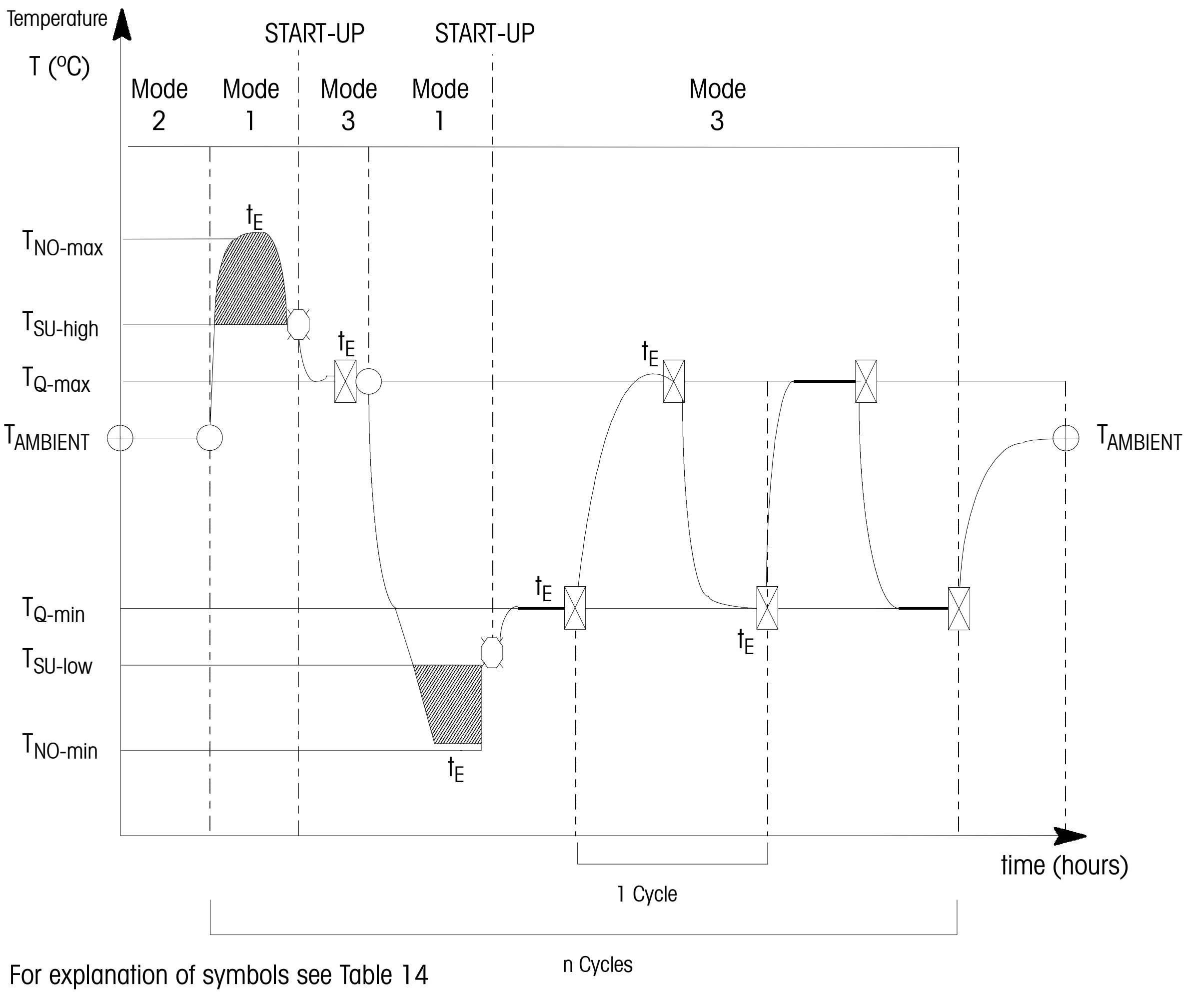


Figure 9: Equipment thermal cycling test sequence

5.1.16.6 Thermal cycling supplementary requirements

a. The equipment shall be subjected to functional test before and after the thermal cycling test at ambient temperature.

b. Functional tests shall be performed at the minimum and maximum operating temperature levels during the first and the last cycles.

c. During the remainder of the test, the equipment shall be monitored to detect intermittent or persistent failures, to the maximum extent possible.

d. Compatibility of valves or propulsion equipment with their operational fluids shall be verified at test temperature extremes during thermal cycling tests. A substitute propellant can be used to ensure operation (open or close) of propulsion units.

5.1.16.7 Thermal vacuum cycling test

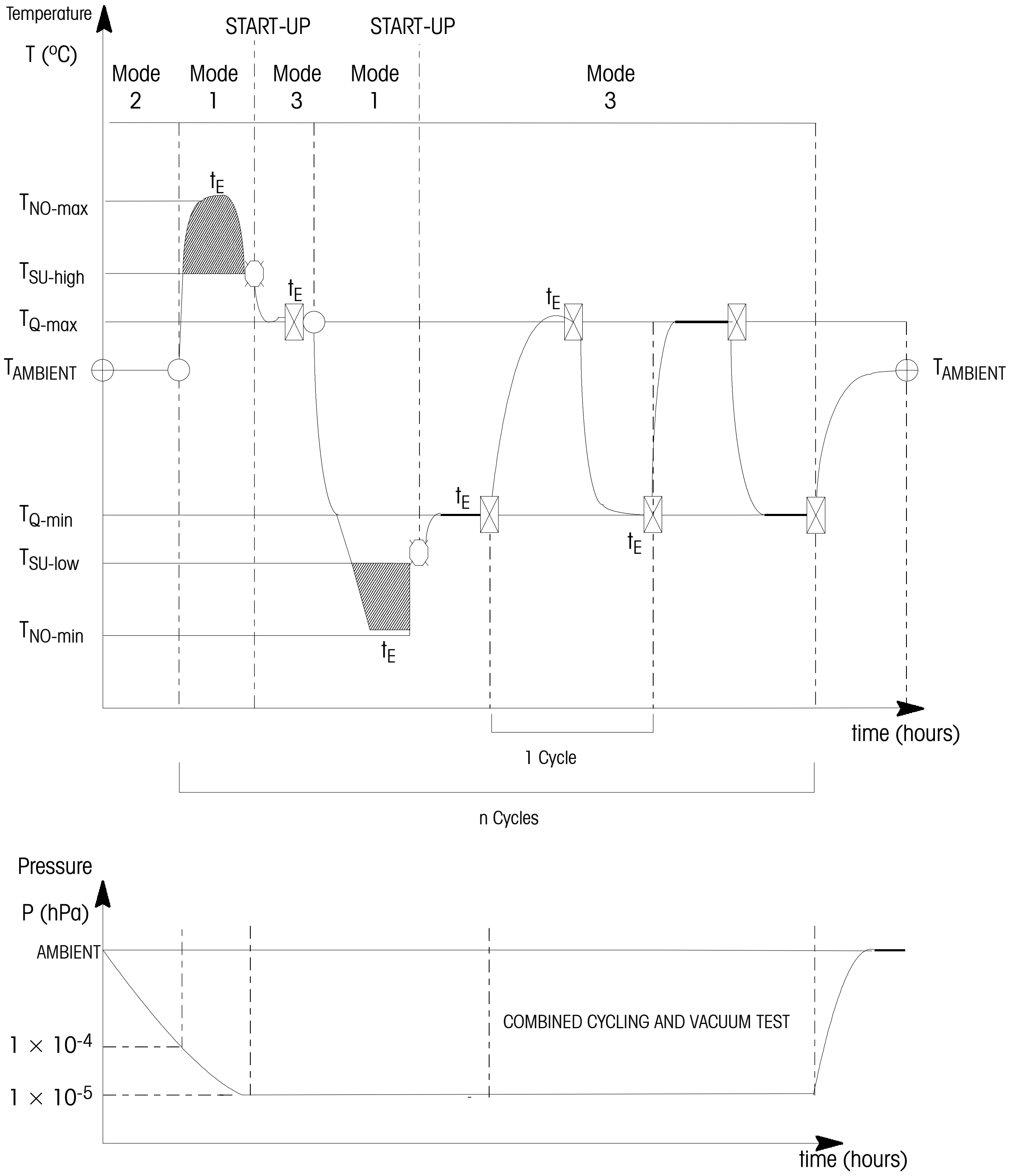
a. For equipment operating in vacuum, the thermal vacuum and thermal cycling tests can be combined.

b. The equipment shall be tested as shown in Figure 10.

c. For the thermal vacuum cycling test the parameters defined in Table 16 shall apply.

d. Test conditions and supplementary requirements shall conform to subclauses 5.1.15.4 and 5.1.15.8

e. Functional and performance tests shall be performed at the minimum and maximum operating temperature levels during the first and the last cycles.



For explanation of symbols see Table 14

Figure 10: Equipment thermal vacuum cycling test sequence

5.1.17 EMC and ESD test, equipment qualification

5.1.17.1 Purpose

The purpose of the EMC and ESD test is to demonstrate that the electromagnetic interference characteristics (emission and susceptibility) of the equipment under worst conditions does not result in malfunction of the equipment and to verify that the equipment does not emit, radiate or conduct interference that can result in malfunction of other system equipment.

5.1.17.2 General

a. The performance of the equipment during and after electrostatic discharge shall be verified both in conducted and radiated modes.

b. The susceptibility margins shall be measured.

5.1.17.3 EMC and ESD test description

a. The test shall be conducted in accordance with EMC and ESD project requirements (see MIL-STD-461D for EMC and ESD procedures and levels of test and MIL-STD-462D for application of the measurement procedures).

b. An evaluation shall be made of each equipment to determine the tests to be performed.

5.1.18 Life test, equipment qualification

5.1.18.1 Purpose

The purpose of the life test is to demonstrate that the equipment can operate during its ground and in-flight life without degradation.

5.1.18.2 Life test description

5.1.18.2.1

a. The equipment (one or more) shall be set up to operate in circumstances that simulate the flight conditions to which they can be subjected.

b. The environmental conditions shall be selected for consistency with end use requirements and the significant life characteristics of the particular equipment.

c. Environments should be either ambient, thermal, thermal vacuum or various combinations of these.

d. The test sample shall be selected at random from production units or shall be a qualification unit.

5.1.18.2.2

a. The test shall be designed to demonstrate the ability of the equipment to withstand the maximum operating time and the maximum number of operational cycles predicted during its service life with a suitable margin.

b. For equipment having a relatively low percentage duty cycle, it shall be acceptable to compress the operational duty cycle into tolerable total test duration.

c. For equipment that 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.

5.1.18.2.3

a. The testing shall accumulate the fatigue damage expected during the service life multiplied by a factor of 4.

b. If environmental conditions contribute to accumulated degradation they shall be simulated during fatigue tests.

5.1.18.2.4

a. The test specimen shall be instrumented with strain and deflection gauges to measure strains and deflections at critical locations.

b. Visual inspection of critical areas shall be performed at regular intervals to confirm that no cracks or flaws initiate (success criteria).

5.1.18.3 Life test levels and duration

a. Pressure

Ambient pressure shall be used except for unsealed units where degradation due to a vacuum environment is anticipated, in which case a pressure of 10^{-4} hPa or less shall be used.

b. Environmental levels

The maximum predicted environmental levels shall be used.

NOTE 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 an equipment life test shall be 4 times that predicted during the service life, including ground testing, in order to demonstrate an adequate margin.

d. Functional duty cycle

1. 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.
2. An abbreviated functional test shall be conducted within specification limits.

5.1.18.4 Life test supplementary requirements

For statistical type life tests, the duration shall be dependent upon the number of samples, subject to the demonstration of confidence and reliability.

5.1.19 Microgravity environment compatibility test, equipment qualification

5.1.19.1 Purpose

The purpose of this test is to verify the microgravity environment compatibility (MEC) at equipment level, with respect to the specified microgravity environment limit response spectrum at the receiver location.

5.1.19.2 General

a. The following measurements shall be performed.

1. Interface dynamic forcing functions of the disturbance source.
2. Mechanical impedance of the test set-up for later system correlation.
3. Unit internal (in case of payload elements) self-induced vibration responses and transfer functions.

b. The forcing functions shall be coupled with the transfer function to predict the vibration responses at the most critical locations.

c. The resulting acceleration response spectrum from above, shall not exceed microgravity environment limit acceleration level.

NOTE For additional MEC details see annex A.4.

5.1.19.3 Microgravity compatibility test description

5.1.19.3.1 General

a. The tests should be conducted under similar ambient conditions as during on-orbit operation with respect to temperature, relative humidity and ambient pressure, for a most realistic simulation of the test article functions.

b. The test article shall be in its nominal operational configuration similar to the on-orbit operational conditions.

c. Earth gravitational mass load compensation devices should be attached to the test article in order to unload moving mechanisms or non-linear springs.

5.1.19.3.2 Quasi-static interface force and torque measurements

a. The test article shall be installed on either

   1. a seismic foundation, or

   2. an interface support structure which is stiff enough to measure interface forces and torques with frequencies below 0,01 Hz.

b. The instrumentation shall consist of force and torque transducers which shall be installed at the interface between the test article and test foundation.

c. The instrumentation shall be able to resolve forces of up to 10^{-3} N and torques up to 10^{-2} Nm from quasi static up to about 0,01 Hz.

d. The test shall consist of, measuring the interface reaction force time histories with frequencies below 0,01 Hz at the test article mounting locations while it is operated according to a selected motion profile, and calculating the resulting disturbance force and torque caused during the test article motion profile.

5.1.19.3.3 Structure-borne vibration-interface force measurements

a. The instrumentation shall consist of force transducer and accelerometer installed at the mounting interface connection points of the test article.

b. The measurement instrumentation should be able to resolve forces of up to 10^{-4} N and accelerations of up to 10^{-4} m/s^2 in the frequency range from 0,01 Hz to 500 Hz.

c. The test article shall be operated according to its operational time line, selecting critical time slices.

d. The test shall be performed according to one of the following methods.

   1. Direct interface force measurement

      Two different set ups shall be used, one with the test article mounted on a seismic foundation, the other one with the test article softly suspended.

   2. Indirect interface force measurement

      (a) Two different set ups shall be used, one with the test article mounted on a reference structure, the other one with the test article dismounted from the reference structure.

      NOTE The reference structure may have any structural property with a high flexibility and high modal density, which can fix the test article mounting interfaces and can carry the test article loads.

(b) The reference test structure shall be softly suspended in order to assure dynamic free or free boundary conditions.

5.1.19.3.4 Airborne vibration-radiated sound power measurements

The measurements shall be performed as specified below:

NOTE This method follows the recommendations given by ISO 3744.

a. The source to be tested should be installed and mounted with respect to the reflecting plane in one or more positions that are typical of the normal usage.

b. The sound pressure level shall be observed over a typical period of operation of the source.

c. Readings of the sound pressure level shall be taken at each measurement point for each frequency band within the frequency range from 31,5 Hz up to 500 Hz.

5.1.19.4 Microgravity environment compatibility test levels and duration

The test level results shall be presented in terms of:

a. Resulting interface force and torque time history with frequency below 0,01 Hz, as a function of the test article operational modes.

b. Frequency spectra with a log-log scale over the frequency range 0,01 Hz to 500 Hz and as 1/3-octave band centre frequency within a dynamic range of 60 dB (three decades).

c. Surface pressure level and sound pressure power level in dB.

5.1.19.5 Microgravity environment compatibility test supplementary requirements

a. The test condition shall be selected depending on the type and function of the disturbance mechanism.

b. The following tests can be omitted:

   1. Quasi-static interface force or torque measurements if no force and torque momentum exchange exist or if the test article is operated under steady state conditions only.

   2. Airborne vibration-radiated sound power measurements if no noise is radiated or if the test article is operated in orbit under vacuum.

5.1.20 Audible noise test, equipment qualification

5.1.20.1 Purpose

The purpose of this test is to verify that the audible noise generated by any equipment inside a manned space vehicle does not exceed the specified noise rating curve.

5.1.20.2 Audible noise test description

5.1.20.2.1 General

a. The equipment induced sound pressure level (L_p) shall be measured when the equipment is fully integrated into the space vehicle.

NOTE The sound pressure level (L_p) inside a space vehicle, as generated by the equipment,is a function of the equipment total sound power (L_{WT}) delivered to the space vehicle and of the sound absorption.

b. For equipment tests, the equipment induced total sound power level (L_{WT}) shall be determined.

NOTE The total sound power delivered by the equipment is the sum of two contributions: the sound power transmitted to the space vehicle via airborne transmission path (see subclause 5.1.20.2.2) and via the structure borne transmission path (see subclause 5.1.20.2.3)

5.1.20.2.2 Equipment airborne sound pressure measurement

a. The equipment sound power measurement should be performed in accordance with the ISO 3740.

b. The airborne sound power levels, as generated by the equipment, shall be determined in octave bands.

c. The noise level and exposure time shall be given for each operational mode of the equipment.

d. The sound level shall be converted into cabin pressure level.

5.1.20.2.3 Equipment structure-borne noise measurement

a. The equipment structure borne noise measurement shall be performed by measurement of the equipment interface disturbance levels.

b. The sound pressure levels shall be converted into cabin pressure levels.

c. The equipment interface disturbance force levels shall be measured both directly or indirectly.

NOTE The indirect measurement method is more favourable, if low levels are measured and the mounting interfaces are complicated.

5.1.20.3 Audible noise test supplementary requirements

The equipment shall be operated during airborne and structure borne noise measurements with the same modes as during the mission phases of the manned space vehicle.

5.2 Subsystem test requirements

5.2.1 Overview

The systematic approach used for equipment testing, defining for each type of equipment a series of required and optional tests, is not feasible for subsystems.

Subsystem testing depends on subsystem type, on project characteristics and verification approach, therefore only general guidelines are given in this subclause.

NOTE With the objectives to increase project flexibility and control, and to reduce cost and schedule, functional qualification of the subsystem may become part of the functional qualification of the element or system, in which they are verified as “functions” of the whole element or system.

5.2.2 General

a. The subsystem qualification should include both functional and performance testing.

b. For the qualification of structures and thermal control subsystems, environmental testing shall be performed.

NOTE These subsystems may be qualified at a higher level, in combination with other subsystems during environmental test campaign.

c. Environmental qualification tests should be conducted at subsystem level where this level of testing provides a more realistic or more practical test simulation (e.g. qualification of optical and telecommunication payloads).

d. For functional testing, each subsystem shall be assembled on a test bench which provides a flight representative environment (hardware and software) in terms of function, interfaces and performances.

e. Dedicated simulators of the external services utilized shall be used.

f. Dedicated simulators should be designed to allow their reuse for higher level testing.

5.2.3 Specific subsystems

a. Communication subsystem

1. Antenna pattern tests should be performed on dedicated spacecraft mock-ups.
2. Performance test of a telecommunication payload shall be carried out in a compact test range facility.

b. Landing subsystem

Qualification tests shall be performed on dedicated space vehicle models or part of them, which shall fully represent the mass properties and landing subsystem.

c. Thermal control subsystem

1. The qualification functional tests shall be conducted on dedicated an active thermal control (ATC) bench.
2. The passive thermal control (PTC) should be qualified at a higher level.

d. Environmental control and life support (ECLS) system

Qualification tests shall be conducted on subsystem dedicated test bench and on boiler plate mock-up.

5.3 Element test requirements

5.3.1 General

Element level testing shall be performed on the following items:

a. Space vehicle,

b. Launcher, and

c. Ground segment.

NOTE Ground segment tests are described in subclause 4.9.

5.3.2 Space vehicle test requirements

a. The space vehicle qualification tests should be performed on ground.

b. When re-entry, aero-thermodynamics characteristics are fundamental for the mission success, in-orbit testing should be performed (see clause 9).

c. The space vehicle qualification test baseline shall consist of the tests specified in Table 17.

d. The optional tests shown in the test baseline (see Table 17) should be considered as required tests, as determined from considerations of design features, required lifetime, environmental exposure, and expected usage.

e. In addition to the tests listed in Table 17, other special tests shall be performed depending upon the project characteristics and life cycle.

f. The following general rules shall be taken into account in defining the space vehicle qualification test program:

1. The functional verification of the space vehicle shall be performed by means of the integrated system test (IST) and the integrated system check (ISC) as follows.

    (a) The IST shall be performed at the start and end of the test campaign.

    (b) The ISC shall be conducted before and after each environmental test, providing the criteria for judging successful survival of the space vehicle in a given test environment.

    (c) The ISC shall also be performed during environmental tests, i.e. while the environment is being imposed.

    NOTE ISC is performed during thermal tests, since the space vehicle is expected to be operative under these conditions.

2. During acoustic or vibration tests the vehicle should be in an operating mode representative of launch and ascent phases.

3. For a small compact space vehicle, acoustic testing does not provide adequate environmental simulation, and random vibration shall substitute the acoustic test.

4. Alignment verifications, for all equipment having alignment requirements, shall be performed throughout the vehicle test campaign, to keep track of any degradation or to ensure that variation of equipment alignment in relationship with the reference axes remains within the specified limits.

5. All deployable items shall be subjected to deployment tests to verify the correct functioning of deployment mechanisms after integration and environmental exposures.

6. Aero-thermodynamic tests shall be used for manned and unmanned vehicles whose missions include re-entry phases.

7. Audible noise, toxic offgassing and human factors engineering (HFE) tests shall be used for manned projects.

5.3.3 Structural qualification tests

5.3.3.1 General

a. The structural qualification tests detailed below shall be complemented with functional tests, alignment checks, visual inspections and leak tests.

b. If the space vehicle has propellant storage tanks these should be filled and pressurized to flight conditions during testing.

    NOTE Simulated propellant may be used.

5.3.3.2 Static load tests

5.3.3.2.1 Purpose

a. The purpose of static load tests is to demonstrate by application of static or quasi-static loads that the space vehicle design is capable of sustaining the launcher, apogee motor, and spin induced static and dynamic accelerations without suffering permanent deformation or failure.

b. There are three methods to impose static loads on a test article:

1. whiffle tree tests (see subclause 5.3.3.2.3),
2. centrifuge tests (see subclause 5.3.3.2.4),
3. acceleration tests (see subclause 5.3.3.2.5).

Table 17: Space vehicle qualification test baseline

Test	Reference subclause	Recommended sequence	Space vehicle
Optical alignment	5.3.4.5	1 [a]	R
Functional and performance	5.3.8.1,5.3.8.2	2 [a]	R
HFE	5.3.9.3	3	R [f]
Physical properties	5.3.4.4	4	R
Pressure	5.3.4.1	5	R
Leakage	5.3.4.2	6 [a]	R
Boost pressure profile	5.3.4.3	7	O
Modal survey	5.3.3.7	8 [c]	O
EMC	5.3.7.1	9	R
Shock	5.3.3.6	10	R
Static load	5.3.3.2	11	R
Spin	5.3.3.3	12	R
Acoustic	5.3.3.4	13 [b]	R [b]
Random vibration	5.3.3.5.3	13 [b]	R [b]
Sinusoidal vibration	5.3.3.5.4	14 [c]	O
Transient	5.3.3.5.5	14 [c]	O
Ambient pressure temperature cycling	5.3.5.3	15	O
Thermal balance	5.3.5.2	16 [d]	R
Vacuum temperature cycling	5.3.5.1	17 [d]	R
Toxic offgassing	5.3.9.1	18	R [f]
Aero-thermodynamics	5.3.6	19	R
Magnetic field	5.3.7.2	20	R
Life	5.3.8.5	21	O
Mission simulation	5.3.8.3	22 [e]	R
End-to-end communication	5.3.8.4	22 [e]	R
Audible noise	5.3.9.2	23	R [f]
Mission specific	5.3.10	24	R

R = Required

O = Optional

Notes

[a] At the start and end of the sequence and before and after environmental tests.

[b] Acoustic or random vibration test.

[c] Transient combined with modal survey can replace sinusoidal vibration.

[d] Can be combined.

[e] Can be combined with functional and performance tests at the end of the sequence.

[f] For manned vehicle.

5.3.3.2.2 General

a. The quasi-static acceleration test levels to be applied shall be derived from the maximum quasi-static loads specified in the respective launcher user's manual or apogee motor manual multiplied by the qualification factor.

b. The static loads to be applied in the whiffle tree tests shall be the same values multiplied by the respective mass which is supported by the structure.

c. Loads in excess of those above shall be applied locally to represent quasi-static loading due to appendages and take into account the qualification factor.

d. If the mission profile indicates thermal stresses during the high acceleration periods, such stresses shall be determined analytically and consider superposition of equivalent stresses.

e. The qualification factor shall be at least 1,25.

5.3.3.2.3 Whiffle tree tests

a. Whiffle tree tests shall be performed for space vehicles that due to its size or mass cannot be acceleration tested.

b. Whiffle tree tests shall be conducted on fully representative structural qualification models as follows:

 1. The model shall be supported by a test fixture that simulates the flight adapter.
 2. The loads shall be applied with a test rig.
 3. Static loads of 100 % (limit load), 110 % (yield load), and 125 % (ultimate or qualification load) of the maximum predicted flight loads should be applied in steps.
 4. Deformations and strains shall be measured at each step of the loading cycle.

5.3.3.2.4 Centrifuge tests

a. For centrifuge tests, the test article shall be mounted in a test fixture that simulates the characteristics of its flight attachment.

b. The capacity of the centrifuge, and the mounting of the test article on it, shall be such that the qualification loads are applied to all relevant parts of the space vehicle structure.

c. Loads of 100 % (limit load), 110 % (yield load), and 125 % (ultimate/qualification load) of the maximum predicted flight loads should be applied in steps.

5.3.3.2.5 Acceleration tests

a. In acceleration tests the test item shall be subjected to a sinusoidal dwell dynamic test at a frequency lower than the frequency of its first mode (i.e. non-resonant response).

b. The test article shall be mounted in a test fixture that simulates the characteristics of its flight attachment.

5.3.3.3 Spin tests

a. If flight operations include a spin that impose acceleration levels of 2 g or more to any part of the space vehicle, a spin test shall be performed.

b. The spin test should envelop the flight case with a qualification margin.

5.3.3.4 Acoustic tests

5.3.3.4.1 Purpose

The purpose of acoustic tests is to demonstrate that the space vehicle can withstand the acoustically induced vibration environment encountered during relevant mission phases.

5.3.3.4.2 Acoustic test description

a. Acoustic tests shall be conducted in a reverberate acoustic chamber, with the test article in the launch configuration mounted on a test fixture simulating the dynamic flight mounting conditions, but being low frequency decoupled from chamber floor and wall structure born vibration.

b. Equipment, that operates during launch, shall be operated and monitored during the test.

c. If equipment is substituted by representative structural dummies, vibration levels shall be measured at the interface between the equipment and structure and compared with the equipment qualification spectra.

d. The qualification test spectrum shall exceed the flight spectrum by 4 dB throughout the frequency range, unless specified differently in the respective launcher user's manual.

   NOTE The sound pressure spectra and overall levels to impose during the acoustic test depend on the launcher; the measured or predicted flight levels and durations are specified in the launcher user's manual.

e. The test duration shall derive from the fatigue equivalent duration in flight, (which depends on the launcher), multiplied by a factor of 4.

   NOTE The test duration is in general 2 minutes.

f. Each qualification level run should be preceded and followed by low level runs, which are used to identify possible modifications of the structural status during the test.

5.3.3.5 Vibration tests

5.3.3.5.1 Purpose

The purpose of vibration tests is to demonstrate that the space vehicle withstands the vibration environment encountered during launch or other high vibration exposures.

There are three types of vibration tests that can be performed for space vehicle qualification:

a. Random vibration tests (see subclause 5.3.3.5.3,)

b. sinusoidal vibration tests (see 5.3.3.5.4), and

c. Transient vibration tests (see 5.3.3.5.5).

5.3.3.5.2 General

a. For launch vibration tests, the test article in its launch configuration shall be mounted on a test table via an adapter that is stiff enough to prevent the generation of anti-resonance.

b. Space vehicle equipped with apogee or retro motors shall be tested for the vibration environment of the motor if

   1. the environment is not enveloped by the launch boost environment, or

   2. the configuration during the apogee is different from the launch configuration, or

3. retro motor burn is different from the launch configuration.

c. Equipment that operates during launch, or during apogee or retro motor burn, shall be operated and monitored during the test.

d. If the equipment is substituted by representative structural dummies, vibration levels shall be measured at the interface between the equipment and structure, and compared with the equipment qualification spectra.

e. Each qualification level run should be preceded and followed by low level runs, which are used to identify possible modifications of the structural status during the test.

5.3.3.5.3 Random vibration tests

a. For random vibration tests, at least one control accelerometer shall be attached to the base of the adapter on which the test article is mounted.

b. With Gaussian random excitation applied at the base of the adapter, the spectrum, as measured by the control accelerometer(s), shall be equalized such that the power spectral density, throughout the frequency range, lies within (-1/+3) dB of the levels specified in the qualification test specification.

NOTE The success criteria is that the overall root mean square acceleration level is within ± 10 % of the specified value.

c. Random excitations shall be applied in three mutually orthogonal directions, one being parallel to the thrust axis.

d. The test duration shall be two minutes per axis (unless specified otherwise in the launcher user's manual).

e. The qualification test specification shall as a minimum cover the relevant launcher user's manual qualification test requirements.

f. Random tests may be waived if acoustic tests are performed that envelop all critical environments and if this is in agreement with the launcher user's manual.

5.3.3.5.4 Sinusoidal vibration tests

a. For sinusoidal vibration tests, at least one control accelerometer shall be attached to the base of the adapter on which the test article is mounted.

b. Sinusoidal excitations shall be applied at the base of the mounting adapter, and shall be swept through at a sweep rate of 2 octaves/min.

c. The test level should be limited in the low frequency range by the physical travel limits (amplitude limitation) of the vibration table.

d. For the higher frequencies, the acceleration levels shall be at least as specified in the launcher user's manual.

e. Vibrations shall be applied in three mutually orthogonal directions, one being parallel to the thrust axis.

f. Sinusoidal vibration qualification tests should be performed between 4 Hz and 150 Hz (Ariane Apex) or 5 Hz and 100 Hz (Ariane 4 and Ariane 5).

g. Low level sinusoidal vibration tests should be performed between 5 Hz and 2 000 Hz.

h. The excitation levels specified in the launcher user's manual can be reduced in certain limited frequency bands to avoid overstressing of the test article, subject to the following:

1. A procedure for such a reduction (notching procedure) is agreed with the launcher authority, on the basis of the latest dynamic coupled load analysis available.

2. This analysis is performed with a mathematical model of the space vehicle verified by dynamic test (shaker or modal survey, the second method being preferred).
3. Launch vehicle forcing function variations are taken into account in the analysis.
4. A safety factor of 1,25 is applied in order to determine the maximum allowable notching levels for protoflight or qualification tests.

5.3.3.5.5 Transient tests

a. As a replacement for sinusoidal vibration tests, transient tests in combination with a modal survey should be used for a more realistic dynamic excitation.

b. The transient excitation signals shall be derived from the space vehicle and launcher loads coupled dynamic analysis (LCDA)

c. The modal survey shall verify the space vehicle mathematical model.

NOTE Although transient test methods are fairly advanced, a number of problems with respect to uncertainties resulting from the analytical process on the test input functions and statistical variations are still to be resolved. Transient tests can relatively easily replace longitudinal tests, but experience is very limited in lateral testing.

5.3.3.6 Shock tests

5.3.3.6.1 Purpose

The purpose of shock testing is to demonstrate that the space vehicle withstands shock levels and frequency spectra as predicted for flight.

Shocks encountered during mission originate from shroud jettison, separation of the launcher and the space vehicle, pyrotechnic devices actuated release of covers or appendages, as well as from the latching of such appendages.

5.3.3.6.2 Shock test description

a. Separation shock tests shall be conducted by actuating the pyrotechnic devices and then verifying the separation, e.g. by suspending the space vehicle together with flight type launcher attachment hardware in a manner that allows separation to occur.

b. Shocks induced by release or latching of appendages should be tested by performing the shock inducing function, for the separation shock test.

c. All explosive devices and other potentially significant shock-producing devices or events, including those from sources not installed on the vehicle under test, shall be activated at least once or simulated.

NOTE Significant shock sources are those that induce a shock response spectrum at any equipment location that is within 6 dB of the envelope of the shock response spectra from all shock sources.

d. The significant shock producing sources shall be activated two additional times to provide variability in the vehicle test and to provide data for predication of maximum and extreme expected shock environments for units.

e. Activation of both primary and redundant devices shall be carried out in the same sequence as they are intended to operate in service.

f. All equipment operating during the shock inducing occurrence shall be operated and monitored during the test.

5.3.3.7 Modal survey

5.3.3.7.1 Purpose

The purpose of the modal survey is to determine experimentally the natural frequencies, mode shapes, and damping factors of the space vehicle throughout the dynamically relevant range, i.e. up to 100 Hz.

The modal survey is not a test in the sense of having acceptance or rejection criteria; its successful completion does not indicate conformance to design requirements, except for the case in which one of the design objectives is to keep certain frequency ranges free from natural frequencies of the space vehicle.

5.3.3.7.2 Modal survey description

a. The modal survey shall be conducted on a structural model, as follows:
   1. Equipment and subsystems can be replaced by dummies which are dynamic representatives of the flight hardware.
   2. Dynamical representativeness shall be provided for the main modes of equipment and subsystems, showing frequencies lower than 100 Hz.

b. Experimental determination of the natural frequencies, mode shapes, and damping factors of the space vehicle shall be accomplished by exciting the natural modes either by single-point or multi-point excitation with transient, random or sinusoidal characteristics and with accelerometers strategically distributed throughout the test article to monitor the response.

c. The results of a modal survey shall be used:
   1. to check whether any natural frequencies fall into an undesirable range, e.g. pogo resonance of the launcher (as mentioned in 5.3.3.7.1), and
   2. to check or adjust the natural frequencies and modes determined analytically with the help of a finite element mathematical model of the structural system.

      NOTE Once the mathematical model is in agreement with the physical system, it can be used to predict structural responses such as displacements, accelerations, and stresses for any force input for the structural system alone, or, if the adjacent element (e.g. launcher, orbital transfer or other motor) is compatibly represented, for the combined structural system. Superimposing of thermal stresses can also be performed.

5.3.3.8 Correlation of structural mathematical model and test results

5.3.3.8.1 General

The correlation of structural mathematical models accounts for frequencies, modes and responses, and strain and deformations when static load tests are performed.

5.3.3.8.2 Frequencies and modes

a. The modal survey configuration and the space vehicle should be supported at the launcher interface to provide information relative to the space vehicle launch configuration.

b. The correlation shall be considered successfully achieved when:
   1. Basic modes with large effective modal masses (> 10 % of total mass) have a frequency deviation of ≤ 3 %.
   2. Other modes up to 50 Hz with dominant responses have a frequency deviation of ≤ 5 % or 2 Hz maximum (whichever is the less).

NOTE Modes having low responses to a base excitation are disregarded in the correlation exercise.

c. If the modal survey is performed in other conditions, the correlation shall be considered successfully achieved when for modes normalised to unit translation displacement:

1. Basic modes with large generalized mass (> 5 % of total mass) have a frequency deviation of ≤ 3 %.

2. Other modes up to 50 Hz with dominant relative responses have a frequency deviation ≤ 5 % or 2 Hz maximum (whichever is the less).

NOTE Modes having low relative responses and low generalised mass are disregarded in the correlation exercise.

5.3.3.8.3 Strain and deformations

For any position where the measured values have significant amplitude, the correlation shall be considered successfully achieved when the difference between the predicted and the measured strain values does not exceed 5 % of the allowable at yield.

5.3.4 Structural integrity

5.3.4.1 Pressure test

5.3.4.1.1 Purpose

The purpose of pressure tests on space vehicle level is to demonstrate that the integrated pressurized (e.g. pneumatic and hydraulic) subsystems achieve the specified requirements for pressure integrity and leakage rate.

5.3.4.1.2 Pressure test description

Pressure tests shall be performed on prototype (qualification models) and protoflight items as follow:

a. The respective subsystems shall be pressurized to proof pressure for at least 5 minutes.

b. The proof pressure shall be 150 % of the maximum design pressure.

c. The pressure shall be reduced to the maximum design pressure.

d. The above sequence shall be performed three times.

5.3.4.2 Leakage test

a. Leakage tests shall be performed by pressurizing the system to its maximum design pressure and holding this pressure for at least 20 minutes.

b. The leak rate shall be established from pressure monitoring.

c. All lines, joints and fittings shall be checked for external leaks.

d. Method for checking leaks include:

1. Using chemically compatible bubble-forming agents, or

2. Using the "tent method" in which the test article is enclosed hermetically during the test and the enclosure content is monitored to determine how much of the trace gas with which the system is pressurized has escaped from it.

5.3.4.3 Boost pressure profile test

5.3.4.3.1 Purpose

The purpose of the boost pressure profile test is to demonstrate that the space vehicle can survive the pressure drop during ascent and the pressure build-up during re-entry.

5.3.4.3.2 Boost pressure profile test description

a. The tests shall be conducted by exposing the test article in the launch and re-entry configuration, respectively, to the pressure profile simulating the analytically determined launch and re-entry histories.

   NOTE The test article may be a protoflight, prototype, or a structural model provided all pressure sensitive components are representative of the flight hardware.

b. A protoflight unit shall not be subjected to a boost pressure profile test more than once.

c. The pumping capability of the test facility shall maintain the pressure within the specified limits at all times.

5.3.4.4 Physical properties measurements

5.3.4.4.1 Purpose

The purpose of the physical properties measurements is to determine the space vehicle mass, centre of gravity location, and moments of inertia around its three co-ordinate axes.

5.3.4.4.2 Physical properties measurement description

a. The principal configurations should be the launch and orbit insertion configurations.

   NOTE Depending upon the mission profile other configurations may be used

b. The tolerances should be the minimum values specified in either Table 18 or in the launcher user's manual.

Table 18: Tolerances for physical properties

Physical property	Tolerance
Mass	0,25 %
Centre of gravity location	10 mm
Moments of inertia (w.r.t. spacecraft centre of gravity)	2,5 %

c. The measurements shall be made using precision scales and balancing benches.

   NOTE Launch configuration balance requirements are stated in the launcher user's manual.

d. For a large space vehicle, element parameters can be calculated from component measurements providing the specified accuracy can be met.

e. Spin balance tests shall be used for spin stabilized systems.

f. If spin balance tests are specified with an empty tank, a correlation with the analytical model (tank full) shall be performed.

   NOTE Operational spin balance requirements vary widely depending on the mission profile and rate of spin; therefore, specific balance requirements and procedures are stated in the space vehicle specifications.

5.3.4.5 Optical alignment

5.3.4.5.1 Purpose

The purpose of optical alignment measurements is to ascertain that space vehicle and experiment sensors alignments are within the tolerances specified for the space vehicle.

5.3.4.5.2 Optical alignment description

The measurements, conducted in a suitable optical alignment facility, shall be performed prior and subsequent to all environmental tests that can affect the alignment, such as thermal vacuum and vibration tests, or after transports.

5.3.5 Thermal qualification tests

5.3.5.1 Vacuum temperature cycling tests

5.3.5.1.1 Purpose

The purpose of vacuum temperature cycling (VTC) tests is to demonstrate that the space vehicle is capable of achieving its design and performance requirements in its operational modes under the vacuum and thermal conditions encountered during the mission.

5.3.5.1.2 Vacuum temperature cycling tests description

a. The test shall be carried out at upper and lower temperature extremes equal to the qualification limits and during transients.

b. A proper selection of test set-up, test modes, and methods shall be made, in order to obtain these limits for selected units.

c. The pressure shall be ≤ 1 hPa.

d. The test sequence should start with the hot exposure (as shown in Figure 11)

   NOTE 1 Starting the sequence with a hot exposure accelerates the outgassing and prevents elevated pressure levels in subsequent hot exposures.

   NOTE 2 The test sequence shown in Figure 11 is given as an example.

e. For protoflight units, an abbreviated sequence with only one hot and one cold 12 hour exposure prior to the extended exposures can be used.

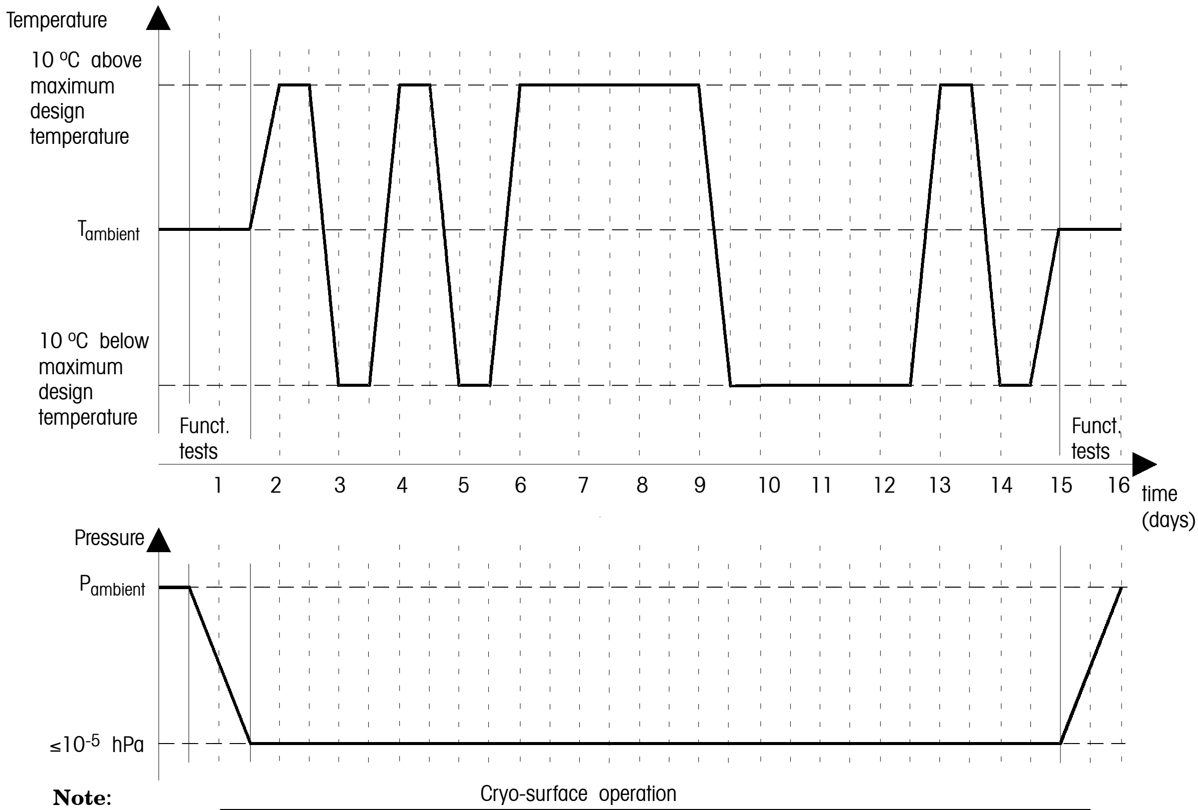
f. In addition to the temperature cycles, the chamber shall be programmed to simulate the various mission phases, and the applicable operational sequences performed by running through a complete cycling of all equipment, including redundant equipment and paths.

g. Cold starts shall be performed at least three times for all components with this capability (one time for protoflight models).

h. The pumping, cooling, and heating speeds should be, as closely as possible, equal to those projected for the mission.

i. Equipment susceptible to corona or other high voltage effects shall be protected by monitoring the pressure and by applying high voltage below critical pressures.

j. Vacuum temperature cycling chambers shall be equipped with cryo-surfaces, which operate at cryogenic temperatures throughout the test to prevent space vehicle contamination.



Note:

The short hot and cold exposures are determined by the duration of the relevant space vehicle tests. The heat-up and cool-down periods are shown to take 12 hours; they can be shorter or longer depending on test item, test equipment and facility.

Figure 11: Example of schematic vacuum temperature cycling test sequence

5.3.5.2 Thermal balance test

5.3.5.2.1 Purpose

The main purpose of thermal balance testing is to demonstrate the ability of the thermal control system to maintain temperatures inside the specified operational limits and to verify that the system performs correctly under vacuum and thermal conditions expected to be encountered during the mission. The test is also used to validate the analytical thermal model of the space vehicle.

5.3.5.2.2 Thermal balance test description

a. The test article shall be placed in a thermal vacuum chamber capable of simulating the thermal and vacuum environments expected during the mission.

b. The thermal state of the test article should be obtained by:

1. simulating the incident radiation (with the help of radiation sources producing the solar and albedo radiation spectra and intensities),
2. simulating the heat absorbed by the test article (with the help of heater blankets and infrared lamps), or
3. combining these two methods.

NOTE The selection of the method, or the degree to which methods are combined depends on the test article configuration and geometry, the relation of internally produced heat to the external heat influx, and the thermal characteristics of the external surfaces.

c. The test article shall be instrumented with suitable temperature measuring devices at critical components and key thermal nodes of the analytical thermal model.

d. The two extreme conditions described below shall, as a minimum, be simulated and evaluated:

NOTE Test conditions and durations for this test depend strongly on the mission profile and on the details of the analytical thermal model to validate.

1. "hot case", i.e. maximum absorbed heat combined with maximum internal power dissipation, and
2. "cold case", i.e. minimum absorbed heat combined with minimum internal power dissipation.

e. The thermal vacuum chamber should be programmed to:

1. simulate all critical thermal mission phases,
2. execute the corresponding operational sequences, and
3. monitor and record temperatures continuously.

f. Success criteria for the thermal balance test shall be:

1. a demonstration of satisfactory operation of the space vehicle within the specified temperature limits; and
2. a satisfactory degree of correlation between temperature measurements and thermal model predictions.

5.3.5.3 Ambient pressure temperature cycling (APTC) test

5.3.5.3.1 Purpose

The purpose of the ambient pressure temperature cycling (APTC) test is to demonstrate the functionality of the test article under ambient pressure conditions and temperature extremes, whilst exceeding the design temperatures by a qualification margin.

NOTE Although components with large power dissipation cannot reach their operational temperature extremes due to convection, these tests are very useful to identify design weaknesses prior to the more complex thermal vacuum tests.

5.3.5.3.2 Ambient pressure temperature cycling (APTC) test description

a. The test should be carried out at upper and lower temperature extremes equal to the qualification limits and during transients.

NOTE Since it is in practice impossible to achieve these levels on all units, a proper selection of test set-up, test modes, and methods is essential in order to obtain these limits for selected units.

b. The test duration shall be at least six hours at stabilized temperatures.

c. Throughout APTC tests, humidity control shall be exercised to maintain the dew point temperature of the test chamber air below the test temperature in order to avoid condensation.

d. Functional tests of all mechanical and electrical components that are expected to operate in the respective environments shall be performed prior to the test, during the temperature stabilized period of the test, and subsequent to the test.

5.3.6 Aero-thermodynamic tests

5.3.6.1 Purpose

The purpose of the aero-thermodynamic test is to verify the aerodynamic and thermal loads on the space vehicle.

5.3.6.2 Aero-thermodynamic test description

The tests shall be performed on dedicated scaled models in wind tunnels for different conditions (e.g. hot and cold hypersonic, low supersonic and subsonic).

5.3.7 Electromagnetic qualification tests

5.3.7.1 Electromagnetic compatibility tests

5.3.7.1.1 Purpose

The purpose of electromagnetic compatibility testing is to determine whether any space vehicle operations can be adversely affected by electromagnetic interference from external sources, and whether the space vehicle itself emits any electromagnetic signals that can adversely affect its own operations, those of its payload, or of external elements.

5.3.7.1.2 Electromagnetic compatibility test description

The space vehicle shall be subjected to electromagnetic susceptibility and emission tests, stated in the EMC specification for the:

a. launch configuration,

b. return configurations, and

c. the most critical and sensitive modes of operation .

5.3.7.2 Magnetic field measurements

The purpose of conducting magnetic field measurements is to determine the permanent, induced, and stray magnetic moments of the space vehicle, and to demonstrate conformance to magnetic cleanliness requirements.

NOTE If there are no stringent magnetic cleanliness requirements, verification by analysis may be sufficient.

5.3.8 Functional qualification tests

5.3.8.1 Mechanical functional tests

5.3.8.1.1 Purpose

The purpose of mechanical functional tests is to check the operation of all mechanisms, deployables, valves, and any other mechanical devices.

5.3.8.1.2 Mechanical functional test description

a. For all mechanical operations that cannot be tested in the Earth's gravity field, suitable ground support fixtures shall be employed to enable operation and evaluation of the devices.

b. If items cannot be tested at element level (e.g. solar arrays) the mechanical functional test shall be performed at subsystem level.

c. Mechanical functional tests shall be performed prior and subsequent to all environmental tests and if specified, during all environmental tests.

5.3.8.2 Electrical functional tests

5.3.8.2.1 Purpose

The purpose of the electrical functional test is to verify the integrity of all electrical circuits and components, as well as their related software.

5.3.8.2.2 Electrical function test description

a. During the electrical functional tests, all components shall be operated, including redundant equipment and paths.

NOTE Pyrotechnic devices may be replaced by simulators that can be energized and monitored.

b. All commands and telemetry parameters shall be exercised based upon the actual database that is used to operate the space vehicle.

c. Preconditioned commands shall be demonstrated that they are not performed unless the relevant preconditions are met.

d. Autonomous functions shall be verified that they are performed only when the conditions for which they are designed are present.

e. Autonomous lockout or shutdown sequences shall be verified to ensure that they do not adversely affect other system operations during or subsequent to the intended lockout or shutdown.

f. Electrical functional tests shall be performed before and after all environmental tests and if specified during all environmental tests

g. At least one complete electrical functional test should be run at both the minimum and maximum bus voltage level.

h. Other equipment performance margins with respect to frequencies, command and data rates, gain and voltage, shall be established.

i. Actual tests of pyrotechnic devices can be conducted at levels below element level.

j. The functional verification of the space vehicle shall be performed by integrated system tests (IST) and integrated system checks (ISC).

k. The IST and ISC should be designed to verify that the performance of the space vehicle conforms with the specification requirements for correct operation in all operational modes, including back-up and degraded modes, and all transients.

l. The IST activities should follow the expected mission sequence, properly involving the interested functions, with the vehicle correct configuration for the particular mission phase.

m. The functional verification should include negative logic testing to assure that no function other than the intended function is performed and no spurious signals or effects are present.

n. The IST should be designed so that a database of critical parameters can be established for trend analysis.

o. The IST shall be performed at the beginning and at the end of the test campaign.

NOTE The IST terminology is used on some projects to identify the integrated subsystem tests (ISST) defined in subclause 6.1 and is known as the system functional test (SFT).

p. The ISC shall:

1. be a sub-set of the IST which can involve all major functions, at the maximum extent automatically performed;

2. be conducted before and after each environmental test, providing the criteria for judging successful survival of the space vehicle in a given test environment;
3. give evidence, with a high degree of confidence, of the correct operation of the space vehicle, in a relatively short time (compared with that one of the IST);
4. provide also data for trend analysis, in order to point out any degradation or trends which can indicate a potential failure.

   NOTE In projects using the terminology SFT, the ISC is known as the abbreviated functional test (AFT).

q. Functional verification of the space vehicle shall be performed by means of ISC, while the environment is being imposed.

r. Functional verification shall be performed for thermal tests, since the space vehicle is expected to be fully operational under these conditions.

5.3.8.3 Mission simulation tests

5.3.8.3.1 Purpose

The purpose of the mission simulation tests are to verify the operation of all space vehicle systems and operations throughout the projected mission.

NOTE Mission simulation tests are a combination of mechanical and electrical functional tests, including flight software, for all mission phases

5.3.8.3.2 Mission simulation test description

During mission simulation tests, the space vehicle shall be put through a simulation of all operations of the entire mission profile with all events occurring in the actual flight sequence, including:

1. final count-down,
2. launch,
3. ascent,
4. separation,
5. orbital operations,
6. apogee motor operations
7. mission operations, and
8. return operations.

5.3.8.4 End-to-end communication tests

The purpose of end-to-end communication testing is to test all space vehicle communication links by simulating the mission communication environment.

NOTE The end-to-end communication test may be performed on payload level if the capacity of existing facilities does not permit test at element level.

5.3.8.5 Life tests

a. Life tests shall be conducted on levels below element one

   NOTE Life tests include fatigue tests, mean-time-to-failure tests and the burn-in tests for electronic components

b. Where a mechanism performs a critical task for the system, life testing shall be performed on a level, where the mechanism function can be performed realistically.

5.3.9 Manned space vehicle specific tests

5.3.9.1 Toxic offgassing test

5.3.9.1.1 Purpose

The purpose of the toxic outgassing test is to verify that flight hardware does not emit toxic vapours that can build up to harmful levels for the crew in the closed loop life support system.

5.3.9.1.2 Toxic offgassing test description

a. The toxic gas test shall be performed in thermal chamber.

b. The test article shall be heated and maintained in temperature conditions, while the emitted gasses and vapours are detected.

c. Appropriate toxic gas markers shall be used on the basis of the materials composing the test article.

5.3.9.2 Audible noise test

5.3.9.2.1 Purpose

The purpose of the audible noise test is to predict that flight hardware does not produce audible noise levels that are detrimental to the crew health and safety.

5.3.9.2.2 Audible noise test description

a. The audible noise test shall be performed in an acoustic shielded chamber.

b. After a background noise measurement, the noise emission during test article operation shall be measured using microphones.

c. The noise measurements shall be performed for the worst-case emission conditions.

5.3.9.3 Human factor engineering (HFE) tests

5.3.9.3.1 Purpose

The purpose of the test is to verify requirements related to human factor engineering (e.g. man-machine interfaces, crew interfaces and accessibility).

5.3.9.3.2 HFE test description

a. The HFE tests shall be performed on:

   1. mock-ups (for accessibility demonstration, layout and verification), and

   2. the engineering model (for man-machine interface and crew operation verifications).

b. HFE tests shall be used for HFE requirement verification in conjunction with analyses.

5.3.10 Mission specific tests

In addition to the space vehicle qualification tests of preceding subclauses, qualification tests should be performed to demonstrate conformance to specific mission requirements such as landing, docking or berthing which, due to their potential variety and complexity, are not covered in this Standard.

5.3.11 Launch vehicle (LV) test

5.3.11.1 General

LV testing specifications depend on the LV type, engine concept, mission and performance definition.

NOTE For a given launcher stage, test specifications can be dependent on materials and equipment characteristics.

5.3.11.2 LV test description

a. Qualification tests should be conducted at both equipment and subsystem level.

b. Complete LV stages shall be submitted to functional qualification tests.

c. The overall approach for equipment qualification tests described in subclause 5.1 should be applied for various categories of LV. However, test levels, duration and cycles should be tailored to the envelope of the LV flight conditions with the addition of specific launch vehicle qualification factors. These parameters are normalized for most equipment.

d. The qualification of LV structures shall conform to the requirements originated from the development phase as follows:

   1. Static tests shall be conducted to qualify the main structures (external and internal loaded structures) and elastomer-base structures.

      (a) One test should be conducted up to effective fracture for each structure.

      (b) For pressurized tanks, effective fracture under internal pressure should be used.

      (c) Thermal conditions applicable in flight shall be taken into account.

   2. Dynamic environment tests (e.g. sinusoidal, acoustic, random and shock) shall be conducted in order to qualify structures including equipment or elastomer-base systems.

      (a) Vibration levels and related fatigue shall be adjusted in order to reflect flight conditions (plus qualification margin).

      (b) Levels applied on equipment supports shall be limited in order to respect the corresponding equipment qualification level.

      NOTE A modal survey test can be conducted on a complete LV stage in order to validate the corresponding dynamic model.

e. Functional qualification tests applicable to major subsystems and to LV stages shall be compatible with the general approach of subclause 5.3.

f. Functional qualification of the whole LV shall be performed:

   1. by ground tests on the launch range, and

   2. by one or several flights with dummy or actual payloads.

5.4 System qualification test

5.4.1 General

System qualification tests cover those aspects relevant to the interrelationship between two or more elements which constitute the system (i.e. space vehicle, launcher and ground segment) and to interfaces with other systems and projects.

5.4.2 System qualification test description

a. Major test requirements for the system qualification shall be included in the lower level product test requirements on elements.

NOTE The qualification tests may be performed at element level.

EXAMPLE See subclause 4.9.5.3 as an example of the compatibility tests between ground segment and space vehicle.

b. The following rules should be applied in performing system qualification tests:

1. Only functional (and software) tests should be performed at this level.

   NOTE In general the elements are not complete models but they can be simulators fully representative of the interfaces to be tested.

2. Qualification tests should be performed at system level only for very complex and new systems and then a better representative of the interfaces should be used.

   EXAMPLE For a new ground segment together with a new complex space vehicle such as a re-entry capsule.

c. System qualification tests shall be performed in a realistic environment to fully demonstrate the capability to meet specific system requirements.

6
Acceptance testing

6.1 Equipment test requirements

6.1.1 Introduction

The requirements provided in this subclause are of two types:

a. General

The general acceptance methodology provides requirements for multi-project utilization (i.e. different launchers and different missions).

With this methodology the equipment is accepted independently from the specific project, assuring the compatibility with the requirements of a specified set of launchers (e.g. ARIANE 4 and 5, and STS) and with a wide range of mission types.

The general methodology makes use of :

1. standard spectra for structural tests, and
2. standard temperature limits for each category of equipment.

b. Specific

The specific acceptance methodology provides requirements tailored for each specific project.

With this approach the equipment acceptance is limited to the requirements of a particular project (e.g. launcher and mission).

The methodology specifies the criteria for the determination of test levels margins to be applied. In principle, the test durations are the same as those specified in the general approach.

6.1.2 General

a. The equipment acceptance test baseline consists of the tests specified in Table 19.

NOTE The categorization of tests into “required” and “optional” is guided by the sensitivity of the type of the equipment to the specific environment and by the probability of encountering the environment.

b. The test baseline and sequencing shall be tailored to the specific item characteristics for each project, giving consideration to both the required and optional tests.

c. Deviations from the baseline requirements for the required tests shall be decided on a case-by-case basis.

NOTE For special items, (e.g. some tape recorders and certain batteries), the specified acceptance test environments can result in physical deterioration of materials or other damage. In those cases, less severe acceptance test environments that still satisfy the system operational requirements are used.

d. All equipment, irrespective of its category, shall be acceptance tested.

e. The test sequence shall include functional tests before and after each environmental test.

f. Specific functional tests shall be performed during the environmental tests.

g. The acceptance test sequence given in Figure 12 should be followed.

NOTE The sequencing is based on a combination of the order in which the environments are encountered during flight and to detect defects as early in the test sequence as possible.

Table 19: Equipment acceptance test baseline

Test	Reference subclause	Recommended sequence	Category/type of equipment											
			a	b	c	d	e	f	g	h	i	j	k	l
Physical properties	6.1.3	1	R	R	R	R	R	R	R	R	R	R	R	R
Functional and performance	6.1.4	2[1]	R	R	R[6]	R	R	R	R	R	R	-	R	R
Leak	6.1.5	3,5,8,11	R[3]	-	R[3]	R	R	R	O	-	-	-	-	-
Pressure	6.1.6	4	-	-	R[3]	R	R	R	O	-	-	-	-	-
Random vibration	6.1.7	6	R	R[3]	R	R	R	R	R	R	R	R	R	-
Acoustic	6.1.8	6	O[9]	R[4]	-	-	-	-	-	-	-	O	O	R[11]
Shock	6.1.9	7	O	-	-	-	-	-	-	-	O	-	-	-
Thermal vacuum[5]	6.1.10	9[6]	R[2]	O	R[6]	R	R	O	R	R	R	O	R	R[11]
Thermal cycling[5]	6.1.11	9[6]	R	O	R[6]	R	R	O	R	R	R	O	R	O
Burn-in[10]	6.1.12	10	R	-	-	O	-	-	O	-	-	-	-	-
Microgravity[7]	6.1.13	12	R	-	-	R	-	-	-	-	-	-	-	-
Audible noise[8]	6.1.14	13	R	R	-	R	R	-	R	-	-	R	R	-

Categories

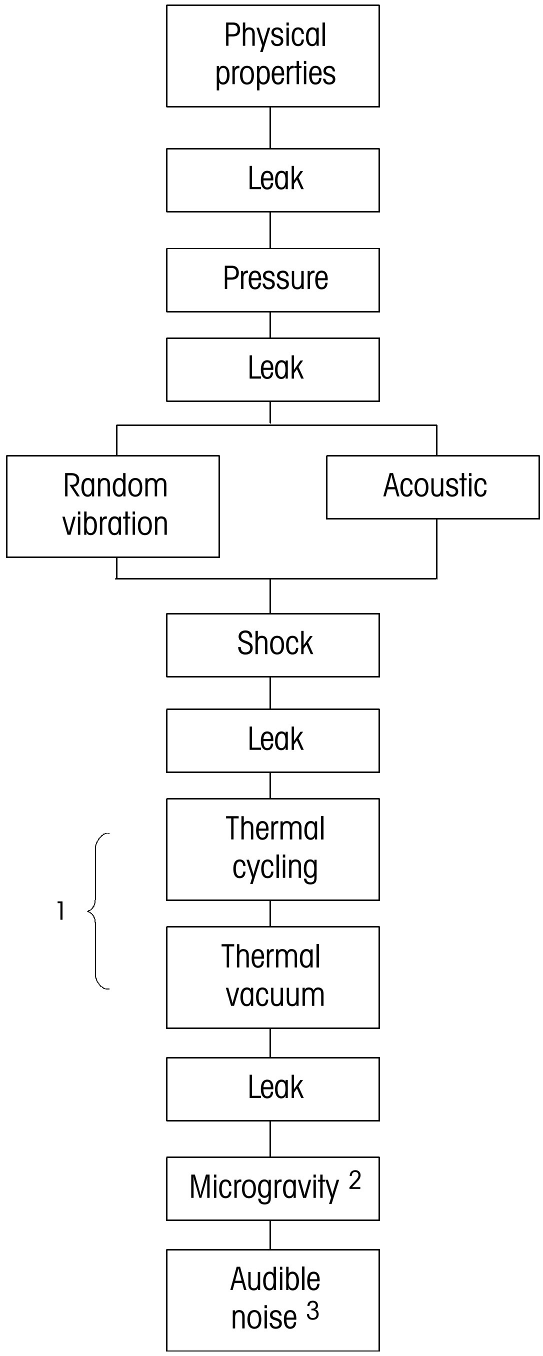
a = Electronic or electrical equipment
b = Antennas
c = Batteries
d = Valves
e = Fluid or propulsion equipment
f = Pressure vessels
g = Thrusters
h = Thermal equipment
i = Optical equipment
j = Mechanical equipment
k = Mechanical moving assemblies
l = Solar arrays

Legend

R = Required O = Optional - = Not required

Notes

1. See subclause 6.1.2 e.
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.
5. Can be combined in thermal vacuum cycling test.
6. Not required for batteries that cannot be recharged after testing.
7. Mission dedicated for in-orbit active mechanical assemblies.
8. Required for equipment for manned spacecraft.
9. If the equipment is sensitive to acoustic environment, the test is performed instead of random vibration test.
10. The test is performed in parallel with other functional and environmental tests.
11. Unless included in element tests.



1 These tests can be combined.

2 Mission dedicated for in-orbit active mechanical assemblies.

3 To be performed in case of manned space vehicle.

Figure 12: Equipment acceptance test sequence

6.1.3 Physical property test, equipment acceptance

The physical property test shall be performed according to subclause 5.1.4.

6.1.4 Functional and performance tests

6.1.4.1 Purpose

The purpose of functional and performance tests is to verify that the electrical and mechanical performance of the equipment meet the requirements of the specifications to detect any anomalous condition.

6.1.4.2 Functional and performance test conditions

The electrical and mechanical functional tests shall be performed at nominal operational conditions.

6.1.5 Leakage test, equipment acceptance

6.1.5.1 Purpose

The purpose of the leakage test is to demonstrate the ability of pressurized equipment to conform to the leakage rates stated in the equipment specifications.

6.1.5.2 Leakage test description and alternatives

a. The equipment leak tests shall be performed before and after exposure to each environmental acceptance test.

b. The test method employed shall have sensitivity and accuracy, consistent with the equipment specified maximum allowable leak rate.

c. One of the methods specified in subclause 5.1.7.2 shall be used.

6.1.5.3 Leakage test levels and duration

The test levels and duration for the leakage test shall conform to subclause 5.1.7.3.

6.1.6 Pressure test, equipment acceptance

6.1.6.1 Purpose

The purpose of the pressure test is to detect material and workmanship defects that can result in failure of the pressure vessel or valves in usage.

6.1.6.2 Pressure test description

The pressure test shall conform to subclause 5.1.8.2 a. and 5.1.8.2 b. with the exception that only one cycle shall be performed and the test at an elevated temperature need not be performed.

6.1.6.3 Pressure test Levels

The test levels for the pressure test shall conform to subclause 5.1.8.3 a. and 5.1.8.3 b.

6.1.6.4 Pressure test supplementary requirements

Safety standards shall be followed in conducting all tests.

6.1.7 Random vibration test, equipment acceptance

6.1.7.1 Purpose

The purpose of the random vibration test is to detect material and workmanship defects prior to installation of the equipment into a space vehicle, by subjecting the unit to a dynamic vibration environment.

6.1.7.2 Random vibration test description

The random vibration test shall conform to subclause 5.1.11.2.

6.1.7.3 Random vibration test levels and duration

a. General

The random vibration test levels for equipment of any mass shall be the minimum envelope between acceptance vibration test (AVT) spectrum given in Table 20 and the qualification spectrum of subclause 5.1.11.3 divided by 2,25 ($1,5^2$ see annex A2)

b. Specific

The test levels shall be the minimum envelope between acceptance vibration test (AVT) spectrum given in Table 20 and the expected flight spectrum.

NOTE No notching for batteries, however, notching is possible for tanks and motors, if the limit load is exceeded at centre of gravity (CoG) in fundamental resonance.

Table 20: AVT spectrum

Mass	Frequency	Level	Remark
$M \leq 50$ kg	(20 - 80) Hz	+ 3 dB/octave	6,06 g_{rms}
	(80 - 350) Hz	0,04 g^2/Hz	
	(350 - 2 000) Hz	-3 dB/octave	
$M > 50$ kg	(20 - 80) Hz	+ 3 dB/octave	
	(80 - 350) Hz	PSD(M) = $[0,05 \times (M+50)/(M+1)] / (1,5)^2]$	
	(350 - 2 000) Hz	-3 dB/octave	
Where:	PSD is in g^2/Hz, M is in kg.		
Duration:	all axis, 2 minutes per axis.		

6.1.8 Acoustic test, equipment acceptance

6.1.8.1 Purpose

The purpose of the acoustic test is to detect material and workmanship defects that are not detected in a static test condition.

6.1.8.2 Acoustic test description

The acoustic test description shall conform to subclause 5.1.12.2.

6.1.8.3 Acoustic test levels and duration

a. **General**

The test levels shall conform to those provided in Table 21

NOTE The test levels are - 4dB with respect to qualification levels in Table 13.

b. **Specific**

The test levels shall be the expected flight levels.

6.1.8.4 Acoustic test supplementary requirements

Supplementary requirements defined in subclause 5.1.12.4 shall be applied to acoustic tests for equipment acceptance.

Table 21: Acoustic acceptance test level and duration

Octaves		1/3 Octaves	
Centre frequency (Hz)	Level (dB) ref. 2×10^{-5} N/m^2	Centre frequency (Hz)	Level (dB) ref. 2×10^{-5} N/m^2
31,5	126	31,5	122
		40	124
		50	125,5
63	131,5	63	127
		80	128
		100	128,5
125	135	125	130
		160	132
		200	134
250	139	250	135
		315	133,5
		400	131
500	134	500	129
		630	127
		800	124
1 000	128	1 000	123
		1 250	121,5
		1 600	120,5
2 000	124	2 000	119
		2 500	117,5
		3 150	116,5
4 000	120	4 000	115
		5 000	113,5
		6 000	112,5
8 000	116	8 000	111
		10 000	109,5
Overall level: 143 dB			
Duration: 2 min			
Spectrum adjustment duration, not to exceed 30 s at full levels			

6.1.9 Shock screening test, equipment acceptance

6.1.9.1 Purpose

The purpose of the shock screening test is to detect intermittent or persistent failures due to conducting particles in electronic equipment.

NOTE 1 The shock screening test is effective for particle detection since shock can dislodge the particle and the extended duration vibration allows the particle to move to a position where a shock may occur and permit detection.

NOTE 2 The shock screening test can detect failures due to cracked or loose dies in electronic parts that are not found in normal acceptance tests.

6.1.9.2 Shock screening test description

a. The shock screening test shall consists of a pyro shock followed by dwell and vibration burst testing.

b. The equipment shall be mounted to a rigid fixture through its normal mounting points.

c. The equipment shall be tested in each direction, on each of the three mutually perpendicular axes (6 tests).

d. The equipment shall be electrically energized and functionally sequenced through all possible operating modes, including redundancy, during testing. Circuits should be monitored to detect failures.

e. A functional test shall be conducted before and after the pyro shock test and after the vibration screening test for each 6 axes.

6.1.9.3 Shock screening test levels and duration

a. A pyro shock test shall be performed once in both directions on each of three mutually perpendicular axes at maximum predicted levels as defined in subclause 5.1.13.3.

b. The screening vibration test level shall be 3 dB below the acceptance test levels specified in subclause 6.1.7, but not lower than 4,5 g r.m.s.

c. Each shock screening test shall consist of a 5 minutes dwell test followed by vibration bursts consisting of 10 s “on” and 10 s “off”.

d. The number of vibration bursts shall be such that all circuits are monitored at least 10 times during the burst sequence with a minimum number of 20 bursts.

6.1.9.4 Shock screening supplementary requirements

a. The shock screening test should be performed where the equipment is critical to mission success.

NOTE This test has proved to be a good method for detecting failures in guidance equipment.

b. A special purpose spectrum can be developed instead of basing the screening vibration test on the expected flight vibration spectrum shape.

c. The screening vibration spectrum imposed on the equipment should cause circuits boards to vibrate at a level of 5 g r.m.s. to 10 g r.m.s. with a flat spectrum from 10 Hz to 450 Hz.

NOTE Input spectrum can be defined from the results of developmental vibration survey tests or analytic response predictions.

d. The equipment qualification vibration levels and durations specified in this Standard shall be assessed to assure that they provide a sufficient margin to safely conduct the screening test.

NOTE 1 The screening test incurs exposure of the equipment to vibration in addition to that experienced in the acceptance vibration test.

NOTE 2 The additional exposure is defined by the screening spectrum based on the expected flight shape, or by the specially developed spectrum.

6.1.10 Thermal vacuum test

6.1.10.1 Purpose

The purpose of the thermal vacuum test is to detect material and workmanship defects by subjecting the equipment to a thermal vacuum environment.

6.1.10.2 Thermal vacuum test description

The thermal vacuum test description shall conform to subclause 5.1.15.3 with the exception that the temperatures shall be controlled, measured and selected such that the test item experiences actual temperatures equal to or beyond the minimum and maximum acceptance temperatures in the test environment.

6.1.10.3 Thermal vacuum test methods

The thermal vacuum test method shall conform to subclause 5.1.15.4.

6.1.10.4 Test temperatures

The equipment temperatures for the minimum and maximum operating acceptance, non-operating acceptance and start-up are defined, according to the following qualification methodologies:

a. General

   1. The standard acceptance temperature limits provided in annex B shall be used for various types of equipment and increased or decreased by test tolerances, as indicated in Table 3.

   2. The range temperature for an effective workmanship screening of electrical and electronic equipment should be at least 56 °C.

b. Specific

   The acceptance temperatures shall be defined, as referenced, in annex B. These values shall be increased or decreased by test tolerances, as indicated in Table 3.

6.1.10.5 Test conditions

The thermal vacuum test conditions shall conform to subclause 5.1.15.6, excluding corona and arcing detection.

6.1.10.6 Test cycle and duration

The thermal vacuum test cycle and duration shall conform to subclause 5.1.15.7. with the exception that $T_{AMBIENT}$ is used instead of T_Q.

6.1.10.7 Supplementary requirements

The thermal vacuum test supplementary requirements shall conform to subclause 5.1.15.8, with the following additions:

a. Thermal vacuum test can be deleted for equipment acceptance and thermal cycling performed at ambient pressure, provided that the lack of vacuum does not compromise equipment acceptance, and these equipment are designed for operation in pressurized modules.

b. If separate thermal cycling and thermal vacuum tests are performed, the thermal vacuum shall follow the cycling test.

6.1.11 Thermal cycling test, equipment acceptance

6.1.11.1 Purpose

The purpose of the thermal cycling test is to detect material and workmanship defects by subjecting the equipment to thermal cycling.

6.1.11.2 Test description

The thermal cycling test description shall conform to subclause 5.1.16.2 with the exception that the temperatures shall be controlled, measured and selected such that the test item experiences actual temperatures equal to or beyond the minimum and maximum acceptance temperatures in the test environment.

6.1.11.3 Test methods

The thermal cycling test method shall conform to subclause 5.1.16.3.

6.1.11.4 Test temperatures

The thermal cycling test temperatures shall conform to the acceptance temperature limits defined in subclause 6.1.10.4 and Table 3.

6.1.11.5 Test cycles and duration

The thermal cycling test cycles and duration shall conform to subclause 5.1.16.5. Characteristic parameters are defined in Table 22.

Table 22: Thermal cycling test parameters (acceptance)

Parameter	Condition or value
Start cycle	Hot
n (number of cycles)	4
t_E (dwell time at T_{hot}/T_{cold}) [a, b]	2 h
dT/dt (temperature rate of change)	< 20 °C/min
Stabilization criterion	1 °C/1 h

[a] $T_{hot} = T_{A\text{-}max}$ or $T_{NO\text{-}max}$.

[b] $T_{cold} = T_{A\text{-}min}$ or $T_{NO\text{-}min}$.

6.1.11.6 Supplementary requirements

The thermal cycling test can be combined with thermal vacuum test described in subclause 6.1.10.

6.1.12 Burn-in test, equipment acceptance

6.1.12.1 Purpose

The purpose of the burn-in test is to detect material and workmanship defects that occur early in the equipment life.

6.1.12.2 Burn-in test description

a. A modified thermal cycling test shall be used to accumulate the additional operational hours required for the burn-in test of electrical components as follows:

   1. While the equipment is operating (power-on) and while parameters are being monitored, the temperature of the unit shall be reduced to the specified low temperature level for 1 hour or longer.

   2. The unit temperature shall be increased to the specified high temperature level and operated for 1 hour or longer.

3. The temperature shall be reduced to ambient to complete one cycle of the burn-in test.
4. The transitions between low and high temperatures shall be greater than 1 °C per minute (average rate).

b. For items such as valves and thrusters, the number of cycles of operation rather than hours of operation shall be used to ensure detecting early failures and functional cycling conducted at ambient temperature.

NOTE For thrusters, a cycle is a hot firing which includes a start, steady-state operation and shut-down.

c. For hot firings of thrusters utilizing hydrazine propellants, flight valves shall be thoroughly cleaned of all traces of hydrazine following the test firings.

d. Devices that have extremely limited life cycles shall be excluded from burn-in test requirements (e.g. positive expulsion tanks).

6.1.12.3 Burn-in test levels and duration

6.1.12.3.1 Pressure

Ambient pressure should be used during the burn-in test.

6.1.12.3.2 Temperature

For cycling of electrical equipment, the extreme temperatures specified in subclause 6.1.10.4 and Table 3 shall be used.

6.1.12.3.3 Duration

a. The total operating time for electrical equipment burn-in shall be 250 hours including the operating time during the thermal cycling and thermal vacuum environment.

b. The minimum number of temperature cycles shall be 18, including those conducted during the thermal cycling acceptance test.

c. Additional test time beyond that required for thermal cycling shall be conducted at either maximum or minimum operating temperature.

d. A 48 hours high voltage test, performed at maximum operating voltage and maximum operating temperature, shall be part of the burn-in procedure.

e. The test shall be considered successful providing the the last 100 hours of the equipment burn-in test are free of failures.

f. For valves, thrusters and other components where functional cycling testing is a better burn-in method, a minimum of 100 cycles shall be conducted.

6.1.12.3.4 Functional test

a. Functional tests shall be conducted at the start of the burn-in test to provide a baseline reference for determining if performance degradation occurs.

b. The functional test shall be repeated after:

1. 150 hours of operation, and
2. during the last 2 hours of the thermal cycling test.

c. Parameters for all circuits, including all redundancy, shall be monitored during the entire test sequence.

d. On-off cycling of the electronic equipment shall be conducted during the test to simulate operational usage.

NOTE The reduction of system level failures by burn-in at the equipment level has a favourable impact on costs and schedules by stabilising the failure rate at or near its minimum and ensuring the highest probability of mission success.

6.1.13 Microgravity environment compatibility test, equipment acceptance

The microgravity environment compatibility test shall conform to subclause 5.1.19.

6.1.14 Audible noise test, equipment acceptance

The audible noise test shall conform to subclause 5.1.20.

6.2 Subsystem test requirements

a. Subsystem level acceptance tests need not be performed.

NOTE Subsystem level acceptance is performed at equipment and element levels.

b. Acceptance tests should be conducted at the subsystem level when the tests at this level provide a better perception of defects than the higher level tests.

c. The suitability of conducting subsystem level acceptance tests should be evaluated considering the relative accessibility of the subsystem and its equipment and the retest time at higher level.

NOTE Often items, such as solar array and apogee boost motors, are identified as subsystems. However, in this Standard they are considered as equipment and follow the requirements of subclause 6.1.

6.3 Element test requirements

6.3.1 General

Element level testing shall be performed on the following items:

a. Space vehicle,

b. Launcher, and

c. Ground segment

NOTE Ground segment tests are described in subclause 4.9.

6.3.2 Space vehicle test requirements

a. The space vehicle acceptance test baseline shall consist of all the required tests specified in Table 23.

b. The test baseline shall be tailored for each project, giving consideration to both the required and optional tests.

NOTE Optional tests are performed when they are considered appropriate with the goals and characteristics of the project.

c. For projects with a prototype model methodology, acceptance testing shall be performed on the vehicle flight models.

d. If protoflight or hybrid model methodologies are used, the protoflight qualification test campaign shall cover the acceptance objectives.

Table 23: Space vehicle acceptance test baseline

Test	Reference subclause	Suggested sequence	Space vehicle
Optical alignment	6.3.4.4	1 [a]	R [b]
Functional and performance	6.3.7.1, 6.3.7.2	2 [a]	R
Physical properties	6.3.4.3	3	R
Pressure and leakage	6.3.4.1, 6.3.4.2	4 [a]	R
EMC	6.3.6.1	5	O
Acoustic	6.3.3.3	6 [c]	R [c]
Random vibration	6.3.3.4.2	6 [c]	R [c]
Vacuum temperature cycling	6.3.5.1	7 [d]	R
Ambient pressure temperature cycling	6.3.5.3	7 [d]	O
Toxic offgassing	6.3.8.1	8	R [e]
Magnetic field	6.3.6.2	9	R [f]
End-to-end communication	6.3.7.4	10 [g]	R
Audible noise	6.3.8.2	11	R [e]
Mission specific	6.3.9	12	R [b]

R = Required

O = Optional

Notes

a At the beginning and end of the sequence and prior and after environmental tests.

b If applicable.

c Acoustic or random vibration test.

d Can be combined.

e For manned vehicle.

f If applicable and in case of stringent requirements.

g Can be combined with functional and performance tests at the end of the sequence.

e. The following general rules shall be taken into account in defining the space vehicle acceptance test program.

1. Thermal tests, which disclose latent defects, should be run towards the end of the sequence.
2. The functional verification of the space vehicle shall be performed by means of the integrated system test and the integrated system check (see 5.3.8.2.2 q.).
3. Functional verification shall be conducted before and after each environmental exposure, by means of the ISC.
4. The IST shall be carried out at the beginning and at the end of the test campaign.
5. The ISC shall be performed while the environment is being imposed, especially during thermal tests, since the space vehicle is expected to be operative under these conditions.
6. Alignment verifications, for all equipment having alignment requirements, shall be repeated during space vehicle acceptance test campaign.

   NOTE Alignment verification is repeated to track any degradation or to ensure that variation of equipment alignment in relationship with the reference axes remain within the specified limits.

7. All deployable items shall be subjected to deployment tests to verify the correct functioning of deployment mechanisms after integration and environmental exposures.
8. Audible noise and toxic offgassing tests shall be performed only for manned projects.

f. The environments used during acceptance tests shall be the maximum environmental exposures expected in service (see 4.8.1.3).

6.3.3 Structural acceptance tests

6.3.3.1 Static load tests

Static load tests need not be performed for space vehicle acceptance.

6.3.3.2 Spin tests

Spin tests need not be performed for space vehicle acceptance.

6.3.3.3 Acoustic tests

a. For acoustic acceptance test operations, the same requirements as for qualification tests (see 5.3.3.4) shall apply with the exception that the acceptance sound pressure spectrum shall be imposed as specified in the relevant launcher user's manual (if not defined, qualification levels minus 4 dB shall be applied).

b. The test duration shall be one minute (unless specified otherwise in the launcher user's manual).

6.3.3.4 Vibration tests

6.3.3.4.1 General

a. Low level vibration tests (also known as "signature tests") should be performed in order to detect latent materials or workmanship defects, and to compare the resonance frequency distribution with that of the mathematical model or modal survey.

b. Automatic notching shall be employed to prohibit excessive resonance build-up.

NOTE Any significant shift in resonance frequencies from those analytically determined is an indication of improper assembly or materials defects.

6.3.3.4.2 Random vibration tests

a. For random vibration acceptance test operations, the same requirements as for qualification tests shall apply (see 5.3.3.5.3) with the exception that the power spectral density shall be adjusted to the acceptance spectrum as specified in the relevant launcher user's manual (if not defined, qualification levels minus 6 dB shall be applied).

b. The test duration shall be one minute per axis (unless specified otherwise in the launcher user's manual).

6.3.3.4.3 Sinusoidal vibration tests

Sinusoidal vibration tests need not be performed for space vehicle acceptance.

6.3.3.4.4 Transient tests

Transient tests need not be performed for space vehicle acceptance.

NOTE Acoustic or random vibration tests are considered to provide sufficient relevant confirmation.

6.3.4 Structural integrity

6.3.4.1 Pressure and leakage

For acceptance pressure tests, the subsystems of the space vehicle shall be pressurized to the maximum expected operating pressures and kept at this pressure for a duration sufficient to establish that the leak rates are within the specified limits.

6.3.4.2 Boost pressure profile

Boost pressure profile tests need not be performed for space vehicle acceptance.

6.3.4.3 Physical properties

The physical property requirements of subclause 5.3.4.4 shall apply with the exception that the moment of inertia determinations need not be performed for space vehicles that are not spin stabilized.

6.3.4.4 Optical alignment

The optical alignment requirements of subclause 5.3.4.5 shall apply.

6.3.5 Thermal acceptance tests

6.3.5.1 Vacuum temperature cycling test

6.3.5.1.1 Purpose

The vacuum thermal cycling test detects material, process, and workmanship defects that respond to vacuum and thermal stress conditions and verifies the thermal control.

6.3.5.1.2 Vacuum temperature cycling test description

For the vacuum temperature cycling acceptance test, the qualification requirements (see 5.3.5.1) shall apply with the following modifications:.

a. The test temperatures shall be at the maximum and minimum design temperatures without any margin.

b. Cold start tests shall be performed only once.

c. The pumping, cooling, and heating speeds should be, as closely as possible, the same as those projected for the mission, but not exceed them.

6.3.5.2 Thermal balance tests

a. Thermal balance tests shall be performed on the flight article for the acceptance under the requirements and conditions defined for qualification testing in subclause 5.3.5.2.

b. Thermal balance tests need not be done for space vehicle acceptance provided such testing has been conducted as part of space vehicle qualification.

6.3.5.3 Ambient pressure temperature cycling test

6.3.5.3.1 Purpose

The purpose of the ambient thermal cycling test is to detect material, process, and workmanship defects by subjecting the vehicle to a thermal cycling environment.

6.3.5.3.2 Ambient pressure temperature cycling test description

For the ambient temperature cycling acceptance test, the qualification requirements (see 5.3.5.3) shall apply with the following modifications.

a. The test temperatures shall be at the maximum and minimum design temperatures without any margin.

b. Cold start tests, shall be performed only once.

c. The pumping, cooling, and heating speeds shall be, as closely as possible, equal to those projected for the mission, but not exceed them.

d. Only one cycle shall be performed in the thermal vacuum acceptance test.

6.3.6 Electromagnetic acceptance tests

6.3.6.1 Electromagnetic compatibility tests

a. If electromagnetic compatibility tests were performed during the qualification process, tests need not be performed for space vehicle acceptance.

b. If no space vehicle level electromagnetic compatibility testing is performed for its qualification, the requirements of subclause 5.3.7.1 shall apply to the acceptance procedure of the space vehicle with the exception of those tests that reduce the life of space vehicle equipment. In which case sufficient verification at lower level shall be performed.

6.3.6.2 Magnetic field measurements

Magnetic field measurements are performed on the space vehicle to demonstrate conformance to magnetic cleanliness requirements.

NOTE If there are no stringent magnetic cleanliness requirements, verification by analysis may be sufficient.

6.3.7 Functional acceptance tests

6.3.7.1 Mechanical functional tests

a. In addition to the functional tests performed with the vacuum temperature cycling test, a final functional test shall be performed under ambient environmental conditions for space vehicle acceptance prior to shipment to the launch site.

b. The final functional test shall conform to subclause 5.3.8.1 with the exception that only testing with nominal operational conditions shall be performed, i.e. no margin testing.

6.3.7.2 Electrical functional tests

a. A final ambient environment electrical functional test shall be performed for space vehicle acceptance prior to shipment to the launch site.

b. The final functional test shall conform to subclause 5.3.8.2 with the exception that only testing at nominal operational conditions shall be performed, i.e. no margin testing, and no testing at minimum and maximum voltage levels.

6.3.7.3 Mission simulation tests

Mission simulation tests shall be performed for space vehicle acceptance if flight software changes were implemented after qualification.

6.3.7.4 End-to-end communication tests

End-to-end communication tests under nominal operational conditions shall be performed for space vehicle acceptance according to the requirements of subclause 5.3.8.4.

6.3.8 Manned space vehicle specific tests

6.3.8.1 Toxic offgassing test

a. The purpose of the toxic offgassing test is to verify that the flight hardware does not produce toxic vapours that can build up to harmful levels for the crew in the closed loop life support system.

b. The toxic offgassing test shall be performed in accordance with subclause 5.3.9.1.

6.3.8.2 Audible noise test

a. The purpose of the audible noise test is to verify that the flight hardware does not produce audible noise levels that are detrimental to the crew health and safety.

b. The test shall be performed in accordance with the requirements in subclause 5.3.9.2.

c. The noise measurements shall be performed in nominal emission conditions.

6.3.9 Mission specific tests

a. For qualification see subclause 5.3.10.

NOTE Specific mission requirements can make additional system acceptance tests essential.

b. Due to the potential variety and complexity of mission specific tests they are not covered in this Standard.

6.3.10 Launcher tests requirements

a. Acceptance tests shall be conducted at equipment level in agreement with the general description of subclause 6.1 with the exception that test parameters (e.g. levels, duration and cycles) shall be tailored to the LV flight conditions.

b. Acceptance tests shall be conducted at subsystem level in line with the general approach of subclause 6.3.2 with the exception of the test parameters.

NOTE The LV equipment bay is the best example of the "element test" approach described in subclause 6.3.2.

c. Acceptance procedures shall be applied to every structural sub-assembly delivered for flight.

NOTE Tests on structural sub-assemblies include:

- proof pressure and tightness tests on pressurized tanks, pipes and valves,
- tests in dynamic environment on sub-assemblies,
- functional tests on mechanisms and access doors, and
- tightness tests on common tank bulkheads.

d. At element level, extensive acceptance functional tests shall be conducted according to the general principles of subclause 6.3.7 and conducted before delivery and during the launch campaign.

6.4 System test requirements

6.4.1 Purpose

The purpose of system acceptance tests is to cover aspects relevant to the interrelationship between two or more elements which constitute the system (i.e. space vehicle, launcher, ground segment) and to interfaces with other systems and projects.

6.4.2 System acceptance tests description

a. System acceptance tests include end-to-end tests between the launcher vehicle and the space vehicle, and space vehicle to ground segment compatibility (see clause 8).

b. Compatibility of all hardware and software elements of the overall command and control network, including the ground station(s), shall be demonstrated by a series of tests with the space vehicle in the various operative configurations.

c. System tests shall verify:

1. the capability of the ground station(s) to command and control the space vehicle;
2. the capability of the ground station(s) to perform tracking and ranging of the space vehicle;
3. the capability of the ground station(s) to receive and process the space vehicle telemetry data.

NOTE Details of the above tests are included in subclause 4.9.

7

Protoflight testing

7.1 Equipment test requirements

a. If the qualification equipment is planned for flight use, a protoflight approach shall be applied.

b. The equipment qualification testing shall be modified from that specified in subclause 5.1 to reduce cyclic stress levels.

c. For protoflight testing, the equipment acceptance tests defined in this Standard need not be used with the exception of the burn-in acceptance tests (see subclause 6.1.12) and the qualification test baseline (see subclause 5.1) shall be applied with the following modifications:

   1. For the equipment thermal vacuum test (see 5.1.15), the temperature extremes shall be 5 °C greater than the minimum and maximum acceptance temperatures (see 6.1.10).

   2. For the equipment thermal cycling test (see 5.1.16), the temperature cycles shall be performed at 5 °C greater than the minimum and maximum acceptance temperature (see 6.1.11).

   3. For the equipment sinusoidal vibration test, the test levels shall conform to subclause 5.1.10 with the exception that the number of sweeps shall be one sweep up and the sweep rate shall be 3 octave/min.

   4. For the equipment random vibration test, the test levels shall conform to qualification levels in subclause 5.1.11 with the exception that the test duration shall be those for acceptance testing on each axis.

   5. For the equipment acoustic test, the test level shall conform to qualification levels in subclause 5.1.12 with the exception that the test duration shall be those for acceptance testing (i.e. one minute).

   6. For the equipment pyro shock test, the shock spectrum shall conform to subclause 5.1.13 with the exception that the number of shocks shall be one per direction.

   7. For the equipment pressure test (see 5.1.8), only proof pressure tests as defined in subclauses 5.1.8.3 a. and 5.1.8.3 b. shall be conducted.

   8. A burn-in test shall be performed if the total duration of the protoflight qualification test sequence is insufficient to detect material and workmanship defect occurring in the equipment life. The total duration time shall be 250 hours minimum.

7.2 Subsystem test requirements

a. Subsystem protoflight testing should be performed according to subclause 5.2.

b. Environmental testing shall be performed for the qualification and acceptance of the structure and thermal control subsystems but may be replaced by protoflight tests of these subsystems at a higher level, in combination with other subsystems during environmental test campaign.

c. Environmental protoflight test campaign can be conducted at subsystem level where this level of testing provides a realistic or practical test simulation (e.g. design qualification of optical and telecommunication payloads).

d. Functional testing shall be performed with each subsystem assembled on a test bench which provides a flight representative environment (hardware and software) covering function, interfaces and performances.

7.3 Element test requirements

7.3.1 Space vehicle tests

The requirements of subclause 5.3 and Table 17 shall apply for the protoflight testing of space vehicle, with the following modifications:

a. The shock test shall be conducted according to subclause 5.3.3.6, with the exception that only 2 repetitions of activated events shall be performed.

b. The acoustic or random vibration tests shall be conducted according to subclauses 5.3.3.4 and 5.3.3.5.3, with the exception that:

   1. the duration factor shall be 2 (instead of 4),
   2. the duration shall be not less than one minute, and
   3. the level margin for the flight environment shall be 3 dB.

c. The thermal vacuum test and the ambient thermal cycling test shall be performed according to subclauses 5.3.5.1 and 5.3.5.3, with the exception that the hot and cold temperature extremes shall be those used for qualification and the number of cycles shall be those used for acceptance (see Table 2).

7.3.2 Launcher tests

a. Protoflight test requirements should be applied at lower level than at the launch vehicle element level and subclauses 7.1 and 7.2 shall apply.

b. Element level tests should not be regarded as protoflight tests.

7.4 System test requirements

Protoflight test requirements need not be applied to systems.

NOTE System tests are not fundamentally environmental tests.

8

Pre-launch testing

8.1 General

8.1.1 Purpose

The purpose of pre-launch testing is to verify by end-to-end tests that each critical path in the system before launch is satisfactory, i.e. there are no out-of-tolerance conditions or anomalous behaviour, and to demonstrate successful integration of the space element with the launch element.

NOTE The pre-launch tests mainly affect space vehicles and launcher elements.

8.1.2 Pre-launch test objectives

a. Pre-launch tests shall verify that:

1. no damage or performance degradation of the space vehicle and its constituents has occurred during shipment or handling;
2. all launch site assembly activities are completed properly, all associated interfaces are verified, and their parameters are within the specified limits;
3. mating with the launch vehicle are completed successfully, i.e. all interfaces between the space vehicle and the launch element, and between the launch element and the ground support facilities are verified.

b. The pre-launch tests shall provide data for trend analysis.

NOTE Trend analysis data can give evidence of a problem, even though all measurements were within tolerances.

c. The pre-launch test flow shall follow a progressive growth pattern to ensure the correct operation of each element involved (e.g. flight, launch and ground) prior to progressing to a higher level of assembly and test.

d. Pre-launch testing should consist of a partial repetition of acceptance test activities.

e. In cases where end-to-end testing cannot be performed with the flight hardware, simulation devices can be used to exercise the flight hardware to the maximum extent possible.

NOTE Not all end-to-end tests can be performed with only flight hardware (e.g. pyro devices and apogee burst motors).

f. Simulation devices should be carefully controlled and used only when there is no feasible alternative for conducting the test.

g. Redundancies shall be verified.

NOTE The extent of pre-launch testing, the appropriate test sequences and the test procedures are unique for each launcher and for each project.

8.2 Functional tests

8.2.1 General

The functional tests performed during pre-launch phase should repeat the mechanical and electrical functional tests performed for acceptance.

8.2.2 Retesting

a. The assembly at launch site shall be retested if it was disassembled for transport or other reasons (e.g. batteries).

b. Comparison with the correspondent results recorded during acceptance testing shall be performed to check for any degradation of the system.

8.2.3 Pyro devices

a. Verification that the correct ignition energy levels are present at each pyro device shall be performed prior to final connection of the firing circuit to the device.

b. A simulator of the pyro device characteristics shall be used during these tests.

c. Circuit continuity and stray energy checks shall be made prior to connection of any firing circuit to any pyro device.

d. Circuit continuity checks shall be repeated whenever that connection is opened and prior to re-connection.

8.3 Propulsion tests

8.3.1 Purpose

The purpose of the functional tests of propulsion subsystem is to verify the correct operation of all equipment after the propellant filling.

8.3.2 Leakage rates

Propulsion subsystem leakage rates shall be verified to be within specified limits by pressure monitoring.

8.4 Integrated launch system test

a. Launch readiness of the system shall be verified by a fully integrated launch system test.

NOTE The functional tests to be performed during this test are generally determined by launch vehicle requirements.

b. The integrated launch system test shall include an evaluation of radio frequency interference between system elements, electrical power interfaces, command and control functions.

8.5 Manned related tests

Final crew interface verification in all operational configurations shall be performed.

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9

In-orbit testing

a. The in-orbit testing shall be performed for projects whose characteristics (e.g. mission and flight operations) require in-orbit verification.

NOTE In-orbit tests supplement ground testing by providing operating conditions that cannot be fully or cost effectively duplicated or simulated on ground.

b. In orbit test shall :

1. Verify the flight performances with the mission requirements and ground element interface performances.
2. State at the end of the tests, the operational readiness of the system together with the recommendations for further actions to achieve the full operational status.

   NOTE For particular types of space vehicles this is known as the commissioning process.

3. Verify particular mission phases, e.g. orbital and re-entry.
4. Verify the requirements prior to the flight using different methods.

   NOTE The flight test is performed only after the ground verification has shown that the in-orbit verification can be successful.

5. Verify crew operations.
6. Verify functional paths not normally used (at periodic intervals).
7. Verify the system after any replacement or repairs.
8. Verify the compatibility for infrastructure elements if not already achieved (e.g. elements developed in different phases of the infrastructure project).
9. Provide the following data to validate mathematical models and simulations for subsequent analysis verification completion using dedicated instrumentation for data collection:
   (a) thermal and structural data for thermal and structural mathematical models updating,
   (b) data on aero-thermal behaviour during re-entry,
   (c) aerodynamic and inertial load data, and
   (d) landing load data.

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10

Post-landing testing

a. Post-landing testing shall be performed for:

   1. multi-mission projects, and

   2. for re-entry vehicles.

b. Post-landing tests should be performed to acquire statistical data on element performances and its degradation (lesson learnt).

c. In the event of long duration storage, the post-landing tests shall verify the items that are recycled between missions.

   NOTE Long duration storage is covered by subclause 4.7.4.

d. The recycled items shall be re-verified between missions to assure their readiness for the subsequent flight as follows:

   1. High level (system or element) functional testing, at the maximum extent possible, should be performed instead of individual low level tests.

   2. All safety critical functions shall be re-verified.

   3. Equipment which was disturbed shall be functionally and environmentally tested (e.g. connectors mated or de-mated and modifications).

e. Items that are not reusable and that have failed or degraded while in orbit, shall be examined, analysed and tested to determine why failure or degradation occurred.

f. Items that failed shall be analysed to determine what can be changed to prevent a recurrence (see subclause 4.7).

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Annex A (informative)

Mechanical and vibration testing for space equipment

A.1 Derivation of qualification test levels for units from mechanical test data analysis

A.1.1 Introduction

For equipment integrated in hardware to be launched on ARIANE or on STS, a first evaluation of test levels based on scrutiny of environmental test specifications and also of NASA Standards was performed (see references. A.1.4.1) in order to construct an "envelope".

To improve the specifications for sinusoidal and random testing at unit level, a second evaluation was based on a statistical analysis using test data obtained from sine, random and acoustic testing at system level (see references A.1.4.2 and A.1.4.3).

For the latter purpose, a database was first prepared using ITS-DIVA software, for which seven major European satellite projects (METEOSAT, MARECS, ISPM, ECS, GIOTTO, TELECOM 1 and SPOT) provided approximately 1500 response curves and associated information.

For the random and acoustic test results, the random response spectrum (RRS) concept was used extensively. A statistical evaluation was then made on the basis of a log-normal distribution for various selections of curves using the parameters available in the database.

The final equipment classification was deduced from these results and the corresponding specifications based on a 95 % confidence level (see Figure A-1) were defined.

The acoustic investigation results highlighted the importance of two particular parameters for unit-level random testing:

- the equipment mass, and
- the external panel in bending compared to other results.

In order to have a limited number of classes, the masses were divided in two categories, leading to four basic classes.

A coefficient of four (4) appeared with the power spectral density(PSD) specifications between smaller masses (less than 3 kg) and the higher ones.

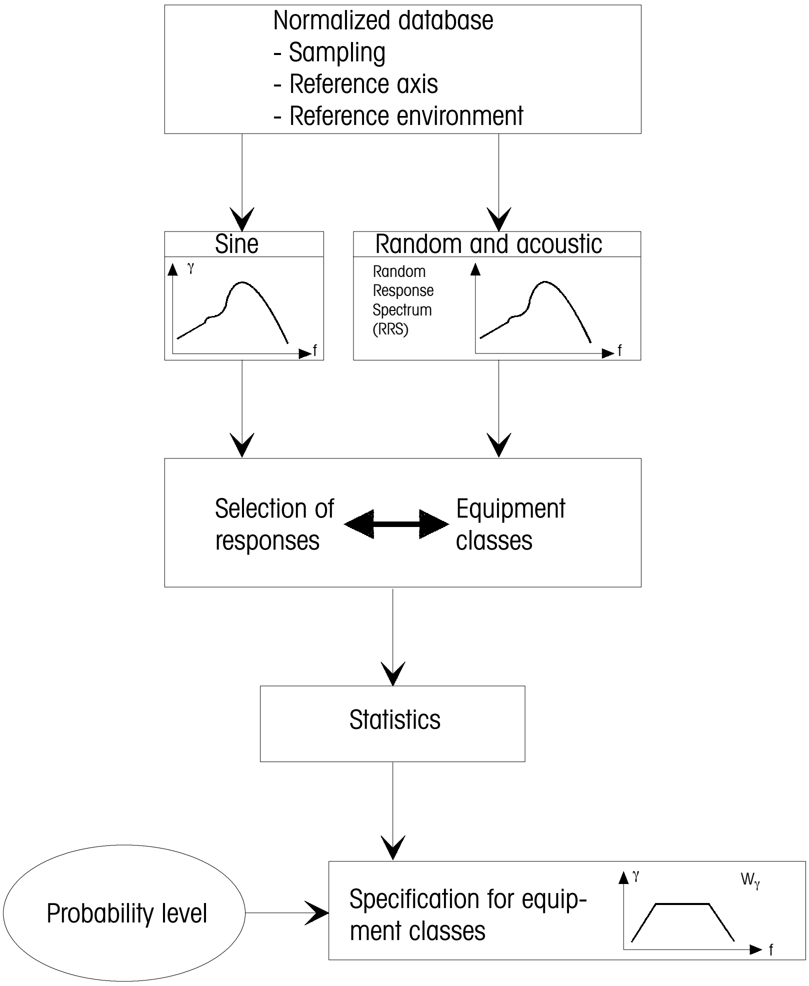

Figure A-1: Data processing arrangement

For unit level sinusoidal testing, statistics for all responses parallel to the excitations showed relatively limited levels in the vicinity of first satellite global mode (longitudinal or lateral) due to the notching procedures, and lower levels at the other frequencies. The specifications were given for the same categories of mass as previously, assuming that the effects of the first satellite mode are limited to 30 Hz for the first lateral mode and 60 Hz for the first longitudinal mode.

All of these results were updated using the database supplemented with data from the HIPPARCOS and OLYMPUS projects (about 300 response curves) and improved data processing (see references A.1.4.2 and A.1.4.4), mainly:

a. a statistical analysis specifically adapted to the sinusoidal test data in the vicinity of the satellite first resonant frequency, for which the previous approach was found to be not well adapted;

b. analysis in more detail of the impact of the main parameters governing the equipment responses, particularly the equipment mass for which it was interesting to extrapolate in the 20 to 50 kg range for the next generation of satellites.

A detailed analysis of the HIPPARCOS and OLYMPUS acoustic response curves showed that only equipment directly mounted on honeycomb panels and in areas with sufficient equipment density can be specified with reasonable reliability in the present context. Equipment located on small brackets (such as thruster, sensor) or isolated from the others on large panels can have higher levels generated by local resonances. The conclusion was to remove all “non-panel” accelerometers from statistical analyses. It was also noted that the test data were clearly valid for equipment qualification purposes only if the accelerometers were very close to the equipment fixation on the panel. Accelerometers mounted on the equipment itself, between equipment items or between distant fixations tended to produce questionable data.

A study was placed by ESA with INTESPACE, with the objective of improving and analysing the mechanical database (see reference A.1.4.5).

The mechanical database was provided with data related to SOHO sinusoidal and acoustic test results and associated data. A statistical analysis was performed on the whole database (13 specimens from 12 satellite projects). The results globally confirmed the validity of the equipment classes selected for specifications in random: "external bending" and "other", and continuous function of mass. In addition, both sinusoidal and random levels results were significantly changed, mainly due to the data from generally heavier satellites.

Evaluation based on the complete database, according to the previous considerations led to the results explained in subclauses A.1.2 and A.1.3.

A.1.2 Unit level random testing

A.1.2.1

For each of the two classes: "external panels in bending" and "others", a statistical analysis on acoustic test data was performed to derive specifications in the form of continuous function of mass. Two complementary approaches were used for better reliability:

a. a method using close masses: for a given mass, statistics were performed using curves corresponding to masses close to the considered value;

b. a method using normalization: statistics were performed on all the curves normalized by a function of the equipment mass representing the dependence of the PSD level; this function was selected from various considerations:

   1. energy and extrapolation techniques,
   2. asymptotic value representing the minimum PSD for large masses,
   3. plots showing levels versus mass,
   4. results from the first method, and
   5. simplicity.

   leading to:

$$\mathrm{PSD}(M) = \mathrm{PSD}(1\ \mathrm{kg}) \times \mathrm{f}(M)$$

where

$$\mathrm{f}(M) = [(1 + m)/(1 + k \times m)] \times [(M + k \times m)/(M + m)]$$

m = driven mass from the panel

k = PSD(0)/PSD(∞)>1 ratio between extreme values.

A.1.2.2

Data evaluation using the method above (A.1.2.1 b.) corroborated by the first method (A.1.2.1 a.) led to m = 1 kg and k = 20, giving the specifications of subclause 5.1.11.3.

These specifications apply only to equipment directly mounted on honeycomb panel and in areas with sufficient equipment density (if not, adequate corrections are used to come to this case).

This represents a significant improvement on the qualification of units by random vibration testing and it is very easy to use. It can be seen that, for lateral axes of equipment located on "external panels" and for all axes of "other" equipment:

a. The maximum values for very small masses, less than 0,5 kg, are close to the relatively high value of 1 g^2/Hz.

b. For medium masses, between 1 kg and 10 kg, the specifications are decreasing significantly (factor of 4 between 1 kg and 10 kg).

Extrapolation to large masses gives a slow convergence towards the asymptotic PSD value of 0,05 g^2/Hz (e.g. .a mass of 30 kg gives 0,08 g^2/Hz). However, the approach is valid only if the equipment is rigid. If some very flexible parts are included in the unit, M does not include the corresponding masses and represents the rigid part driven by the fixations. 50 kg seems to be an extreme limit in the present context, giving a minimum value of 0,07 g^2/Hz.

Significantly higher values exist in vertical axis for equipment located on "external panels" (asymptotic value of 0,12 g^2/Hz).

A.1.3 Unit level sinusoidal testing

During sinusoidal tests at system level, as opposed to random or acoustic tests, the responses depend mainly on the overall characteristics of the satellite and also on the notching criteria. Consequently, a very large number of parameters are involved in the equipment behaviour and a statistical approach can only give global results, which were treated with caution in the present context.

Concerning the influence of notching on the satellite responses, leading to a serious problem for database normalization, among the possible assumptions, the following one was selected for its logic and its simplicity: "the levels occurring during notching cannot be exceeded", leading to relatively limited levels in the vicinity of the first satellite global mode. In these frequency bands (up to 30 Hz in lateral and 60 Hz in longitudinal), selecting the resonance levels and performing statistics on these particular data was found to be a better approach than using statistics at each sampled frequency (better log-normality and more efficient analysis).

Data evaluation led to the specification of subclause 5.1.10.3. The 20 g below 60 Hz, independent of the equipment mass, are governed completely by the first satellite longitudinal mode.

In addition, the normalization approach was used to determine the influence of the parameters on the resonance levels. The only case showing no ambiguity was the distance to interface for lateral levels. If specifications had to be derived in that case for equipment having a known location, it can be the following:

Distance $z < 1$ m	6 g
Distance $z \geq 1$ m	7 g $\times$ z

In fact, extrapolation above 3 m, and even 2 m, can be pessimistic: the available results show limited levels above 2 m which can be a direct effect of notching. The levels at large distances from interface strongly depend on the design of the payload and on its notching philosophy.

A.1.4 References

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INTESPACE DO 94.186 EI/ED, December 16, 1994.

A.2 Vibration test philosophy

The random acceptance vibration spectrum (AVT) is derived from the successful test approach used for GEMINI and APOLLO, applying standard qualification (QAVT) and acceptance (AVT) test levels. This approach was used on shuttle and Spacelab with success[1)]. The philosophy is to use for acceptance a standard random vibration flight environment.

The AVT approach is different to the ARIANE test approach where flight acceptance test levels are as high as the expected flight environment.

For the GEMINI and APOLLO programmes the qualification level (QAVT) were higher by an appropriate factor, see the Table A-1 and the Figure A-2.

Table A-1: QAVT and AVT levels

Frequency (Hz)	QAVT	AVT
20	0,017 g^2/Hz	0,01 g^2/Hz
20 – 80	3 dB/octave	3 dB/octave
80 – 350	0,067 g^2/Hz	0,040 g^2/Hz
350 - 2000	-3 dB/octave	-3 dB/octave
2000	0,012 g^2/Hz	0,007 g^2/Hz
Test levels (r.m.s.)	7,87 $g_{r.m.s.}$	6,06 $g_{r.m.s.}$

The QAVT was slightly modified for Spacelab, the expected vibration levels, which were based on the Orbiter generated acoustic noise of 145 dB, were calculated by using the extrapolation method of "Barret" and the QAVTs were considered for the different zones within Spacelab. In the case where the calculated level was below the QAVT and the QAVT remained valid. Figure A-3 shows an example of the calculated test level for the zone "pallet hardpoints" plus the QAVT on Spacelab.

The random acceptance vibration test (AVT) has no relationship to the expected mission. The test level (6,06 $g_{r.m.s.}$) and test time 2 minutes per axis were selected since experience from GEMINI, APOLLO and Spacelab programmes showed that equipment exposed to these levels incurred a minimum number of check-out and flight problems and is sufficient to verify the workmanship.

As a compromise it is proposed that an AVT level of 0,04 g^2/Hz is the minimum test level limited to the qualification levels given in subclause 5.1.11.3 divided by the qualification factor $(1,5)^2$. These levels are sufficient to detect failures in workmanship as it was proved on the verification studies of the Spacelab program.

1) Study on Spacelab AIV programme generally applicable to other spacecraft programmes.
ESTEC Contract number 4980/82/NL/PP(sc), November 1982.

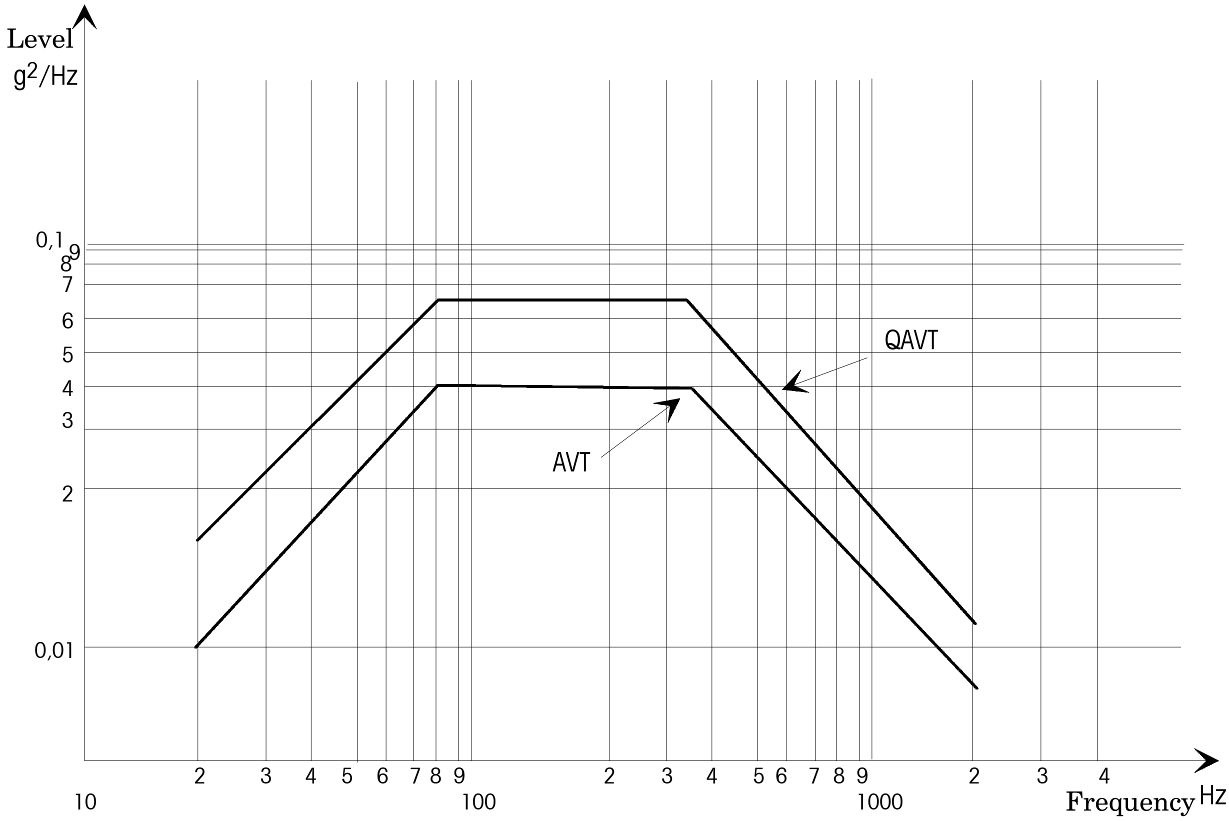


Figure A-2: Qualification and acceptance vibration test curves (QAVT/AVT)

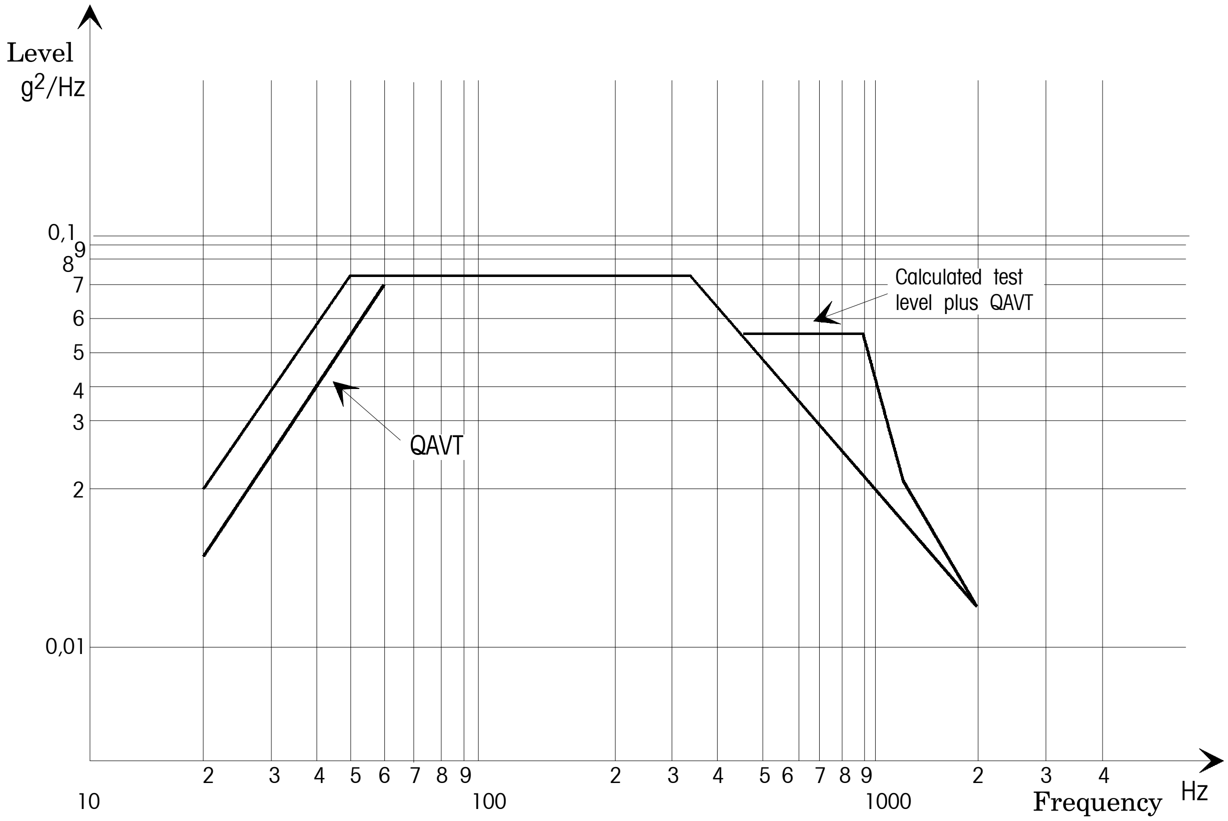


Figure A-3: Example of a calculated test level for zone “pallet hardpoints” plus QAVT (Spacelab)

A.3 Technical information on mechanical test factors

A.3.1 Introduction

This Standard is not intended for design but it is worth explaining the definition and approach on mechanical test factors used at system level.

For example in ESA the mechanical design requirements are defined by the system project engineers in a document called generally: "Mechanical and thermal design requirements specification". This document states the design philosophy adopted and the references.

In ESA, spacecrafts are designed to quasi-static loads, but this can change due to the evolved complexity in payloads and the structural components leading to a decrease in the stiffness.

With the present methods, designers defined for flight components the most critical quasi-static loads, which are the superposition of static loads and maxima or minima of the dynamic loads in the different directions. The validity of the dimensioning to the load is verified by a coupled satellite and launcher analysis. The following subclauses define the references, loads and margin of safety.

A.3.2 References

As a consequence of this design philosophy, designers consider a reference for quasi-static loads and a different reference for dynamic loads. The starting point is to consider the limit loads defined as:

$$\text{Limit Load} = Q_l$$

Where Q_l is the maximum expected load during the satellite life on the ground, during launch and in orbit. If a Gaussian law is assumed for the load distribution, the limit load is at least equivalent to a load having a probability not exceeding 99 % (2,3 σ) The limit loads are deducted from the first payload and launcher coupled analysis.

The launcher user's manual considers the flight worst case load which is the extreme flight load due to the launcher. If ARIANE 4 or 5 User's Manual is used, it is defined as:

$$\text{Flight worst case loads} = Q_f$$

Where Q_f is extreme flight loads which are applied to a payload and are evaluated as being the loads, which are not exceeded with a probability of 97,7 % at 2 σ estimates considering a Gaussian distribution for the loads.

In the case of ARIANE for dynamic testing, it appears that the flight worst case loads are considered to be the flight acceptance loads (Q_a).

NOTE The definition of Q_f can change with launchers other than ARIANE 4 or 5. Problems can happen when determining the flight loads to be applied on a payload which can be launched by different launchers. It becomes very important to state the approach taken for the reference.

A.3.3 Loads

Starting with the limit loads, the following loads are defined:

The qualification loads $Q_q = K_q\ Q_l$

There are the loads for testing for qualification of design.

The yield loads $Q_y = K_y\ K_q\ Q_l$

The ultimate loads $Q_u = K_u\ K_q\ Q_l$

where

K_q is the safety factor for qualification

K_y is the safety factor for yield

K_u is the safety factor for ultimates

NOTE 1 No deformation above 0,2 % is introduced when applying the yield loads, no buckling or rupture is introduced when applying the ultimate loads.

NOTE 2 In the relevant cases K_y and K_u for the secondary structure, an additional factor is applied.

NOTE 3 It is generally accepted that $K_q = 1{,}5$, $K_y = 1{,}1$, $K_u = 1{,}25$ for design and testing of the primary structure.

NOTE 4 For the dynamic testing it is generally adopted in ESA that $Q_l = Q_a$ and therefore:

$$Q_q = K_q\ Q_a \text{ with } K_q = 1{,}5.$$

A.3.4 Margin of safety

All structural elements are designed to exhibit a positive margin of safety (MS) with respect to yield and ultimate loads. The margin of safety is defined as the ratio of the allowable load (or stress) to the applied load (or stress):

MS = [(Allowable load (or stress))/(Applied load (or stress))] – 1

A.4 Microgravity environment compatibility (MEC)

A.4.1 Introduction

The microgravity environment is created by inertial forces due to residual accelerations. These accelerations are caused by natural perturbations (such as the gravity gradient field and the remaining atmospheric drag) and self-induced perturbations (such as the attitude control, orbital manoeuvre, on-board machinery operation and crew handling and motion).

Self-induced accelerations can also be caused by the operation of mechanisms via momentum exchanges and other perturbation forcing functions interacting with the spacecraft system.

The spacecraft (system, subsystem and equipment) and payload designated for the performance of experiments and processes in microgravity environment are designed such that the natural and induced acceleration levels do not exceed the specified limit acceleration level at sensitive micro-g payload locations during their operation.

A.4.2 Disturbance source definition

A.4.2.1 Mechanisms

The operation of mechanisms during microgravity sensitive periods can introduce disturbance forcing functions which drive the spacecraft rigid body and the flexible structure with quasi-static and dynamic accelerated motions. Any mechanical component which changes its velocity creates a reaction force according to the basic Newton's law:

$$F = m\,a$$

In addition to the momentum exchange during accelerated motions, parasitic forces exist, created by for example friction effects, non-linearity, hysteresis, unbalances, misalignments of drive elements and torque variations.

The mechanism disturbance sources are grouped in three categories:

a. Category A: rotating mechanisms (e.g. motors and centrifuges);
b. Category B: mechanisms with dominating translation motion (e.g. sample exchange and manipulator);
c. Category C: mechanisms for support functions (e.g. cooling fluid and air systems).

A.4.2.2 Crew operations

The typical crew handling activities (e.g. reconfiguration, servicing and maintenance) can introduce disturbances to the microgravity environment.

A.4.3 MEC control approach

A.4.3.1

The payload design parameters that have an impact on the microgravity quality are primarily:

a. accelerated motion interaction with the spacecraft rigid body;
b. internal mechanism forcing functions due to mechanical and fluid sources, exciting flexible spacecraft structural vibrations with harmonic, stochastic and transient accelerations;
c. mechanism direct acoustic radiation exciting vibro-acoustic susceptible structures within pressurized compartments of the spacecraft with stochastic vibrations.

A.4.3.2

The MEC verification is based on standard methods, such as:

a. review of design,
b. analysis, to be performed where applicable, and
c. test, to be performed where required.

The tests are performed in any case when other verification methods do not provide reliable results.

The outputs of single source impacts gained in the time domain and in the frequency domain are superimposed separately. Time domain superimposition is made with linear superimposition, considering the phase, if possible.

Frequency domain superimposition is made on the basis of the root-sum-squared (RSS) values.

Quasi-static micro-g disturbance impacts in the time domain are analysed considering the interaction of disturbance forces and torques with the spacecraft as a rigid body at frequencies below 0,01 Hz.

Microgravity dynamic disturbances are controlled in the frequency domain over a range from 0,01 Hz to 500 Hz. The transmission of dynamic disturbances within the spacecraft from the source to the receiver locations is described by input to output transfer functions, covering both the structure-borne and the air-borne transmission paths between the source and the receiver locations.

The definition of source interface forces and torques in the time and frequency domain, together with the radiated sound power limit, as derived from the acceleration budget limits after application of spacecraft system transfer functions, simplifies the design and verification process for the source.

Transfer functions are given as accelerations to force relations for structure-borne vibration transmission paths or acceleration to sound power relations for air-borne transmission paths as worst case envelope functions between any source and any receiver location.

The vibro-acoustic disturbance transfer functions are under spacecraft system level control and need update when first dynamic response calculations are available or first structural test models are prepared for transfer function measurements.

The determination of disturbance sources forcing functions is the most important aspect of the microgravity environment control approach , and are determined by analysis or test.

Analysis can select design and performance parameter and calculate interface reaction forces and torques.

MEC tests provide measurements of interface forcing functions from quasi-static up to higher frequencies, as far as structure-borne transmission path is concerned, and the measurements of radiated sound power, as far as the air-borne transmission path is concerned.

Microgravity dynamic disturbance forcing functions in the frequency domain 0,01 Hz to 500 Hz are summed over the source mounting feet according to

$$F_i(f) = \sqrt{\sum F_i^2(f)}$$

such that a root-sum-squared (RSS) level results for the source over all mounting feet and directions.

The dynamic disturbance acceleration spectrum due to source excitation and a single transmission path results after multiplication with the transfer function for a forced vibration response:

$$a_i(f) = H_i(f)\, F_i(f)$$

and for a vibration response due to sound power:

$$a_{wi}{}^2(f) = H_{wi}{}^2(f)\, L_{wi}(f)$$

Superposition of single source responses applies the root-sum-square (RSS) approach:

$$a(f) = \sqrt{\sum (a_{i,w})^2(f)}$$

The inverse application of transfer function approach with the allowable acceleration limits as input leads to the derivation of allowable interface force spectra and the allowable radiated sound power of the disturbance source.

The spacecraft system transfer functions to take be considered are:

- Acceleration-to-force functions for structure borne vibration transmission paths, valid for any source-to-receiver location and direction. It represents a worst case envelope over any transfer function, with equal input and output point impedances.
- Acceleration-to-sound power functions for airborne vibration transmission paths, valid for any source-to-receiver location and direction.

Annex B (informative)

Temperature limits and test levels for space equipment

B.1 Introduction

Several studies were performed in the field of the thermal environment for system and equipment tests.

The objectives of these studies were to standardize the test conditions for equipment thermal vacuum and cycling test with the goal to reduce failure rates at system level tests and in orbit and qualify equipment independently from their application in later programmes.

In order to achieve the objectives and to produce a standard environment test specification, two main tasks were considered:

a. The analysis of the influence of the temperature limits on the thermal design and the philosophy of temperature uncertainty application and determination.
   An approach for standard unit temperature limits was made, with the goal to extend the thermal control design margins and allow a reduction of the thermal verification test at system level in combination with other preconditions.

b. An approach was made to standardize the test conditions. The test parameters investigated were:
   1. temperature test duration imposed by the number of cycles and dwell time, and
   2. test environment conditions.

B.2 References

B.2.1 J. WEYDANDT "Standardization of Temperature Limits and Test Levels for Spacecraft Equipment"
MBB/ERNO – AIV STUDY – April 6, 1984.

B.2.2 J. WEYDANDT, L. KERSTEIN "Applicability of the Spacelab Approach for Equipment and System Test for Ariane 4 Class Satellites"
MBB/ERNO – AIV STUDY – Prepared for ESA – 1983.

B.2.3 Institute of Environmental Sciences
Environmental Stress Screening Guidelines
ESSEH 1981.

B.2.4 USAF
MIL-STD-1540 B
Military Standard: Test Requirements for Space Vehicles.

B.3 Influence of equipment temperature limits on thermal design

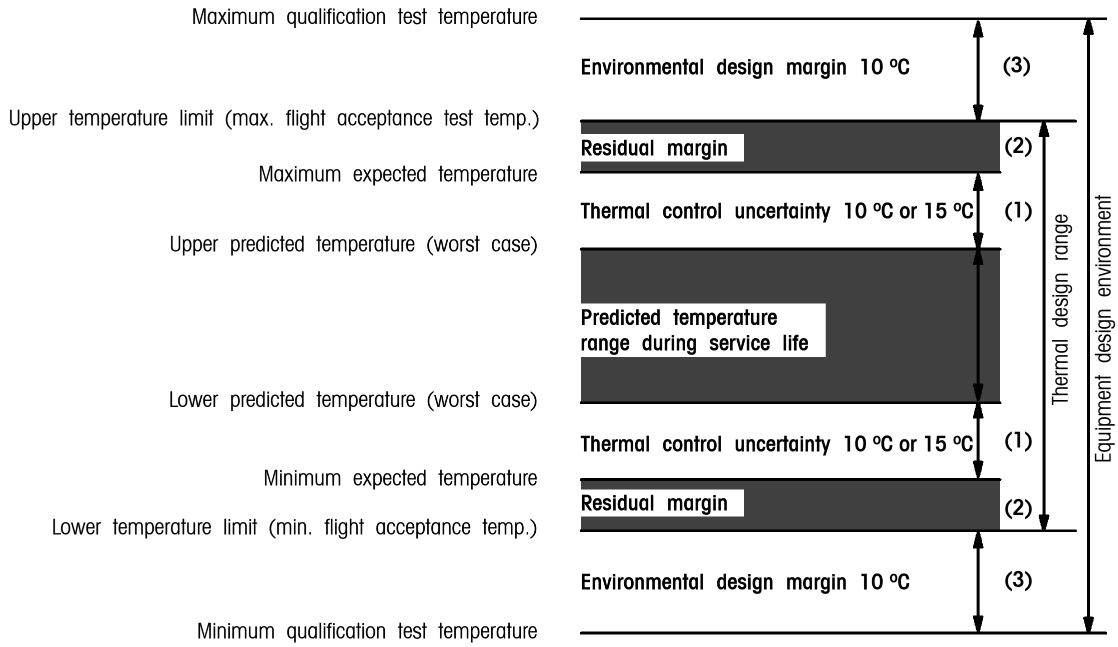
The general definition of temperature limits and margins of a spacecraft thermal design are summarized in Figure B-1. The bandwidth between the upper and lower temperature limits defines the thermal design range and the type of thermal design to be used. The smaller the bandwidth, the greater the analytical effort is performed to ensure the limits, and the more likely the requirement for an active or semi-active control.

The approach in defining limits in certain cases is very different between US and European programmes, with European programmes usually having the more restrictive temperature limits. Further complications arise when a programme uses certain equipment from a previous project that cannot exactly complement the limits pertaining to the other equipment.

Ideally, a trade-off is made early in each project, either to

a. allow large temperature limit bandwidths, with the resulting costs and effort to qualify the components, in conjunction with a simple passive thermal control subsystem, or

b. to impose a narrow temperature limit bandwidth and transfer the costs and the effort to the thermal control subsystem to provide an active control system to maintain equipment within limits.



Notes

b Thermal control uncertainty depends on type of equipment:
10 °C uncertainty for thermal control verified by test,
15 °C uncertainty for thermal control unverified.

3. Residual thermal design margin can be zero.

4 10 °C margin does not include test condition tolerances

Figure B-1: Temperature limits and margins definitions

B.4 Verification by analysis concerning accuracy and level of confidence

B.4.1 General philosophy of uncertainty application

Performance of electronic and electromechanical spacecraft equipment and subsystems can be verified on the ground, by measurements of input and output signals.

The performance of the thermal control, i.e. its capability to maintain temperatures of the spacecraft subsystems within the desired limits, under all operating and environmental conditions, can only be predicted by thermal analytical methods and is verifiable only in orbit.

Consequently any thermal design can provide only a limited assurance that the predicted temperatures encompass all the events and conditions of the spacecraft lifetime. To minimize this uncertainty thermal engineers apply analysis techniques involving the simulation of the satellite by a thermal mathematical model which is usually "tuned" by a series of ground tests to a specific level of accuracy before being adapted for predicting in-orbit performance. The limitations of this method are recognized and accepted by the aerospace industry as the uncertainties with the input parameters.

The input parameters can be categorized in three main classifications:

a. Environmental parameters

   The environment of the spacecraft (which is temperature controlled) during its life from launch to end of operation is not perfectly known due to the uncertainty of solar intensity.

b. Spacecraft physical parameters

   The temperature of spacecraft is controlled through conductive and radiative heat transfer paths. All parameters that describe these transfer paths are subject to uncertainties that are due either to

   - measurement tolerances, or
   - fabrication tolerance, or
   - measurement and fabrication tolerances.

   The uncertainties range from tolerances of thermal conductivity of material lots to measurement accuracy of thermo-optical properties through cross section and dissipation tolerances.

   Whereas these uncertainties can be limited by applying boundary requirements at the beginning of a project (e.g. sheet thickness tolerances), other groups of parameters cannot be predetermined at all for a particular spacecraft. Instead they are deduced from experiences with previous spacecraft and from tests conducted, e.g. the overall thermal conductance through MLI assemblies and the interface conductance from units to structures.

   Arguably the most important uncertainty for satellites in orbit is the change of material thermo-optical properties during the mission life.

c. Mathematical model uncertainties

   The heat balances of components within the spacecraft are governed by a set of nonlinear differential equations for which no closed integration solution is available. Therefore numerical solutions are employed based on finite difference methods.

The employed techniques induce a wide variety of uncertainties including:

- The definition of finite, isothermal elements is physically incorrect.
- The methods to determine the conductive and radiative exchange between these isothermal elements are only approximate.
- The numerical solution to the system of differential equations has only a finite accuracy.

In addition, a special set of uncertainty parameters is allocated to account for test conditions during and after solar simulation test (SST) when correlation and mathematical evaluation of the thermal mathematical model (TMM) is performed.

These uncertainty figures are mainly the result of assumptions for the chamber wall temperature and for example:

- its thermo-optical properties,
- solar simulation intensity,
- distribution and spectrum composition,
- test adapter temperatures, and
- temperature sensor reading accuracy.

This uncertainty figure is replaced after test evaluation by the orbit uncertainties for improved orbit temperature predictions.

It is difficult to determine with high accuracy the magnitude of each individual uncertainty, and even given this information, the accumulation of these individual contributors to the overall uncertainty still remains uncertain.

One of the main tasks in thermal design is to make the unit as insensitive to these parameter uncertainties as possible, and to set up a verification programme that defines and reduces uncertainties by measurement of important parameters. It is equally apparent, that to determine the total uncertainty, only the statistical approach can be reasonably adopted.

B.4.2 Temperature uncertainty determination

At the start of a project, uncertainties are large, since much of the data used is not secured and still under development together with the spacecraft.

The thermal designer at this stage determines which uncertainty parameters can have a major influence on the temperature, and sets up a development plan which allows, through a programme of tests and measurements, a successive reduction of the uncertainties (see Figure B-2).

A prerequisite for this technique, is to incorporate into the initial spacecraft thermal design the capability to trim the temperatures in either direction in order to compensate for those initial errors.

Based on European space industry experience and information, the total uncertainty for 1,6 σ (90 % probability) for unverified TMM temperature predictions is at least $\pm$15 K.

Experience has shown that this value can normally be reduced to $\pm$(8 to 10) K following a thermal verification programme. In some special cases where these reduced values are still too large to guarantee a unit to stay within specified limits, extended supporting tests are performed, possibly on a component basis or very detailed analysis in order to reduce the analytical uncertainty. The minimum achievable value appears to lie at $\pm$5 K at least for spacecraft in geostationary orbit (based upon the long experience with this type of spacecraft).

The classical approach has up until now, almost always been a space simulation test at spacecraft level on which the final clearing up of uncertainties which can be reduced on the ground is performed in the TMM.

The following alternative design and verification methods are a minimum prerequisite for the approach to have no space simulation test at spacecraft level:

a. Alternative test methods to validate the TCS design as there are modular tested or IR tested on the entire spacecraft.
b. Active thermal control design using fluid loops, heat pipes, louvers and automatic heater control systems in order to make the TCS design insensitive to analytical uncertainties.
c. Standard temperature limits which are larger and more unified than before to suit thermal requirements and provide a wide thermal design range for spacecraft TCS. Also standard thermal environment tests for equipment, which makes the application and allocation within space programmes less critical.

B.4.3 Quantification of uncertainty values

To understand the various associated uncertainties in the modelling parameters Figure B-1 contains some typical general values based on reference B.2.1.

As a general rule, the uncertainty of the individual parameters are accumulated in a statistical nature to provide an overall temperature uncertainty with the exception of mathematical modelling errors, some of which can be considered as systematic.

The method of statistically determining uncertainties is already well-documented in the technical library and is not reproduced here. Each different spacecraft or system design reacts differently to variances in parameters, however, experience has shown that in specific terms the major effects are likely induced by:

a. uncertainty in external MLI conductances (effective emissivity),
b. effects of thermo-optical material property of external radiator materials (lack of reliable degradation data over long mission life, e.g. > 5 years),
c. interface conductance unit/structure, and
d. uncertainty in certain dissipations, e.g. TWTA, batteries.

The first two have a global effect on the spacecraft whereas the latter mainly affect individual units and equipment. The uncertainty associated with the latter two parameters can be further refined in combinations of conductance and dissipation values over the full range found in spacecraft TCS.

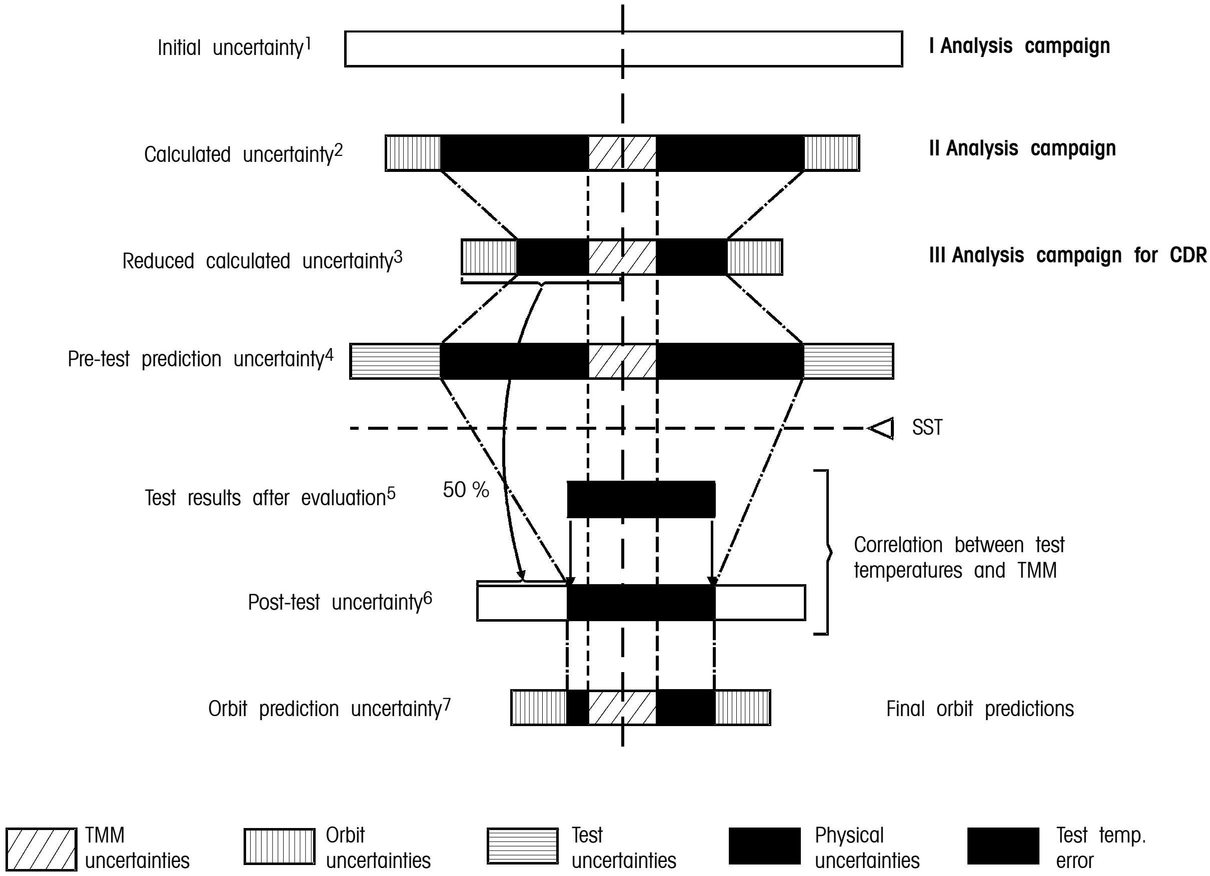
It is then a normal procedure to select at least one of the units from each of the combinations and subject it to a sensitivity analysis in which the applicable parameters are independently varied, such as:

- contact conductance,
- dissipation,
- radiating surface,
- local limit environment temperature, and
- thermal capacity of unit.

This computation can be carried out for example by a simple TMM of the unit and local structure and environment in which the different parameters are separately varied and the results then compared with the nominal case results.

To arrive at a value for the cumulated uncertainty, statistical methods are employed in so much as the environmental and physical parameters (see Table B-1) are usually combined in a RSS fashion with the mathematical modelling errors arithmetically added, i.e.:

$$\text{Uncertainty } \Delta T = \frac{\sqrt{\sum_{I=1}^{n} (\Delta T_i)^2}}{(n-1)} + \Delta T_{modelling\ uncertainty}$$



Notes

c Assessment based on experience.

5. These figures are calculated as described in A.4.3.

6. In order to prevent temperature excursions due to the relatively large initial uncertainties, the band width has been reduced by the expected uncertainty reduction during test evaluation. This procedure is a common practice and was confirmed during several satellite projects.

7. Made up of reduced calculated predictions – orbit uncertainties and test uncertainties.

8. This temperature band represents the residual temperature differences between test temperatures and analytical predicted temperatures.

9. As a useful rule during various satellite projects the following uncertainty modification has been established and confirmed by orbit temperature evaluation for various communication satellites: to add to the test temperature band (5) half of the calculated uncertainty band (3).

10. This figure is obtained by replacing the test uncertainty by the previous estimated orbit uncertainty.

Figure B-2: Typical progression of uncertainty in spacecraft programmes

Table B-1: Example of typical parameter uncertainties for 1,6 σ (90 %) value

	Parameter	Initial uncertainty variation [a]
A.	**Environmental parameters**	
	Solar intensity	±5 W/m^2
	Earth radiation	±65 W/m^2
	Albedo factor	±0,1
B.	**Spacecraft physical parameters**	
	Radiation absorbance [b]	+0,04/-0,02
	Radiator emittance	±0,03
	Radiator emittance (< 0,2)	±002
	Radiating area	±5 %
	Effective MLI conductance	±50 %
	Thermal conductivity	±10 %
	Shape (view) factors [c] (simple geometry)	±10 %
	Shape (view) factors [c] (complex geometry)	±50 %
	Contact resistance unit-structure (by similarity)	+100 %/- 50 %
	Contact resistance for units supported by conductance tests	±25 %
	Dissipation (for absolute value > 10 w)	±20 %
	Dissipation (for absolute value < 10 w)	±10 %
	Thermal capacity of electronic boxes	±15 %
C.	**Test parameters**	
	Chamber wall temperature	±10 °C
	Chamber wall emittance	±0,03
	Test adaptor temperature (mean value)	±2 °C
	Test adaptor interface conductance	±50 %
	Solar intensity distribution and spectrum	±3 %
	Test configuration effect	±1 °C
	Temperature sensor measurement accuracy	±1,5 °C [d]
	Nodel and sensor position error	±1 °C [d]

a These parameter variances are considered to present a 90 % probability or a 1,6 σ value.

b Applicable for ground tests, certain mechanisms causing degradation of materials in the spatial environment are not completely understood.

c Not including effect of inter-reflection.

d Added to the test uncertainty figure.

B.4.4 Test correlation criteria

The verification of the TMM is performed by a thermal test that normally reduces the initial uncertainties in the area of the spacecraft physical parameters. The uncertainties resulting from the environment and from the mathematically modelling are naturally unchangeable. A minimum correlation between test temperature results and TMM can assure to some extent an improvement of the above mentioned uncertainties.

Based on the experience in this area statistical correlation criteria can be defined which fulfils, from the engineering point of view, with sufficient accuracy the conformity between test results and analytical predicted temperatures.

As an adequate correlation criteria, an acceptable difference in the maximum temperature level between test temperatures (selected sensors) and TMM (corresponding nodes), a value for the mean deviation is:

$$\Delta T_o = \frac{1}{n}\sum_1^n \Delta T_i$$

of between 1 °C and 2 °C, and a standard deviation

$$\Delta T_s = \sqrt{\sum_1^n \frac{\Delta T_i^2}{n-1}}$$

of between 3 °C and 5 °C is normal practice.

B.5 Standardization of equipment temperature limits

B.5.1 Evaluation and classification of equipment data

The equipment thermal data of several spacecraft projects were evaluated in the frame of the study in reference. B.2.1, such as ECS, MCS, L-SAT, HIPPARCOS, ERS-1 DFS, TV-SAT/TDF 1, SPACELAB and, EURECA.

The classification was made in close approach with the type and function of the units in the spacecraft system, associated with the subsystems, and the unit location in the spacecraft plus thermal aspects. The classification is a guide line and overlooks for the “standardization” of so many different units, therefore some exceptions exists, that are discussed in more details in reference B.2.1.

The classification of equipment concerning temperature limits covers:

a. Servicing equipment (e.g. power, AOCS, TT&C and data handling),

b. Payload units (e.g. repeater, TWT amplifier and remote sensing instruments),

c. Externally located equipment (e.g. antennas and solar arrays), and

d. Propulsion units.

B.5.2 Standardization approach

The evaluation of the satellite programme thermal data was made under the aforementioned classification guidelines and the equipment temperature limits for:

a. operational mode in orbit,

b. non-operational mode, and

c. qualification level in test.

The bandwidths of temperature limits of all equipment per subsystems were plotted in summary sheets (see reference B.2.1).

The best approach, to establish a common temperature limit for all units, was made. The assumption was that programmes under development at the time of the study, such as MCS, ECS and HIPP, were “conservative”, and programmes under design, such as L-SAT, TV-SAT and DFS, were more progressive in defining more unified and wider temperature limit bandwidths (e.g. the ECS SHF transponder was qualified for OTS and does not fit in the established bandwidth).

The results are presented in Table B-2 in the order of the above classification.

The detailed thermal data of all equipment are also presented in reference B.2.1 for the above mentioned space programs.

Table B-2: Nominal temp. limits for various space vehicle equipment

Equipment		$T_{Q\text{-}OP}$ Qualif. limits (°C)[a]		$T_{A\text{-}OP}$ Accept. limits (°C)[b]		T_{SU} Start-up temp. (°C)[c,d]		T_{NO} Non-oper. limits (°C)[e]	
		min	max	min	max	min	max	min	max
1.1	TT&C electronic units	-35	+70	-25	+60	-45	+80	-45	+80
1.2.1	AOCS El. units + sensors	-30	+60	-20	+50	-40	+70	-40	+70
1.2.2	AOCS wheels	-20	+55	-20	+45	-25	+60	-25	+60
1.2.3	AOCS SAS sensors	-40	+80	-30	+70	-80	+80	-80	+80
1.2.4	AOCS gyro packages	-20	+55	-10	+45	-25	+60	-25	+60
1.3.1	Power subsystem units	-35	+70	-25	+60	-45	+80	-45	+80
	Power subsystem shunt electr.	-35	+70	-25	+70	N/A	N/A	-40	+80
1.3.2	Batteries								
a)	short term operation [f]	-10	+30	-5	+25	N/A	N/A	stor.	stor.
b)	long term operation [g]	N/A	N/A	-5	+20	N/A	N/A	-5	+25
1.3.4	Power								
a)	subsystemBAPTA(SADM)	-60	+100	-50	+90	N/A	N/A	-50	+90 [h]
b)	Drive mechanism flange	N/A	N/A	-25	+55	-25	+55	-25	+55 [h]
	Motor housing								
1.4	Any other equipment	-35	+70	-25	+60	-45	+80	-45	+80
2.	Payload equipment								
2.1	Communication S/S units								
2.1.1	Repeater TWT's low power	-20	+90	-10	+80	-40	+90	-40	+90
2.1.2	Repeater TWT's high power								
a)	Collector radiating	-55	+350	-45	+340	-60	+340	-60	+340 [i]
b)	Body/helix	-20	+80	-10	+70	-40	+70	-40	+70 [i]
2.1.3	Repeater units, general incl. waveguides at repeater units	-35	+70	-25	+60	-40	+70	-40	+70
2.2	Remote sensing instruments	-15	+50	-5	+40	-45	+80	-45	+80
2.3	Data handling units	-35	+70	-25	+60	-45	+80	-45	+80
2.4	Any other equipment	-35	+70	-25	+60	-45	+80	-45	+80
3.	Externally located units								
3.1	Waveguides at antennas	-180	+110	-170	+100	N/A	N/A	N/A	N/A
4.	Propulsion subsystem units								
4.1	Hydrazine system (excl. thrusters)	+4	+65	+9	+50	+9	+50	+9	+50 [h]
4.2	Cold gas system	-35	+70	-25	+30	-45	+80	-45	+80 [h]
4.3	Bi-propellant system	-7	+50	0	+40	0	+40	0	+40 [h]
4.4	Solid ABM	-10	+40	N/A	N/A	N/A	N/A	0	+30

a Qualification limits are qualification test temperatures for operating equipment.

b Acceptance limits are acceptance test temperatures for operating equipment.

c The start-up temperatures are equal to the non-operating temperature as long as possible. A special switch-on temperature is defined for equipment intended for full performance immediately after switch-on.

d The high start-up temperature cannot be adequate to ensure a proper functioning at high qualification temperature due to overheating. In this case to start the equipment at a temperature slightly less than the high qualification temperature is a good practice.

e Non-operating limits are qualification and acceptance test temperatures for non-operating equipment.

f Battery short-term operation ≤ 100 h (during qualification test and transfer orbit).

g Battery long-term operation > 100 h (during mission life time).

h Due to intermittent operation or physical limitations, the non-operating is identical to the operating limits.

i Due to high dissipation, the upper non-operating limits are below the maximum qualification limits with the operating unit.

B.6 Standardization of thermal vacuum and cycling test conditions

B.6.1 Evaluation of test requirements

The environmental test requirements were evaluated concerning temperature test parameters of eight European space programmes and of different organizations.

The purpose was to find out the practice of temperature test conductance with respect to:

a. number of cycles,

b. dwell time at extreme temperatures,

c. temperature rate of change, and

d. overall test sequence and temperature profile.

The results are summarized in reference B.2.1 and compared with the standardized parameters. The parameters differ much from one project to another and no common test philosophy is evident. In general all parameters are tailored to the project requirements and contractors experience. Therefore the different environments in which the equipment operates (active or passive thermal control), the duration of the mission (e.g. Spacelab: 10 days and communication satellite: 10 years), and the location of the equipment on the spacecraft (internally or externally of the spacecraft) are parameters to consider by evaluation.

B.6.2 Standardization approach

The sampling and summarizing of programme parameters only does not allow any conclusions how effective these test parameters were to uncover failures during equipment tests and to reduce the failure rate at system level testing. The total failure history and test effectiveness of those programmes are evaluated. Therefore, an existing study of test effectiveness was evaluated concerning temperature test parameters that provide the optimum of failure detection at equipment level testing (see reference. B.2.2).

To support the specification of standard temperature test requirements the recommendations of the environmental stress screening guidelines (ESSEH 1981, reference B.2.3, sponsored by the Institute of Environmental Science) were investigated in detail. The results of this investigation indicate unit screening parameters for the temperature range, the number of cycles (Figure B-5), the dwell time at the extreme temperatures (Figure B-4) and temperature rate of change. The parameters cover the range between minimum and maximum recommended stresses for the temperature cycle test with the optimum of failure rate reductions. These parameters were used as a guideline (plus other projects) for defining the test levels for the standard environment test specification excluding the temperature level ranges. These are defined in reference B.2.1.

No thermal vacuum requirements were investigated in reference B.2.3. A practical method to compare the various project test parameters is to calculate the test duration, which is the time when the equipment is exposed to the extreme temperatures. The test time was calculated:

$$t = 2\, n\, t_E \text{ (hours)}$$

where

t_E = dwell time of temperature extreme;

n = number of cycles.

NOTE t does not include the transient time.

In general, the maximum requirements of ESSEH guideline were not used, because it covers mainly military, airborne avionics equipment with higher screening stresses than for standard spacecraft equipment. The temperature profile and test sequence of the ECS, MCS, L-SAT and HIPPARCOS programmes were adapted as standard procedure because of the good experience accumulated with it.

The classification of the equipment and the applicable test parameters are made with regard to the environment of the equipment in which it is located. The environment conditions can be vacuum or atmospheric pressure depending upon the spacecraft location (satellite and SL pallet of SL module and IGLOO container). The temperature environment can be actively controlled with negligible temperature fluctuations (e.g. fluid loop and heat pipes) or passive by controlled spacecraft interior with nominal temperature variations. For special, externally located equipment it can result in large, frequent temperature fluctuations (e.g. solar arrays and antennae). The matrix in Figure B-3 gives an indication which type of test and parameters are applicable. This is a guideline only, the final allocation depends on the type of equipment and mission and specified in the equipment test matrix.

Equipment temperature test parameters (2 × number of cycles × dwell time)			
Thermal environment	**Test level**	**Atmospheric environment**	
		Atmospheric pressure	**Vacuum**
Active controlled		Cycling test	TV test only
		2 × 8 × 4 h	
		2 × 4 × 4 h	2 × 1,5 × 12 h
Passive controlled spacecraft interior		Cycling test	Combined cycling + TV test
	Qualification	2 × 8 × 4 h	2 × 8 × 4 h
	Acceptance	2 × 4 × 4 h	2 × 4 × 4 h
Passive controlled exterior			Combined cycling + TV test
	Qualification	N/A	2 × 12 × 2 h
	Acceptance		2 × 4 × 2 h

Figure B-3: Equipment thermal test matrix

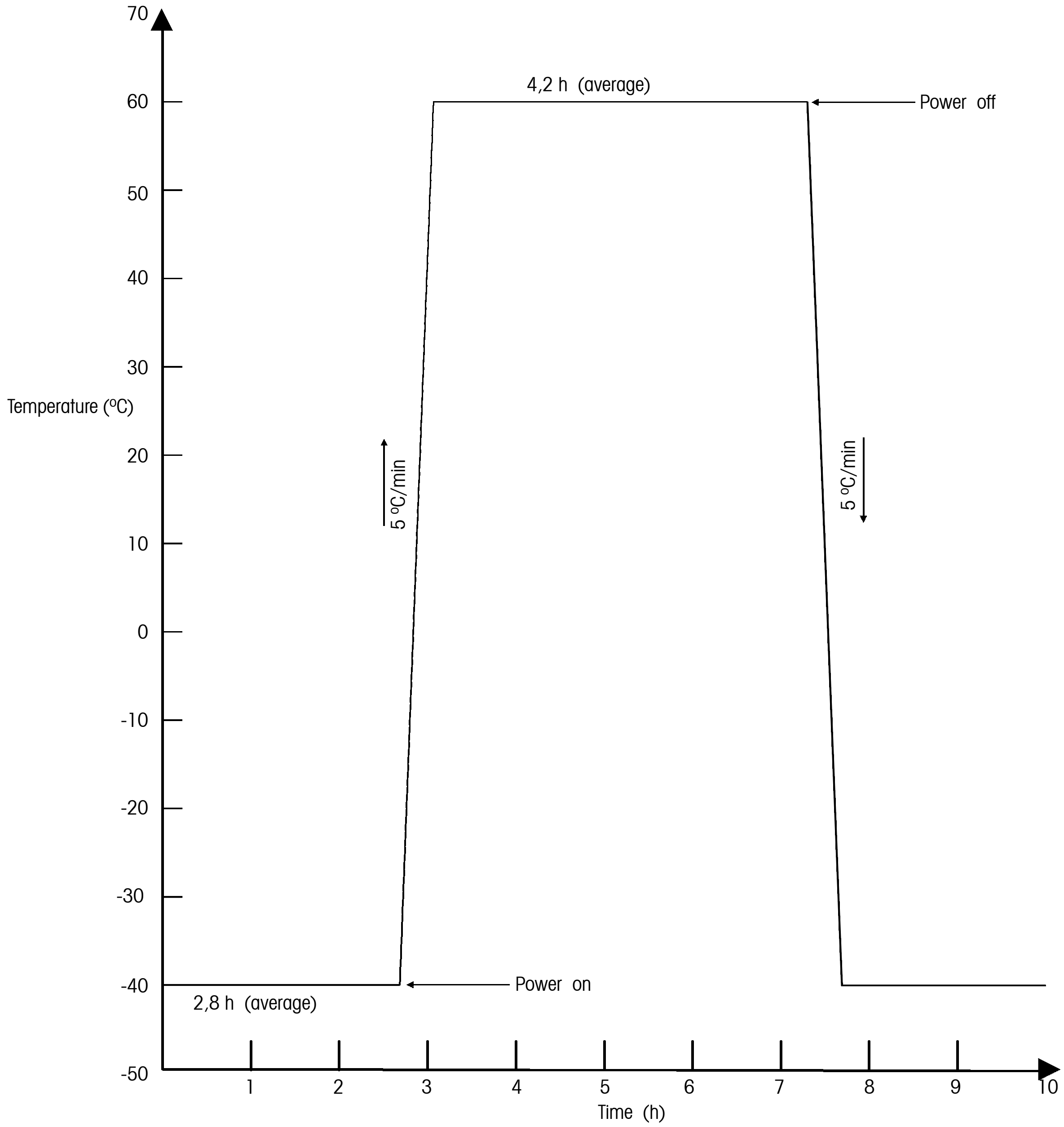


Figure B-4: Example of a typical temperature profile for unit level thermal cycling

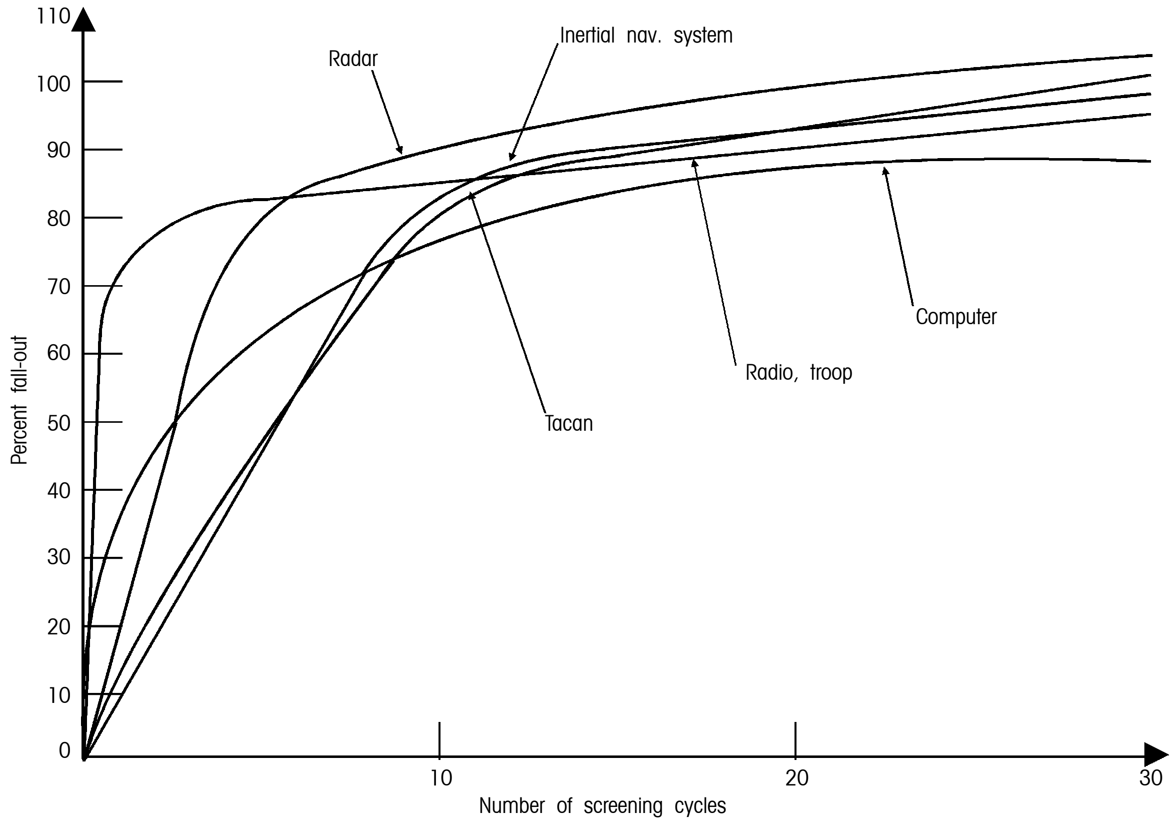


Figure B-5: Percent fall-out vs. number of cycles increased for unit level screening

B.7 Conclusions

The verification method for a given satellite is dependent on the overall project aspects such as the number of spacecrafts to be manufactured, the payload function, the spacecraft configuration and the state-of-the-art technology. The evidence collected from previous satellite projects indicated that in almost every case, anomalies were detected in the spacecraft thermal design, which if undetected, can lead to a major reduction in the spacecraft performance or part failure.

The present status and future trends in the design and production of satellites is tending towards higher payload densities and more specialized equipment, which in general leads to greater attention to thermal aspects.

Thermal control design is also still mostly based on passive methods and only when relatively large excesses of energy are made available, a heater based active control system can further reduce the residual uncertainties from thermal verification. Without a definite move to more active thermal control systems and more relaxed operational temperature limits the dependence on verification of thermal environment by testing at system level is maintained, for at least projects in the near future.

In conclusion it can be stated that sufficient analytical methods exist so that it is theoretically possible to perform a purely analytical verification of a satellite. A thermal control design relying mainly on active control methods and large bandwidth between equipment temperature limits is essential. In practice, however, there is no alternative to the thermal vacuum test for determining workmanship defects or local anomalies in the thermal control hardware. Purely analytical verification is therefore considered unsuitable, however, it is obvious that the effort devoted to thermal testing can be in future significantly reduced to the level of one or two steps at equipment and sub-assembly level.

The standardization of equipment temperature limits as proposed, significantly improves the thermal control design margin and makes the design less sensitive to temperature uncertainties.

The extension of the operating temperature bandwidth to the current possible limits (example L-SAT) and the unification of the temperature limits (e.g. all units of a unit functioning in one compartment are at same minimum and maximum temperature level) result in a considerable improvement of the thermal control bandwidth. Compared to the past projects, an improvement by a factor of 1,5 to 2 is achieved, based on the study in reference B.2.1, the thermal control bandwidth (range between highest minimum temperature and lowest maximum temperature within a unit) is increased for the unit equipment.

The standardization of temperature test levels and conditions leads to unified test procedures for qualification and acceptance of “standard” equipment independently of its further application in space programs. The extended temperature ranges, the increased test durations (number of cycles × dwell time at extreme temperature) and more rapid temperature rate of change are considered as a result of following investigations:

- U.S. long experience with a multitude of space programmes that were studied for their test effectiveness (see references B.2.3, B.2.4).
- The (future European) approach to delete or to decrease the effort on the system test (e.g. SPACELAB and EURECA) and to qualify the equipment without the consideration of a later system test. (see reference B.2.1).
- The extended mission life-time of spacecraft up to 10 years and the consequent higher reliability.

The test description is based on the accumulated equipment test experience in the European space industry with the successful European communication satellite and Spacelab programmes.

Annex C (normative)

Test requirement specification – DRD

C.1 Introduction

This document provides the main rules for the preparation of a general test requirement specification for test activity(ies) by tailoring this Standard (ECSS-E-10-03) for a specific project.

This document ensures a homogeneous common basis to the overall test processes definition from the lower level activities to the higher ones.

C.2 Scope and applicability

C.2.1 Scope

This document requirements definition (DRD) establishes the content for the test requirement specification.

This DRD does not define format, presentation or delivery requirements for the test requirement specification.

C.2.2 Applicability

This DRD is applicable to all projects using the ECSS Standards.

C.3 References

C.3.1 Glossary and dictionary

This DRD uses terminology and definitions controlled by:

ECSS-P-001	ECSS Glossary of terms
ECSS-E-10-02	Space engineering — Verification

C.3.2 Source document

This DRD defines the data requirements of a test requirement specification as specified in this Standard

C.4 Definitions, abbreviations and symbols

C.4.1 Definitions

For the purposes of this DRD the definitions given in ECSS-P-001A, in the ECSS-E-10-02 and in this Standard apply.

C.4.2 Abbreviations

The following abbreviations are defined and used within this DRD.

Abbreviation	Meaning
AIV	assembly integration and verification
DRD	document requirements definition
GSE	ground support equipment.

C.5 Description and purpose

The test requirement specification describes the general methodological test requirements applicable to a specific project. For the various tests it defines the general constraints, conditions and sequence that can be considered programme dependent.

The document, together with the system test specifications, is used as input for the system test procedures, as applicable requirement document. It also provides the specific requirements for the lower level subcontractors, as applicable document for the lower level test activities.

The document provides evidence to the customer and an overall picture of the complete testing approach at all levels of the project with some details of the test activities in advance of the activity itself.

C.6 Application and interrelationship

The document contains the general requirements for all tests of every verification level.

The DRD is generated using the information contained in this Standard taking into account the detailed information contained in the other specific project documentation (e.g. drawings and ICDs).

This DRD is used to write the test specification (DRD), see ECSS-E-10-02A Annex F and test procedures (DRD) see ECSS-E-10-02A Annex G.

C.7 Test requirement specification preliminary elements

C.7.1 Title

a. This document shall be titled "[insert a descriptive modifier] test requirement specification".

b. The descriptive modifier shall be selected to clearly identify the program.

EXAMPLE "Columbus test requirement specification".

C.7.2 Title page

The title page for this document shall identify the project document identification number, title of the document, date of release and release authority.

C.7.3 Contents list

The content list shall identify the title and location of every clause and major subclause, figure, table and annex contained in the document.

C.7.4 Foreword

A foreword shall be included in the document that describes the following items:

a. identification of which organizational entity prepared the document;

b. information regarding the approval of the document;

c. identification of other organizations that contributed to the preparation of the document;

d. a statement of effectivity identifying which other documents are cancelled and replaced in whole or in part;

e. a statement of significant technical differences between this document and any previous document;

f. the relationship of the document to other standards or documents.

C.7.5 Introduction

An introduction may be included to provide specific information or commentary about the technical content.

C.8 Content

C.8.1 Scope and applicability

C.8.1.1 Introduction

This clause shall be numbered 1, and shall describe the scope, purpose and applicability of the test requirement specification.

C.8.1.2 Scope

This subclause shall be numbered 1.1, and shall contain the following statements:

> "This test requirement specification defines the test methodology and the general test requirements for the [insert project identifier] project".

C.8.1.3 Purpose

This subclause shall be numbered 1.2, and shall contain the following statements:

> "This test requirement specification is aimed to specify the general test conditions and requirements representing the minimum set to associate to the various stages and levels of the testing process."

NOTE 1 The document is used as input for the test specifications and procedures, as applicable requirement document, and for lower level subcontractor.

NOTE 2 It provides evidence to the customer of the overall picture of the complete testing approach at all levels of the project with some details of the test activities in advance of the activity itself.

C.8.2 References

C.8.2.1 Introduction

This clause shall be numbered 2 and shall contain the subclauses C.8.2.2 and C.8.2.3.

C.8.2.2 Normative references

This subclause shall be numbered 2.1 and shall contain the following statements followed by the list of normative references:

> "The following normative documents contain provisions which, through reference in this text, constitute provisions of this document. For dated references, subsequent amendments to, or revisions of any of these publications do not apply. However, parties to agreements based on this document are encouraged to investigate the possibility of applying the most recent editions of the normative documents indicated below. For undated references, the latest edition of the normative document referred to applies.
>
> [insert document identifier] [insert document title]".

NOTE The reference documents are usually the AIV plan and the product specification.

C.8.2.3 Informative references

This subclause shall be numbered 2.2 and shall contain the following statement followed by the list of informative references:

> "The following documents, although not a part of this test requirement specification, amplify or clarify its contents:
>
> [insert document identifier] [insert document title]".

C.8.3 Definitions and abbreviations

C.8.3.1 Introduction

This clause shall be numbered 3 and shall contain the subclauses C.8.3.2 and C.8.3.3.

C.8.3.2 Definitions

This subclause shall be numbered 3.1, and shall list any project dictionary or glossary, and all unusual terms, or terms with a meaning specific to the test requirement specification, with the definition for each term.

a. If a project dictionary or glossary is applied, insert the following sentence:

> "The definitions of [insert title and identifier of applicable dictionaries or glossaries] apply to this document".

b. Insert the following sentence:

> "The following terms and definitions are specific to this document:
>
> [insert term] [insert definition]".

C.8.3.3 Abbreviations

This subclause shall be numbered 3.2, and shall list all abbreviations used in the test requirement specification with the full spelled-out meaning or phrase for each abbreviation.

C.8.4 General test conditions and requirements

This clause shall be numbered 4 and shall specify the general test conditions (including ambient conditions, cleanliness levels, accuracy of instruments and test equipments) and test condition tolerances.

C.8.5 Qualification testing

a. This clause shall be numbered 5 and shall provide the minimum set of requirements for qualification tests with the application to all levels (e.g. equipment, subsystem and system) of the testing process.

b. The following categories of tests shall be considered:

   1. functional tests,
   2. environmental tests (e.g. thermal vacuum, thermal cycling, vibration, pyrotechnic shock, acceleration, humidity, pressure, leakage and EMC),
   3. static structural load,
   4. modal survey, and
   5. physical properties.

c. For each test,the purpose of the test shall be stated, followed by a test description that illustrates the sequence of the events including test levels and duration. This includes also considerations on the various supplementary requirements to obtain the maximum information from the test (e.g. performing a functional test or monitoring before, during and after thermo-mechanical tests).

C.8.6 Acceptance testing

a. This clause shall be numbered 6 and provide the minimum set of requirements for acceptance tests in the application to all levels (e.g. equipment, subsystem and system) of the testing process.

b. The following categories of tests shall be considered

   1. functional tests,
   2. environmental tests (e.g. thermal vacuum, thermal cycling, vibration, pyrotechnic shock, pressure and leakage), and
   3. physical properties.

c. For each test,the purpose of the test shall be stated, followed by a test description that illustrates the sequence of the events including test levels and duration. This includes also considerations on the various supplementary requirements to obtain the maximum information from the test (e.g. performing a functional test or monitoring before, during and after thermo-mechanical tests).

C.8.7 Protoflight testing

This clause shall be numbered 7 and shall define, for the various verification levels, the variations to be applied to the verification philosophy in order to reduce stress levels for qualification assemblies that are planned for flight use, combining acceptance and qualification campaign.

C.8.8 Re-testing methodology

This subclause shall be numbered 8 and shall define the methodology for failure analysis and repetition of the test due to a discrepancy or other factors, such as design changes, related to the items previously tested.

C.8.9 GSE testing

a. This clause shall be numbered 9 and shall provide the minimum set of requirements for tests of the ground support equipment used in integration and testing process.

b. The following categories of tests shall be considered

1. functional tests,
2. environmental tests (e.g. vibration, drop shock, pressure, leakage, rain and solar),
3. static proof, and
4. physical properties.

c. For each test,the purpose of the test shall be stated, followed by a test description that illustrates the sequence of the events including test levels and duration. This includes also considerations on the various supplementary requirements to obtain the maximum information from the test (e.g. performing a functional test or monitoring before, during and after thermo-mechanical tests).

Bibliography

ECSS-E-30-01	Space engineering — Fracture control
ECSS-M-00-02	Space project management — Tailoring of space standards
ECSS-Q-20-09	Space product assurance — Nonconformance control system
MIL-STD-461D	Requirements for the Control of Electromagnetic Interference Emissions and Susceptibility
MIL-STD-462D	Test Method Standard for Measurement of Electromagnetic Interference Characteristics
MIL-STD-1540C	Test Requirements for Launch, Upper-stage and Space Vehicles
MIL-STD-1541	Electromagnetic Compatibility Requirements for Space Systems
MIL-STD-1833	Test Requirements for Ground Equipment and Associated Computer Software Supporting Space Vehicles

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ECSS Document Improvement Proposal

1. Document I.D. ECSS-E-10-03A	**2. Document date** 15 February 2002	**3. Document title** Testing
4. Recommended improvement (identify clauses, subclauses and include modified text or graphic, attach pages as necessary)		
5. Reason for recommendation		
6. Originator of recommendation		
Name:	Organization:	
Address:	Phone: Fax: e-mail:	**7. Date of submission:**
8. Send to ECSS Secretariat		
Name: W. Kriedte ESA-TOS/QR	Address: ESTEC, P.O. Box 299 2200 AG Noordwijk The Netherlands	Phone: +31-71-565-3952 Fax: +31-71-565-6839 e-mail: Werner.Kriedte@esa.int

Note: The originator of the submission should complete items 4, 5, 6 and 7.

This form is available as a Word and Wordperfect-file on internet under
http://www.estec.esa.nl/ecss

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