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Space systems — Design qualification and acceptance tests of small spacecraft and units

Systèmes spatiaux — Qualification de la conception et essais de réception des petits véhicules spatiaux

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Foreword

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The procedures used to develop this document and those intended for its further maintenance are described in the ISO/IEC Directives, Part 1. In particular, the different approval criteria needed for the different types of ISO documents should be noted. This document was drafted in accordance with the editorial rules of the ISO/IEC Directives, Part 2 (see www.iso.org/directives).

Attention is drawn to the possibility that some of the elements of this document may be the subject of patent rights. ISO shall not be held responsible for identifying any or all such patent rights. Details of any patent rights identified during the development of the document will be in the Introduction and/or on the ISO list of patent declarations received (see www.iso.org/patents).

Any trade name used in this document is information given for the convenience of users and does not constitute an endorsement.

For an explanation on the voluntary nature of standards, the meaning of ISO specific terms and expressions related to conformity assessment, as well as information about ISO's adherence to the World Trade Organization (WTO) principles in the Technical Barriers to Trade (TBT), see the following URL: www.iso.org/iso/foreword.html.

This document was prepared by Technical Committee ISO/TC 20, *Aircraft and space vehicles*, Subcommittee SC 14, *Space systems and operations*.

Introduction

There is an increasing demand for small/micro/nano/pico satellite development and utilization worldwide; yet, there is no clear and globally accepted definition of what is considered "small", "micro", "nano" or "pico" satellites. These satellites are often built with emphasis on low cost and fast delivery. They are characterized by extensive use of non-space-qualified commercial-off-the-shelf (COTS) units. For the sake of convenience, the term "small spacecraft" is used throughout this document as a generic term to refer to these satellites.

A small spacecraft is a satellite that utilizes non-traditional risk-taking development and management approaches to achieve low cost and fast delivery with a small number of team. To achieve these two points, low cost and fast delivery, satellite design relies on the use of non-space-qualified commercial-off-the-shelf (COTS) units, making satellite size inherently smaller. The design accepts a certain level of risk associated with the use of COTS.

A certain set of tests is necessary to ensure the mission success of small spacecraft. Applying the same test requirements and methods as those applied to traditional large/medium satellites, however, will nullify the low-cost and fast-delivery advantages possessed by small spacecraft.

This document is meant to improve the reliability of small spacecraft, especially those with commercial purpose, while maintaining the low-cost and fast-delivery nature of small spacecraft. This document intends to promote worldwide trade of small spacecraft products by providing a minimum level of assurance that a product made of non-space-qualified commercial-off-the-shelf parts and units can work in space. This document also aims to serve as a testing guideline for those who intend to enter satellite manufacturing through development of small spacecraft products.

Space systems — Design qualification and acceptance tests of small spacecraft and units

1 Scope

This document provides test methods and test requirements for design qualification and/or acceptance of small spacecraft or units. It provides the minimum test requirements and test methods to qualify the design and manufacturing methods of commercial small spacecraft and their units and to accept the final products.

This document places emphasis on achieving reliability against infant mortality after satellite launch to orbit while maintaining low cost and fast delivery.

This document is applied to satellites whose development methods are different from the ones used for traditional satellites that have little room for risk tolerance, as shown in Figure 1. The scope of this document encompasses different categories of small spacecraft, so-called mini-, micro, nano-, pico- and femto-, as well as CubeSat, spacecraft. Therefore, for the sake of convenience, the term "small spacecraft" is used throughout this document as a generic term.

This document includes CubeSat, as long as it is developed with the untraditional processes.
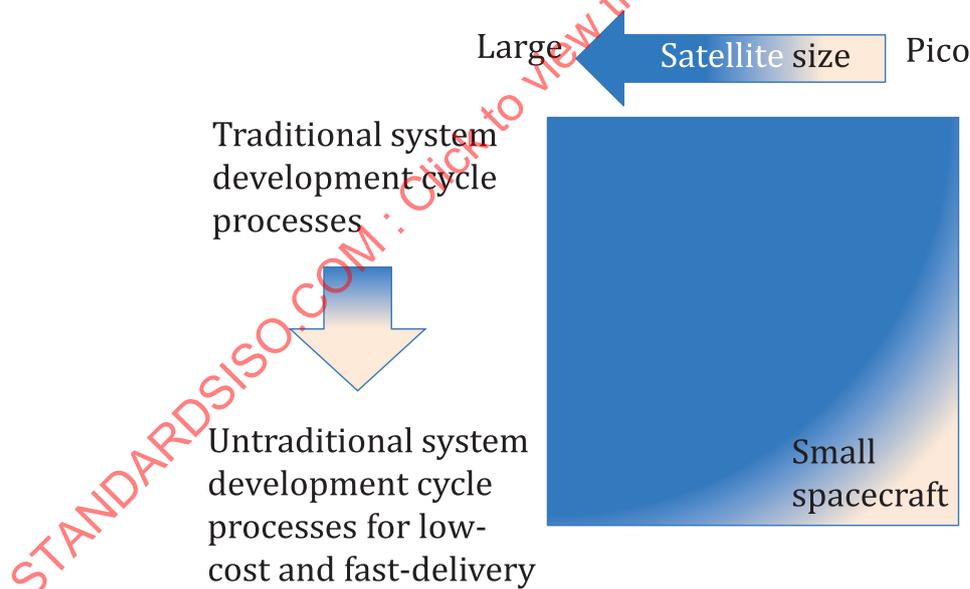
Figure 1 — Applicability of this document

This document does not cover satellite deployment mechanisms, such as POD, as the verification requirements are defined in the Interface Control Document (ICD) with the launcher, such as ISO 26869.

This document does not cover software testing, although some tests such as functional test, mission test and end-to-end test are inherently used to test the software installed in the hardware being tested. General requirements and processes of satellite software testing can be found in various references, such as ECSS-E-ST40.

This document does not cover requirements regarding safety nor debris mitigation. Appropriate documents such as ISO 14620–1 or ISO 24113 can be referred to. Other common requirements for small spacecraft can be found in Reference [14].

2   Normative references

The following documents are referred to in the text in such a way that some or all of their content constitutes requirements of this document. For dated references, only the edition cited applies. For undated references, the latest edition of the referenced document (including any amendments) applies.

ISO 11221:2011, *Space systems — Space solar panels — Spacecraft charging induced electrostatic discharge test methods*

ISO 14302, *Space systems — Electromagnetic compatibility requirements*

ISO 15864:2004, *Space systems — General test methods for space craft, subsystems and units*

ISO 17566:2011, *Space systems — General test documentation*

3   Terms and definitions

For the purposes of this document, the following terms and definitions apply.

ISO and IEC maintain terminological databases for use in standardization at the following addresses:

—   IEC Electropedia: available at http://www.electropedia.org/

—   ISO Online browsing platform: available at http://www.iso.org/obp

3.1
flat-sat
table-sat
configuration where only *units* (3.4), sometimes bare circuit boards only, are laid out in atmosphere on a table while not being mounted to the satellite structure

3.2
flight model
satellite or unit model dedicated to launch and operate in orbit and subjected to acceptance testing

3.3
test article
satellite or *unit* (3.4) on which a test is conducted

3.4
unit
lowest level of hardware assembly for which acceptance and qualification tests are required

3.5
POD
box housing *CubeSats* (3.6) during launch

3.6
CubeSat
box-shaped satellite whose volume is composed of "N" 10 cm × 10 cm × 10 cm sub-volumes

EXAMPLE   1U = 10 cm × 10 cm × 10 cm, 2U = 20 cm × 10 cm × 10 cm, and 3U = 30 cm × 10 cm × 10 cm.

Note 1 to entry: See ISO 17770 for further definition.

4 Abbreviated terms

AT   Acceptance test

COTS   Commercial-off-the-self

CVCM   Collected volatile condensable materials

EMC   Electromagnetic compatibility

ESD   Electrostatic discharge

ESS   Environment stress screening

EED   Electroexplosive devices

EM   Engineering model

EMC   Electromagnetic compatibility

FM   Flight model

FMEA   Failure Mode and Effects Analysis

FMECA   Failure Mode, Effects, and Criticality Analysis

IC   Integrated circuit

ICD   Interface control document

LEO   Low Earth Orbit

MEMS   Micro Electro Mechanical Systems

MMA   Moving mechanical assembly

PFT   Proto-flight test

PFM   Proto-flight model

QM   Qualification model

QT   Qualification test

QCM   Quartz Crystal Microbalance

RF   Radio frequency

PSD   Power spectral density

SAA   South Atlantic Anomaly

SE   Single event

SEE   Single event effect

SRS   Shock response spectrum

STM   Structural thermal model

TID   Total ionization dose

TML   Total mass loss

5   General requirements

5.1   Tailoring

Specifications described in this document are tailorable upon agreement between the customer, the manufacturer and the launch provider. Annex A provides tailoring and waiver guides of test requirements.

5.2   Qualification test

For satellite system level qualification tests, there is little difference between small spacecraft and traditional satellite qualification tests in terms of objectives and requirements. Qualification tests demonstrate that items meet design requirements and include proper margin. Qualification tests can serve as good practice for personnel who are inexperienced in testing toward acceptance tests to be carried out later. If the same design is used for many satellites, such as satellites of a constellation program, the qualification tests are not necessary except for the first satellite. ISO 15864:2004, 4.4 provides additional requirements for system qualification test.

Unit level qualification tests dealt in this document is different from those done for traditional satellites. The unit QT in this document provides a minimum guarantee that a given unit sold as "a satellite unit" has a certain level of tolerance against the space environment. Therefore, the unit QT in this document does not include proper margin against the maximum predicted environment stress, which depends on each satellite. This document provides numeric values for the test level and duration of unit QTs as much as possible with their rationale given in Annex B. The satellite developers who purchase the COTS-unit tested according to this document shall make consistent decisions about how to obtain the margin. The satellite developers may purchase a dedicated test model in addition to the flight model and carry out another QT with the margin. They may carry out PFT using a flight model or only AT taking the risk of little margin. The satellite developer shall provide the test levels and duration of the additional QT, AT or PFT. See C.1 for additional note.

5.3   Acceptance test

There is little difference in terms of objectives and requirements of acceptance tests between small spacecraft and traditional satellites. ISO 15864:2004, 4.5 provides requirements for acceptance test.

5.4   Proto-flight test

There is little difference in terms of objectives and requirements of proto-flight tests between small spacecraft and traditional satellites. ISO 15864:2004, 4.6 provides requirements for proto-flight test.

5.5   Retest

Situations that may require retest are described in ISO 15864:2004, 4.8.

5.6   Test documentation

In order to minimize program cost, the amount of paper work should be reduced as much as possible. The documents used inside the developing organization can only be simplified considering the small size of the team. At the same time, however, the test documentation shall be detailed enough to ensure

traceability from the later stages of satellite development or operation. The importance of the test procedure document should not be underestimated, as well-prepared tests will eventually save both time and money.

For the unit QT, the test documentation shall provide the important information necessary to prove that the COTS-based units have adequate durability against the space environment. See C.1 for additional notes.

5.6.1   Test plan, specification and procedure

The simplicity of a small spacecraft allows the combination of the test plan, specification and procedure documents for the system test or unit test into one document as recommended by ISO 17566. In the following cases, it is recommended to separate the test plan/specification and the test procedure into two separate documents.

a)   The test is fairly complex and requires many discussions and iterations within the development team to define the test specification.

b)   The test is carried out at a location outside the developing organization and requires consultation with the test institution well before the test.

c)   The test has to be approved by the customer. The test plan/specification may be used as the document for approval.

d)   Others.

The contents of the test plan, specification and procedure shall be based on ISO 17566:2011, Table 2 or Table 3. Some of the content in the specification, such as test facility requirements and procedural test requirements, may be moved to the test procedure document. The documents may be revised as test preparation progresses. It is important to keep track of the version number, the revision points and the revision dates.

5.6.2   Test report

The test report content shall be based on ISO 17566:2011, Annex D. Any anomaly during the test and its disposition shall be reported in the test report. The test report shall clearly describe the following information or refer to the test documents that contain the information.

a)   Temperature measurement points and the measured temperature profile in the case of thermal tests, e.g. thermal cycle, thermal vacuum, thermal balance, etc.

b)   Acceleration measurement points in the case of mechanical tests, e.g. vibration, shock, etc.

c)   The points of reference (thermo-couples or accelerometers) used to control the test levels, e.g. temperature, vibration, acoustic, etc.

d)   The measured pressure profile during the test in the case of tests conducted in a vacuum, e.g. thermal vacuum, vacuum functioning, multipaction, etc.

e)   The power spectrum density waveform of the acceleration measured at the reference points in the case of random vibration test or modal survey.

f)   The shock response spectrum calculated from the acceleration measured at the reference points in the case of shock test.

g)   The source of radiation particles, their energy and the total fluence in the case of SEE test.

h)   The radiation source and energy, total dose and shielding effects considered to derive the total dose, dose rate and temperature during testing in the case of TID test.

i)   The result of functional performance measurements before, during and after the environment test.

If the test results in failure, the test report shall be precise enough to assist with the root cause analysis or other investigation. See Reference [22] for root cause analysis.

5.6.3   Datasheet for unit test results

The following information shall be included in the datasheet for unit test results.

a)   Random vibration spectrum in power spectral density, root-mean-square value of acceleration and duration for each axis if random vibration test was carried out.

b)   Vibration level, sweep rate and frequency range for each axis if sinusoidal vibration test was carried out.

c)   Radiation and conduction emission spectrum if EMC test was carried out.

d)   Temperature profile, number of cycles, hot and cold soak temperatures and their duration, temperature ramp rate (up/down) and gas environment if thermal cycle test in atmospheric pressure was carried out.

e)   Temperature profile, number of cycles, hot and cold soak temperatures and their duration, temperature ramp rate (up/down) and pressure profile if thermal vacuum cycle test was carried out.

f)   Shock response spectrum for each axis and the method of applying the shock acceleration if shock test was carried out.

j)   The source of radiation particles, their energy and the total fluence if SEE test was carried out.

k)   The radiation source and energy, total dose, shielding effects considered to derive the total dose, dose rate and temperature during testing if TID test was carried out.

l)   The power spectral density of the measured force for various rotational speeds if microvibration acceptance test was carried out.

5.7   Test conditions, tolerances and accuracies

The requirements in ISO 15864:2004, 4.10 shall apply.

5.8   Functional test

Complete functional tests shall be performed at the beginning and end of the test sequence. Partial functional tests shall be conducted before and after each environmental exposure.

5.9   Design, verification and testing philosophy

Annex C describes the difference of small spacecrafts from traditional satellites in terms of design, verification and testing philosophy. Tables C.1 and C.2 summarize the characteristics inherently associated with the satellite program/design and the corresponding verification strategy when low cost and fast delivery are the primary drivers.

6 Satellite system tests

6.1 Test items

The satellite system test items are listed in [Table 1]. [Annex D] provides test selection logic flows.

Table 1 — System test items

Test items	QT	AT	PFT
Electrical interface	R	R	R
Functional test	R	R	R
Mission test	R	R	R
Total Ionization Dose (TID) test	O[a]	—	O[a,b]
Single Event Effects (SEE) test	O[a]	—	O[a,b]
Spacecraft Charging Induced Electrostatic Discharge (ESD) test	To be done in the unit level.		
Electromagnetic Compatibility (EMC) test	R	O[c]	R
Deployment test	R	R	R
Magnetic field test	O[d]	O[d]	O[d]
Antenna pattern test	To be done in the unit level.		
Alignment measurement	O[e]	O[e]	O[e]
Physical property measurement	R	R	R
Launcher/Spacecraft interface test	R	R	R
Quasi-static load test	O[f]	O[f]	O[f]
Modal survey	O[g]	—	O[g]
Sinusoidal vibration test	O[h]	O[h]	O[h]
Random vibration test	R[i]	R[i]	R[i]
Acoustic test	O[i]	O[i]	O[i]
Shock test	O[j]	O[j]	O[j]
Thermal balance test	O[k]	—	O[k]
Thermal vacuum test	R[l]	O[m]	R[l]
Functional test in vacuum		—	
Thermal cycle functional test		O[n]	
Cold/hot start test	O[o]	O[o]	O[o]
Thermal cycle endurance test	To be done in the unit level.		
Pressure test	O[p]	O[p]	O[p]
Leakage test	O[p]	O[p]	O[p]
Microvibration test	To be done in the unit level.		
Burn-in and wear-in test	To be done in the unit level.		

Table 1 *(continued)*

Test items	QT	AT	PFT
End-to-end mission simulation	—	R	R
Bake out and outgas	—	Oq	Oq

R Required; O Optional

a To be done in the flat-sat configuration without satellite structure. See Figure D.1.

b Use of dedicated test model.

c To be done in anechoic chamber if radio licence is not obtained yet. See Figure D.2.

d Required when a magnetic sensor is on-board the satellite.

e Required for an observation (earth or astronomy) satellite.

f Required if launcher ICD requires or if structural analysis cannot replace testing. See Figure D.3.

g Required if very high accuracy of the structural analysis is necessary. See 8.15.

h Required if Launcher ICD requires. See Figure D.4.

i When acoustic test is also required by launcher ICD, either random vibration or acoustic test is recommended, whichever is more appropriate, with the other discretionary. See Figure D.5 for the selection logic of acoustic test.

j Required if launcher ICD requires. See Figure D.6.

k Required if design is new and thermal accuracy does not have enough accuracy. See Figure D.7.

l Either thermal vacuum or a combination of thermal cycle function and functional test in vacuum is required. See Figure D.7.

m Required if temperature distribution is large. See Figure D.7.

n Required if thermal vacuum test is not done. See Figure D.7.

o Required if separation temperature after cold launch is much higher or lower than the max/min temperature. See Figure D.8.

p Required if pressure vessel is on-board the satellite.

q Required if launcher ICD requires or if a payload sensitive to contamination is on-board the satellite.

6.2 Test level and duration

The test level and duration of system tests shall be specific to each satellite, depending on the environments to be encountered and the mission objectives. In the mechanical and thermal tests, the test level of AT shall cover the maximum temperature range and acceleration predicted. For the mechanical and thermal QT or PFT, appropriate margin shall be added.

7 Unit tests

7.1 Test items

The unit test items are listed in Tables 2, 3 and 4. Annex D provides test selection logic flows.

Table 2 — Unit test items (QT)

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
Electrical interface test	To be done at the assembly and integration to the system level.									
Functional test	R	R	R	R	R	R	R	R	R	—
Mission test	To be done in the system level.									
Total Ionization Dose (TID) test	O^a	—	—	—	—	—	—	—	—	—
Single Event Effects (SEE) test	O^a									
Spacecraft Charging Induced Electrostatic Discharge (ESD) test	O^b	—	—	O^b	—	—	—	—	—	—
Electromagnetic Compatibility (EMC) test	R	O^c	O^c	—	—	—	—	—	—	—
Deployment test	—	O^d	O^d	O^d	—	—	—	—	O^d	O^d
Magnetic field test	—	—	O^e	—	—	O^e	O^e	O^e	O^e	—
Antenna pattern test	—	R	—	—	—	—	—	—	—	—
Alignment measurement	To be done in the system level.									
Physical property measurement	R	R	R	R	R	R	R	R	R	R
Launcher/Spacecraft interface test	To be done in the system level.									
Quasi-static load test	To be done in the system level.									
Modal survey	O^f	O^f	O^f	O^f	O^f	O^f	O^f	O^f	O^f	O^f
Sinusoidal vibration test	—	O^g	—	O^g	—	—	—	—	O^g	O^g
Random vibration test	R	R	R	R	R	R	R	R	R	R
Acoustic test	—	—	—	—	—	—	—	—	—	—
Shock test	O^h	O^h	O^h	O^h	O^h	O^h	—	O^h	O^h	O^h
Thermal balance test	—	O^i	O^i	O^i	—	—	—	—	O^i	O^i
Thermal vacuum test	O^j	O^l	O^j	O^l	O^l	O^l	O^l	O^j	O^j	—
Functional test in vacuum	O^k	O^l	O^k	O^l	O^l	O^l	O^l	O^k	O^k	—
Cold/hot start test	O^m	—	O^m	—	—	O^m	—	—	—	—
Thermal cycle functional test	O^k	O^l	O^k	O^l	O^l	O^l	O^l	O^k	O^k	—
Thermal cycle endurance test	R	O^n	O^n	O^n	—	—	—	—	O^n	O^n
Pressure test	—	—	O^o	—	—	R	R	R	—	—
Leakage test	—	—	O^o	—	R	R	R	O^o	—	—
Microvibration test	—	—	O^p	—	—	—	—	—	—	—

Table 2 *(continued)*

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
Burn-in and wear-in test	—	—	Oq	—	—	—	—	—	—	—
End-to-end mission simulation	To be done in the system level.									
Bake out and out-gas test	To be done in the system level.									

R   Required; O   Optional

[a]   May be done when a manufacturer wants to demonstrate radiation tolerance. See Figure D.9.

[b]   Required if a unit is exposed to outer space and high voltage is used. See Figure D.10.

[c]   Required when unit is equipped with electrical and electronic circuits.

[d]   Required for deployable units.

[e]   Required when units use magnet.

[f]   Required if very high accuracy of the structural analysis is necessary. See 8.15.

[g]   Required if fundamental frequency is below 200 Hz. See Figure D.11

[h]   Required if sensitive material or parts to high frequency acceleration are used. See Figure D.11.

[i]   May be done at unit level if a test facility cannot house the entire system. See 8.20.2.

[j]   Required if temperature distribution is large. See Figure D.12.

[k]   Required if thermal vacuum test is not done. See Figure D.12.

[l]   Either thermal vacuum or thermal cycle plus functional test in vacuum is required (see Figure D.12) if electronics or moving parts are involved.

[m]   Required if unit is turned on immediately after separation.

[n]   Unit exposed to outer space only.

[o]   Required when unit is sealed or pressurized.

[p]   Required when need is identified by analysis.

[q]   Required when unit is not a one-shot device.

Table 3 — Unit test items (AT)

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
Electrical interface test	To be done at the assembly and integration to the system level.									
Functional test	R	R	R	R	R	R	R	R	R	—
Mission test	To be done in the system level.									
Total Ionization Dose (TID) test	—	—	—	—	—	—	—	—	—	—
Single Event Effects (SEE) test	—	—	—	—	—	—	—	—	—	—
Spacecraft Charging Induced Electrostatic Discharge (ESD) test	—	—	—	—	—	—	—	—	—	—
Electromagnetic Compatibility (EMC) test	—	—	—	—	—	—	—	—	—	—

Table 3 *(continued)*

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
Deployment test	—	Oa	Oa	Oa	—	—	—	—	Oa	Oa
Magnetic field test	—	—	—	—	—	—	—	—	—	—
Antenna pattern test	—	R	—	—	—	—	—	—	—	—
Alignment measurement	To be done in the system level.									
Physical property measurement	R	R	R	R	R	R	R	R	R	R
Launcher/Spacecraft interface test	To be done in the system level.									
Quasi-static load test	To be done in the system level.									
Modal survey	Ob	Ob	Ob	Ob	Ob	Ob	Ob	Ob	Ob	Ob
Sinusoidal vibration test	—	—	—	—	—	—	—	—	—	—
Random vibration test	R	R	R	Rc	R	R	R	R	R	R
Acoustic test	—	—	—	Oc	—	—	—	—	—	—
Shock test	—	—	—	—	—	—	—	—	—	—
Thermal balance test	—	—	—	—	—	—	—	—	—	—
Thermal vacuum test	Od	Oe	Od	Oe	Oe	Oe	Oe	Od	Od	—
Functional test in vacuum	—	—	—	—	—	—	—	—	—	—
Cold/hot start test	Of	—	Of	—	—	Of	—	—	—	—
Thermal cycle functional test	Og	Oe	Og	Oe	Oe	Oe	Oe	Og	Og	—
Thermal cycle endurance test	—	—	—	—	—	—	—	—	—	—
Pressure test	—	—	Oh	—	—	R	R	R	—	—
Leakage test	—	—	Oh	—	R	R	R	Og	—	—
Microvibration test	—	—	Oi	—	—	—	—	—	—	—
Burn-in and wear-in test	R	—	Oj	—	—	R	—	R	—	—

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Table 3 *(continued)*

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
End-to-end mission simulation	To be done in the system level.									
Bake out and outgas test	To be done in the system level.		R	To be done in the system level.						

R Required; O Optional

a Required for deployable units.

b Required if very high accuracy of the structural analysis is necessary. See 8.15.

c When acoustic test is also required by satellite system ICD, either random vibration or acoustic test is recommended, whichever is more appropriate, with the other discretionary. See Figure D.5 for the selection logic of acoustic test.

d Required when large temperature distribution exists inside the unit.

e Either thermal vacuum or thermal cycle is required if electronics or moving parts are involved.

f Required if unit is turned on immediately after separation and separation temperature after cold launch is much higher or lower than the max/min temperature. See Figure D.8. Can be combined with thermal vacuum test or thermal cycle functional test.

g Required when thermal vacuum test is not done.

h Required when unit is sealed or pressurized.

i Required when need is identified by analysis.

j Required when the total duration of the acceptance test sequence is insufficient.

Table 4 — Unit test items (PFT)

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
Electrical interface test	To be done at the assembly and integration to the system level.									
Functional test	R	R	R	R	R	R	R	R	R	—
Mission test	to be done in the system level									
Total Ionization Dose (TID) test	O[a,b]	—	—	—	—	—	—	—	—	—
Single Event Effects (SEE) test	O[a,b]									
Spacecraft Charging Induced Electrostatic Discharge (ESD) test	O[a,c]	—	—	O[a,c]	—	—	—	—	—	—
Electromagnetic Compatibility (EMC) test	R	O[d]	O[d]	—	—	—	—	—	—	—
Deployment test	—	O[e]	O[e]	O[e]	—	—	—	—	O[e]	O[e]
Magnetic field test	—	—	O[f]	—	—	O[f]	O[f]	O[f]	O[f]	—
Antenna pattern test	—	R	—	—	—	—	—	—	—	—
Alignment measurement	To be done in the system level.									
Physical property measurement	R	R	R	R	R	R	R	R	R	R
Launcher/Spacecraft interface test	To be done in the system level.									

Table 4 *(continued)*

Test items	Electrical and electronic	Antenna	MMA	Solar array	Battery	Valve	Pressure vessel	Thruster	Optical	Structural
Quasi-static load test	To be done in the system level.									
Modal survey	O[g]	O[g]	O[g]	O[g]	O[g]	O[g]	O[g]	O[g]	O[g]	O[g]
Sinusoidal vibration test	—	O[h]	—	O[h]	—	—	—	—	O[h]	O[h]
Random vibration test	R	R	R	R[i]	R	R	R	R	R	R
Acoustic test	—	—	—	O[i]	—	—	—	—	—	—
Shock test	O[j]	O[j]	O[j]	O[j]	O[j]	O[j]	—	O[j]	O[j]	O[j]
Thermal balance test	—	O[k]	O[k]	O[k]	—	—	—	—	O[k]	O[k]
Thermal vacuum test	O[l]	O[n]	O[l]	O[n]	O[n]	O[n]	O[n]	O[l]	O[l]	—
Functional test in vacuum	O[m]	O[n]	O[m]	O[n]	O[n]	O[n]	O[n]	O[m]	O[m]	—
Cold/hot start test	O[o]	—	O[o]	—	—	O[o]	—	—	—	—
Thermal cycle functional test	O[m]	O[n]	O[m]	O[n]	O[n]	O[n]	O[n]	O[m]	O[m]	—
Thermal cycle endurance test	—	O[p]	O[p]	O[p]	—	—	—	—	O[p]	O[p]
Pressure test	—	—	O[q]	—	—	R	R	R	—	—
Leakage test	—	—	O[q]	—	R	R	R	O[p]	—	—
Microvibration test	—	—	O[r]	—	—	—	—	—	—	—
Burn-in and wear-in test	R	—	O[s]	—	—	R	—	R	—	—

Table 4 *(continued)*

Test items	Electrical and electronic	Antenna	MMA	Solar array	Bat-tery	Valve	Pres-sure vessel	Thrust-er	Optical	Structur-al
End-to-end mission simulation	To be done in the system level.									
Bake out and outgas test	To be done in the system level.			R	To be done in the system level.					

R Required; O Optional

a Use dedicated test model.

b May be done when a manufacturer wants to demonstrate radiation tolerance. See Figure D.9.

c Required if a unit is exposed to outer space and high voltage is used. See Figure D.10.

d Required when unit is equipped with electrical and electronic circuits.

e Required for deployable units.

f Required when units use magnet.

g Required if very high accuracy of the structural analysis is necessary. See 8.15.

h Required if fundamental frequency is below 200 Hz. See Figure D.11.

i When acoustic test is also required by satellite system ICD, either random vibration or acoustic test is recommended, whichever is more appropriate, with the other discretionary. See Figure D.5 for the selection logic of acoustic test.

j Required if sensitive material or parts to high frequency acceleration are used. See Figure D.11.

k May be done at unit level if a test facility cannot house the entire system. See 8.20.2.

l Required if temperature distribution is large. See Figure D.12.

m Required if thermal vacuum test is not done. See Figure D.12.

n Either thermal vacuum or thermal cycle plus functional test in vacuum is required (see Figure D.12) if electronics or moving parts are involved.

o Required if separation temperature after cold launch is much higher or lower than the max/min temperature. See Figure D.8. Can be combined with thermal vacuum test or thermal cycle functional test.

p Required when units are exposed to outer space. Determined after consultation with the satellite developer.

q Required when unit is sealed or pressurized.

r Required when need is identified by analysis.

s Required when unit is not a one-shot device.

7.2   Test levels and duration

Table 5 lists the test level and duration for unit QT. The test level and duration for unit AT or PFT shall be determined after consultation between the satellite developer and the unit manufacturer. See Annex E for AT or PFT of electrical and electronic units.

Table 5 — Test level and duration of Unit QT

Test item	Test level and duration	Requirement	Note
Radiation (Total dose)	Total dose	10 kRad(= 100 Gy) or higher	The shielding effect due to the unit housing shall be taken into account.
	Dose rate	0,01 Gy/s or lower	
EMC	Power bus conducted interference, load induced, frequency domain	10 kHz to 10 MHz	Equivalent to MIL-STD.461 CE102[23].
	Radiated electric field emissions	10 kHz to 1 GHz or higher	Equivalent to MIL-STD.461 RE102[23].
Sinusoidal vibration	Vibration amplitude	8,4 G_{0-p} or higher	
	Frequency	approximately 5 Hz to 100 Hz	
	Sweep rate	4 Oct/min up and down	
	Number of applications	Once in each axis	
Random vibration	Root mean square	13,3 Grms or higher	
	PSD		This PSD may be tailored according to the satellite mass for which the manufacturer expects to use the unit under test.
	Lower tolerance limit on PSD	0 dB	
	Duration	1 min for each orthogonal axis	
	Frequency	20 Hz to 2 000 Hz	
	Number of applications	Once in each axis	
Shock	SRS		
	Number of shocks	Once in each axis	
	Q factor	10	
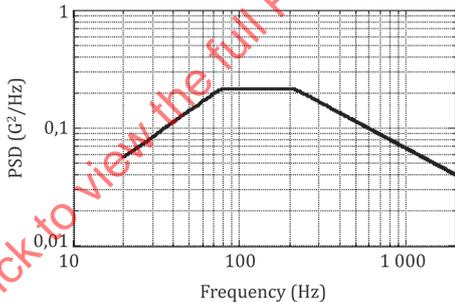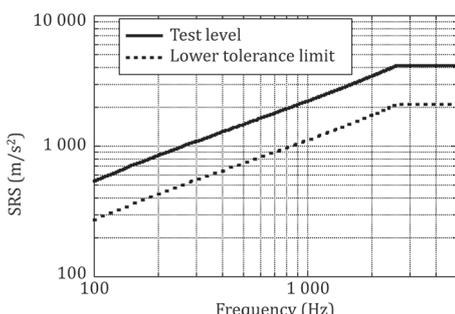
Table 5 *(continued)*

Test item	Test level and duration	Requirement	Note
Thermal vacuum (internal unit only)	Temperature range	−15 °C to +50 °C	Specify the measurement points in the datasheet.
	Number of cycles	2 or more	
	Operational soak duration	1 h or longer	
	Thermal dwell	1 h or longer	
	Tolerance limit	3 °C	
	Temperature ramp rate	+5 °C/min or slower −5 °C/min or slower	
	Chamber pressure	$1{,}0 \times 10^{-3}$ Pa or lower	
	Temperature profile at high temperature limit		
	Temperature profile at cold temperature limit		
Vacuum function	Elevated temperature	+50 °C or higher	
	Soak time at elevated temperature	60 min or longer	
Cold start	Starting temperature	−35 °C or lower	Not applicable to battery and reaction wheel.
Hot start	Starting temperature	+70 °C or higher	Not applicable to battery.

Table 5 *(continued)*

Test item	Test level and duration	Requirement	Note
Thermal cycle functional (internal unit only)	Temperature range	−15 °C to +50 °C	
	Number of cycles	2 or more	
	Operational soak duration	1 h or longer	
	Thermal dwell	1 h or longer	
	Tolerance limit	3 °C	
	Temperature ramp rate	+5 °C/min or slower	
		−5 °C/min or slower	
	Temperature profile at high temperature limit		
	Temperature profile at high temperature limit		
Thermal cycle endurance	Temperature range	−70 °C (or lower) to +100 °C (or higher) for an external unit −25 °C (or lower) to +60 °C (or higher) for an internal unit	Not applicable to battery.
	Number of cycles	24 or more	
	Soak duration	5 min or longer	
	Ramp rate	2 °C/min or slower	
	Functional test	Only before the first cycle and after the last cycle	

8 Test requirements

8.1 Electrical interface

8.1.1 Purpose of test

The electrical interface test verifies that all interface signals are within acceptable limits of applicable performance specifications. The test is important to reduce the risk of schedule delay and cost increase due to defects found during/after integration.

8.1.2   Test facilities and setup as basic requirements

There is no requirement.

8.1.3   Test article configuration

There is no requirement.

8.1.4   Monitoring during test

There is no requirement.

8.1.5   Test levels and duration

There is no requirement.

8.1.6   Test conditions and guidelines

The test shall be conducted before the integration of an assembly, unit, or subsystem into the next higher hardware assembly. Prior to mating with other hardware, electrical harnessing shall be tested to verify proper characteristics such as routing of electrical signals, impedance, isolation and overall workmanship.

8.2   Functional test

8.2.1   Purpose of test

The functional test verifies that the electrical and mechanical performance of the test article meet specification requirements.

In a small spacecraft project, environment tests are often done at a location outside the developer's institution and the test time is very limited. Anomalies found on-site are a major cause of schedule delay. Thorough testing at the developer's institution before moving to the test site will help identify whether any anomaly found on-site is due to system integration, a change of the testing place (from a laboratory to a vacuum chamber for example), or the actual environmental stress imposed to the test article. Troubleshooting on-site often takes an unexpectedly long time due to limitation of tools, personnel, etc. A thorough functional test after system integration at the developer's home institution will save the overall testing time and improve quality.

8.2.2   Test facilities and setup as basic requirements

The requirements in ISO 15864:2004, 7.2.2 shall apply.

8.2.3   Test article configuration

The test article shall be in the operational configuration corresponding to each function to be tested. For the system level test, all units or their simulated article shall be integrated in a flight-representative manner as much as possible. The simulated article may range from a mere mass dummy to an exact copy of flight hardware. The test may be conducted in atmosphere unless the functionality under test requires vacuum environment.

8.2.4   Monitoring during test

The requirements in ISO 15864:2004, 7.2.4 shall apply.

8.2.5   Test levels and duration

There is no requirement.

8.2.6 Test conditions and guidelines

The requirements in ISO 15864:2004, 7.2.6 shall apply.

8.3 Mission test

8.3.1 Purpose of test

The mission test verifies that the integrated satellite can function in all the operational modes, including contingency to satisfy the specification requirements.

8.3.2 Test facilities and setup as basic requirements

Mission tests may be conducted in atmosphere. Operational modes that require vacuum environment, such as thruster firing, may be simulated by other means. Communication with the satellite may be done either through cables or RF airlink.

8.3.3 Test article configuration

The test article shall be in the operational configuration corresponding to each operational mode as much as possible.

8.3.4 Monitoring during test

The satellite conditions shall be monitored in as much detail as possible and compared with the telemetry data.

8.3.5 Test levels and duration

Nominal operational modes shall be tested long enough to detect any latent software error that manifests only after a certain run time. The duration shall be agreed upon with the customer.

8.3.6 Test conditions and guidelines

Mission tests shall include simulation of all operational modes in nominal situations, (e.g. separation, early orbital operations, mission operations and decommissioning) within the constraints of what can be simulated on the ground. The actual sequence from separation to early orbital operation shall be simulated in actual flight sequence. If hot-launch is anticipated, the modes during final countdown, launch and ascent shall be tested.

Mission tests shall also include simulation of contingency operations within the constraints of what can be simulated on the ground. The contingency operations, which are time critical, are for example those when the satellite is or could be in danger, to recover from a safe mode, to reconfigure the satellite after a major failure, etc.

8.4 Total Ionization Dose (TID) test

8.4.1 Purpose of test

TID test verifies that the test article can withstand the radiation environment imposed during the test article mission lifetime in orbit.

For system TID test, the purpose is to identify the system behaviour when the effect of radiation damage begins to appear and to derive and/or verify a possible mitigation scheme if the expected radiation damage during the mission life may exceed the design limit.

8.4.2   Test facilities and setup as basic requirements

The test article shall be placed in a radiation test facility, such as Co^{60} (gamma ray).

8.4.3   Test article configuration

The entire test article shall be exposed to radiation such that the cumulative dose becomes as uniform as possible. The housing if any shall be removed and the circuit board shall be exposed.

For system TID test, the satellite structure and the unit housing shall be removed. The unit circuit boards shall be interconnected with each other, i.e. in the so-called flat-sat or table-sat condition and exposed to the radiation simultaneously. The units shall be made of electronics parts from flight spares. Although the boards shall be as flight representative as possible, the test article is not appropriate for flight due to cumulative damage by radiation and is regarded as a dedicated qualification test model. If some units are already proven to have enough radiation tolerance, they may be heavily shielded so that they can be used for flight. The test article may be operated by an external power supply. The power supply is controllable outside the radiation area so that the hardware can be reset by power cycling during the test without entering the radiation area.

8.4.4   Monitoring during test

The test article shall run continuously in the nominal orbit operational mode while it is being irradiated. Critical parameters to indicate are the health condition of the test article and the behaviour of the test article and these parameters shall be monitored continuously.

8.4.5   Test levels and duration

For the system TID test, the total dose during the test shall be equivalent to or greater than the expected total dose in orbit. Shielding effects due to the satellite structure shall be taken into account.

For the unit TID test, the total dose shall be at least 100 Gy. The dose rate shall be 0,01 Gy/s or less to avoid excessive positive charge build up during irradiation compared to the orbital environment where a typical dose rate is µGy/s or less. If the unit has housing, the test dose may be reduced from 100 Gy by taking into account of the shield effect.

8.4.6   Test conditions and guidelines

Before and after TID test, a complete set of functional tests shall be carried out to check the functionalities of the parts not used in nominal operation during irradiation.

8.5   Single Event Effect (SEE) test

8.5.1   Purpose of test

SEE test in this document is not intended to derive the cross-section of a single event per LET nor susceptibility of each electronic part to radiation particles. Its purpose is to understand the behaviour of the entire satellite system or an individual unit when a single event, especially single event latch-up, occurs in orbit and to verify that the designed recovery methods work as intended.

8.5.2   Test facilities and setup as basic requirements

The test article shall be placed in a radiation test facility, such as proton beam, heavy ion beam, Californium-252, etc.

8.5.3   Test article configuration

The test article shall be circuit boards and applicable flight software. The unit housing shall be removed. The satellite structure is not included. The boards shall be interconnected with each other, i.e. the so-

called flat-sat or table-sat condition. The boards shall be made of electronics parts from flight spares. Although the boards shall be as flight representative as possible, the test article is not appropriate for flight due to cumulative damage by radiation and is regarded as a dedicated qualification test model.

The energetic particles (proton, heavy ions, fission fragments) may be concentrated on particular electronics parts. A mechanism to change the irradiated area *in situ* shall be employed. The electronics parts of concern may be decapped so that the underlying IC can be directly exposed to the radiation.

The test article may be operated by external power supplies that are controllable outside the radiation area so that the hardware can be reset by power cycling during the test without entering the radiation area.

8.5.4 Monitoring during test

The test article shall run continuously in the nominal orbit operational mode while it is being irradiated. The health condition of the test article and a sign of SEE shall be monitored continuously. Especially the power consumption shall be monitored to detect latch-up phenomena.

8.5.5 Test levels and duration

Many small spacecraft operate in LEO where protons are the major source of SEE. If a proton beam is used as the radiation source of a SEE test, test duration shall be set to deliver the total fluence equivalent to that expected in orbit. If heavy ion particles are used as the radiation source of a SEE test, the total fluence shall be 100 000 particles/cm^2.

NOTE   Assuming a typical single event cross section of 0,001 cm^2 [8], a total of 100 single events occur with 100 000 particles/cm^2.

8.5.6 Test conditions and guidelines

If the design includes automatic recovery from SEE, especially single event latch-up, the recovery process shall be monitored. The radiation source may be turned off during the recovery process as it is very rare that SE occurs consecutively in a short time in orbit.

If the design includes a contingency operation from the ground, such as a reset command, the command shall be sent from the simulated ground network to the test article to demonstrate that the test article can recover from SEE.

8.5.7 Test conditions and guidelines

There is no requirement.

8.6 Spacecraft Charging Induced Electrostatic Discharge (ESD) test

8.6.1 Purpose of test

The spacecraft charging induced electrostatic discharge test verifies that the test article is free from ESD caused by spacecraft charging or meets the functional and performance requirements even after suffering ESD.

8.6.2 Test facilities and setup as basic requirements

The test article shall be placed in a vacuum chamber capable of simulating charging conditions of the test article in space in accordance with the requirements of ISO 11221:2011, 6.2.

8.6.3 Test article configuration

The test article shall be as flight representative as possible. As the test may damage the test article, a non-flight item shall be used as the test article.

8.6.4 Monitoring during test

Discharge occurrence shall be closely monitored by an appropriate detection method, such as a voltage/current probe inserted in the test article and connected to an oscilloscope or a video camera to identify a flash of light associated with ESD. The best effort shall be made to measure the discharge current waveform.

8.6.5 Test levels and duration

The maximum charging condition of the test article expected in orbit shall be simulated.

8.6.6 Test conditions and guidelines

The test article shall be charged to a certain potential positive or negative with respect to the electrical ground of the test article. The charging potential shall be measured by a high impedance voltage probe including a non-contact type surface potential meter. For each charging condition, a certain waiting time shall be taken to judge whether ESD occurs or not. The charging potential shall be raised until the threshold potential for ESD inception is found.

Once the ESD inception threshold is found, repeated ESD may be imposed on the test article to see whether ESD may trigger detrimental effects on the test article, such as transition to arc discharge, material degradation, electromagnetic interference, etc. The number of repeated discharges may be derived from the expected number of charging and discharge events in orbit in accordance with the requirements of ISO 11221:2011, 6.6.

8.7 Electromagnetic Compatibility (EMC) test

8.7.1 Purpose of test

The system EMC test demonstrates satellite compatibility with the self-induced electromagnetic environment and compliance with interface documents (launch manual, etc.). The unit EMC test demonstrate that the electromagnetic interference (EMI) emission and susceptibility of the test article comply with the requirements. The unit QT is to characterize the conduction and radiation emission noise.

8.7.2 Test facilities and setup as basic requirements

External RF signal shall be shielded. Spurious RF reflected signals shall be minimized. To prevent external conduction noise from interfering, it is the best to operate the satellite by its internal battery and communicate with the satellite using airlink.

The units EMC (emission and susceptibility) test shall comply with ISO 14302.

8.7.3 Test article configuration

The on-orbit configuration shall be tested.

8.7.4 Monitoring during test

For the system EMC test, selected critical parameters shall be monitored while the test article is operated.

8.7.5   Test levels and duration

The units EMC (emission and susceptibility) test shall comply with ISO 14302 requirements.

8.7.6   Test conditions and guidelines

For the system EMC test, if the battery permits, the satellite shall be operated at full power. The test shall demonstrate satisfactory electrical and electronic equipment operation in conjunction with expected electromagnetic radiation and conduction from other internal units in various operational modes. The electromagnetic radiation and conduction from the satellite shall not disturb the launcher if it is not cold-launched.

An electrostatic arc discharge susceptibility test shall be conducted if the spacecraft has the possibility of disturbance that may arise from electrostatic discharge on orbit. Inert EEDs shall be installed and monitored during all tests.

8.8   Deployment test

8.8.1   Purpose of test

The deployment test verifies that the deployment mechanisms, such as antenna, solar panel, etc., can function according to the design requirements.

8.8.2   Test facilities and setup as basic requirements

The test may be done in atmosphere. The difference of environment such as vacuum, temperature, microgravity, and others shall be properly assessed during test planning. If possible, the deployment should be performed in vacuum environment at the coldest temperature expected to simulate the worst-case condition.

8.8.3   Test article configuration

If the deployment involves one-shot devices, a dedicated test model shall be used. Not only the function of individual parts related to the deployment, but also the entire sequence of deployment shall be tested. Any hardware units and software involved in the deployment shall be included in the test article as much as possible. If the deployment sequence is initiated by a command from the ground station network, the RF link compatibility between the ground station network and the satellite shall be also tested.

8.8.4   Monitoring during test

The motion of the deployment shall be monitored. If possible, the power consumption and the command data flow during the deployment sequence shall be monitored.

8.8.5   Test levels and duration

There is no requirement.

8.8.6   Test conditions and guidelines

Many small spacecraft utilize deployment mechanisms to compensate their small size. Deployment is often critical to mission success. Testing deployment repeatedly may be necessary if the deployment mechanism involves parts whose quality variation is uncertain. The number of repeated deployments shall be determined from the viewpoint of securing enough confidence and the cost/schedule associated with the test. For small spacecraft, deployed elements are often held by synthetic fibre thread. Technical properties of the fibre thread such as temperature and/or creeping shall be obtained.

8.9 Magnetic field test

Requirements in ISO 15864:2004, 7.4 shall apply.

8.10 Antenna pattern test

Requirements in ISO 15864:2004, 7.5 shall apply.

8.11 Alignment measurement

Requirements in ISO 15864:2004, 7.6 shall apply. The alignment measurement test applies not only to the optical sensors, but also to the alignments of antennae, nozzles and thrusters.

8.12 Physical property measurement

Requirements in ISO 15864:2004, 7.7 shall apply.

8.13 Launcher/Spacecraft interface test

Requirements in ISO 15864:2004, 7.9 shall apply.

8.14 Quasi-static load test

8.14.1 Purpose of test

The quasi-static load test demonstrates the ability of the primary satellite structure to sustain the quasi-static loads induced by the launch vehicle.

8.14.2 Test facilities and setup as basic requirements

The test article shall be placed in a vibration test facility capable of generating a sine burst for the test requirement. The test article shall be mounted to a fixture at the normal mounting points. The sine burst shall be applied in three mutually orthogonal directions, one being parallel to the thrust axis.

8.14.3 Test article configuration

The test article shall be in launch configuration. Propellant tanks may be empty during spacecraft testing, provided the filled tank configuration has been test-validated. Antennae and other devices that extend or change position after orbital injection shall be in launch configuration during the test article response.

In the case of the CubeSat system test, the CubeSat shall be tested after being integrated with POD.

8.14.4 Monitoring during test

The test article shall be in the same condition as in the launch phase. For the cold launch case, chattering of the power inhibit switches shall be monitored independently for each switch. Before and after each quasi-static load test, a signature check (low level random vibration or sine sweep starting from 5 Hz) shall be conducted to check any change in the structural integrity. During the functional tests before and after the shaker test series (see Figure D.4), battery health shall be checked using the open voltage, the charging/discharging characteristics, or others.

8.14.5 Test levels and duration

The test levels shall be derived from the respective launch vehicle user manual.

8.14.6 Test conditions and guidelines

See Reference [19] for details of the sine burst test.

8.15 Modal survey

8.15.1 Purpose of test

The modal survey test determines the natural frequency to verify the result of structural analysis. The structural analysis result, once verified with sufficient accuracy, can be used to demonstrate the strength of the primary structure against quasi-static load. For small spacecraft, structural analysis is often very simple and does not include details such as mode shapes and damping factors. Therefore, the test methodology is not as detailed as the one for traditional satellites. Often identifying only the natural frequencies suffices the purpose.

8.15.2 Test facilities and setup as basic requirements

The test article shall be placed in a vibration test facility. The test article shall be mounted to a fixture at the normal mounting points.

8.15.3 Test article configuration

Requirements in ISO 15864:2004, 7.12.3 shall apply.

In the case of the CubeSat system test, the CubeSat shall be tested after being integrated with POD.

8.15.4 Monitoring during test

Requirements in ISO 15864:2004, 7.12.4 shall apply.

8.15.5 Test levels and duration

Requirements in ISO 15864:2004, 7.12.5 shall apply.

8.15.6 Test conditions and guidelines

Low-level random vibration or sine-sweep is applied.

8.16 Sinusoidal vibration test

8.16.1 Purpose of test

Requirements in ISO 15864:2004, 7.13.1 shall apply.

8.16.2 Test facilities and setup as basic requirements

Requirements in ISO 15864:2004, 7.13.2 shall apply.

8.16.3 Test article configuration

Requirements in ISO 15864:2004, 7.13.3 shall apply.

In the case of the CubeSat system test, the CubeSat shall be tested after being integrated with POD.

8.16.4 Monitoring during test

During the system vibration test, the test article shall be in the same condition as in the launch phase. For the cold launch case, chattering of the power inhibit switches shall be monitored independently for each switch. Before and after each vibration test, signature check (low level random vibration or sine sweep starting from 5 Hz) shall be conducted to check any change in the structural integrity. During the functional tests before and after shaker test series (see Figure D.4), battery health shall be checked using the open voltage, the charging/discharging characteristics, or others.

8.16.5 Test levels and duration

For the system vibration test, test levels and duration shall be derived from the respective launch vehicle user manual. For QT of large flexible structural unit such as solar panel, see Table 5. For unit AT or PFT, the vibration level and duration shall be determined after consultation between the satellite developer and the unit manufacturer.

8.16.6 Test conditions and guidelines

Requirements in ISO 15864:2004, 7.13.6 shall apply.

8.17 Random vibration test

8.17.1 Purpose of test

Requirements in ISO 15864:2004, 7.14.1 shall apply.

8.17.2 Test facilities and setup as basic requirements

Requirements in ISO 15864:2004, 7.14.2 shall apply.

8.17.3 Test article configuration

Requirements in ISO 15864:2004, 7.14.3 shall apply.

In the case of the CubeSat system test, the CubeSat shall be tested after being integrated with POD.

8.17.4 Monitoring during test

During system vibration test, the test article shall be in the same condition as in the launch phase. For the cold launch case, chattering of the power inhibit switches shall be monitored for each switch independently. Before and after each vibration test, signature check (low-level random vibration or sine sweep starting from 5 Hz) shall be conducted to check any change in the structural integrity. During the functional tests before and after shaker test series (see Figure D.4), battery health shall be checked using the open voltage, the charging/discharging characteristics, or others.

8.17.5 Test levels and duration

For the system vibration test, test levels and duration shall be derived from the respective launch vehicle user manual. For unit QT, see Table 5. For unit AT or PFT, the vibration level and duration shall be determined after consultation between the satellite developer and the unit manufacturer. If the unit AT or PFT is done as ESS, the vibration level shall encompass the maximum predicted environment.

8.17.6 Test conditions and guidelines

Requirements in ISO 15864:2004, 7.14.6 shall apply.

8.18 Acoustic test

8.18.1 Purpose of test

Requirements in ISO 15864:2004, 7.15.1 shall apply.

8.18.2 Test facilities and setup as basic requirements

Requirements in ISO 15864:2004, 7.15.2 shall apply.

8.18.3 Test article configuration

Requirements in ISO 15864:2004, 7.15.3 shall apply.

In the case of the CubeSat system test, the CubeSat shall be tested after being integrated with POD.

8.18.4 Monitoring during test

Requirements in ISO 15864:2004, 7.15.4 shall apply.

For the cold launch case, chattering of the power inhibit switches shall be monitored for each switch independently.

8.18.5 Test levels and duration

Requirements in ISO 15864:2004, 7.15.5 shall apply.

8.18.6 Test conditions and guidelines

Requirements in ISO 15864:2004, 7.15.6 shall apply.

8.19 Shock test

8.19.1 Purpose of test

Requirements in ISO 15864:2004, 7.16.1 shall apply.

8.19.2 Test facilities and setup as basic requirements

The test article in a flight configuration shall be installed in a shock test facility capable of generating shock response spectra simulating the environment. Where feasible, flight structure and flight duplicate shock-producing hardware may be employed.

8.19.3 Test article configuration

Requirements in ISO 15864:2004, 7.16.3 shall apply.

In the case of the CubeSat system test, the CubeSat shall be tested after being integrated with POD.

8.19.4 Monitoring during test

During the system shock test for fairing separation or satellite separation, the test article shall be in the same condition as in the launch phase. When the fairing separation is simulated for a cold-launched satellite, chattering of the power inhibit switches shall be monitored independently for each switch.

After the system shock test for satellite separation, the test article shall be operated and monitored for malfunctions.

During the functional tests before and after a shock test, battery health shall be checked using the open voltage, the charging/discharging characteristics, or others.

8.19.5 Test levels and duration

For the system shock test, test levels and duration are derived from the respective launch vehicle user manual.

For unit QT, see Table 5. For unit AT or PFT, the SRS level and the number of tests shall be determined after consultation between the satellite developer and the unit manufacturer.

8.19.6 Test conditions and guidelines

Test conditions shall be determined by the type of shock and shock phase. Rehearsals using STM, a dummy satellite, or a dummy mass are recommended to find appropriate test conditions to provide the required SRS. It should be noted that SRS might differ significantly if the structural type of the dummy used in the rehearsal is different from the test article.

8.20 Thermal balance test

8.20.1 Purpose of test

Requirements in ISO 15864:2004, 7.17.1 shall apply.

8.20.2 Test facilities and setup as basic requirements

The thermal balance test shall be conducted on the satellite. For deployable units, the thermal balance test may be conducted at unit level due to restrictions on physical dimensions of the test facility.

8.20.3 Test article configuration

Requirements in ISO 15864:2004, 7.17.3 shall apply.

8.20.4 Monitoring during test

The test article shall be operated and monitored throughout the test. Functional tests shall be conducted before and after the test.

8.20.5 Test levels and duration

The approach to thermal balance test shall be based on simulating the thermal conditions that are predicted to be most critical for the thermal control subsystem or for the unit that has the narrowest acceptable temperature range, such as the battery. The test profile depends on the mission, satellite design, satellite operating modes, and times required to reach stabilization. At least, several cases shall be tested to verify the thermal analysis. To see the functionality of the thermal control subsystem, the worst hot and/or cold case shall be tested. The exposure shall be long enough for the test article to reach stabilization or steady periodic variation so temperature distributions may be verified.

8.20.6 Test conditions and guidelines

Success criteria depend on

a) thermal control hardware that meets the design requirement, and

b) obtaining test data that correlates with mathematical models.

Lack of correlation with theoretical models may indicate a deficiency in the test mode, test setup, test article hardware, or mistakes in the mathematical model itself. The correlated thermal mathematical model shall be used to make the final temperature predictions for the various mission phases.

8.21 Thermal vacuum test

8.21.1 Purpose of test

Requirements in ISO 15864:2004, 7.18.1 shall apply.

8.21.2 Test facilities and setup as basic requirements

The test article shall be placed in a thermal vacuum chamber and a functional test shall be performed to ensure readiness for chamber closure. The chamber may be programmed to achieve the required thermal extreme sequence. Execution of the sequence shall be coordinated with expected environmental conditions, and a complete cycling of all hardware shall be performed, including the operation and monitoring of redundant units and paths. Strategically placed witness plates or other instrumentation shall be installed in the test chamber to measure the outgassing from the satellite and units. It is advised, in case of emergency, to be aware of the difficulty of stopping the test and removing the satellite from the chamber.

8.21.3 Test article configuration

Requirements in ISO 15864:2004, 7.18.3 shall apply.

8.21.4 Monitoring during test

The test article shall be operated and monitored throughout the test. A functional test shall be conducted before, during, and after the test. The functional tests performed during the thermal vacuum test shall include the following minimum requirements:

a)   electrical discharge check during pumping down if the satellite is operated during the launch phase;

b)   operation at minimum cold temperature stress conditions;

c)   operation at maximum hot temperature stress conditions;

d)   transitions between temperature extremes while operating;

e)   hot and cold turn-on capabilities shall be demonstrated at extreme conditions.

8.21.5 Test levels and duration

For the system test, the test temperature shall be determined on the following conditions:

a)   QT: the design temperature and qualification margin (minimum ±5 °C) with two cycles or more;

b)   AT: the design temperature with two cycles or more.

The soak duration, test profile and test configuration shall be specified in the test specification based on a number of factors such as the test article design (test operation modes, heat inertial characteristics, etc.). Test facility characteristics (heat capacity, heater configuration, etc.) can also affect configuration.

For the unit QT test, see Table 5. For unit AT or PFT, the test level and duration shall be determined after consultation between the satellite developer and the unit manufacturer.

8.21.6 Test conditions and guidelines

Requirements in ISO 15864:2004, 7.18.6 shall apply.

8.22 Functional test in vacuum

8.22.1 Purpose of test

The functional test in vacuum test demonstrates the ability of the test item to meet the design requirements under vacuum conditions and high temperature.

8.22.2 Test facilities and setup as basic requirements

A vacuum chamber shall be used to adequately evaluate the effects of vacuum environment on electrical and mechanical functions of the test article. The vacuum level shall be 0,01 Pa or better to simulate operation after separation.

NOTE   For 300 K nitrogen gas, the free molecular flow condition (Knudsen number >1) for characteristic length of 50 cm is satisfied by 0,013 Pa.

8.22.3 Test article configuration

The test article shall be placed in a vacuum chamber. For the system level test, all the units or their simulated article shall be integrated in a manner as flight representative as possible. The simulated article ranges from a mere mass dummy to an exact copy of flight hardware. Functionality at high temperature shall be checked. The temperature may be raised by an external heat source or by deliberately turning on the internal equipment to check the functionality of the electronics under a heat concentrated situation at high temperature.

8.22.4 Monitoring during test

A database of critical parameters shall be established for trend analysis. The trends shall be compared with those obtained in atmosphere to elucidate the effects of vacuum environment.

If the satellite operates during the ascent phase, corona and multipacting shall be checked during pumping down.

8.22.5 Test levels and duration

Two temperature levels, room temperature and an elevated temperature, shall be imposed. Regarding the elevated temperature, see Table 5 for unit QT. For unit AT or PFT, the temperature shall be determined after consultation between the satellite developer and the unit manufacturer. For system QT or PFT, qualification margin shall be added. For AT, the design temperature shall be used.

8.22.6 Test conditions and guidelines

Units that operate during ascent shall be operated and monitored for corona and multipacting during chamber pressure reduction. Radio frequency (RF) units shall be operated at the maximum power and design frequency to demonstrate the absence of corona and multipaction.

8.23 Cold/Hot start test

8.23.1 Purpose of test

Cold/Hot start test verifies that the test article can be turned on and initiate the starting sequence following separation from a launcher.

8.23.2 Test facilities and setup as basic requirements

The test article shall be placed in a thermal chamber (or a thermal vacuum chamber) and cooled or heated to a pre-determined temperature. After confirming that the temperature becomes steady, the test article shall be turned on.

8.23.3 Test article configuration

The on-orbit configuration immediately after separation shall be represented as much as possible. Special care should be taken so that the temperatures of units that can be damaged due to low or high temperature, such as the battery, do not exceed their limit.

8.23.4 Monitoring during test

Make sure that the test article goes through the starting sequence. If the test article involves any deployment mechanism within 30 min from separation, confirm that the key parts and units for deployment are correctly activated. Actual deployment is not necessary.

NOTE   It is enough for a satellite to exit eclipse in 30 min.

8.23.5 Test levels and duration

For the system level test, the starting temperature shall be determined after consultation with the satellite developer and the launch provider based on the result of thermal analysis.

For the unit level qualification test, the starting temperature shall be −35 °C for cold test and 65 °C for hot test. For the acceptance or proto-flight test, the condition shall be determined after consultation with the satellite developer and the unit manufacturer.

NOTE   The design of internal electronics is often based on a temperature range of −15 °C to +45 °C, with a margin of ±20 °C added.

8.23.6 Test conditions and guidelines

The test does not intend to verify steady operation at the given low or high temperature as the test article will soon reach a temperature within the predicted operational envelope due to heat generated internally and the external heat input (for cold case), or radiation toward outer space (for hot case).

8.24 Thermal cycle functional test

8.24.1 Purpose of test

The thermal cycle functional test demonstrates the ability of the test article to meet all functional and performance requirements over a temperature range at ambient pressure. The thermal cycle functional test is carried out to replace the thermal vacuum test.

8.24.2 Test facilities and setup as basic requirements

The test article shall be placed in a thermally controlled chamber. The chamber may be programmed to achieve the required thermal extreme sequence. Execution of the sequence shall be coordinated with expected environmental conditions, and a complete cycling of all hardware shall be performed, including the operation and monitoring of redundant units and paths.

8.24.3 Test article configuration

The test article shall be placed in a thermally controlled chamber having the capability to expose the test article to environments equal to or beyond the minimum and maximum test temperatures. It may be necessary to achieve temperature limits at certain locations by altering thermal boundary conditions locally or by altering the operational sequence to provide additional heating or cooling. Not only on-board thermal sensors but also monitoring thermal sensors for testing shall be installed as indicated by the thermal analysis for monitoring and acquiring thermal data.

8.24.4 Monitoring during test

The test article shall be operated and monitored throughout the test. A functional test shall be conducted before, during, and after the test. The functional tests performed during the thermal vacuum test shall include the following minimum requirements:

a) operation at minimum cold temperature stress conditions;

b) operation at maximum hot temperature stress conditions;

c) transitions between temperature extremes while operating;

d) hot and cold turn-on capabilities shall be demonstrated at extreme conditions.

8.24.5 Test levels and duration

For the system test, the test temperature and the number of cycles shall be determined on the following conditions:

a) QT: the design temperature and qualification margin (minimum ±5 °C) with two cycles or more;

b) AT: the design temperature with two cycles or more.

The soak duration, test profile and test configuration shall be specified in the test specification based on a number of factors such as the test article design (test operation modes, heat inertial characteristics, etc.). Test facility characteristics (heat capacity, heater configuration, etc.) can also affect configuration.

For the unit QT test, see Table 5. For unit AT or PFT, the test level and duration shall be determined after consultation between the satellite developer and the unit manufacturer. If the unit AT or PFT is done as ESS, the temperature range shall encompass the maximum predicted environment.

8.24.6 Test conditions and guidelines

Throughout the test, humidity control shall be exercised to maintain chamber test air temperature above the dew point temperature, in order to avoid condensation. Auxiliary heating and cooling may be employed for selected temperature-sensitive units (e.g. batteries).

8.25 Thermal cycle endurance test

8.25.1 Purpose of test

The thermal cycle endurance test demonstrates the ability of the test article to withstand the stress imposed by thermal cycles in orbit.

8.25.2 Test facilities and setup as basic requirements

The test article shall be placed in a thermally controlled chamber.

8.25.3 Test article configuration

The test article shall be placed in a thermally controlled chamber having the capability to expose the test article to environments equal to or beyond the minimum and maximum test temperatures.

8.25.4 Monitoring during test

A functional test and visual inspection shall be performed before and after the test. The temperature of the test article shall be monitored constantly by multiple thermocouples attached to the test article.

8.25.5 Test levels and duration

For the unit QT test, see Table 5. For unit AT or PFT, the test level and duration shall be determined after consultation between the satellite developer and the unit manufacturer.

8.25.6 Test conditions and guidelines

Throughout the test, humidity control shall be exercised to maintain chamber test air temperature above the dew point temperature, in order to avoid condensation. One complete thermal cycle is a period beginning at ambient temperature, transitioning to one temperature extreme and stabilizing, then transitioning to the other temperature extreme and stabilizing, and finally returning to ambient temperature.

8.26 Pressure test

Requirements in ISO 15864:2004, 7.20 shall apply.

8.27 Leakage test

Requirements in ISO 15864:2004, 7.21 shall apply.

8.28 Microvibration test

8.28.1 Purpose of test

The microvibration test characterizes the mechanical disturbance from a unit that can be a source of microvibration, such as a reaction wheel or other rotatory/translatory mechanism. Microvibration may cause mechanical jitter on the satellite, which has detrimental effects on missions requiring precise pointing accuracy. The microvibration test is carried out by the unit manufacturer and the results are included in the datasheet.

8.28.2 Test facilities and setup as basic requirements

The test article shall be placed on a defined base configuration, for example a seismic mass table that is decoupled from the laboratory environment and isolated from external disturbance.

8.28.3 Test article configuration

The test article shall be placed on the defined base configuration. The vibration caused by the test article shall be measured by a set of suitable sensors. The best practice is to use force measurement sensors. The background vibration noise of the test assembly environment shall be known prior the test article characterization.

8.28.4 Monitoring during test

The output of the used sensors shall be monitored and analysed in the time and frequency domain. For example in the frequency domain FFT and power spectral density plots indicate test article noise and characteristic eigenfrequencies.

8.28.5 Test levels and duration

The test duration shall be chosen by consideration of measurement accuracy, environmental conditions and lifetime aspects.

8.28.6 Test conditions and guidelines

Mechanisms shall be actuated according to their mission profiles with consideration on environmental and lifetime aspects (e.g. gravity, vacuum, temperature). In the case of a reaction wheel, the rotational speed shall be varied between zero and the guaranteed rotation speed or more.

8.29 Burn-in and wear-in test

Requirements in ISO 15864:2004, 7.22 shall apply.

8.30 End-to-end mission simulation

8.30.1 Purpose of test

The end-to-end tests verify that the satellite can be controlled and operated by the specified ground network. The emphasis is to verify the flight software on board the satellite and communication between the satellite and the ground station.

8.30.2 Test facilities and setup as basic requirements

The setup shall include hardline and RF airlink for the telemetry/telecommand signal. If the licence permits and sufficient signal strength is obtained without antenna deployment, the compatibility of the RF airlink between the satellite and the ground network shall be tested at least once before launch. The satellite may be put in atmosphere. The operational modes that require vacuum environment, such as thruster firing, may be simulated by other means.

8.30.3 Test article configuration

The flight configuration is mandatory except for deployable units and fuelled propulsion subsystems.

8.30.4 Monitoring during test

Test data shall be compared with previously obtained data to identify trends in performance parameters.

8.30.5 Test levels and duration

Nominal operational modes shall be tested long enough to detect any latent software error that manifests only after a certain run time. The duration shall be agreed upon with the customer.

8.30.6 Test conditions and guidelines

End-to-end tests shall include simulation of all operational modes in nominal situations (for example: early orbital operations, mission operations, and decommissioning) within the constraints of what can be simulated on the ground. End-to-end tests shall also include simulation of contingency operations. The contingency operations, which are time critical, are for example those when the satellite is or could be in danger, to recover from a safe mode, to reconfigure the satellite after a major failure, etc.

End-to-end tests serve as an important simulation of satellite operation and control. The actual personnel who are responsible for satellite operation and control shall participate in the test to check the user interface of the ground network and its software and become familiar with the expected data during the nominal situation and the procedure during contingency operations.

The responsibility of satellite tracking from launch to separation shall be defined in the ICD with the launch provider. If the responsibility is at the hand of the satellite developer, the end-to-end test to check the communication compatibility during those phases shall comply with the ICD.

8.31 Bake out and outgas test

8.31.1 Purpose of test

The bake out and outgas test verifies that the satellite flight model can meet the outgas requirement in case of piggy-back launch.

8.31.2 Test facilities and setup as basic requirements

The test article shall be placed in a vacuum chamber capable of achieving vacuum level at least $1,0 \times 10^{-3}$ Pa or better. A method to measure outgas deposition shall be incorporated in the chamber, such as a QCM or contamination witness plates. The chamber shall be equipped with a controllable heating unit, such as IR lamp.

8.31.3 Test article configuration

The test article shall be flight hardware.

8.31.4 Monitoring during test

The chamber pressure and the test article temperature shall be monitored continuously.

8.31.5 Test levels and duration

The temperature level and duration shall be defined in the ICD with the launch provider.

8.31.6 Test conditions and guidelines

Functional tests shall be carried out before and after the bake out and outgas test to ensure that no detrimental effects are caused by the test. A procedure to prevent contamination of the test article from outgassed material shall be incorporated.

8.32 Tailoring and waiver guides

Table 6 lists tailoring and waiver guides of each test. Annex A or ISO 15864 shall be referred.

Table 6 — Tailoring and waiver guides

	Test items	Tailoring guide
1	Electrical interface test	No requirements
2	Functional test	No requirements
3	Mission test	A.1
4	Total Ionization Dose (TID) test	A.2
5	Single Event Effects (SEE) test	A.3
6	Spacecraft Charging Induced Electrostatic Discharge (ESD) test	A.4
7	Electromagnetic Compatibility (EMC) test	A.5
8	Deployment test	A.6
9	Magnetic field test	ISO 15864:2004, 7.4.7
10	Antenna pattern test	ISO 15864:2004, 7.5.7
11	Alignment measurement	ISO 15864:2004, 7.6.7
12	Physical property measurement	ISO 15864:2004, 7.7.7
13	Launcher/Spacecraft interface test	No requirements
14	Quasi-static load test	A.7
15	Modal survey	A.8
16	Sinusoidal vibration test	A.9
17	Random vibration test	A.10
18	Acoustic test	A.11
19	Shock test	A.12
20	Thermal balance test	A.13
21	Thermal vacuum test	A.14
22	Functional test in vacuum	A.15
23	Cold/Hot start test	A.16
24	Thermal cycle functional test	A.17
25	Thermal cycle endurance test	A.18
26	Pressure test	ISO 15864:2004, 7.20.7
27	Leakage test	ISO 15864:2004, 7.21.7
28	Microvibration test	A.19
29	Burn-in and wear-in test	ISO 15864:2004, 7.22.7
30	End-to-end mission simulation	No requirements
31	Bake out and outgas test	A.20

Annex A
(normative)

Tailoring and waiver guides

A.1 Mission test

Mission tests except separation, deployment and thruster firing may be conducted as a part of the end-to-end mission simulation test.

A.2 Total Ionization Dose (TID) test

Whether the system TID test is required or not depends on the expected total dose. If the expected total dose during the mission lifetime (typically from months to three years for small spacecraft seeking low cost and fast delivery) is less than 100 Gy including the shielding effect due to the satellite structure, TID test is not required. See Reference [20] to calculate the expected total dose for a given orbit. If all units sensitive to TID are already tested at the unit level, the system TID test is not required. See Figure D.1 for the logic flow.

A.3 Single Event Effect (SEE) test

When the manufacturer wants to demonstrate the product's tolerance against SEE, a SEE test should be carried out. The SEE test should verify that the test article has either a negligible probability of single event occurrence or a capability to recover from SEE by itself. If the unit SEE test is not carried out, the satellite system should be equipped with a capability to recover the unit by external means, such as power cycling.

A.4 Spacecraft Charging Induced Electrostatic Discharge (ESD) test

Spacecraft charging induced ESD becomes a concern when

a) the satellite will encounter energetic electrons that make at least one of the following phenomena possible:

1) the satellite potential with respect to the plasma becomes negative, a few hundred volts or more,

2) the electrons penetrate dielectrics and bury negative charge,

3) the electrons charge the internal circuit board,

b) the satellite flies in LEO regardless of its orbital inclination and satisfies at least one of the following conditions:

1) the solar panel generates at a voltage of 70 V or higher,

2) there is an exposed conductor whose potential is more positive than the satellite chassis by 55 V or higher,

c) the satellite carries a charge emission device, such as an electric propulsion system,

d) a unit is capable of supplying current with more than 1 A at 30 V or higher, e.g. a solar panel, an electrical power unit, a slip ring, etc., that has a possibility of arc discharge sustained by its own power once triggered by an ESD, and

e) the mission, such as scientific measurement, requires strict control of satellite charging.

Whether charging induced ESD test is required or not should be determined by assessing the detrimental effects caused by ESD. The effect of ESD is defined by the capacitance of the charged material, insulator or floating metal. See Figure D.10 for the decision logic flow.

If the discharge current can be characterized and the effect of ESD is limited to electromagnetic interference only, the ESD test may be carried out as a part of EMC susceptibility test where electromagnetic pulses expected from ESD are either injected into the test article as conduction noise or radiated toward the test article as radiation noise.

A.5 Electromagnetic Compatibility (EMC) test

The system EMC test may be carried out as a part of the functional tests.

A.6 Deployment test

The deployment test may be carried out as a part of the functional tests, functional test in vacuum, or thermal vacuum test.

A.7 Quasi-static load test

Notching is permitted so that acceleration at the test article does not exceed the maximum quasi-static load provided by the launcher ICD to prevent overstress.

It is convenient to carry out the quasi-static load test in conjunction with the system vibration test using the same shaker machine.

A.8 Modal survey

The modal survey test is not generally required if natural frequencies are identified by other mechanical environment tests. The test is only required for special cases when very high accuracy of structural analysis is needed.

During the quasi-static, sinusoidal vibration, and random vibration tests, before and after the high level mechanical stress is applied, a low-level vibration is applied to see any change in the natural frequencies. This procedure is called "signature check" and included in the procedures of those mechanical tests.

A.9 Sinusoidal vibration test

Notching is permitted to prevent overstress. These cases concern items having fundamental frequencies below 100 Hz.

Most of the small spacecraft units have fundamental frequencies far above 100Hz. If the result of modal survey and/or structural analysis shows that the first fundamental frequency is more than 200Hz, the sinusoidal vibration test is not necessary for the unit.

The sinusoidal vibration test becomes necessary when the random vibration test cannot verify the strength of the test article at low frequencies.

NOTE   For a damping vibration system with single degree of freedom, the square of transmission factor at 100 Hz becomes less than 2 (1,7 to 1,8) if the resonant frequency is 200 Hz or more.

A.10 Random vibration test

Notching is permitted to prevent overstress. For a vibration test, at the resonance frequency of the test article, an excessive stress that won't occur in reality may be applied due to the dynamic absorber effect if the test level derived by taking an envelope of various spectra is used. To protect the test article from such unrealistic stress, the test level may be notched at particular frequencies. See Reference [21] for more detail.

For small spacecrafts, the random vibration test is carried out instead of acoustic test due to their small size.

A.11 Acoustic test

The random vibration test may be conducted instead of an acoustic test for small, compact test articles that can be exited more effectively through interface vibration than by an acoustic field.

Assuming the sound pressure level of 138 dB at 200 Hz with 1/3 octave band[17], the equivalent PSD is approximately given by:

$$(PSD)_a = \frac{(2\times10^{-5})^2 \times 10^{\frac{138}{10}}}{0,232\times200}\left(\frac{S}{M}\right)^2 = 548\left(\frac{S}{M}\right)^2\left[\left(m/s^2\right)^2/Hz\right]$$

where

S is the surface area, in m^2;

M is the mass, in kg.

Comparing this with a typical random vibration $(PSD)_v$ of 0,04 g^2/Hz at 200 Hz[17], which is 3,8 in $(m/s^2)^2$/Hz, the acoustic PSD exceeds the vibration PSD under the following condition:

$$\left(\frac{S}{M}\right) > \sqrt{\frac{3,8}{548}} = 0,083 \quad [m^2/kg]$$

A unit with large surface-to-mass ratio, such as a solar array, can be excited more effectively through the acoustic field. For a satellite of 50 cm × 50 cm × 50 cm volume and 50 kg mass, the surface-to-mass ratio is 0,025 m^2/kg, which makes the interface vibration more important.

For many cases, the shaker test such as modal survey, quasi-static load and sinusoidal vibration, is required. Then, the random vibration test is preferred as the test can be done easily as a part of the shaker test series as shown in Figure D.4.

On the other hand, the acoustic test can avoid overstress generated by the random vibration test due to the dynamic absorber effect. The acoustic test may be chosen when the conditions in Figure D.5 are met.

A.12 Shock test

If the specified shock spectrum is enveloped by the random vibration spectrum and propagation path and damping characteristics are well known, the shock test may be eliminated.

Relays and crystals are generally sensitive to shock. If the units contain such parts, unit QT or PFT should be done.

A.13 Thermal balance test

See Figure D.7 for the assessment on whether thermal balance test is required or not.

A.14 Thermal vacuum test

See Figure D.7 or Figure D.12 for the assessment on whether the thermal vacuum test is required or not. See Annex F for further explanation.

A.15 Functional test in vacuum

The functional test in vacuum is meant to fulfil some objectives of the thermal vacuum test by focusing only on the effects of vacuum, i.e. less heat transfer, outgas, leakage, lubrication, etc. A particular concern is concentration of heat at specific electronics parts that may not be possible to simulate by a thermal cycle chamber in atmospheric pressure. A thermal vacuum test dealing with low extreme temperature needs a thermal vacuum chamber that is not often readily available to the developers of small spacecraft. Unless significant temperature distribution develops within a satellite body, functionality at hot and cold temperature extremes may be tested by thermal cycle tests. See Annex F.

The functional test in vacuum is not necessary if thermal vacuum test is carried out.

The functional test in vacuum as system AT/PFT may be combined with bake out and outgas.

A.16 Cold/Hot start test

The cold test should be considered when a satellite is separated from a launcher in a shadowed condition. The satellite may be placed in the shadow of other satellites and may reach extremely cold temperatures if it stays in shadow for a significantly long time. The hot test should be considered when a satellite is separated from a launcher in an illuminated condition. The satellite may be heated due to aerodynamic heating within the fairing and may reach extremely hot temperatures if it stays in the illuminated side for a significantly long time. This case may be applied to a small auxiliary satellite.

The cold test is required if the thermal analysis indicates that temperature at the time of separation is less than the minimum predicted operational temperature by 10 °C or more. The hot test is required if the thermal analysis indicates that temperature at the time of separation is greater than the maximum predicted operational temperature by 10 °C or more. Both or either of the cold and hot tests may be required.

The cold/hot test may be combined with either thermal vacuum or thermal cycle test.

The unit cold/hot test is not applicable to the battery.

A.17 Thermal cycle functional test

See Figure D.7 or Figure D.12 for the assessment on whether the thermal cycle test is used to replace the thermal vacuum test or not. See Annex F for further explanation. Note that the order of the vacuum function test and the thermal cycle test in Figure D.7 and Figure D.12 does not matter.

A.18 Thermal cycle endurance test

Unit AT or PFT test may be carried out as a part of the thermal vacuum test or the thermal cycle test.

A.19 Microvibration test

The microvibration test may be required for a unit to be used for a satellite with optical mission, earth observation, optical communication, astronomy, etc. The requirement on the maximum mechanical jitter allowable should be derived from the system design through analysis. See Reference [7] and references therein regarding derivation of the microvibration requirement.

A.20 Bake out and outgas test

Bake out and outgas test may be combined with thermal vacuum test or the cumulated high temperature soak time. Baking before the thermal vacuum test may be included in the total bake out time.

Annex B
(informative)

Basis of test levels and duration

B.1 Radiation

In Table 5, the total dose of 10 kRad corresponds to three years total dose of silicon in a simple aluminium sphere of 2 mm orbiting in 700 km circular orbit at 98° inclination. See Reference [20] for an example calculation. Typical small spacecrafts have external panels of 2 mm or thicker.

B.2 Thermal

In Table 5, the number of cycles in the thermal vacuum test or the thermal cycle function test unit QT was specified as two or more. If it is only one cycle, we may have some singularities. Therefore, at least two cycles are required. Other standards require four or eight cycles. But to reduce cost, two is enough as the minimum starting number.

The temperature ranges of internal units on-board various 50 kg-class satellites are listed in Table B.1. They are either flight data or analysis results verified by the thermal balance test. The temperature range of approximately −15 °C to +45 °C in Table 5 for thermal vacuum, thermal cycle and vacuum function tests correspond to the coldest and the hottest cases in Table B.1.

Table B.1 — Temperature range of internal units on-board 50 kg-class satellites

Satellite	Lowest (°C)	Highest (°C)	Orbit type	Altitude (km)	Flight or Analysis
A	−15	+45	SSO	520	Analysis
B	+10	+45	SSO	555	Analysis
C	−6	+23	SSO	680	Flight
D	−10	+50	SSO	550	Analysis
E	0	+40	SSO	500	Analysis

The temperatures of cold and hot starts in Table 5, −35 °C and +65 °C, were derived by adding a margin of 20 °C to both limits. Thermal cycle endurance test limits for external units, approximately −70 °C to +100 °C, corresponds to a typical temperature range of solar panels at SSO. For an internal unit, Table 5 specifies approximately −25 °C to +55 °C. This is a non-operational temperature range. This range was derived by adding 10 °C margin to the operational temperature range of approximately −15 °C to +45 °C.

B.3 Mechanical

In Table 5, the PSD spectrum of the random vibration spectrum was derived based on analysis of test data from various 50 kg-class satellites and basic research carried out for this project. Figure B.1 shows the PSD spectrum obtained from Reference [12], where statistical analysis was carried out to derive the upper and lower limits of the amplification factors at several frequency ranges based on log-normal distribution. The spectrum in Figure B.1 was obtained by multiplying PSD spectrum of a typical launcher by the amplification factors. As the PSD of Reference [12] is difficult to realize by a shaker, it was smoothed by three straight lines to derive the PSD spectrum in Table 5. It should be noted that the PSD shown in Figure B.1 assumes a satellite of 50 kg size. If a unit under test is intended to be used for much heavier satellite, the PSD level may be tailored to lower values.
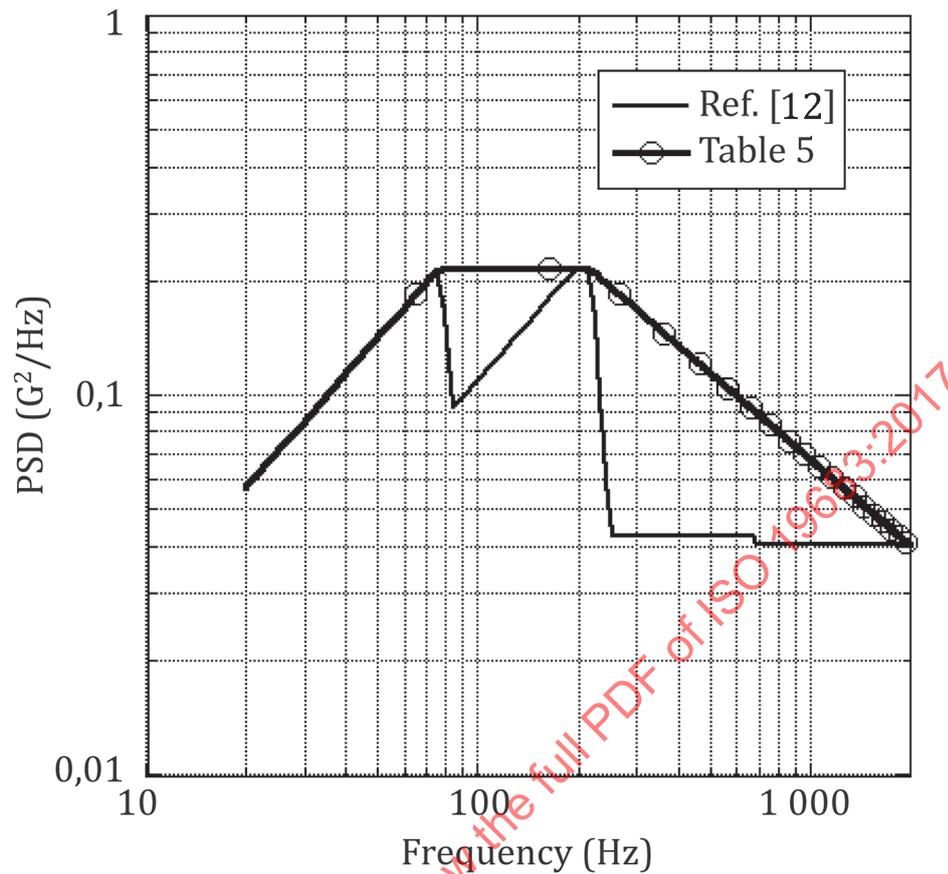
Figure B.1 — Unit random vibration QT PSD levels proposed in Reference [12] and adapted in Table 5

The sinusoidal amplitude was derived by multiplying a sinusoidal vibration AT level defined by a typical launcher, 2,0 $G_{0\text{-p}}$, by a factor of 4,2. The test data of various satellites was analysed in Reference [12].

The shock response spectrum of Table 5 was derived using Formula (B.1)[18]:

$$\text{SRS}(D_2) = \text{SRS}(D_1)\exp\left\{-8 \times 10^{-4} f_n^{\left(2,4 f_n^{-0,105}\right)}\left(D_2 - D_1\right)\right\} \tag{B.1}$$

where

f_n is the frequency;

D_1 is the distance from the pyrotechnic source to the reference, in m;

D_2 is the distance from the pyrotechnic source to the new location, in m;

$\text{SRS}(D_1)$ is the shock response spectra for the responses at the reference;

$\text{SRS}(D_2)$ is the shock response spectra for the responses at the new location.

Formula B.1 is an empirical equation found in Reference [18].The SRS given by a typical launcher for fairing separation (shown as Rocket A in Figure B.2) was used for $\text{SRS}(D_1)$. $\text{SRS}(D_2)$ was calculated assuming that the distance between D_2 and D_1 is 50 cm. The result is shown as "Attenuated" in Figure B.2. For simplicity, the high frequency part was made flat and proposed as UNIT QT SRS, along with the lower tolerance limit of −6 dB.
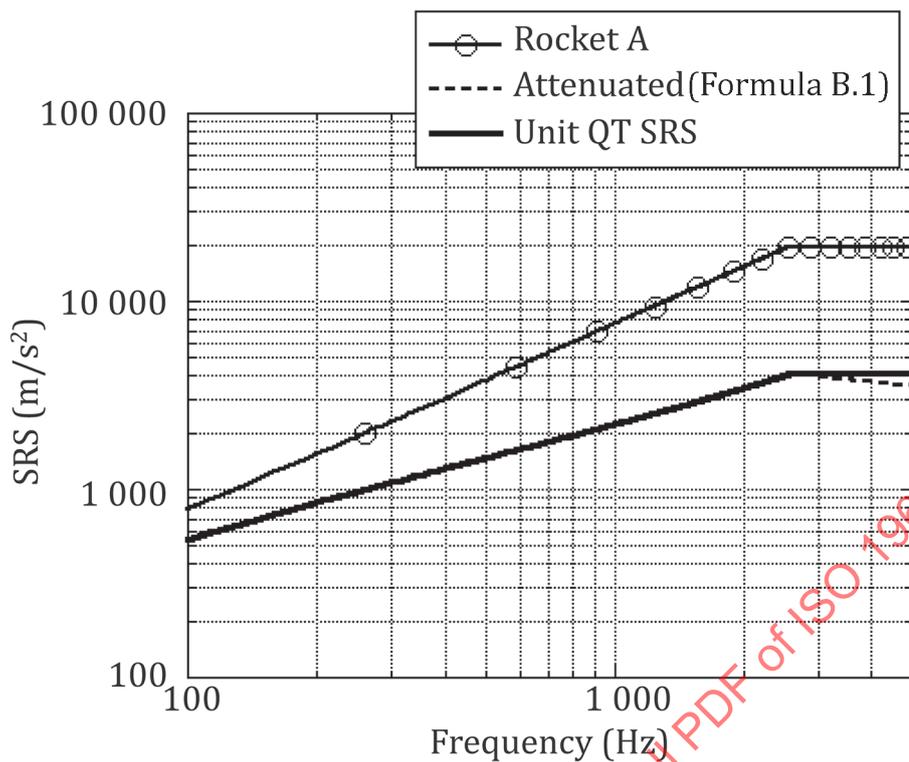
Figure B.2 — Derivation of unit shock QT SRS

Annex C
(informative)

Design, verification and testing philosophy for small spacecrafts

C.1 COTS

COTS, in this document, refers to either parts, units, components and technologies that are readily available and are not manufactured, inspected or tested in accordance with military or space standards.

The unit QT in 5.2 and 7.2 is carried out while the unit manufacturers develop their product. The unit manufacturer does not develop a custom-made product. The satellite developers select the units from inventory that best fit their needs. Even if two COTS units have the same product number, there is no guarantee that the parts inside belong to the same production lot or have the same part number, if the production times are different. Spacecraft developers should be aware that the COTS unit they buy may not be identical to the one tested by the manufacturer up to the exact detail.

The unit procurement method for small spacecrafts is different from that of traditional satellites. Small spacecraft developers may purchase the COTS-based units from the open market, even without any prior knowledge of the manufacturer. Therefore, the test documentation in 5.6.2 and 5.6.3 shall provide the information necessary to provide the minimum level assurance that the COTS-based unit has durability against the space environment. The COTS unit manufacturers should be aware that the test documentation is a crucial source of information for their customers to consider purchase.

C.2 Design

To achieve low cost and fast delivery, small spacecrafts use COTS units and COTS technologies extensively. Procurement of the units from the market is often chosen over in-house development. Interfaces and the overall design should be simplified as much as possible, considering the fact that human resources of the development team are often limited. Unproven technologies are sometimes permitted to play key roles in the satellite bus and mission payloads to incorporate state-of-the-art COTS technologies in electronics, sensors, MEMs and other fields if the stakeholders agree on taking the risks and a certain level of testing is performed on the ground. Due to resource limitations, redundancy is limited. Innovative ways of thinking not bound by traditional satellite design methodology should be encouraged.

C.3 Verification

Verification is often very time-consuming and can become the bottleneck in a satellite lifecycle. Even in the case of small spacecraft development, a verification plan should be made during an early phase of the lifecycle. Because the system is simpler than the traditional satellite, the documentation associated with verification should be more minimal. Due to the small size and simplicity, the error associated with analysis of small spacecraft may be kept small compared to that of traditional satellites. Therefore, analysis ("use of analytical data or simulations under defined conditions to show theoretical compliance"[25]), instead of testing, can be used extensively in various phases of verification if it saves time or monetary cost.

Several statistics show that small spacecrafts suffer higher rates of infant mortality in orbit compared to traditional large/medium satellites[9]. Defects in design or manufacturing not detected during the pre-launch verification process manifest as premature death. Therefore, thorough verification will help prevent infant mortality by detecting latent defects. Once satellites pass the infant mortality stage, long-term survival in orbit is significantly more likely. In fact, there are many small spacecrafts that have operated or are operating for five years or longer even though they were made of COTS parts.

Small spacecrafts can be utilized as a constellation. Since large numbers of identical satellites can be produced, verification should be planned in the context of a satellite program, instead of individual satellites. Extensive testing to verify the design, including orbital demonstration by launching pathfinder satellites, may be worthwhile[11]. Once the design is verified, verification of the flight hardware can be streamlined with minimum effort. At the same time, however, care should be taken when using the method of qualification by similarity[10]. Quality management over parts, materials and manufacturing processes of the small spacecraft system and units is not as rigorous as traditional satellites. As procurement spans over time, COTS parts purchased in different time periods may be different. Therefore, there is no guarantee that the satellites and units actually built are identical even if built using the same design. This caution applies not only to a constellation program but also to any small spacecraft program that tries to inherit its design from previous satellites.

Whether a particular method, such as testing, analysis, demonstration, inspection and similarity, is chosen as a verification method or not should be assessed based on discussion as quantitative as possible. Analysis often helps the assessment process. The assessment reason should be transparent to the customer and written clearly in the verification document. In this document, the rationale behind the given numeric criteria will be given in the annex.

Verification is different from validation (see ISO 10795 for definition of each term). Testing may be used for validation to confirm that the requirements for a specific intended use or application have been fulfilled. The test requirements and test methods described in this document are all those of testing for verification purposes.

C.4 Testing

Testing is one verification method out of many methods, such as analysis, demonstration, inspection, or review of design (and verification by similarity). Testing can be categorized into environment test, functional test and measurement test. Once testing is adopted as a verification method, there is often little difference in the method itself depending on the satellite size. However, as the expected reliability of small spacecraft is different from that of traditional satellites, the test level, duration and precision may not be the same as those applied during the testing of traditional satellites.

The test level, duration and precision are different for each satellite depending on the environments each satellite encounters and the mission objectives. Appropriate margin should be added to the predicted environmental conditions when the test level and duration are derived. The margin level added affects the cost, delivery time, and reliability of the satellite. Therefore, it is not necessary to seek the same margin level as traditional satellites.

C.5 Small spacecraft program/design characteristics and verification strategy

Tables C.1 and C.2 summarize the characteristics inherently associated with the satellite program/design and the corresponding verification strategy when low cost and fast delivery are the primary drivers. In the tables, comparisons are made against program/design characteristics of traditional satellites. Due to the uniqueness of small spacecrafts, the verification methods may be different from those of traditional satellites. In some cases, analysis may be chosen instead of testing.

Table C.1 — Typical small spacecraft program characteristics and verification strategy

Program factor	Traditional large/medium satellite	Small spacecraft		Verification criteria
	Characteristics	Characteristics	Verification strategy	
Satellite mass/volume	— Mass is limited only by capability of available launchers. — Size is limited by payload fairing.	— Often dictated by maximum size and mass of piggyback launch specification. — For example, for H2A, 50cm × 50cm × 50cm, 50 kg — Approximately 400 kg/m³ to 1 500 kg/m³		
Orbit	— Any orbit	— Mostly earth orbit (depends on communication capability) — Depends on main payloads or transfer orbit of main payloads in case of launch as an auxiliary payload.		
Program cost	— No limit, often exceeding USD 100 million	— Typically, less than USD 3,5 million for a single satellite (satellite cost at USD 2 million, launch cost at USD 1 million, operations cost at USD 0,5 million).	— Spend less than USD 0,5 million at maximum in tests, including personnel cost	
Delivery time	— Typically 2 years or longer	— Less than 2 years	— Spend less than 1 year (sometimes 6 months) in design verification — Minimize the transportation by continuous test at a single location.	
System renewal time (time-to-launch of next-generation satellites)	— Long — Sometimes, takes nearly 5 years to 15 years	— Short — Small step-by-step design change	— For next-generation satellites, do QT only for new units and system AT only for the other units.	
Consequence of failure	— Failure of a single satellite jeopardizes the overall satellite program.	— The satellite program may be saved by building and launching a replenishing satellite easily.	— Verify that sufficient amount of telemetry data is available even if a satellite fails to reflect the lessons to the subsequent satellites.	

Table C.1 *(continued)*

Program factor	Traditional large/medium satellite		Small spacecraft		
	Characteristics		**Characteristics**	**Verification strategy**	**Verification criteria**
Developer size	— Often exceeding 100 people		— About 20 people, sometimes less than 10 people — One person works on multiple issues and subsystems. — Simplified satellite system is adopted to allow management by small number of people.	— Minimize the scattering of human resources by continuous tests at a single location. — Test at the upper level (sub-system rather than unit, system rather than subsystem)	
Developer infrastructure	— Often equipped with basic test facilities, especially at units or subsystem level. — System test can be done at a different location.		— Limited infrastructure — Need to do a test at a different location.	— Outsourcing the tests — Continuous tests at a single location — Thorough functional test before transporting the test article to the outside.	— The system should function well in the integrated state.

Table C.1 *(continued)*

Program factor	Traditional large/medium satellite — Characteristics	Small spacecraft — Characteristics	Small spacecraft — Verification strategy	Small spacecraft — Verification criteria
Developer infrastructure			— The test specification and procedure should be prescriptive enough — for better communication between the testing people and the developers, and — for traceability of test so that the test can be done in the same way by different people.	
Procurement of units	— Procure the made-to-order product from reliable subcontractors.	— Procure the ready-made product from the market and limit the in-house development as much as possible. The unit data sheet shall contain enough test results for use in space.	— Confirm that the product has tolerance to the minimum level of thermal cycle, vacuum and vibration before procurement of the flight product.	— The product should withstand the minimum level of thermal cycle, vacuum and vibration.
Manufacturing	— Rigorous manufacturing processes and material control specific for space products.	— Use of non-space manufacturing processes and materials	— Functional tests in high temperature, low temperature and vacuum — Thermal cycle endurance — Vibration — Shock	— No workmanship defect should be found. — Performance meets specification requirements under hot, cold, vacuum conditions. — Endure vibration, shock and thermal cycle.
Outsourcing	— Outsource to subcontractors to reduce the cost — Specific subcontractors experienced with space products — Rigorous control over subcontractors' work	— Outsource the manufacturing of units to non-space manufacturer to cover the shortage of human resource. — Simplified interface — Outsource to non-space companies	— Use companies with a certain level of quality control — Workmanship/material check using burn-in and wear-in test — Interface test — Clear documentation for ordering specification	— No workmanship/material defect should be found. — The outsourced units work in conjunction with other units.

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Table C.1 *(continued)*

Program factor	Traditional large/medium satellite — Characteristics	Small spacecraft — Characteristics	Small spacecraft — Verification strategy	Small spacecraft — Verification criteria
Testing	— Test in-house for unit/subsystem level — Well-established test procedure and documents based on international/domestic/in-house standard	— Test at different places — Satellite size is too small — Test procedure not well established — If possible, share the test among multiple satellites or units to reduce the cost.	— Use a test facility appropriate for satellite size. — Use personnel experienced in testing.	
Safety and hazard control regarding launch	— Strict — Many items related to safety and hazard control	— The safety and hazard control in launch phase is the same as traditional satellite. — Crew safety should be maintained when the satellite goes through habitable module.	— Safety verification through testing can be expensive. Verify using design inspection or simple analysis.	
Safety and hazard control regarding satellite handling	— Strict — Large internal energy including thruster propellant, battery and pyrotechnic devices	— For launch, safety requirement is the same as large/medium satellites. — Small internal energy — Avoid designs that raise safety concerns (e.g. hydrazine) so that the verification of safety hazard control can be skipped. — Designs to avoid safety hazards should be flight-proven	— Safety verification through testing can be expensive. Verify using design inspection or simple analysis. — Hazard analysis — Inspection and analysis only — Demonstration/testing costs time and money	— Satisfy the safety requirement by the launcher. — Satisfy the internal safety standard.
Time from delivery to launch	— Launcher waits for the arrival of the satellite — Battery charging at the launch site	— Could be long (as long as two months) — No charging of battery may be allowed at launch site — Preserve battery charging state — Selection of low self-discharge battery	— Characterization of battery self-discharge rate — Battery charging test — Deployment test	— Battery should have low self-discharge rate. — Demonstrate that satellite can recover from zero battery state. — Demonstrate that deployment is possible at low battery voltage.

Table C.1 *(continued)*

Program factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Launch method (shared launch)	— Hot launch — Dedicated launcher	— Cold launch — Multiple inhibits against accidental power-on and/or deployment — Outgas limits	— Check of drawing for clearance if a large fairing rocket is used — Fit-check for clearance if a small fairing rocket is used — Demonstration of inhibits using vibration and shock tests — Cold/hot start of after-separation sequence — Outgas test	— Meet requirements specified in launcher/satellite ICD.
Launch method (dedicated launch by a small rocket)	— Hot or cold launch	— Hot launch	— Fit-check for clearance	— Meet requirements specified in launcher/satellite ICD.
Operation continuity	— Operated above the ground station — The use of data relay satellites or satellite-to-satellite communication is possible.	— Operated only above the ground station. — No use of data relay satellites or satellite-to-satellite communication — Autonomous functions in satellites	— Mission test	— The satellite can work autonomously for a specified duration without ground control.
Operation style	— Many people, high-cost control — Operation and control by specialists — Large and high-cost ground station	— Less people, low-cost control — Autonomous operation — Operation and control by users — Small and low-cost ground station — Operation through network from remote offices if it is legal.	— End-to-end mission simulation test	— The satellite can be controlled and operated by the specified ground station network by actual personnel who are responsible for control and operation.

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Table C.2 — Typical small spacecraft design characteristics and verification strategy

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Interface	— Complicated	— Simple interface — Every unit connected by the same type of interface — Common mechanical interfaces (bolt type, bolt separation, etc.) to reduce the number of test attachments.	— Interface test — Mission test — End-to-end mission simulation test — Test at least once with a full configuration	— Signal and power transmitted through the interface as specification. — Meet the specification requirements of impedance matching and insulation. — Mechanically attached with strength as specification. — Performance in the full configuration meets specification requirements. — Satellite communication is compatible with the specified ground network.
Design life (mission life)	— As long as 15 years	— Short (less than 3 years) compared to traditional satellites — Depends on orbit	— For missions over 3 years, evaluation at EOL regarding degradation of surface material, solar cell, electronics parts, wear of mechanical parts.	— Performance at EOL meets specification requirements.
Reliability calculation	— Based on random failure rate	— Emphasis on infant mortality prevention — Do not consider random failure rate of each part.	— No need of test to characterize the random failure rate of each part, e.g. single-event probability. — Emphasis on detecting design/workmanship defects — Thorough FMEA/FMECA	

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Satellite availability (continuity)	— Shall be continuous — Use of redundancy, on-orbit spare, channels of other satellite	— Allow temporal outage due to anomaly. — Recover using commands from the ground — Use on-orbit spare satellites if available	— Demonstration of satellite recovery after temporal shortage using functional tests. — Thorough FMEA/FMECA — End-to-end test	— Satellite can recover using command from the ground. — Outage duration is within acceptable limits.
Mission assurance	— Targeted very high	— Allow a certain rate of failure — Prompt launch of substitutes — Sufficient telemetry to identify the failure causes through minimum level of FTA	— Thorough FMEA/FMECA to identify the mission failure scenario. — Test of mission critical operation in flight representative situation — Deployment test — end-to-end mission simulation test — Mission test	— Sufficient amount of telemetry data is available. — Mission critical operation should be demonstrated.
Single point of failure	— Basically not acceptable	— Not acceptable in principle, but sometimes unavoidable.	— Thorough FMEA/FMECA to identify the single points of failure. — Thorough tests of parts/units related to the single points of failure.	— Parts/units related to single points of failure should have sufficient reliability.

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Table C.2 (continued)

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Failure mitigation	— Redundancy — Safe mode	— Switching to safe mode — Soft reset — Power reset — Hardware rebooting — Use a legacy unit as redundancy	— Identify failure modes — Thorough FMEA/FMECA — Mission test — Radiation test — Single Event — TID if the expected total dose is higher than 100 Gy — Demonstration of anomaly/failure detection, safe mode transition, reset/rebooting mechanisms and switching to the redundant unit. — Anomaly/failure simulation should be as flight representative as possible.	— Detection of anomaly/failure meets the specification requirement. — Demonstrate recovery from radiation induced anomaly. — Performances in contingency operation mode meet the specified requirement.
Time from launch to operation	— Hot or cold launch	— Power-on soon after separation — Communication has to wait as long as 30 min to avoid interference with launcher.	— Corona test inside power control unit — Multipactoring test of RF transmitter and cables	— Discharges do not occur under the worst case.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite — Characteristics	Small spacecraft — Characteristics	Small spacecraft — Verification strategy	Small spacecraft — Verification criteria
Parts selection	— Space-qualified — Radiation hard electronics parts	— Low-cost and fast delivery — Easy availability of parts from the market — Substitutes with similar functionalities available — Easy to handle — Desired material properties	— Function in high and cold temperature limits — Function in vacuum, especially battery and units with lubricant such as reaction wheel — Vibration test — Shock test	— Performance meets specification requirements under hot, cold, vacuum conditions. — Endure vibration, shock and radiation.
		— Mechanical strength — Thermal property — Temperature limits — Radiation tolerance	— Surface property measurement, such as thermo-optical property — Radiation test (TID, single event) if possible	
Radiation tolerance	— High — Radiation hard parts selection — Single event cross-section known or measured — Strict control of the manufacturing lot	— Not strong — Selects parts with little knowledge of radiation tolerance property — Does not lose functionalities even at single event (recovery, switching to the redundancy) — Difficult to control the manufacturing lot of COTS parts	— Thorough FMEA/FMECA — Identify the possible results of radiation induced anomalies — Single event test — Table-sat (flat-sat) level to identify the behaviour at single event — TID test if the expected dose is higher than 100 Gy	— The satellite can fulfil the mission even under the radiation environment.
Mounting of electronics parts	— Space-qualified soldering	— The same mounting process as consumer electronics — Narrow temperature range inside the satellite body — Vibration/shock/thermal cycle mitigation	— Thermal cycle — Vibration — Shock	— Performance meets specification requirements under hot, cold, vacuum conditions. — Endure vibration, shock and thermal cycle.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Number of parts	— Very large	— Minimize the number of parts to lower the assembly/integration cost, management cost, and procurement time — Use one-chip IC — Increased degree of integration per parts — More vulnerable to failure/anomaly of single parts	— Radiation test on parts susceptible to single event e.g. CMOS IC, digital IC, analogue IC with latch circuit	— Identify the behaviour when a single event occurs.
Outgas limitation	— Adopts outgas limitations, such as TML, CVCM — Very strict outgas control depending on payload	— As a piggy-back satellite, need to comply with the demand of main satellite — Accept contamination to itself — May need to start optical observation soon after launch	— Baking before delivery — Preparation of material list — Coating of small parts	— The material outgas properties should comply with the launcher requirement. — Contamination should not affect the mission purpose.
Contamination control	— Work in a clean room	— Work in a clean room — Impossible to be free from foreign particle intrusion — Coating of print-circuit-board pattern or metal terminals	— Clean room inspection — Inspection of design and assembly procedure	— The effect of foreign particles intrusion should be minimized.
Material selection	— Use material with known outgas property or measure if unknown	— Use material with known outgas property	— Obtain CVCM and TML from material table — Avoid material property measurement as much as possible — Vacuum baking	— Outgas property is within acceptable limits.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite Characteristics	Small spacecraft Characteristics	Verification strategy	Verification criteria
Debris mitigation	— Use of on-board propulsion for de-orbiting — Passivation operation	— Same requirement as traditional satellites — Larger fraction of resources is occupied by the deorbit mechanism compared to a large satellite — Passivation function	— Compliance with deorbiting and passivation regulation should be demonstrated by analysis — The malfunction (accidental switch-on) of the passivation should be thoroughly verified through tests	— The satellite can deorbit within 25 years. — At EOL, the satellite can be passivated by the command from the ground. — The passivation mechanism should not activate accidentally.
Wire harness	— Space-qualified wire harness	— Use of non-space grade wire harness	— Vacuum baking	— Outgas is within the specified acceptable limits.
Power control unit	— High bus voltage (as high as 100 V) — Multiple power routing — Solar paddle rotation — Solar array power output is relatively constant	— Low bus voltage (typically 28 V or less) — Single power routing — No rotation of solar paddle — Fluctuating solar array power	— Charging/discharging test — Over-charging/over-discharge protection — Functional test at low and cold temperature limits — Arc checks during switch on/off — Single event test of PCU circuit board to identify the anomaly mode	— Charging/discharge battery of a specified cycle. — Performance meets specification at hot and cold temperature. — No arc discharge. — Can recover from single event.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite		Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Solar array	— High power, sometimes more than 10 kW — High performance solar cells — Large number of strings, sometimes more than 100 string circuits — Concern over solar cell degradation — Large area	— Low power (approximately 10 W to 100 W) — High performance solar cell. — One of the cost-drivers — Connecting and laying down solar cells are more costly than the cost of cell itself (lay-down is done by developers or non-space manufacturers). — Small number of strings. — Loss of one string has a large effect — Less concern over solar cell degradation (depends on orbit) — Small area (less thermal expansion/shrinkage in thermal cycle)	— Vacuum baking — Thermal cycle endurance test — VI measurement of solar cell — Vibration — Shock	— Endure thermal cycle. — Endure shock and vibration. — Solar array output meets specification.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Battery	— Space-qualified battery — Safety regulation — Charging allowed at launch site — Cell-balancing circuit	— COTS battery available at a general appliance store — Safety regulation with Li-ion battery — No charging allowed at launch site — Screening of batteries to select ones with characteristics similar to each other when no cell-balancing is adapted.	— Function test in vacuum — Charging and discharging — Overcharging/over-discharge protection — Function test at low and cold temperature limits — Leak test — Characterization of self-discharge rate — Demonstration of battery charging from the empty condition — Thermal analysis of battery	— Charging/discharge battery of a specified cycle. — Charging/discharge is within the specified acceptable depth-of-discharge. — Performance meets specification at hot and cold temperature. — No leak in vacuum. — Self-discharge during the launch standby is within the specified acceptable limits. — Battery can be charged from the empty state. — The battery temperature range is within the specified acceptable limits.
Command and data handling	— Radiation hard units — High computing capability — Reprogrammable in orbit	— Use COTS processor — Not necessarily state-of-art — Data compression — On-board processing — Reprogrammable in orbit	— End-to-end mission simulation test — Mission test	— Communication with the specified ground station network is possible. — Can function long-time without bug. — Performances in contingency operation mode meet the specified requirement.

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Table C.2 *(continued)*

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Telemetry data rate (beacon and house-keeping)	— High-speed, large amount of data available — Many monitoring points	— Low-speed, limited amount of data, wide beam angle — Fewer monitoring points compared to large satellites — Sufficient telemetry for anomaly identification desired	— FMEA/FMECA — Functional test in hot, cold and vacuum environment — Antenna pattern — End-to-end mission simulation test	— Telemetry is sufficient to identify possible anomalies. — The correct HK data should be collected and transmitted within high and cold temperature limits. — The link budget should guarantee continuous data recovery at the ground with enough margin. — Communication with the specified ground station network should be demonstrated.
Mission data rate	— High speed, directed transmission	— High speed, directed transmission	— Antenna pattern — Link budget derivation — End-to-end mission simulation test	— The link budget should guarantee the mission data downlink with sufficient margin. — Mission data can be down-linked and recovered at the specified ground station network.
Radio-transmitter	— Use of TWTA (travelling wave tube amplifier) for high power amplifier	— Use of SSPA (solid state power amplifier) — Power concentration in small volume	— Functional test in hot, cold and vacuum — Characterize the trend of power and frequency over temperature — Multipactoring discharge — Thermal analysis	— Power and frequency variation is within the specified acceptable range. — No multipactoring discharge occurs. — The temperature is within the specified acceptable range.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Antenna	— Deployable antenna	— Body mount antenna — Deployable antenna for high speed mission data transmission — Failure of deployment may lead to the loss of mission	— Antenna pattern — End to-end mission simulation test — Deployment test — Attitude control hardware-in-the loop test	— The link budget should guarantee communication between the satellite and the ground station within the specified margin. — Antenna can be deployed after separation under the worst case condition.

Deployment	— Deployment of solar paddle and antenna is common
	— Deployment of boom structure for mission payloads
	— Deployment using pyrotechnic devices
		— Multiple deployments to compensate for the small area
		— Rare usage of pyrotechnic devices
		— Little need of deployment shock test
			— Deployment test in vacuum and at low temperature
			— Demonstration of sequence from separation to deployment
			  — Cold welding
			  — Vibration and shock test
			  — Coating endurance (wear, peel-off, etc.), bearing endurance
				— Deployment should be possible under the worst case condition.
				— Deployment mechanism endures the launch environment.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite Characteristics	Small spacecraft Characteristics	Small spacecraft Verification strategy	Small spacecraft Verification criteria
Structure	— Launch load is sustained by central core — Acoustic vibration	— Launch load is sustained by frames and panels — Resonance frequency higher than that of a large satellite	— Quasi-static load either by analysis or test — Modal survey + analysis for resonant frequencies — Modal survey + random vibration for structural integrity — Allow notching — Sinusoidal vibration — Allow notching — Use of STM to verify analysis models and to help shift the risk of identification of critical failures to an earlier project stage. Selected tests can be done only on the STM and burden on PFM can be reduced.	— The structure can withstand the static load within a margin. — The resonant frequencies are different from those of the launcher. — No deformation due to vibration.
Separation mechanism (Marman clamp)	— Use of pyrotechnic devices — Shock on separation	— Strong safety demands from the launcher compared to POD style — Shock at satellite separation and fairing separation	— Shock test	— Satellite can endure the shock environment. — No accidental turn-on of the satellite for a cold launch case.
Separation mechanism (POD style)	— Not applicable	— Resonant frequencies of the launcher, POD and satellite should be separated — Shock can be attenuated	— Satellite shock test only for fairing separation (Can be skipped if the shock acceleration is attenuated.) — No satellite separation shock test	— Satellite can endure the shock environment. — No accidental turn-on of the satellite for a cold launch case.

Table C.2 *(continued)*

| Design factor | Traditional large/medium satellite | Small spacecraft | | |
	Characteristics	Characteristics	Verification strategy	Verification criteria
Separation switch	— Hot launch	— Inserted in power lines for cold launch — Need very high reliability — May be exposed to outer space	— Vibration test — Shock test — Inspection of harness routing that may be exposed to space	— No chattering (contact bouncing) should occur during the launch environments. — Switch should work as specified after separation. — Harness is protected from space environment exposure.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite	Small spacecraft		
	Characteristics	Characteristics	Verification strategy	Verification criteria
Thermal	— Large heat capacitance — Complex area — Use of thermal radiator, active thermal control — Large temperature difference within a satellite — Large internal heat generation — Rigorous temperature control (depending on mission)	— Small heat capacitance. Wider range of high and low temperature — Limited thermal radiation area or thermal radiation from whole surface area — Less temperature difference within satellite compared to a large satellite — Less internal heat generation — Little need for active thermal control such as heat pipe — Not enough power resource to do active thermal control for all units — Passive thermal control (radiation, conduction) is mostly used — Easier thermal conduction inside a satellite — No rigorous temperature control (depending on mission)	— Derivation of hot and cold temperature limits by thermal analysis. — Material data for heat conduction, IR radiation and solar absorptance. — Use of STM to verify analysis models and to help shift the risk of identification of critical failures to an earlier project stage. Selected tests can be done only on the STM and burden on PFM can be reduced.	— The discrepancy between thermal balance test and thermal analysis should be within the specified acceptable limits.
Attitude control (Passive)	— Gravity gradient	— Passive attitude control using gravity gradient or permanent magnet	— Deployment test (gravity gradient)	— The boom can be deployed under the worst case condition.

Table C.2 *(continued)*

Design factor	Traditional large/medium satellite — Characteristics	Small spacecraft — Characteristics	Small spacecraft — Verification strategy	Small spacecraft — Verification criteria
Attitude control (Active)	— Reaction wheel — Propulsion	— Magneto-torquer + magneto sensor — Reaction wheel	— Magnetic field measurement if magneto sensors are used — Verification of attitude control algorithm by hardware-in-the-loop test	— The remaining magnetic field is within the specified acceptable limits. — Control algorithm performs as specified.
MMA	— Microvibration may be an issue for payloads	— Microvibration may be an issue for high-precision pointing	— Analysis of microvibration generation — Characterization of microvibration source — Measurement of microvibration transmittance — Microvibration susceptibility test	— The effect of microvibration is within specified acceptable limits.
Propulsion	— Large thrust — Multiple thruster heads — Chemical or electric propulsion	— Small thrust — Need to miniaturize the volume of propulsion system — Integrated propellant feed — MEMS thruster	— Pressure test — Thruster firing test — Alignment test	— Structural failure or excessive deformation of the pressure vessel should not occur. — Thruster performance meets the specified requirements. — The thrust vector is aligned to the specified direction within the specified acceptable limits.
EMC	— Large current in circuit boards and cable harness — Long cable harness — Strict control over EMC	— Distance between circuit boards is small — Less freedom in harness routing — Use of COTS DC/DC converter	— EMC test — Functional test after integrating all the subsystems with no external connection — Analysis for compliance with launcher requirement	— The satellite functions as specified under the self-induced electromagnetic environment. — The satellite should not malfunction during the launch phase due to external electromagnetic noise.

The reliability of a satellite extensively using COTS is not guaranteed at the parts level, which is different from a traditional satellite. Therefore, contingency operation where the satellite system detects anomaly and recovers either automatically or by an uplink from the ground is important for mission success. Rather than performing detailed tests on units or in subsystem levels, more emphasis is put on tests under the fully integrated state, such as the mission test and the End-to-end mission simulation test where the contingency operation should be fully verified. To design the contingency operational modes, thorough FMEA/FMECA should be carried out. Mission critical operation, such as deployment, should also be fully tested when the entire operational sequence is tested with all the hardware and software involved, possibly in the fully integrated state. The small satellite size makes such tests much easier compared to traditional large/medium satellites.

When the satellite is integrated for the first time a thorough functional test and a thorough EMC test to detect defects is important, in order to distinguish between anomalies found in later environmental tests and anomalies truly caused by the environment or the ones caused by incompatibility among the units. Small Spacecraft developers often use test facilities outside their own institution. Troubleshooting at the test site is very time consuming.

C.6   Testing strategy

The most appropriate test strategy for a given small spacecraft depends on the nature of the satellite program, such as single satellite, a series, or a constellation. A single satellite based on a new design is most likely to go through QT+AT or PFT. For a series of satellites with design modifications, some QTs, especially at unit levels, may be skipped. For a constellation of identical satellites, after QT of the first satellite, the remaining satellites may be launched after passing AT only.

For a single satellite program, even if QT+AT strategy is chosen, the AT model may not be exactly the same as the QT model due to uncertainty of quality management. Even if the flight model passes AT, it may have failed if a margin equivalent to QT had been added to the test level. PFT has the advantage of testing the flight model within a certain margin. It has the additional advantages of reducing the procurement cost as there is no need to buy hardware for a QT model, shortening the test schedules, and lowering the test cost. It, however, carries the risk of lending latent fatigue to the flight model. To avoid this risk, there is the option of carrying out only AT, even for a single satellite program. This option, however, incurs another risk of overlooking some design defects that would have manifested at higher test levels. Overall, for a single satellite program, there are advantages and disadvantages (risk) for the three strategies listed in Table C.3. The most appropriate test strategy selected depends on how much risk the program accepts.

Table C.3 — Test strategy advantages and disadvantages for a single satellite program

	QT+AT	PFT	AT only
Design verification with margin	A[a]	A	D
Testing flight model with margin of less strict quality management	D	A	D
Procurement cost	D	A	A
Test schedule	D	A	A[b]
Test cost	D	A	A[b]
Fatigue to flight model	A	D	A
A Advantage, D Disadvantage			
a More advantage due to larger margin than that of PFT.			
b More advantage due to shorter test duration than that of PFT.			

For a satellite series or constellation, flight data gained by the early generation of satellites can be used to clarify the uncertainty of environmental conditions, such as temperature range. For satellites of later generations, more precise environmental predictions can be used to define the test level. Then, PFT may be a good choice by adding margin to the environmental precision. The margin in this case can be smaller than that used for PFT in the single satellite program because the margin associated

with uncertainty of the flight prediction, which is 11 °C for a thermal test of US military programs (see Reference [15] for an example), is no longer needed. Therefore, the risk of giving latent fatigue to the flight model due to PFT can be mitigated to a certain degree.

Even if PFT or AT only strategy is chosen, STM testing is very useful. By doing preliminary tests using STM, the risk of damaging the flight model can be mitigated. STM can serve many other purposes, such as wire harness routing, fit-checks of jigs, etc.

Skipping some unit tests leads to cost reduction of testing and shortening of schedule. Of course, this strategy increases the risk of defects at the system level, which are more costly than when the defects are at a lower level. The cost and schedule required to integrate the system, disintegrate it to remove the failed unit and reintegrate the repaired or new unit to the system may be significant. Proper assessment based on the maturity of knowledge about each unit should be made as to whether or not the unit test is skipped.

Small spacecrafts are often tested at locations outside the unit manufacturer or the system integrator. The testing cost can be divided into the facility fee, the personnel cost and expendables such as liquid nitrogen and jigs. The main cost driver is the personnel cost. Therefore, reducing the test schedule also leads to cost savings. Communication with the test facility personnel using well-written test plans, specifications and procedure documents is crucial to efficient testing. Anomalies are often found during the checking phase of field test preparation. It is necessary to clarify whether the anomaly is due to the difference of the test environment, i.e. the test centre or the developer's laboratory, or not. A thorough functional test prior to test centre arrival is also crucial as trial and error in the field is very time consuming.

Testing intensively in one location leads to savings in cost and schedule associated with logistics. Reliability is increased by guaranteeing test result traceability and removing ambiguity in the test results caused by the use of different facilities. Proper usage of the testing facility leads to reduction of testing cost and schedule. The operational cost of a vacuum chamber increases more than linearly as its size becomes larger. One way to reduce testing cost is to share chamber space. Sharing is especially appropriate if the satellite program is a constellation.

A series of testing can be regarded as reliability growth through testing repetition, defect detection, correction and testing. Generally speaking, defects are found at an early phase of testing, such as the first several cycles of thermal vacuum. For example, Reference [16] shows that half of the units that suffered failure during an 8-cycle thermal test at ΔT = 85 °C suffered the failure in the first cycle. Repeating short duration tests rather than one long duration test may lead to higher reliability per time spent in testing (see Reference [13]). Carrying out tests that can easily reveal defects at an early stage of the testing series also helps to save the overall testing time.