İSTANBUL TECHNICAL UNIVERSITY ★ INSTITUTE OF SCIENCE AND TECHNOLOGY

INTEGRATION AND TESTING
OF
ITUpSAT1

M.Sc. Thesis by
Özgün SARI

Department : **Aeronautical and Astronautical Engineering**

Programme : **Interdisciplinary Program**

JANUARY, 2010

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<u>İSTANBUL TEKNİK ÜNİVERSİTESİ ★ FEN BİLİMLERİ ENSTİTÜSÜ</u>

İTÜpSAT1 UYDUSUNUN ENTEGRASYONU VE TESTLERİ

YÜKSEK LİSANS TEZİ

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FOREWORD

I would like to express my appreciation for my advisor Assist. Prof. Dr. Gökhan İnalhan and Astronautical Engineering Department Chair Prof. Dr. A. Rüstem Aslan for their support throughout the ITUpSAT1 project. I would also like to thank to my colleague Barış Toktamış and ITUpSAT1 team members.

January 2010 Özgün Sarı

Aerospace Engineer

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ABBREVIATIONS

ΔT_A	: Acceptance Temperature Range
ΔT_Q	: Qualification Temperature Range
AO	: Atomic Oxygen
CCD	: Charge Coupled Device
CME	: Coronal Mass Ejection
DM	: Development Model
ECSS	: European Cooperation on Space Standardization
EMC	: Electromagnetic Compatibility
ESD	: Electrostatic Discharge
EQM	: Engineering Qualification Model
FM	: Flight Model
GCR	: Galactic Cosmic Ray
GEO	: Geostationary Orbit
ITUpSAT1	: İstanbul Technical University Picosatellite 1
LEO	: Low Earth Orbit
MEO	: Medium Earth Orbit
N_A	: Required Number of Acceptance Cycles
ΔT_{AMAX}	: Maximum Allowable Number of Acceptance Cycles
N_Q	: Required Number of Qualification Cycles
PSLV	: Polar Satellite Launch Vehicle
RH	: Relative Humidity
S/C	: Spacecraft
SEU	: Single Event Upset
SPE	: Solar Particle Event
SPL	: Single Picosatellite Launcher
TC	: Thermal Cycle
TV	: Thermal Vacuum
TVC	: Thermal Vacuum Chamber

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INTEGRATION AND TESTING OF ITUpSAT1

SUMMARY

Artificial satellites orbiting around the Earth are essential since the start of the space age, which begun with Sputnik. Today, they perform critical missions and serve humanity to make life easier. Communication, remote sensing, data collection, navigation, exploration, scientific experiments are the most important fields where they utilized. Investments in space technology result in advancements in engineering and science, and these advancements also lead to developments in the space technology. If one compares the first spacecrafts and today's, a huge diffence can be observed clearly. As in the other technologies, today's satellites are smaller, lighter and more compact than their first ancestors. This is a very important factor when considering risk and cost requirements of a space project. The satellites that weight hundred of kilograms or several tons is also a heavy weight for the mission budgets. Therefore, small satellites attract attention, since they are easier to handle, to test, to verify and to launch to space. The space technology of the future will be driven by this point of view.

In 2001, a small satellite (picosatellite) project, called CubeSat, has been started by the association of California Polytechnic State University and Stanford University. The main objective was the education of college and university students, however the concept spread out to other areas. Today, the project continues effeciently with the contribution of universities, companies and governments; and around a hundred CubeSats have been launched to space.

Istanbul Technical University, as the leading educational establishment of Turkey, decided to have hand on the development of small satellites technologies by introducing a Cubesat project called ITUpSAT1 by the year 2005. The project resulted in 23 Sept, 2009 by the launch of the satellite from India to space.

This thesis is the product of the ITUpSAT1 project and presents the test standards tailored for the small satellites, specifically for pico and nano scales.

İTÜpSAT1 UYDUSUNUN ENTEGRASYONU VE TESTLERİ

ÖZET

Yapay uydular, Sputnik ile başlayan uzay çağının en önemli elemanları olmuşlardır. Güzümüzde bu uydular, gerçekleştirdikleri kritik görevlerle insanlığa hizmek etmektedirler. İletişim, uzaktan algılama, veri toplama, navigasyon, keşif ve bilimsel deneyler kullanıldıkları alanlardan sadece bir kısmını oluşturmaktadır. Uzay teknolojisine yapılan yatırımlar hiç bir zaman boşa gitmemiş, mühendislikte ve bilimdeki ilerlemeleri tetikleyen bir unsur olmuştur. Ayrıca bu gelişmeler de uzay teknolojisinin ileriye gitmesinde ve gelişmesinde büyük rol oynamıştır. Günümüzün uzay araçları ile onların atalarını karşılaştırdığımızda, bugünün uydularının çok daha küçük, hafif ve kompakt yapılar olduklarını görürüz. Uzayla ilgili bir projenin risk ve maliyet gereksinimlerini dikkate alırsak, bu durum çok önem arz etmektedir. Yüzlerce, hatta binlerce kilogramlık uydular, aynı zamanda görev bütçeleri için de ağır bir yük teşkil etmektedirler. Bu nedenle, küçük uydular tasarımları, testleri, onaylaması ve fırlatma maliyetlerindeki avantajlardan dolayı dikkat çekmektedirler.

2001 yılında, California Polytechnic Üniversitesi ve Stanford Üniversitesi tarafından 'Cubesat' isimli bir küçük (piko) uydu projesi başlatıldı. Asıl amaç lise ve üniversite öğrencilerinin eğitilmesiyken, tasarım diğer alanlara doğru yayıldı. Üniversitelerin, şirketlerin ve devletlerin desteği ile proje verimli bir şekilde devam etmektedir ve bugüne kadar yaklaşık yüz Cubesat uzaya fırlatılmıştır.

İstanbul Teknik Üniversitesi, Türkiyenin önde gelen üniversitelerinden biri olarak, 2005 yılında ITUpSAT1 isimli Cubesat projesine el atarak küçük uydu teknolojilerinde söz sahibi olmak istediğini gösterdi. Proje, 23 Eylül 2009'da uydunun Hindistan'dan uzaya fırlatılmasıyla son buldu.

Bu tez, ITUpSAT1 uydusunun tasarlanması sürecinde gerçekleştirilmiş olup, test standartlarının piko ve nano seviyelerdeki küçük uydulara uyarlanmasını konu edinmektedir.

1. INTRODUCTION

During the integration and testing process of ITUpSAT1, it has been understood that testing of space systems is a very essential part of a space mission. Every component of a spacecraft must perform adequatly and reliably to be succeed in the mission, and during the mission the spacecraft can’t be maintained or fixed, except for software updates or electrical resets. Therefore, testing of a spacecraft is one of the most important issues of space mission planning.

During the ITUpSAT1 project, which started at 2006, a testing approach have been develop as in all space projects. The MIL and ECSS test standards have been examined and literature searches have been performed. Rationally but unfortunatly for us, the test standards present general requirements and philosophies by containing all the space projects, such as satellites, planetary missions and lauch vehicles, etc. On the other hand, throughout a project, more specific test standards, procedures and plans are needed. For this purpose, in this tesis; the general testing standards of MIL and ECSS have been reduced and tailored to small satellites, especially for pico and nano scales.

Thoughout this chapter , adaptation of the general testing concept, model and testing philosophies to the small satellites will be discussed. To apply the general concept to small satellites, firstly we have to introduce them and state differences than other larger satellites.

1.1 Small Satellite Concept and Weight Categories

By speaking of a small satellite, we actually define the weight interval that it has to be. In fact, these satellites cover a wide range of weight interval and there is no generally accepted definition of small satellites. However, we can classify a spacecraft that is lighter than 500 kg as a small satellite. In addition, small satellites can also be categorized according to their weight. Table 1.1 shows small satellite types.

Table 1.1: Small Satellite Categories

Category	Weight (kg)
Mini	100 to 500
Macro	10 to 100
Nano	1 to 10
Pico	0.1 to 1
Femto	< 0.1

Small satellite concept is present since the outbreak of the space age. The need for smaller systems was constantly in minds and the technological development is always in that way, making it smaller and efficient. Miniaturization of the spacecraft elements and advancements in the micro-electro-mechanical systems lead us to a new idea. [1] Small satellites started to find new application areas like remote sensing, scientific experiments and even communications. The most remarkably advantage of a small satellite arises in the mission cost trades; design, testing, launch and operation costs are pulled down by a significant margin. For example, since they are lighter, they can be launched as a secondary or tertiary payload by a launch vehicle, meaning that a sharp fall in the launch expenditures.

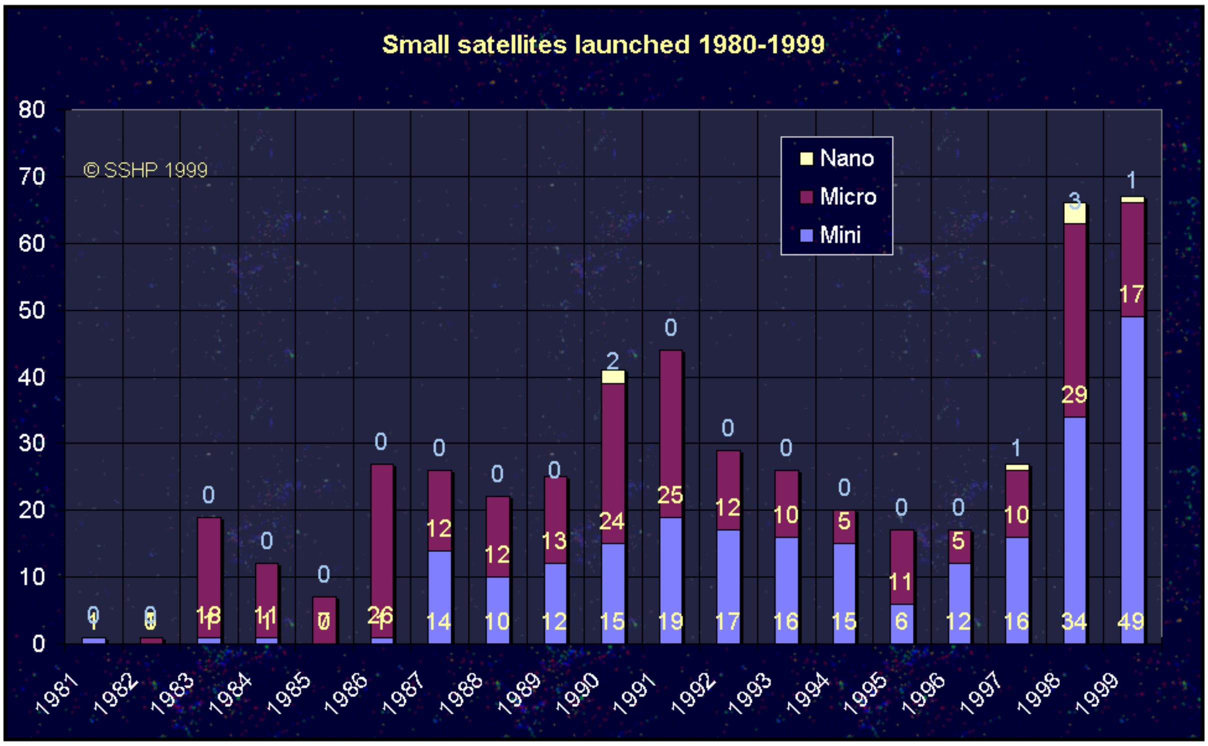

Figure 1.1: Small Satellite History [1]

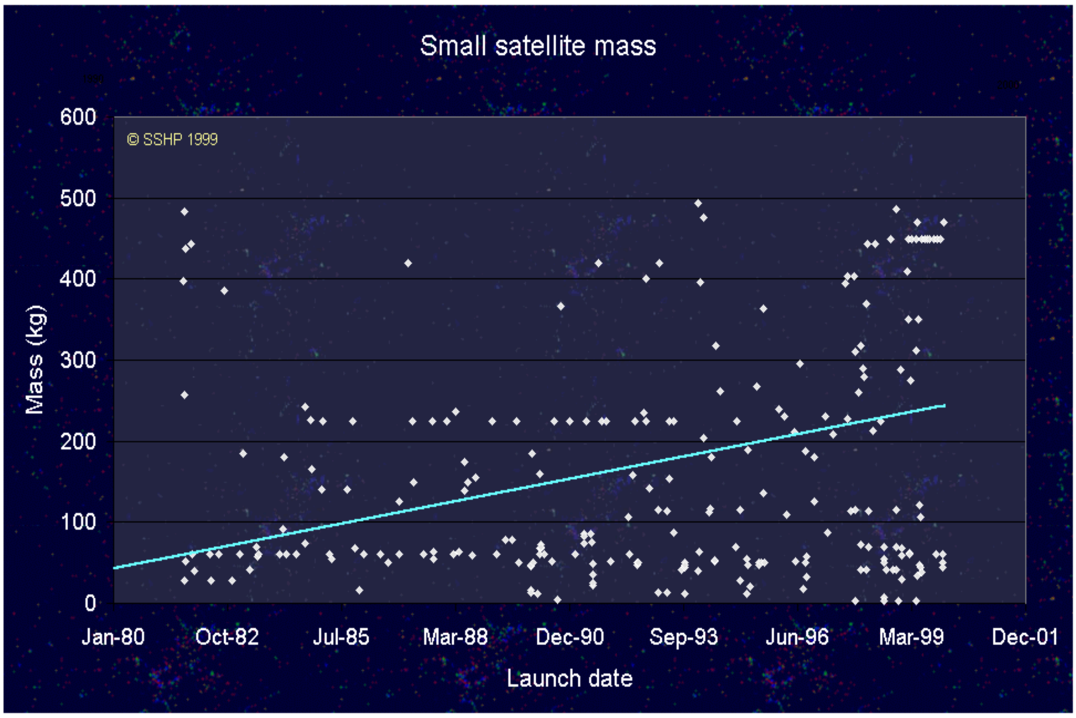

Figure 1.2: Launched Small Satellite Mass distribution [1]

Beginning by 2001, The Cubesat Project brought a new way of thinking, for the space missions. The project was started by California Polytechnic State University and Stanford University to provide low-priced space education and experiments. Satellites in the pico and nano scales drew attention of governments for scientific experiments, universities as an education platform and even big commercial firms as a cost-effective test bed. Today, over 60 universities and high schools are participating to the Cubesat program, government organizations and private companies benefiting from the advantage to test space elements in space in a very cost-effective way [2,3].

1.2 Specifications of Satellite To Be Tailored

In the previous section we mentioned that the weight of small satellites ranges between 0 to 500 kg. This is a very wide range to compose a specific spacecraft project. Design, production, testing and verification can be totally different between a mini- and pico-satellite. Instead of this, focusing a specific class is a more effective approach.

This document is dedicated to the testing standards of the satellites in pico and nano scale. In other words, we are stating the test requirements for a satellite that has a mass of up to 10 kg.

While tailoring the requirements to a satellite, another important factor to be considered is the environment where the satellite will operate. Examining the previous and the future pico and nano satellite launches, it can be concluded that it will be a low Earth orbit, specifically between 600 and 800 km. In that altitude atmospheric drag, gravitation, solar pressure, radiation effects are present.

2. DEFINITIONS

The technical terms used in this work are defined as used in the standards of ECSS (European Corporation for Space Standardization) [4] and MIL (USA Department of Defense Standardization) [5]. The definitions of the most important terms are listed below giving their specific meaning in the literature of this thesis.

Part: A part is one single piece or the connected pieces that can’t be disassembled without vandalizing the design.

Examples: resistor, integrated circuit, relay, roller bearing.

Subassembly: A subassembly is an element that has multiple parts with the ability of disassembly or part substitution.

Examples: printed circuit board with parts installed, gear train.

Unit: A unit is an operational element that is utilized for the aim of manufacturing, maintenance, or record keeping.

Examples: hydraulic actuator, valve, battery, electrical harness, transmitter.

Subsystem: A subsystem is a fusion and assembly of operationally related units and contains multiple units. A subsystem may also contain items such as cables or tubes, structures or mechanisms that are used for interconnection.

Examples: electrical power, attitude control, telemetry, thermal control, and propulsion subsystems.

Vehicle: Any vehicle defined in this section may be termed expendable or recoverable, as appropriate.

Launch Vehicle: A launch vehicle is one or more of the lower stages of a flight vehicle capable of launching upper-stage vehicles and space vehicles, usually into a suborbital trajectory. A fairing to protect the space vehicle, and possibly the upper-stage vehicle, during the boost phase is typically considered to be part of the launch vehicle.

Space Vehicle: A space vehicle is an integrated set of subsystems and units capable of supporting an operational role in space. A space vehicle may be an orbiting vehicle, a major portion of an orbiting vehicle, or a payload which performs its mission while attached to a launch or upper-stage vehicle. The airborne support equipment (3.2.1), which is peculiar to programs utilizing a recoverable launch or upper-stage vehicle, is considered to be a part of the space vehicle.

System: A system is a composite of equipment, skills, and techniques capable of performing or supporting an operational role. A system includes all operational equipment, related facilities, material, software, services, and personel required for its operation. A system is typically defined by the System Program Office or the procurement agency responsible for its acquisition.

Launch System: A launch system is the composite of equipment, skills, and techniques capable of launching and boosting one or more space vehicles into orbit. The launch system includes the flight vehicle and related facilities, ground equipment, material, software, procedures, services, and personnel required for their operation.

Maximum and Minimum Expected Temperatures: The maximum and minimum expected temperatures are the highest and lowest temperature levels that a spacecraft will be subjeted during its service life on orbit. These levels can be determined analitically and adding a uncertainty margin on it.

Extreme and Maximum Expected Sinusoidal Environment: The sinusoidal vibration environment is defined as an acceleration amplitude in g over the frequency range for which amplitudes are significant.

Extreme and Maximum Expected Acoustic Environment: The acoustic environment is given by a 1/3-octave-band pressure spectrum in dB (reference 20 micropascal) for center frequencies spanning a range of at least 31 to 10,000 Hz. The extreme and maximum expected acoustic environments are the bases for qualificaiton and acceptance test spectra, respectively, subject to workmanship-based minimum spectra [5].

Extreme and Maximum Expected Random Vibration Environment: The random vibration environment is defined as an acceleration spectral density (commonly power spectral density or PSD) in g^2/Hz over the frequency range of at least 20 to 200 Hz.

Extreme and Maximum Expected Shock Environment: The shock environment is defined as the derived shock response spectrum in g, based upon the maximum absolute acceleration or the equivalent static acceleration induced in an ideal, viscously damped, single-degree-of-freedom system.

Ambient Environment: The ambient environment is defined as room conditions with a temperature of $23 \pm 10^{\circ}$C, RH of 50 ± 30 percent and pressure of 101 +2/-23 kilopascals.

Service Life: The service life of an item starts with the production and ends with disposal or recovery from orbit.

Thermal Soak Duration: Thermal soak duration of a unit at the hot or cold exteme temperatures is the time that the unit is operating while the baseplate is maintaining its temperature within stabilization margins.

3. GENERAL REQUIREMENTS

3.1 Testing Philosophy

Testing is an important tool for the verification process of a space vehicle design for part, unit, subassembly, and subsystem or vehicle level. To test a design, some issues must be taken into the consideration. Firstly, the expected performance and requirements for the desired design must be decided. This subject determines the scope of the spacecraft design and drives the required properties that the environment enclosing the spacecraft must have. The other subject to be considered is the environmental effects of the space medium on the operations and system functionality of the spacecraft. Test planning, test requirements and test criteria are derived from these two topics.

To describe the testing process specifications for part, unit, subassembly, subsystem and vehicle levels; the specifications and verification method of the mentioned design elements must be clearly stated. Using this information an overall test plan must be established. This test plan must consist of sequences of the test process, objectives and scope of the test, test facilities description and implementation of all tests.

A complete test program for a spacecraft consists of development, qualification, acceptance and prelaunch validation tests respectively. [5]

Development tests are carried out to validate new designs, to help the design process of an item or to implement proven designs to new configurations. The same test levels for the loads shall be applied as in qualification, acceptance, prelaunch validation and operational tests.

Qualification tests are performed to demonstrate that design concept and manufacturing of the desired design can satisfy the requirements and withstand environment loads. The qualification load levels that will be applied during the test must exceed the expected environment loads by a safety margin.

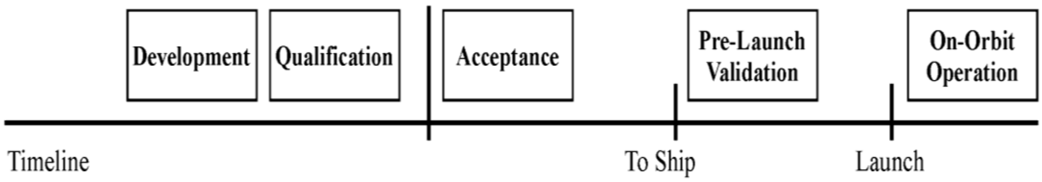

Figure 3.1 Testing Timeline For a Space Project

Acceptance tests are used to approve the design and production, to detect manufacturing faults, workmanship error and start of the failures and functional anomalies. The acceptance test levels shouldn’t exceed the predicted environmental levels during the life of the satellite.

Prelaunch validation tests aim to verify the readiness of the manufactured system for the launch and orbital life.

In the process of implementation of the test program; test methods, environments, measured parameters during testing and results of the each tests must be comprehensively analyzed and the data obtained in the test process of one element must be taken into the account by considering the effects on the other elements of the spacecraft.

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

As stated before, testing is a powerful and precise tool to verify the validity of a desing with lowest risk; however the expense of the test must be accurately estimated and it must be minimized as much as possible. Therefore, a test engineer shall understand and analyze the requirements and the scope of the design accurately. He/she shall distinguish that which tests are critical, which tests are unnecessary for a project.

3.2 Model Philosophy

After establishing the basis for the testing philosophy, the next step is composing a model approach that will be used during the testing process. Model approach should explain how the qualification, acceptance and protoflight test activities will be performed, e.g. which models will be used. Model philosophy is important since an accurately stated model approach makes the testing process more effective.

Next topic explains the models that can be used for development, qualification, acceptance and protoflight tests.

3.2.1 Description of Models

Below and also on Table 3.1, you can find models that can be employed for pico and nano satellites.

Development Model (DM):

Development model is used to validate a new design or a redesign and to assist the development process.

Development models may be employed for all kind of equipment like parts, units and even subsystems. Functional and environment test shall be performed on development models.

Integration Model (IM):

Integration model is chosen generally for functional and interface tests of the electronic hardware and software.

Suitcase:

The suitcase model aims to test the performance of data handling and communication equipment.

It is used to test link budgets with the ground station or other networks and also command and telemetry formats, bitrates and packets.

Structural Model (SM):

Structural model is employed to demonstrate the qualification of the structural design and to correlate mathematical models.

The structural model should represent the end item in structural aspects. For this purpose, structural dummies can be used for the SM of a system.

Thermal Model (TM):

Thermal model is used to show the qualification of the thermal design and to correlate mathematical models. The thermal model should represent the end item in thermal aspects. For this purpose, thermal dummies can be employed for the TM of a system.

Structural - Thermal Model (STM):

The combination of SM and TM can be used to validate the structural and thermal qualification together. The structural-thermal model of system represents the end item with thermal and structural dummies.

Engineering Model (EM):

The engineering model is flight representative in form, fit and function, without full redundancy and hi-rel parts. [6]

It may be used to demonstrate the functional qualification of the item.

Engineering Qualification Model (EQM):

The engineering qualification model fully reflects the design of the end item, except for the parts standard. [6]

The engineering qualification models are used for functional performance qualification and EMC testing. Environmental tests may also be performed on the EQM, if it is suitable.

Qualification Model (QM):

The qualification model is identical to the end item design in all properties. All functional and environmental tests should be carried out on the qualification model to demonstrate the qualification of a part, unit, subsystem or integrated system.

Flight Model (FM):

The flight model is the end item configured to be launched to space. Functional and acceptance tests should be applied on the flight model.

Protoflight Model (PFM:)

The protoflight model is the flight end item on which a partial or complete protoflight qualification test campaign is performed before flight. [6]

The test levels and duration that will be applied on the protoflight model should be carefully analyzed and determined considering the life-time of the test object.

Flight Spare (FS):

The flight spare is a spare end item that can be used for flight. Acceptance testing should be applied on the FS.

Refurbished qualification items may be used as flight spares, however the item that went through the qualification testing should never be used as flight spare.

3.2.2 Model Philosophies Description

According to the requirements of the project, several models philosophies may be applied ttesting. Most applicable model approaches are given below.

Prototype Approach:

In the prototype approach, one or more qualification models may be used for qualification testing with qualification levels and durations. For the tests that use more than one QM, each qualification model may be subjected to different tests according to its configuration and representativeness.

The acceptance testing shall be applied on the flight model.

Protoflight Approach:

In the protoflight approach, all of the required tests are performed on the same model that will be launched. The tests should be conducted with qualification levels and acceptance durations. Testing on the protopflight model should be very carefully carried out to avoid from wearout, drift and fatigue related problems.

Table 3.1: Model Definition (Adapted from reference [6])

Model	Objectives	Representativeness	Applicability	Remarks
Development Mode (DM)	Confirmation at design feosibility	Total conformity with functional electrical & S/w req. in agreement with verif. objectives (size, shape & I/Fs could not be representative)	All levels	Development testing Sometime if is also called breodboard
Integration Model (IM)	Functional development SW development Procedure validation	Functional representativeness Commercial ports Simulators of missing parts	All levels	Development testing It could be considered something in between o mock-up and on EM Sometime s called also Bectrical Model
Suitcase	Simulation of functional & RF performances	Right design Commercial parts Functional representativeness	Equipment level System level	Qualification testing
Structural Model (SM)	Qualification structural design Validation of structural mathematical model	Flight standard with respect to structural parameters Equipment structural dummies	Ss level (Structure) Sometime it could be considered system level if involves other SS or is merged with the system test flow	Qualification testing
Thermal Model (TM)	Qualification thermal design Validation of thermal mathematical model	Flight standard with respect to thermal parameters Equipment thermal dummies	Sslevel (thermal control) Sometime it could be considered system level if involves other SS or is merged with the system test flow	Qualification testing
Structural-Thermal Model (STM)	SM&TM objectives	SM & TM representativeness Equipment thermo structural dummies	System level	Qualification testing
Engineering Model (EM)	Functional qualification failure survival demonstration & parameter drift checking	Fight representative in form fit function Right design without redundancies and hi-rel parts	ALL LEVELS	Partial functional qualification testing
Engineering Qualification Model (EQM)	Functional qualification of design & I/Fs EMC	Full Flight Design MiL- Grode parts procured from the some manufacturer of hi-rel parts	ALL LEVELS	Functional qualification testing
Qualification Model (QM)	Design qualification	Full Flight Design & Flight Standard	Equipment level SS level	Qualification testing
Flight Model (FM)	Flight use	Full Flight Design & Flight Standard	All levels	Acceptance testing
Proflight Model (PFM)	Flight use design qualification	Full Flight Design & Flight Standard	All levels	Protoflight qualification testing
Flight Spare (FS)	Spare for flight use	Full Flight Design & Flight Standard	Equipment level	Acceptance testing

Hybrid Approach:

A combination of the protoflight and prototype approach may also be used. The models, which are explained in detail in the previous sections may be employed to confirm the performance of the required parts, units or subsystems of the satellite.

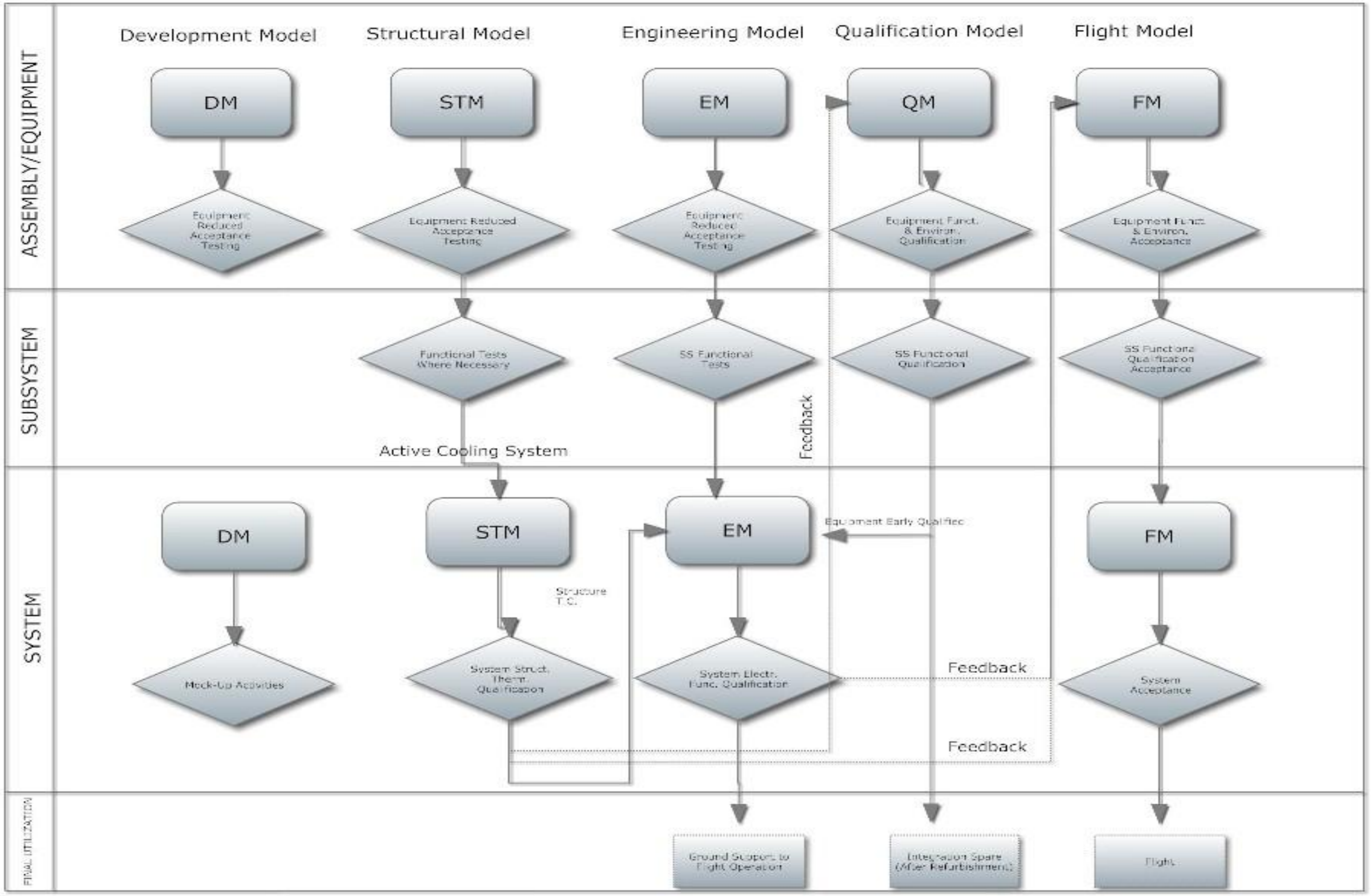

Figure 3.2: Example Model Philosophy Diaram [6]

3.3 Test Condition Tolerances

The test performance parameters stated in the frame of the test philosophy should be evaluated considering the maximum allowable test tolerances demonstrated in Table 3.2.

The project or testing authority can specify different tolerances and can be rigid or flexible if required.

Table 3.2: Test Parameter Tolerances [5]

Temperature		± 3°C
-54°C to +100°C		
Relative Humidity		± 5 percent
Acceleration		+10/-0 percent
Static Load and Pressure		+ 5/-0 percent
Atmospheric Pressure		
Above 133 pascals (>1 Torr)		±10 percent
133 to 0.133 pascals (1 Torr to 0.001 Torr)		± 25 percent
Below 0.133 pascal (<0.001 Torr)		±80 percent
Test Time Duration		+ 10/-0 percent
Vibration Frequency		± 2 percent
Sinusoidal Vibration Amplitude		±10 percent
Random Vibration Power Spectral Density		
Fraquency Range	Maximum Control Bandwidth	
20 to 100 Hz	10 Hz	± 1.5 dB
100 to 1000 Hz	10 percent of midband frequency	± 1.5 dB
1000 to 2000 Hz	100 Hz	± 3.0 dB
Overall		± 1.0 dB
Sound Pressure Levels		
1 /3-Octave Midband Frequencies		
31.5 to 40 Hz		± 5.0 dB
50 to 2000 Hz		± 3.0 dB
2500 to 10000 Hz		± 5.0 dB
Overall		± 1.5 dB
Note: The statistical degrees of freedom shall be at least 100.		
Shock Response Spectrum (Peak Absolute Acceleration) Natural *Frequencies* *So* aced at 1/6-Octave *Interval* s		
At or below 3000 Hz		± 6.0 dB
Above 3000 Hz		+ 9.0/-6.0 dB
Note: At least 50 percent of the spectrum values shall be greater than the nominal test specification.		

3.4 Test Plans and Procedures

The test plans and procedures documents shall consist of sufficient detail to present the frame for identifying and interrelating all of the specific tests and test procedures required [5].

3.4.1 Test Plans

Test plans should contain the objective and general description of each test, and also the test conditions. When forming the test plan, test requirements and mission operations must be clearly stated and analyzed.

The test plans should contain some important information about tests.

- A short background information about the project and information about the items to be tested.
- The test philosophy, testing approach, and test objective for each item, tailoring of the test requirements to a specific item, if any.
- The several different test areas on the design.
- The classification of different modes and levels of environment that the item or satellite will be subjected. For example, the load level during launch and during orbit mission.
- The information about the specifications of environmental test areas.
- Test equipment and test facility information.
- The proof of the test tools and test beds that they can achieve the actual operational environment during tests.
- The formats and standards of the test data that will be recorded.
- The review and verification method for test plans and procedures.
- The detailed schedule of the test process.

3.4.2 Test Procedures

Test procedures are prepared as a walkthrough to perform required test according to their test objectives and test plans. The test objectives, testing criteria, and pass-fail criteria shall be stated clearly in the test procedures [5]. It is recommended that the procedures explain the test process step by step, and including also implementation of each step.

The test procedure for each item shall include, as a minimum, descriptions of the following:

- Criteria, objectives, assumptions, and constraints.
- Test setup.
- Initialization requirements.
- Input data.
- Test instrumentation.
- Expected intermediate test results.
- Requirements for recording output data.
- Expected output data.
- Minimum requirements for valid data to consider the test successful.
- Pass-fail criteria for evaluating results.
- Safety considerations and hazardous conditions [5].

4. ENVIRONMENTS AND EFFECTS

The environment describes the medium that a spacecraft will encounter from the beginning to the end of its life. This process starts with the production of the spacecraft and ends with its re-entry. We can classify the environments that a spacecraft will be exposed in three categories: The earth, launch and space environment.

The spacecraft is designed and manufactured to operate and survive in launch and space environments; however, the earth environment has also great importance, since more severe loads may impose on the vehicle during its handling on Earth.

The range and level of the loads in launch and space environment draw the outline of the environment tests to be implied on the spacecraft. Derived qualification test levels should be threated as actual environment conditions to demonstrate the validity of the design.

4.1 Earth Environment

During the accomadation of the spacecraft on Earth, it is subjected various environments that degrades the performance of the spacecraft. Atmosphere is an important phenomena that effects the life and performance of a spacecraft. Consisting water and oxygen, it causes corrosion on the structural and mechanic parts and on the circuits boars. This may likely result in malfunction or performance degradation. To avoid this, the component or spacecraft shall be handled in a controlled relative humidity environment. In general, 40-50% RH is a preferable compromise.

Particulate contamination is another problem that affects spacecraft performance. Dust particles in the environment may accumulate on the surfaces of s/c components. Particles may reside on the solar cells and degrade the potential electrical production of the solar cells. or, dust particles on the lens of a star tracker camera may be perceived as a star on the map and this will cause performance degradation or malfunction of the attitude determination subsystem [7]. To avoid contamination, assembly and handling of the space equipment is held in “cleanroom” environments. In a clean room RH, temperature, and particle contamination is controlled in a desired margin. Cleanroom workers shall wear special clothing that restricts the contamination caused by regular clothing. Cleanroom clothing includes gloves, smocks frock or bunny-suits, head covering and foot covering [7]. Cleanrooms are classified according to the particle number per cubic meter in the environment. Table 4.1 demonstrates the ISO standards for cleanroom contamination levels. Most achievable level is class 10,000 and this is a typical standard for spacecraft operations. For comparison, an ordinary room is approximately 1,000,000 class (ISO Class 9).

Table 4.1: ISO (and FED STD 209E Equivalent) Cleanroom Standards

Class	maximum particles/m³						**FED STD 209E equivalent**
	≥0.1 µm	**≥0.2 µm**	**≥0.3 µm**	**≥0.5 µm**	**≥1 µm**	**≥5 µm**	
ISO 1	10	2					
ISO 2	100	24	10	4			
ISO 3	1,000	237	102	35	8		Class 1
ISO 4	10,000	2,370	1,020	352	83		Class 10
ISO 5	100,000	23,700	10,200	3,520	832	29	Class 100
ISO 6	1,000,000	237,000	102,000	35,200	8,320	293	Class 1000
ISO 7				**352,000**	**83,200**	**2,930**	**Class 10,000**
ISO 8				3,520,000	832,000	29,300	Class 100,000
ISO 9				35,200,000	8,320,000	293,000	Room air

Static electricity is an important issue that can cause damage on the space equipments via triboelectric effect. Integrated circuits and components including metal-oxide semiconductor technology are highly vulnerable by the voltage differences. To avoid this, cleanroom workers shall be grounded when handling the hardware. Conductive flooring, conductive shoes and ankle straps, grounded wristbands shall be used. As stated before, high relative humidity is an undesirable case. On the other hand, too dry air is also posing a problem causing static charge accumulation. Considering both situations, RH in a cleanroom is 40-50% in general.

Transportation of the equipment or spacecraft may cause unnecessary vibration and shock. In addition, the levels of these forces may exceed the flight level of a vehicle. Considering higher vibration and shock levels and longer exposure durations, an appropriate carrier structure must be supplied or manufactured. During ground or air transportation, the s/c shall be properly secured to the casing. Cleanness, humidity and other mentioned constraints must be taken into the consideration also in transportation. Passive humidity, temperature and acceleration sensors may be used to see the transportation history of the spacecraft.

4.2 Launch Environment

Launch environment presents highly stressful load levels for a relatively short time period according to earth and space environments. Axial loads are generated due to the acceleration of the launch vehicle, lateral loads from steering and wind gusts. Strong mechanical vibration and acoustic energy input, and the signifiant pressure drop at the initial phase of launch are the other conditions to be considered. Additionally, aerodynamic heating may enforce thermal loads on the launch vehicle, and stage shotdown, seperation and fairing jettison will produce shock transients [7].

Do the these effects mentioned, acceleration, sinusoidal vibration, random vibration, acoustic and shock environment must be analyzed and clearly stated to perform environmental tests. For the qualificaiton and acceptance tests, a margin of safety should be imported to verify the validity of the design. To compose a preliminary design, launch vehicle manuals specify required parameters. Since the spacecraft and launch vehicle interact, they must be analysed as a coupled system to form the actual environment loads.

4.3 Space Environment

“The space environment is characterized by a very hard vacuum, very low gravitational acceleration, possibly intermittent or impulsive nongravitational accelerations, ionizing radiation, extremes of thermal radiation source and sink temperatures, severe thermal gradients, micrometeorids, and orbital debris. Some or all of these features may drive various aspects of spacecraft design.” [7]

It should be noted that, for every orbit (LEO, MEO, GEO) the space environment and its effects are different. In this subclause, the effects and risk for a LEO satellite will be considered.

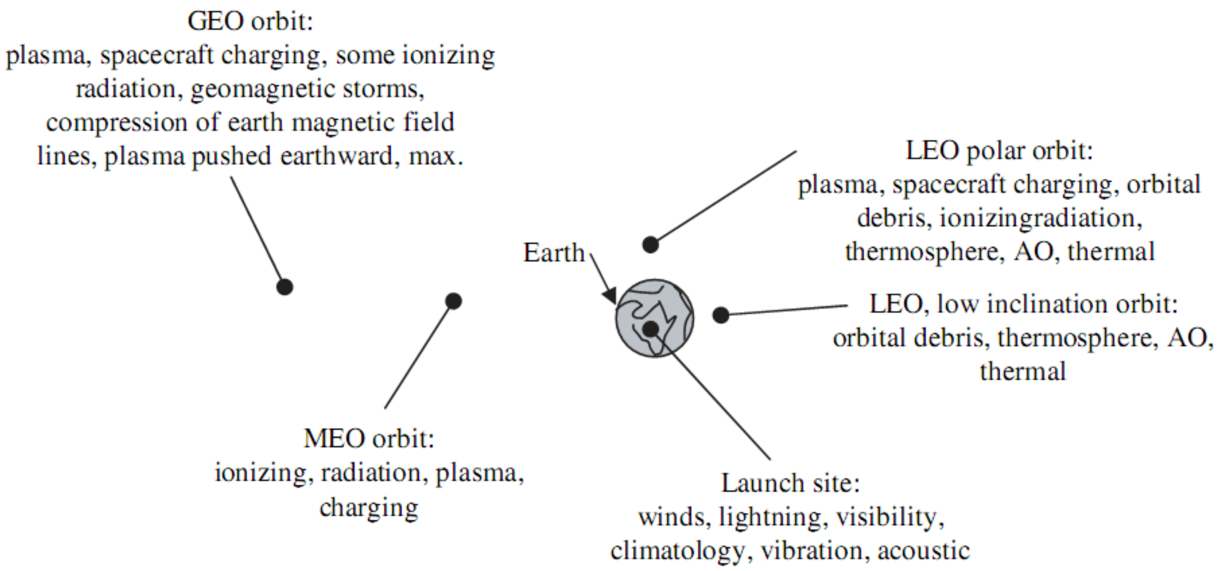

Figure 4.1: Dominant Space Environment Effects Due to Altitude [8]

4.3.1 Vacuum Environment

The vacuum environment may be defined as the absence of the usual atmosphere which means low pressures. The standard atmospheric pressure is 101.325 kPa and at the 350 km altitude it is over ten orders of magnitude less than sea level on Earth. This vacuum(like) condition brings serious consideraitons on the thermal environment, because in vacuum the heat transfer mechanisms are radiation and conduction.

A spacecraft will be exposed to solar ultraviolet radiation during its orbit. The energy of a photon can sever organic chemical bonds and change characteristics of the materials. If oxygen atoms are removed from bonds, this may be resulted in darkening of the outer surfaces of the spacecraft. Darker surfaces absorb more heat and this will cause an increase in the innner operating temperatures of the spaceraft elements. The safe operating temperature limits for equipments or electronics may be exceeded. To ensure the UV degradation safety, exterior materials must be selected carefully and protective films and coating may be utilized.

The spacecrafts orbiting in the space are cooler than the surrounding environment, so it absorbs the incoming energy, related to its solar absorptance (α_s) and it dissipates energy according to its emittance coefficient (ε). Excessive energy inside the vehicle, is dissipated generally by the thermal radiators with high α_s/ε ratio. Altough some of the degradation in solar absorptance values are caused by solar ultraviolet radiation, most of the degradation is due to molecular contamination. If a thin film reside on the surface, this will accumulate heat by absorbing some of the incoming radiation [10]. The cleaning of the radiators are especially important since they keep the temperature of the spacecraft in an operational temperature range. Contamination may also cause solar panel degradation and optical signal attenuation.

Another problem involved with the vacuum environment is molecular outgassing of the materials. Outgassing may also cause contamination and even harmful effects on optics of the spacecraft. Therefore, thermal vacuum bakeout shall be applied on the spacecraft.

Table 4.2: Vacuum Environment Design Guideline [9]

Materials selection	Hoose UV resistant, and low outgassing, materials and coatings
Configuration	Vent outgassed material away from sensive surfaces
Margin	Allow for degradation in thermal/optical properties on orbit
Materials pre-treatment	Consider vacuum bakeout of materials before installation in vehicle
Flight & ground operations	Provide time for on orbit bakeout during early operations; provide cryogenic surfaces the opportunity to warm up and outgas contaminant films

4.3.2 Neutral Environment

Although the space environment is described as a vacuum environment, its not totally vacuum. In LEO, MEO and even in GEO there is an atmosphere, which can not be ignored. It has severe mechanical and chemical effects on the spacecraft. Due to relatively high velocities, neutral atmosphere causes aerodynamic drag on the vehicle and it may physically sputter material from surfaces. As we are dealing with a specific orbit LEO, the richest element here is atomic oxygen. AO has a very reactive nature and may cause erosion on surfaces and a visible glow where interacts with the spacecraft.

Drag caused by the neutral atmosphere is an important parameter for the orbiting spacecraft. In some cases, it would become the biggest orbital disburbance that the spacecraft expose. At 600 km altitute, density of the atmosphere is 9.89x10-14 kg/m3 [10]. Although it seems as a relatively small number, it alters the attitude and altitude of the spacecraft. “In addition to the drag force on surfaces oriented normal to flow direction, there is also a possibility for drag on laterally oriented surfaces. The thermal velocity o an atomic oxygen in LEO is ~ 1 km/s and the veleocity of the spacecraft is around 8 km/s. The lateral speed of the AO is big enough to impart momentum to the spacecraft laterally.” [9] When the altitude is increased the density of th atmosphere drops and the aerodynamic drag force decreases. However, for a space mission, it is impossible to plan the orbit and altitude according to this phenomena. Instead of this, the normal area to the RAM direction (direction of travel) should be minimized. For this purpose, a commonly used technique is orienting the solar panels to an appropriate angle, while also considering the power requirements.

Another physical effect of the neutral environment is the sputtering. The neutral molecules in the medium has big enough impact energies to collide and sever chemical bounds on the surface. This causes a change in the characteristics of he surface and it may be manged by choosing appropriate material.

In addition to the mechanical effects, there is also chemical interaction between AO and spacecraft surfaces. AO related erosion or oxidation may lead to very serious damages on the system, since the space materials are generally selected according to their thermal properties and made as thin as possible to decrease weight. AO degradation is a essential issue for the long term missions in LEO.

Table 4.3: AO erosion rates (mm/year) [8]

Material	Silver	Mylar	Kapton	Epoxy	Carbon	Teflon	Aluminum
Erosion rate (mm/yr)	$\sim 10^{-1}$	$\sim 10^{-1}$	$\sim 10^{-1}$	$\sim 10^{-2}$	$\sim 10^{-2}$	$\sim 10^{-3}$	$\sim 10^{-5}$

Lastly, the spacecraft glow may degrade the spacecraft performance. Especially optical sensors and remote sensing equipment may malfunction because of the glitter of the surface. Glow phenomena has been reported in some NASA's missions.

Table 4.4: Natural Environment Design Guideline

Materials	Choose materials that (a) are resistant to AO, (b) do not glow brightly (if optical instruments presents), and (c) have high sputtering thresholds
Configuration	Aerodynamic drag may be minimized by flying the vehicle with a low cross-sectional area perpendicular to ram. Orient sensitive surfaces and optical sensors away from ram.
Coatings	Consider protective coatings for surfaces that are susceptible
Operations	If possible, fly at altitudes that minimize interactions

4.3.3 Plasma Environment

Plasma includes combination of free electrons and ions - atoms that have lost electrons. Energy is needed to strip electrons from atoms to make plasma [11]. More formally, a plasma can be defined as a gas of electrically charged particles, where electrons have enough kinetic energy to remain free from ions [9].

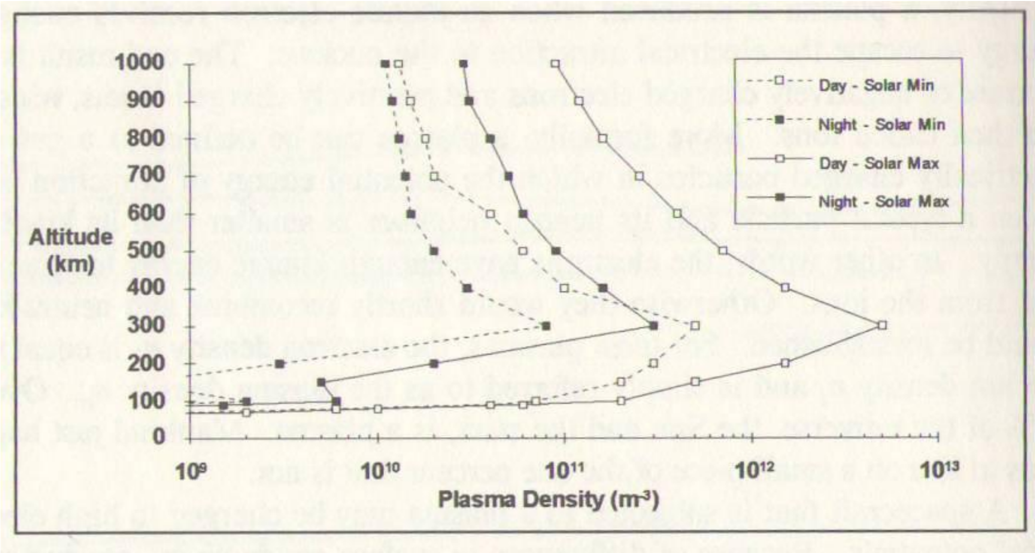

Figure 4.2: The LEO Plasma [9]

Main engineering concerns related with the plasma environment on LEO orbit are power leakage, possible discharges, high spacecraft ground potential, sputtering and surface charging. Communications of the spacecraft with the ground station may also be perturbed or jammed [12].

A spacecraft that is exposed to a plasma environment may be charged with high electrical potentials, high potential differences may be occurred between spacecraft components. Therefore, the s/c shall be grounded. There are three options for grounding. Connecting the spacecraft to the end of the array that has no interaction with plasma is called as *negative grounding*. Connecting the spacecraft to the end of the array that moves above the plasma is called as *positive grounding*. Making no electrical ground on the spacecraft is called *floating ground.* Negative grounding is usually preferred, since this convention accommodates current flow through standard npn transistors [9].

To control the spacecraft charging; grounding, surface material selection, shielding, filtering and testing are required precautions. Grounding all of the conductive surfaces and components to a common ground will efficiently reduce the potential differences. Selection of appropriate surface materials and shielding and filtering the input to circuits will be a good precaution to electrostatic discharges. In addition, testing and verification on the ground is very important to cancel the undesired interactions.

Table 4.5: Plasma Environment Design Guideline [9]

Uniform Surface Conductivity	Make exterior surfaces of uniform conductivity if possible
ESD Immunity	Utilize uniform spacecraft ground, electromagnetic shielding, and filtering on all electronic boxes
Active Current Balance	Consider flying a plasma contractor or a plasma thruster

4.3.4 Radiation Environment

There are basically three source of radiation for a spacecraft in space. These are trapped radiation belts around the Earth, galactic cosmic rays (GCRs) and solar particle events (SPEs). The trapped radiation belts, or Van Allen Belts consist of energetic particles (electrons and protons), which gyrate around the Earth's magnetic field lines. Radiation belts affect high orbits as well as the low altitudes. Van Allen Belts includes trapped electrons up to a few MeV and protons up to several hundred MeV of energy. Inner and outer radiation belts are peaking around 4000 km and 24,000 km.

Another important distortion in low earth orbit is caused by South Atlantic Anomaly. South Atlantic Anomaly depends on the offset between the Earth's magnetic field and geographical poles by about 11 degrees. This results that the radiation belts to reach lower altitudes over the South Atlantic region and LEO satellites expose high amount of energetic particles during their passes [13].

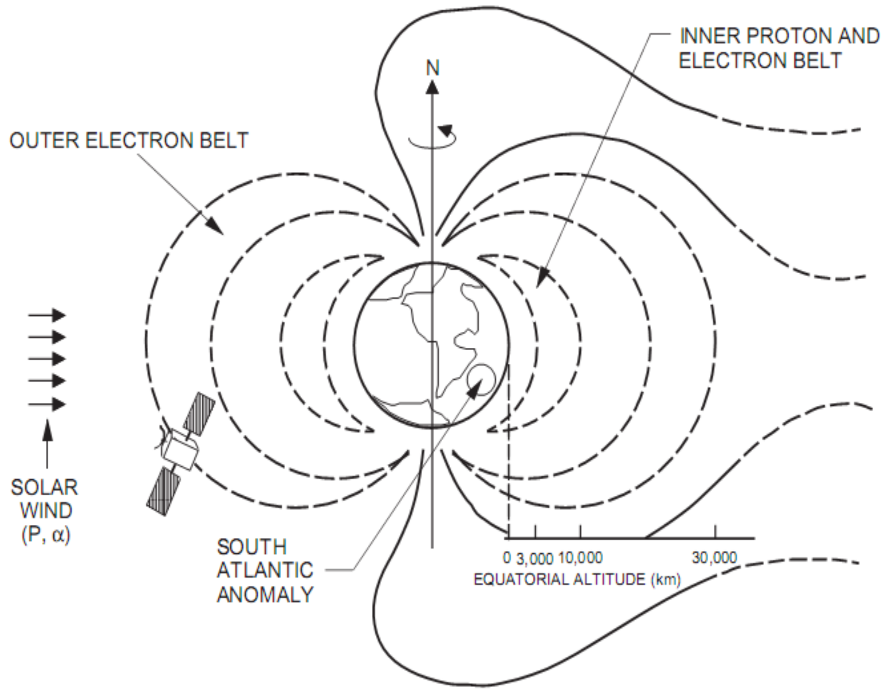

Figure 4.3: Trapped Radiation Belts and South Atlantic Anomaly [13]

Another constituent to radiation environment is solar energetic particles, which are originated by coronal mass ejections (CMEs) from the Sun. Sun ejects through the CMEs protons, alpha particles and also heavier elements, however, the flux of ejected protons are very high related to other elements. The effects of these so called solar particle events (SPEs) may last from a few hours to a few weeks. The earth magnetic field provides a magnetic shielding for satellites, especially for lower orbits.

For the spacecraft, there is also a continuous flux of Galactic Cosmic Ray (GCR) ions. The flux is low, a few ions per cm2 per second, it includes energetic heavy ions that may cause problems especially on the electronics.

The radiation environment elements described above may have severe effects on the systems or components of the spacecraft. Energetic particles may have hazardous consequences on the solar arrays, electronics and materials. They may easily penetrate through the walls of the satellite and accumulate doses of hundred kilorads during the orbital life. Energetic ions from GCRs and SPEs may cause ionizing radiation by losing energy in materials. Ionizing radiation can harm the solar arrays or memory elements, leading to single event upset (SEU). Energetic electron can pass through the thin walled structures and may cause static electricity on the elements like cables circuit boards an ungrounded metallic boards. In addition to the ionizing damage, energetic particles may bring on displacements damage by dislocating the particles in material from their original sites. This will alter the characteristics of the mechanical, electrical and optical materials and may harm electro-optical components like solar cells, and detectors such as CCD’s.

Table 4.6: Radiation Tests [12]

Radiation effect	Parameter	Test means
Electronic component degradation	Total ionizing dose	Radioactive sources (e.g. ^{60}Co), particle beams (e^-, p^+)
Material degradation	Total ionizing dose	Radioactive sources (e.g. ^{60}Co), particle beams (e^-, p^+)
Material degradation (bulk damage)	Non-ionizing dose (NIEL)	Proton beams
CCD and sensor degradation	Non-ionizing dose (NIEL)	Proton beams
Solar Cell degradation	Non-ionizing dose (NIEL) and equivalent fluence	Proton beams (low energy)
Single event upset or latch up for example	LET spectra (ions) proton energy spectra, explicit SEU/L rate	Heavy ion particle beams, proton particle beams
Sensor interference (background signal)	Flux above energy threshold, flux threshold Explicit background rate	Radioactive sources, particle beams
Internal electrostatic charging	Electron flux and fluence Dielectric E-field	Electron beams Discharge characterization

In LEO, the galactic cosmic rays and solar particle events are mostly filtered by electromagnetic shielding of the Earth. The low altitude environment is characterized by high radiation belt trapped energetic protons. However, for the low altitude polar orbits at high latitudes the spacecraft expose the unattenuated dose of GCRs and SPEs.

To protect the spacecraft from the radiation environment, radiation shielding is necessary. To take precautions for the energetic particles, the radiation environment must be modeled and maximum and minimum doses should be determined to calculate required shielding thickness for materials.

The models for trapped radiation belts are available on Community Coordinated Modeling Center. There are AP8MIN and AP8MAX models for proton flux in the solar minima and solar maxima, and AE8MIN and AE8MAX model for electron flux in solar minima and maxima. For the SPE, JPL-1991 model is used generally and GCRs levels may be calculated using CREME96 model. In addition, codes are present like Shieldose and Spacerad to determine the shielding depth using environmental radiation doses.

Flash X-ray machines are often used to simulate transient ionizing radiation. Electron beam or brehmsstrahlung X-rays may be used for the tests. Doses of 1 MRad (Si) can be obtained from an electron beam, where X-rays can provide 1 KRad (Si). Dose and Dose rate can be adjusted by changing the distance between object and source. [9]

Concentrated source of radioactive materials can be used to subject samples to doses of either alpha, beta, or gamma radiation, depending on the nature of the source itself. Co^{60} and Cs^{137} are both gamma-emitting sources that are often used in immersion studies. Single event effects may be simulated with Cf^{252}, which emits alpha particles [9].

5. DEVELOPMENT TESTS

5.1 Development Test Concept

Development tests shall be performed in case of:

- Validate new design concepts or the application of proven concepts and techniques to a new configuration.
- Assist in the evolution of designs from the conceptual phase to the operational phase.
- Reduce the risk involved in committing designs to the fabrication of qualification and flight hardware.
- Validate qualification and acceptance test procedures.
- Investigate problems or concerns that arise after successful qualification. [5]

Development test is the vital part of the whole testing philosophy for the programs that plan to test items to be launched to space, by reducing the levels and durations of the qualification testing. Development tests can be applied on breadboard equipments, prototype hardware mock-ups, development and integration models.

Throughout this chapter, development testing for pico and nanosatellites will be stated as in the order of MIL-STD-1540C test standard document. The requirements for each test will be discussed.

Development test approach can be different for a project; actually it depends on the maturity of the design. Moreover, development tests may not require the same levels with the qualification, acceptance, prelaunch validation tests.

5.2 Part, Material and Process Development Tests

Development tests are applied for new parts, materials, and processes to assure the feasibility of them in the implementation of the design. Using development tests, design and manufacturing alternatives can be demonstrated.

5.3 Subassembly Development Tests

Subbassemblies can be subjected to development tests to minimize the design risk and to demonstrate the feasibility of the items for an efficient design.

5.4 Unit Development Tests

Units can be subjected to development tests to minimize the design risk, to show the manufacturing feasibility, to confirm the mechanical or electrical performance. The development tests can also be applied to show that the unit can survive under the environmental conditions of launch and space medium.

In the test process, more severe load levels than the qualification requirements can be applied on the unit to identify the weak elements and to show assurance of the design.

New designs should be subjected to worst-case voltage or frequency variations. The designs can also be tested in a thermal scenario, within minimum and maximum temperature limits, and the effects of the vacuum environment can also be investigated.

Resonance survey tests can also be conducted and analyzed the resonance modes of the unit to avoid undesired response that will lead to harmful results.

5.4.1 Thermal Development Tests

For the units, especially f**o**r the electrical and electronic items, which will perform under a vacuum environment below 0.133 Pascal; development tests shall be necessary to demonstrate the conformity of the design. The temperature range for critical items, where they function properly must be determined to assist to the qualification tests by forming a basis for the requirements.

For the systems that have an active thermal control subsystem, heat transport tests may be required in unit level to show that the unit is able to accomplish to keep the system between operational temperature limits.

The heat conductance of the units may be also necessary, which have parts like thermal isolators, vibration isolators, harness, etc.

5.4.2 Shock and Vibration Isolator Development Tests

For newly designed or existed units that will be mounted on a shock or vibration isolator, development tests must be performed to confirm the effectiveness of the isolator. The performance of the isolator in the extreme thermal and chemical effects should be determined validate the design. During this development test, the unit itself or the identical dummy of unit with mass properties shall be utilized. The test should be conducted at all three axis and the responses should be measured at all corners of the unit or its dummy to show the effectiveness of the isolator and to form a basis for the qualification and acceptance testing requirenments.

5.5 Integrated System and Subsystem Development Tests

Structural and thermal models of a vehicle or subsystem should be subjected to the development tests using dynamic and thermal environment levels to verify the resistance of the thermal or structural design to the expected loads and to establish qualification and acceptance criteria.

In subsystem and vehicle level, the aim of the development tests is to verify the mechanical interfaces and interactions, to assist to the performance check of deployment mechanisms and thermal subsystems.

5.5.1 Mechanical Fit Development Tests

Development tests to investigate the compatibility of mechanical interfaces, where the vehicle is to be mounted are an important part of the whole test concept. For example, if a launch adaptor, which will be mounted on the launch vehicle, is to be designed then the mechanical fit development tests becomes more essential, because the designer or producer of the adaptor has to prove the physical fitness of the mechanical equipment and assemblies to the launch service provider.

5.5.2 Modal Survey Development Tests

Before the qualification modal survey tests, modal survey development tests should be performed to form basic information for the further testing. Once the natural modes of a subsystem or a vehicle designed is early determined, the design can be reviewed or qualification or acceptance test levels can be rearranged to avoid undesired responses. Mechanisms, structural or control subsystem should be characterized, since they consist more vulnerable parts or units to the dynamics.

5.5.3 Structural Development Tests

For the structures that expose to redundant loads like solar panels, extension booms and nonlinear bearings it is important to determine the member loads, stress distributions, deflections and the other structural responses. These data can be utilized to verify the load transformation matrix and the deflection transformation matrix. In addition to the dynamic model of the design, these matrices can be used to calculate the axial forces, bending moments, shears and torsion moments, and various stresses and deflections. This set of data will help to load limits that can be applied during a test and will help to determine safety margins for a design.

5.5.4 Acoustic and Shock Development Tests

The high frequency vibration and shock responses are difficult to predict by analytical techniques or analyses. Therefore acoustic and shock development tests may be required to demonstrate the sufficiency of the dynamic design criteria.

For a pico- or nanosatellite, especially that has a small-sized symmetrical body; acoustic development tests may not be required. Furthermore, pico- and nanosatellites don’t use pyrotechnical equipments due to their small size. This means they do not generate a reasonable shock on the system. The only shock that will affect the satellite will be originated from the launch vehicle. While performing the shock development test, these levels form the datasheet of the rocket may be applied with a reasonable margin.

5.5.5 Thermal Balance Development Tests

A thermal balance development test may be required for a vehicle or subsystem if thermally induced structural distortions or functionality losses are critical for mission success. This test may also be constituted to validate the effectiveness of the thermal control subsystem.

The test model can be flight hardware or a thermally identical structure with addition of all the equipments on it, and thermally identical electrical, electronic, mechanical and pneumatic units.

If applicable, the test should be performed in a space simulation chamber, to expose the subsystem or satellite to the launch and on-orbit thermal vacuum environments.

5.5.6 Transportation and Handling Development Tests

Although the vibration and shock levels during transportation of the equipment will be less than the dynamic loads that the satellite will be subjected during launch, analyzing the loads to be originated by transportation and handling may be taken into consideration. A drop and impact test may be applied to the vehicle with appropriate casing and equipment.

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6. QUALIFICATION TESTS

6.1 General Qualification Test Requirements

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

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

The qualification test levels shall exceed the maximum predicted levels by a factor of safety which assures that, even with the worst combination of test tolerances, the flight levels shall not exceed the qualification test levels. [5]

6.1.1 Qualification Hardware

The qualification hardware to be tested should be produced from the same drawings and same materials, should undergo the same manufacturing process as the flight hardware to be launched to space.

Minor changes with respect to the flight models can be made to the QM used for qualification testing, assuring that they do not jeopardize the qualification objectives [5]. For instance, during a thermal cycle test, thermal sensors may be used for the data acquisition.

Wear-in test is required before the qualification test phase, to identify material defects or workmanship errors. This will provide a more smooth test process and more reliable test results.

6.1.2 Qualification Test Levels and Durations

The environment levels and durations of the loads to be applied during qualification tests should be more than the maximum expected levels and durations during acceptance testing, retesting, launch and orbital life. However, care must be taken not to exceed the design safety margins determined from the development tests. Table 6.1 shows the margin levels and duration which can be applied during qualification testing.

Table 6.1: Typical Qualification Test Level Margins and Durations [5]

Test	Units	Vehicle
Shock	6 dB above maximum expected environment, 3 times in both directions of 3 axes	1 activation of all shock- producing events; 2 additional activations of controlling events (5.2.3)
Acoustic*	6 dB above acceptance for 3 minutes	6 dB above acceptance for 2 minutes
Vibration*	6 dB ebove acceptance for 3 minutes, each of 3 axes	6 dB above acceptence for 2 minutes, each of 3 axes
Thermal Vacuum (Tables 5.2, 5.3)	10°C beyond acceptance temperatures for 6 cycles	10°C beyond acceptance temperatures for 13 cycles
Combined Thermal Vacuum and Thermal Cycle (Tables 5.2, 5.3)	10°C beyond acceptance temperatures for 25 thermal vacuum cycles and 53% thermal cycles	10°C beyond acceptance temperatures for 3 thermal vacuum cycles and 10 thermal cycles
Static Load	1.25 times the limit load for unmanned flight or 1.4 times the limit load for manned flight, for a duration close to actual flight loading times	Same as for unit, but only tested at subsystem level

6.1.3 Thermal Vacuum and Thermal Cycle Tests

Thermal temperature ranges and number of thermal cycles are demonstrated on Table 6.2 and 6.3. The number of cycles applied on the test specimen is generally unrelated to the cycles to be exposed during mission.

For the electrical and electronic units or electrical and electronic elements containing units may be exposed to multiple thermal vacuum and thermal cycles to detect the workmanship faults as soon as possible.

For the units that have not electrical and electronic components, only thermal vacuum cycle test is adequate to show the unit conformance.

Table 6.2: Temperature Ranges for Thermal Cycle (TC) and Thermal Vacuum (TV) Tests [5]

Required Testing	Unit	Vehicle	
	TC &TV	TC	TV
Acceptance (ΔT_A)	105°C[1]	≥ 50°C	note 3
Qualification (ΔT_0)	125°C[2]	≥ 70°C[2]	note 4

Notes:

1. Recommended, but reduced if impracticable or increased if necessary to encompass operational temperatures.
2. $\Delta T_0 = \Delta T_A + 20°C$.
3. Governed by the unit that first reaches its hot or cold acceptance temperature limit.
4. Like note 3, but for qualification temperature limit.

6.1.4 Acoustic and Vibration Qualification Tests

For the acoustic and vibration test levels and durations, accelerated testing approach may be used for pico- and nanosatellites. The testing is accelerated by replacing the actual test duration with a reduced duration at the qualification level. For this purpose, time reduction factor that is derived from a mathematical equation is used (Table 6.4).

For example, fatigue equivalent duration for a flight is 7.5 minutes, than the acceptance level testing duration is 4 times of the fatigue equivalent duration that equals to 30 minutes. When the qualification margin is 6 dB and the test tolerance on the spectrum is 1.5 dB, than the time reduction factor is 15. This means that, 30 minutes of acceptance level test can be accelerated to 2 minutes of testing at qualification level. For the qualification level vibration or acoustic test, the duration can be extended with a typical 1 minute duration, resulting in 3 minutes testing.

Table 6.3: Numbers of Cycles[1] for Thermal Cycle (TC) and Thermal Vacuum (TV) Tests [5]

Required Testing	Unit			Vehicle	
	Acceptance (Table XIII)		Qualification (Table X)	Acceptance (Table XII)	Qualification (Table VIII)
	N_A[3]	N_{AMAX}[4]	N_Q[5]	N_a	N_Q[5,6]
Both: TC[2]	8.5	17	53.5 25	4 1	10 3
TV	4	8			
Only TV	1	2	6	4	13
Only TC	12.5	25	78.5		

Notes:

1. Numbers of cycles correspond to temperature ranges in Table 5.2.
2. Tests may be conducted in vacuum to be integrated with TV.
3. For tailoring: N_A - $10(125/\Delta T_A)^{1,4}$ for TC only and for the sum of TC and TV when both conducted.
4. $N_{AMAX} = 2N_A$, but can be changed to allow for more or less retesting.
5. $N_Q = 4N_{AMAX}(\Delta T_A/\Delta T_0)^{1.4}$. assuming temperature cycling during mission or other service is insignificant; if significant, additional cycling shall be required using the same fatigue equivalence basis.
6. $N_{AMAX} = N_A$, assuming that vehicle-level acceptance retesting will not be conducted.

Table 6.4: Time Reduction Factors, Acoustic and Random Vibration Tests [5]

Margin M (dB)	Maximum Test Tolerance on Spectrum, T (dB)	Time Reduction Factor
6.0	±1.5	15
6.0	±3.0	12
4.5	±1.5	7
4.5	±3.0	4
3.0	±1.5	3
3.0	±3.0	1

Note: In general, the time reduction factor is $10^{M/5}[1 + (4/3)\sinh^2(T/M)]^{-1}$, where T is sum of the absolute value of the negative tolerance for the qualification test and the positive tolerance for the acceptance test.

6.2 Integrated System Qualification Tests

Table 6.5 shows the required tests for a satellite in pico and nano scales. Additional tests may be conducted to this table as required. Note that, in the qualification testing, on-board computer and the other processors should have the flight version software.

Table 6.5: Integrated System Qualification Test Baseline [5]

Test	Suggested Sequence	Space Vehicle
Inspection	1	R
Functional	2	R
EMC	3	R
Shock	4	R
Acoustic or Vibration	5	R
Thermal Cycle	6	O
Thermal Balance	7	R
Thermal Vacuum	8	R
Modal Survey	Any	R

6.2.1 Functional Tests, Integrated System Qualification

Purpose: The functional test validates the mechanical and electrical functionality and all of the software algorithms of the satellite according to the requirements specified.

Test Description: Mechanical devices or deployable items should be tested during functional testing. Fit and alignment checks may also be performed using proper gages.The performance of the electrical and electronic equipments like sensors, devices, coded algorithms and processors should also be observed and their performance should be stated. Functional tests at the room conditions may be performed before all the test sequences. This provides identification of the problem and its source more easily. Functional tests shall also be performed during thermal vacuum tests to validate functionality at the space environment.

6.2.2 Electromagnetic Comapatibility Test, Integrated System Qualification

Purpose: Electromagnetic compatibility test is performed to validate electromagnetic compatibility of the satellite and to assure that the satellite will withstand under the electromagnetic environment during launch and orbital life.

Test Description: The test shall demonstrate satisfactory electrical and electronic equipment operation in conjunction with the expected electromagnetic radiation from other subsystems or equipment, such as from other satellite elements and ground support equipment. The satellite should be tested in launch and in orbit configurations and in all operational modes.Potential electromagnetic interference from the test and ground support equipment should be determined before the test.

6.2.3 Shock Test, Integrated System Qualification

Purpose: Shock test aims to demonstrate that the satellite will survive in the shock environment during launch.

Test Description: For the test, a suitable adaptor should be designed and used tomount the satellite on the exciter table. Shocks are usually originated by the engine transients and payload fairing separations. Accelerometers shall be installed on the 3 orthogonal axes of the satellite to measure the dynamic responses due to the shock. The levels and durations of the shock test is driven by the launch vehicle characteristics for a satellite in pico and nano scale.

6.2.4 Acoustic Test, Integrated System Qualification

Purpose: The acoustic test demonstrates the ability of the satellite to endure acoustic acceptance testing and meet requirements during and after exposure to the extreme expected acoustic environment in flight.

Test Description: The satellite should be mounted on a flight-type support structure during the test. The adaptor designed for the vibration and shock testing may be used for this purpose. Control microphones shall be placed at a minimum of 4 locations, but no closer than 0.5 meter to both the test specimen and chamber wall. The test levels and durations should be determined using section 6.1.4 and table 6.4.

6.2.5 Vibration Test, Integrated System Qualification

Purpose: The vibration test is carried out to confirm the ability of the satellite to withstand the acceptance level loads and the extreme dynamic environment that will occur during the flight.

For compact pico and nanosatellites, the vibration test may be applied instead of the acoustic testing, because they are exited more efficiently by interface vibration rather than acoustic waves.

Test Description: The satellite shall be exited in each 3 orthogonal axis and one axis must be parallel to the thrust direction of the launch vehicle. The acceleration sensors shall be used to measure the responses at all 3 axes.The spectrum to be applied on the satellite can be produced using section 6.1.4. The spectrum may be modified to avoid the undesired responses, however, it shouldn't be under the minimum spectrum for a satellite. For the test margins and tolerances Table 6.4 shall be utilized. The qualification spectrum should be applied for 2 minutes as a typical test for each axis.

6.2.6 Thermal Cycle Test, Integrated System Qualification

Purpose: The aim of the thermal cycle test is to validate that the satellite can withstand the flight vehicle thermal cycle acceptance testing.

Test Description: The thermal cycle test is carried out in a thermal chamber at the ambient temperature. Before the start of the test, mechanical and electrical functionality test can be performed to assure the readiness of the satellite. The system shall be observed during the entire test.

The test starts with ambient temperature, then the next step is cycling to one extreme (max or min) temperature and stabilization at that level, then to the other extreme (min or max) temperature and stabilization at that level and finally returning to ambient temperature. This constitutes one thermal cycle.

The satellite may be operational during the entire test as desired or it may be turned off in some cycles. It recommended that, during the cold cycling the system is turned off to assist the stabilization process.

Operational cycles (for the satellite) can be selected according to the requirements. During the last cold and hot cycles the satellite shall be tested for the functionality.

During the test, thermal sensors may be placed on the system, especially on the critical parts like batteries, electronics, processors, motors, etc.

The minimum temperature range that the satellite will expose should be 70°C. For example after approval by analysis, a pico or nanosatellite can be tested between the temperatures of -35°C and +35°C. Within the range of 70°C, the number of thermal cycles can be selected as 10. The rate of change of the temperature must be as rapid as possible, of course it is determined by the constraints of the test equipment. The temperature range and number of cycles shall be selected from Table 6.2.

Thermal cycling tests can be combined with the thermal vacuum tests and reduced-cycle thermal vacuum test may be conducted (Table 6.3).

6.2.7 Thermal Balance Test, Integrated System Qualification

Purpose: Thermal balance test is carried out to verify the validity of the thermal design and the thermal model and to verify the effectiveness of thermal subsystem by proving required thermal data. Test may also be combined with the thermal vacuum test.

Test Description: The aim of the test is to simulate the vacuum environment that the satellite will be exposed during its orbital life. This simulation may include specific and extreme conditions like different solar angles, sun and eclipse conditions, and heat dissipation anomalies due to the bus voltage variations.

Thermal sensors should be used to observe the critical equipments and subsystems. If any, heater and cooler should be also electrically tested to demonstrate their efficiency.

If a thermal vacuum balance test is to be performed, then the test system should provide a pressure of less than 0.0133 Pa for an on orbit system. In the test thermal environment of the satellite may be provided by the following sources:

a. Absorbed Flux: The absorbed solar, earth and planetary radiation may be simulated by heater panel or infrared lamps, or by electrical resistance heaters attached to the surface of the satellite.

b. Incident Flux: The intensity, spectral content and angle of the solar, earth and planetary radiation may be simulated.

c. Equivalent Radiation Sink Temperature: The equivalent radiation sink temperature may be simulated using infrared lamps and calorimeters.

d. Combination: Thermal environment may be achieved by the combination of a, b and c.

Test durations and conditions depend on the thermal design, configuration of the subsystems and other elements and mission details. To decide on the durations and levels, below topics must be analyzed.

a. Maximum external flux absorption and maximum internal heat dissipation.

b. Minimum external flux absorption and minimum internal heat dissipation**.**

6.2.8 Thermal Vacuum Test, Integrated System Qualification

Purpose: Thermal vacuum test is performed to demonstrate the ability of the satellite to verify the qualification vacuum conditions plus extreme temperatures during launch and orbital life.

Test Description: The satellite should be placed in the thermal vacuum chamber and a functionality test should be performed on the test specimen to prove its readiness to the test process.

Thermal cycle begins with the satellite at ambient temperature; the temperature is then raised to maximum level and stabilized. After the hot soak, the temperature level is reduced to lowest temperature (minimum level) and stabilized. Then the satellite is returned to the ambient condition and this concludes one thermal cycle. Note that, the cycle may be start with cold case and may end with the hot case as required. The functionality check shall be performed in first and the last thermal cycle at the both maximum and minimum temperatures.

The test can be modified or extended with further testing, by performing operational sequences coordinated with expected environment.

If possible; witness plates, quartz crystal microbalances or other equipments shall be used to measure the outgassing of the materials during the test.

During the test external and internal environment of the satellite should be carefully observed. During the hot or cold case, if any unit of an equipment reaches to the qualification thermal testing limit, it means the temperature limit is reached. The test data should be carefully analyzed during the test to avoid unnecessary excessive thermal loading of equipments for qualification. The pressure must be under 0.0133 Pa during the test. For the test durations (cycles) and levels, Table 6.3 provides necessary approach.

6.2.9 Modal Survey Test, Integrated System Qualification

Purpose: The modal survey test is performed to develop a dynamic model of the satellite or to provide data to verify an analytical model. Once analytical model is established according to test results, using the expected loading during flight, structural margins and adequacy of the structural static loading conditions can be determined.

Test Description: The test specimen shall consist of flight-quality structure with assembled units, payloads, and other subsystems. If any, the liquids shall also be contained within the system. In addition, wire harness may be installed as desired. The data acquired from the modal survey test should define the resonant frequencies and associated mode shapes and damping values for all modes in the frequency range of interest. [5] Additionally, the primary modes should be determined for each of the 3 axes. The load levels to excite the system during the test are generally low compared to the estimated actual flight levels.

If an analytical model is to be verified according the test data, frequencies of the analytical model have to be within ± 3 percent range of the test frequencies.

6.3 Subsystem Qualification Test

Subsystem qualification tests shall be conducted on subsystems to verify their design, to qualify the subsystems that are subjected to environmental acceptance tests. These tests may also be performed for the units, on which subsystem level tests will give more realistic results.

For specific units like wiring, qualification may be more efficiently achieved at subsystem level compared to unit level testing.

6.3.1 Structural Static Load Test, Subsystem Qualification

Purpose: The structural static load test is constituted to determine whether the strength and stiffness requirements of the subsystem are met or not. Requirements are modified with a desired qualification margin considering critical environments that the satellite will undergo during its on-orbit life.

Test Description: Static loads representing the design yield load and the design ultimate load shall be applied to the structure and measurements of the strain and deformation shall be recorded. Strain and deformation shall be measured before loading, after removal of the yield loads, and at several intermediate levels up to yield load for post-test diagnostic purposes. The test conditions shall encompass the extreme predicted combined effects of acceleration, vibration, pressure, preloads, and temperature. These effects can be simulated in the test conditions as long as the failure modes are covered and the design margins are enveloped by the test. For example, temperature effects, such as material strength degradation and additive thermal stresses, can often be accounted for by increasing mechanical loads.

Analysis of flight profiles shall be used for subsequent design modification effort, and to provide data for use in any weight reduction programs. Failure at design yield load means material gross yielding or deflections, which degrade mission performance. Failure at design ultimate load means rupture or collapse.

Unless otherwise specified, the load level for the ultimate static load test is 1.25 times limit load. Loads shall be applied as closely as practical to actual flight loading times, with a dwell time not longer than necessary to record test data such as stress, strain, deformation, and temperature.

6.3.2 Vibration Test, Subsystem Qualification Test

Purpose: Same as 6.2.5.

Test Description: Same as 6.2.5.

6.3.3 Acoustic Test, Subsystem Qualification Test

Purpose: Same as 6.2.4.

Test Description: Same as 6.2.4.

6.3.4 Thermal Vacuum Test, Subsystem Qualification Test

Purpose: Same as 6.2.8.

Test Description: Same as 6.2.8.

6.4 Unit Qualification Test

Unit qualification testing concept can be demonstrated in Table 6.6. This test should normally be performed on the items at the unit level. However, if the test is not applicable at unit level because of the properties of the equipment, then the test may be conducted at higher equipment levels such as subsystem or even system (satellite) level. If applicable, in the end of all qualification tests, the unit should be disassembled and examined. For the units that belong to multiple categories in Table 6.6, the tests in each category shall be applied

6.4.1 Functional Test, Unit Qualification

Purpose: The aim of the functional test is to verify the performance of the mechanical, optical and electrical units according the operational requirements.

Test Description: As the specifications and sequences expected in flight is not the same for all units, test requirement parameters are different among units. The electrical tests shall include investigation of the expected properties of voltage, impedance, frequencies, pulses and waveforms. The performance criteria also consist of electrical continuity, stability, response time, alignment, pressure, leakage. Performance parameters of the unit should be monitored during the entire test process and the anomalies should be stated.

6.4.2 Thermal Cycle Test, Unit Qualification

Purpose: The thermal cycle test is conducted to verify that the electrical and electronic units will endure and operate under the qualification temperature range.

Test Description: The test control temperature should be monitored from a suitable location. The mounting point of the unit to the base plate is a representative location for this purpose. At the extreme temperatures, namely hot and cold cases, the temperature should be stabilized according to the stabilization requirements. Figure 6.1 represents a typical thermal cycle scenario for unit qualification level.

The test starts at the ambient temperature, then the temperature is raised to the maximum temperature and waiting for the stabilization. Following that, the temperature is decreased to the minimum temperature level and waiting stabilization. After the stabilization at the cold case, the temperature is raised again to the ambient condition. This process forms a thermal cycle.

During the hot case, the unit shall be turned off and then hot started. Similarly at the cold case, the unit shall be powered off to provide stabilization at low temperature level. After the stabilization at the cold case, the unit shall be cold started.

For the test, normally ambient pressure is used. However, the test may also be performed under vacuum conditions. For the cold case at low temperatures, the condensation must be taken into the consideration before the test.

During the transitions from hot to cold or cold to hot, the temperature change rate should be 3˚C to 5˚C per minute on average, and it shouldn’t be slower than 1˚C/min.

Table 5.3 represents the required number of thermal cycles for qualification. Thermal soak durations for hot and cold cases should be at least 6 hours for the first and last cycles, and at least1 hour for intermediate cycles.

6.4.3 Thermal Vacuum Test, Unit Qualification

Purpose: The thermal vacuum test is conducted to demonstrate the ability unit to perform under the qualification thermal vacuum environment and to withstand the acceptance and flight thermal vacuum conditions.

Test Description: The unit should be mounted on a base plate in a temperature controlled vacuum chamber. It is recommended that the base plate is temperature controlled to keep the unit in the desired temperature range. Unit surface finishes should also thermally identical with the flight configuration. Before the beginning of the test, the unit should be functionality checked by observing the performance parameters. For the test implementation, test conditions and test duration, the information provided in thermal cycle unit qualification test section shall be utilized.

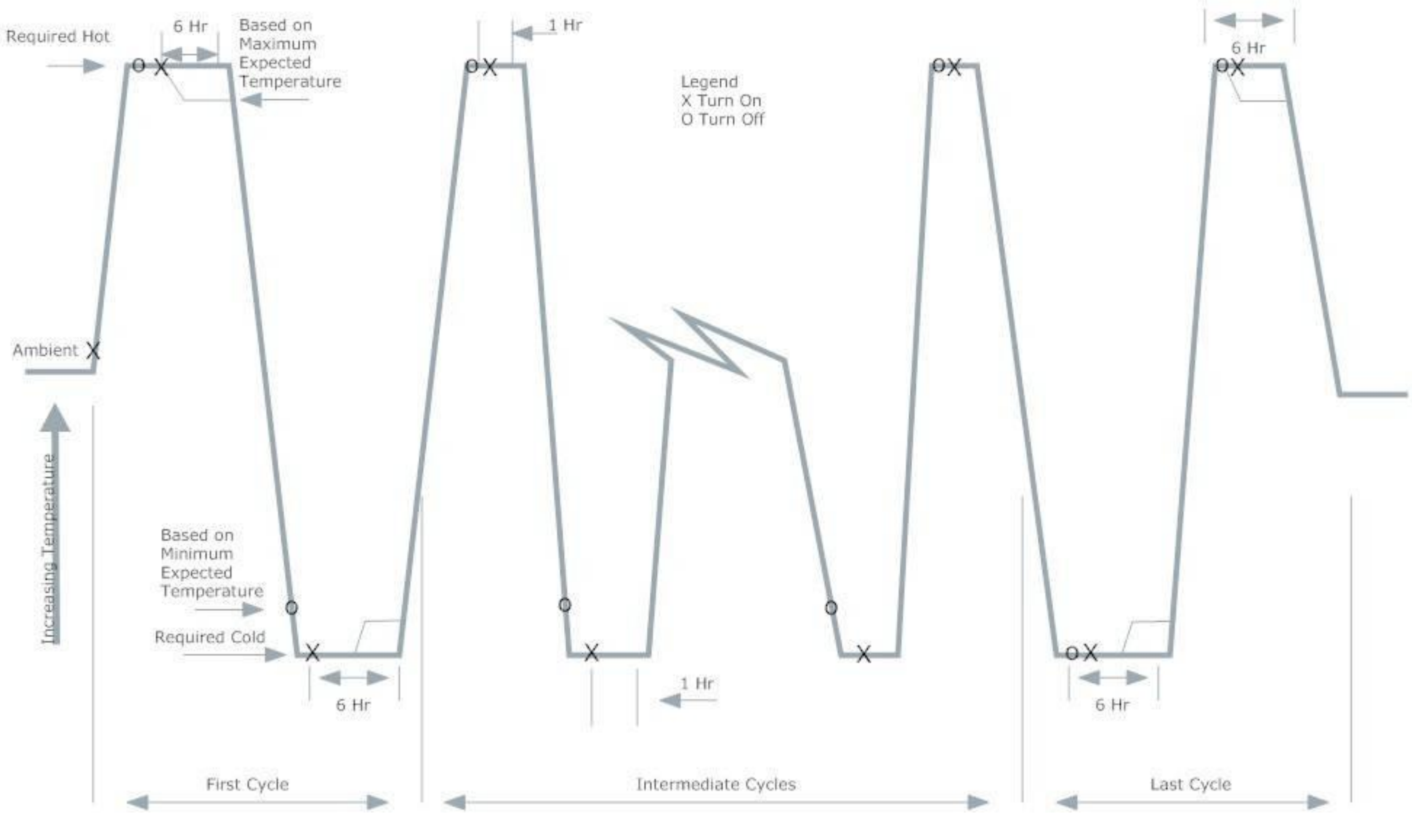

Figure 6.1: Typical Component Thermal Cycle Profile

6.4.4 Vibration Test, Unit Qualification

Purpose: The vibration test is conducted to show that the unit is able to endure maximum duration of the acceptance testing vibration levels and maximum expected dynamical loads during launch.

Test Description: The unit should be mounted to a fixture through the normal mounting points. The fixture used for the qualification testing should be used also for the acceptance testing for consistency. The parts that constitute the unit should be in flight configuration. The unit shall be excited in each of 3 orthogonal axes and responses at all directions should be measured and recorded. Test levels and durations should be applied as explained in the section

Throughout the test, all electrical and electronic equipments may be powered on and performance parameters of these equipments may be observed to detect failures.

6.4.5 Acoustic Test, Unit Qualification

Purpose: The unit qualification acoustic test aims to validate the ability of the test units that have large surface areas (and whose dynamic responses are originated dominantly form acoustic excitations) to endure the maximum level acoustic acceptance level testing and the maximum expected flight dynamics. For the compact units the acoustic test is optional and may be neglected. Contrarily, for the units that have large surface areas, the acoustic testing is more critical to demonstrate the endurance of the unit and vibration test may be neglected.

Test Description: The satellite should be mounted on a flight-type support structure during the test. The adaptor designed for the vibration and shock testing may be used for this purpose. Control microphones shall be placed at a minimum of 4 locations, but no closer than 0.5 meter to both the test specimen and chamber wall. The test levels and durations should be determined using section 6.1.4.

6.4.6 Shock Test, Unit Qualification

Purpose: The objective of the unit qualification shock test is to validate the ability of the unit to perform and function properly after exposing to extreme expected shock during the launch phase.

Test Description: The unit should be mounted to a fixture through the normal mounting points. The fixture used for the qualification testing should be used also for the acceptance testing for consistency. The parts that constitute the unit should be in flight configuration. Mounting the unit on a dynamically similar platform (as in the satellite) will give more realistic results than mounting on a rigid body.

The shock spectrum applied on each of 3 axes should be at least qualification level.Throughout the test, all electrical and electronic equipments may be powered on and performance parameters of these equipments may be observed to detect failures.

6.4.7 Acceleration Test, Unit Qualification

Purpose: The aim of the unit qualification acceleration test is to demonstrate the ability of the unit to endure or operate in qualification level acceleration.

Test Description: The test unit should be mounted to a fixture and be subjected to acceleration in proper directions related with the launch vehicle design and location on the rocket.

For each direction, acceleration level should be minimum 1.25 times the maximum expected acceleration level during launch phase, and the test duration should be at least 5 minutes for each direction.

6.4.8 Life Test, Unit Qualification

Purpose: The life test is performed for the units that have wearout, drift or fatigue-type failure mode, or performance degradation like batteries. The test aims to shows that the unit is capable to perform at maximum duration including ground tests and orbit mission life.

Test Description: During the test more than one unit shall be tested under service conditions. The test environment in life test includes ambient, thermal and thermal vacuum to investigate wearout and drift failures; pressure, thermal and vibration to identify fatigue-type failures.

The test also identifies the maximum operational duration or cycle number of the tested unit.For the tests to be performed in thermal vacuum conditions, the pressure should be no more than 0.0133 Pa. For the load levels extreme environment conditions shall be used as in other qualification tests. The duration of the test should be at least two times the expected service life of the unit including ground operations.

6.4.9 Electromagnetic Compatibility (EMC) TEST, Unit Qualification

Purpose: The electromagnetic compatibility test is conducted to demonstrate that the electromagnetic interference characteristics of the unit does not affect the functions of the unit, does not result in malfunction of the unit and the other neighboring units.

Test Description: The test should be carried out in accordance with the military standards MIL-STD-1541.

7. ACCEPTANCE TESTS

7.1 General Acceptance Requirements

Acceptance tests are applied on a system, subsystem or unit level to demostrate the validity of the item to perform in launch and space environment. These tests are also intended to determine the failures originated from the workmanship errors and structural defects. The test level shall not cause unrealistic failures by exceeding design safety margins.

The general requirements for thermal ranges, thermal cycles, acoustic and vibration tests are given in the subsequent clauses.

7.1.1 Temperature Ranges and Thermal Cycles

The temperature range should be between -35˚C to 70˚C for unit level tests. The number of cycles should be determined using Table 6.3. If in qualificaiton tests; 105˚C plus 10˚C hot and cold extension did cause unrealistic failures, then the range may be narrowed and the number of thermal cycles according to the new range should be rearranged.

During the system level thermal vacuum tests, at least one unit shall reach its acceptance hot temperature limit at hot soaks and at least one unit shall reach its acceptance cold temperature limit at cold soaks.

7.1.2 Acoustic Environment

The acoustic test at acceptance levels shall be conducted according to the maximum expected environment. The test level also shouldn’t be less than the spectrum in Figure 7.1. The minimum duration for the acceptance acoustic tests are 1 minute.

7.1.3 Vibration Environment

The random vibration test at acceptance levels shall be conducted according to the maximum expected environment. The test level also shouldn't be less than the spectrum in Figure 7.2. The minimum duration for the acceptance acoustic tests are 1 minute for each of 3 orthogonal axes. The sinusoidal vibration test level shall also be determined according to the maximum expected sinusoidal vibration.

Table 7.1: Acceptance Test Levels and Durations

Test	Units	Vehicles
Shock	Maximum expected spectrum, achieved once in both directions of 3 axes. Discretionary if spectrum is low	1 activation of significant shock-producing events
Acoustic	Same as for vehicles.	Envelope of maximum expected spectrum and minimum spectrum, 1 minute.
Vibration	Envelope of maximum expected spectrum and minimum spectrum, 1 minute in each of 3 axes.	Same as for units.
Thermal Vacuum	1 cycle, -35 to +70°C . Vacuum at 13.3 millipascals (10^{-4} Torr).	4 cycles, -35 to +70°C . Same pressure as for units.
Thermal Cycle	12.5 cycles, -35 to +70°C.	Min. Temp. Range 50°C, 4 Cyle
Combined Thermal Vacuum and Cycle	8.5 thermal cycles and 4 thermal vacuum cycles, -35 to +70°C.	Min. Temp. Range 50°C, 4 Cyle for both

Curve Values			
1/3-Octave-Band Center Frequency (Hz)	Minimum Sound Pressure Level (dB)	1/3- Octave-Band Center Frequency (Hz)	Minimum Sound Pressure Level (dB)
31	121	630	125
40	122	800	124
50	123	1000	123
63	124	1250	122
80	125	1600	121
100	125.7	2000	120
125	126.5	2500	119
160	126.7	3150	118
200	127	4000	117
250	127	5000	116
315	126.7	6300	115
400	126.5	8000	114
500	125.7	10000	113
		Overall	138

Figure 7.1: Minimum Acoustic Spectrum, System and Unit Acceptance [5]

7.2 Integrated System Acceptance Tests

Integrated System tests shall include the frame in Table 7.2. If the vehicle is controlled by on-board data processing, the flight version of the software to be used shall be loaded for the acceptance tests.

Frequency (Hz)	Minimum PSD (g^2/Hz)
20	0.0053
20 to 150	+ 3 dB per octave slope
150 to 600	0.04
600 to 2000	-6 dB per octave slope
2000	0.0036
The overall acceleration level is 6.1 grms	

Figure 7.2: Minimum Random Vibration Spectrum, Unit Acceptance Tests [5]

7.2.1 Functional Tests, Integrated System Acceptance

Purpose: The functional test validates the mechanical and electrical functionality and all of the software algorithms of the satellite according to the requirements specified.

Test Description: The performance of the electrical and electronic equipments like sensors, devices, coded algorithms and processors should also be observed and their performance should be stated.Functional tests at the room conditions may be performed before and after all the test sequences. This provides identification of the problem and its source more easily.Functional tests shall also be performed during thermal vacuum tests to validate functionality at the space environment.

Frequency (Hz)	Minimum PSD (g^2/Hz)
20	0.002
20 to 100	+ 3 dB per octave slope
100 to 1000	0.01
1000 to 2000	-6 dB per octave slope
2000	0.0025
The overall acceleration level is 3.8 grms	

Figure 7.3: Minimum Random Vibration Spectrum, System Acceptance Tests [5]

Table 7.2: System Acceptance Test Framework [5]

TEST	SUGGESTED SEQUENCE	SPACE VECHILCE
Inspection	1	R
Functional	2	R
EMC	3	O
Shock	4	O
Acoustic or Vibration	5	R
Thermal Cycle	6	O
Thermal Vacuum	7	R
Storage	any	O
R: Required, O: Optional		

7.2.2 Electromagnetic Compatibility Test, Integrated System Acceptance

For accpetance level, limited EMC testing shall be performored to demostrate that the major changes have not occurred in the system. These tests shall include power bus ripple and peak transients and critical circuit parameters monitoring.

7.2.3 Shock Test, Integrated System Acceptance

Purpose: The shock tests are conducted to apply the shock environment on the item to fly. It also aims to determine the workmanship errors and structural weaknesses.

Test Description: Same as 6.2.3.

7.2.4 Acoustic Test, Integrated System Acceptance

Purpose: The acoustic tests are conducted to simulate the acoustic environment on the item to fly. It also aims to determine the workmanship errors and structural weaknesses that are not detected during static tests.

Test Description: Same as 6.2.4. Test levels and durations shall be as defined in7.1.2.

7.2.5 Vibration Test, Integrated System Acceptance

Purpose: The vibration tests are conducted to simulate the sinusoidal and random vibration environment on the item to fly. It also aims to determine the workmanship errors and structural weaknesses that are not detected during static tests. If the vehicle have a compact design, then vibration tests may be carried out in lieu of the acousting testing.

Test Description: Same as 6.2.5. Test levels and durations of random vibration test shall be as defined in 7.1.3.

7.2.6 Thermal Cycle Test, Integrated System Acceptance

Purpose: The aim of the test is to detect the material, process and workmanship errors.

Test Description: Same as 6.2.6. The minimum temperature range to be applied on the vehicle should be 50˚C. The rate of change of the temperature during cycling shall be as rapid as possible. Minimum number of thermal cycles is 4 according to tables 6.2 and 6.3.

7.2.7 Thermal Vacuum Test, Integrated System Acceptance

Purpose: Thermal vacuum tests aim to detect the material, process and workmanship errors that would appear in thermal vacuum conditions.

Test Description: Same as 6.2.8. During the test, the pressure shall be maintained under 1.33x10-2 Pa (10-4 Torr). The number of cycles shall be at least 4 with hot and cold soaks. First and last cycles shall include minimum 8 hours of hot and cold soaks. For intermediate cycles the soak duration is minimum 4 hours.

7.3 Subsystem Acceptance Tests

For a satellite, the subsystem level acceptance tests are optional, except for the pressurized subsystem s such as thermal control subsystem. The application of the test may be useful to detect the failures and errors, however, time and cost trade-offs should be carried out to determine necessity of the test.

For a pressurized system, proof load and proof pressure tests may be performed. For these and the other tests to be considered the test levels are based on the system test requirements.

7.4 Unit Acceptance Tests

The unit acceptance tests may be consist of inspection, functional, shock, vibration, acoustic, thermal cycle, themal vacuum, wear-in and EMC tests. The necessity of the tests should be considered according to the requirements of the unit element. These tests shall be generally performed in uit level, where some of them may be carrried out at the higher level of assembly, such as sollar arrays and integrated circuits.

7.4.1 Functional Test, Unit Acceptance

Purpose: The aim of the functional test is to verify the mechanical, optical or electrical performance of the units under given operational environments.

Test Description: Same as 6.4.1.

7.4.2 Thermal Cycle Test, Unit Acceptance

Purpose: The thermal cycle test aims to determine material and workmanship errors by subjecting the unit to the thermal cycles.

Test Description: The test should be carried out in normal atmospheric conditions. If reduced pressure condition is desired, then the test should be combined with thermal vacuum tests.

For thermal cycle test, the minimum number of cycles shall be 12.5, and the last two cycles shall be failure free. The temperature soak duration for hpt and cold shall be minimum 3 hours for the first and last cycle and minimum 1 hours for intermediate cycles. During the test, the unit may be turned off to stabilize the temperature. The change of the temperature between hot and cold soaks should be minimum 1˚C per minute, and it should be as high as possible.

7.4.3 Thermal Vacuum Test, Unit Acceptance

Purpose: The thermal cycle test aims to determine material and workmanship errors by subjecting the unit to the thermal vacuum environment.

Test Description: The test shall be performed under high vacuum environment, namely 0.0133 Pa (10^{-4} Torr). Units that are prove to operate succesfully under vacuum environment may be subjected to thermal cycle tests under ambient pressure.

The hot and cold temperatures shall be the acceptance temperature limits as given in Table 7.1. Electrical and electronic units shall be 4 thermal vacuum cyclesThe hot and cold soak duration shall be at least 3 hours for the first and last cycle and 1hour for intermediate cycles. For the other units, one cycle with 3 hours hot and cold soak durations are enough.

7.4.4 Vibration Test, Unit Acceptance

Purpose: The vibration test inspects the material and workmanhip errors by subjecting the unit to a vibration environment.

Test Description: Same as 6.4.4. The vibration environment shall be as defined in Figure 7.2.

7.4.5 Acoustic Test, Unit Acceptance

Purpose: The acoustic test inspects the material and workmanhip errors by subjecting the unit to a vibration environment.

Test Description: Same as 6.4.5. The acoustic environemnt shall be as defined in Figure 7.1.

7.4.6 Shock Test, Unit Acceptance

Purpose: The shock test detects the material and workmanship errors by subjecting the item to high level shock environment.

Test Description: During the test the unit doesn't have to be electrically operational. It is really important to perform functional checks before and after all shock tests to reveal the problem due to the shock environment.

The test shall be applied in each 3 orthogonal axes and at least once per axis.

7.4.7 Wear-in Test, Unit Acceptance

Purpose: The wear-in test aims to detect the material and workmanship errors that arise early in unit life and that effect the mission time of the unit.

Test Description: The pressure and temperature should be at ambient levels. The duration should the 5 to 15 percent of the expected unit life.

7.4.8 EMC Test, Unit Acceptance

EMC test shall only be performed on the units to observe their characteristic change in critical units. In other cases, EMC test is restricted and qualification level EMC tests are considered satisfying.

8. TAILORED APPLICATION: INTEGRATION AND TESTING OF ITUpSAT1

8.1 Model and Testing Philosophy

Integration and testing is the most serious step for the verification of the strength, functionality and performance of the satellite. Test procedures are determined by the environmental conditions. Procedures are classified as vibration tests and thermal vacuum test. Vibration tests are performed to simulate the static and dynamic loads during the launch of the satellite and its load levels are provided by the launch service provider. Thermal vacuum tests are performed to simulate the environmental conditions (pressure and temperature level) of the satellite during its orbital life.

The testing procedure of ITUpSAT1 is divided into two parts: Engineering Qualification Model Tests and Flight Model Tests. In Flight Model, the satellite contains the final configuration and integration of the hardware and software. Differently from the engineering model, the software can be updated before launch. In Engineering QualificationModel, the satellite consists of the components that won’t be launched to space, but it is identical to the flight model as mentioned.

For the engineering and flight model of ITUpSAT1, there will be acceptance and qualification level tests.

In the acceptance testing, the aim is to show conformance and to identify manufacturing defects, workman-ship errors, the start of failures and other performance anomalies, which are not readily detectable by normal inspection techniques. Acceptance testing is carried out under environmental load levels, which don’t exceed the expected levels during mission.

In qualification testing, the aim is to guarantee that the spacecraft performs satisfactorily under the environmental conditions during flight. Moreover, a factor of safety shall be used to assure that the environmental loads during flight will not exceed the qualification test levels.

Firstly, vibration tests will be carried out on the engineering model of ITUpSAT1 on the qualification levels to prove the survival of the satellite during the launch environment. If this model passes the tests applied, then tests will be carried out on the flight model with the final configuration of hardware and software used. However, only the acceptance levels will be applied on the flight model of the satellite to prevent the unnecessary loading caused by the qualification testing.

Thermal vacuum cycling test and bake-out procedure are to be performed according to the information provided by the launch service provider and according to the examination of the orbit characteristics of ITUpSAT1. On the EQM, only the thermal vacuum cycling test is performed. On the other hand, on the FM, the full procedure for thermal vacuum tests is applied.

In this section, firstly, the required test procedures and then the report are presented.

Table 8.1: Tests to be performed on EQM and FM

	Vibration Testing (Acceptance Level)	**Vibration Testing (Qualification Level)**	**Thermal Bake-out**	**Thermal Vacuum Cycling**
EQM		X		X
FM	X		X	X

8.2 Vehicle Acceleration Test (Quasi-Static Loads)

Quasi-static acceleration test is performed to verify the design of the satellite under the acceleration levels produced by the launcher. The test represents an equivalent static acceleration with the combination of spacecraft launch, steady-state longitudinal and lateral accelerations and low frequency transient loads. There are several test methods for the quasi-static acceleration test to simulate the loads. We have used sine-burst test to apply required load levels. Sine-burst test is a simple method to apply a quasi-static load using a vibration shaker and it is more economical than acceleration (centrifuge) or static tests. The method on a shaker simulates a uniform and homogeneous static load on the whole satellite and very closely simulates the actual flight loads.

As the test procedure demands to apply a quasi-static load to the satellite, the test frequency must be lower than the first natural frequency of the satellite. As a common practice, the applied test frequency must be below the one-third of the satellite natural frequency to avoid resonance.

The acceleration levels for the PSLV are represented on the table below. In the test, the acceleration loads are applied using a load factor of 1.25.

Table 8.2: Vehicle Acceleration Levels

Vehicle Quasi-Static Acceleration levels	
Longitudinal Axis (X)	+11 g
Lateral Axis (Y,Z)	+/- 6 g
Load Factor	1.25

8.3 Vibration Testing

Although a satellite is also affected by the structural loading during the life on its orbit, the most serious forces that act on the satellite appear at the period of launching phase. Therefore, the vibration test scenario is formed using the vibration levels of India's Polar Satellite Launch Vehicle, which are given in PSLV manual. During the launch and trajectory of PSLV, ITUpSAT1 will be subjected to various dynamic loads. These estimated loads form the acceptance and qualification load levels for vibration testing.

Through the test process, ITUpSAT1 is located in the Single Picosatellite Launcher (SPL). To connect SPL and ITUpSAT1 to the shake table, an aluminum adaptor designed for SPL is used. SPL is mounted on the adaptor with six M4 screws and the adaptor is mounted to the shake table with four M10 screws.

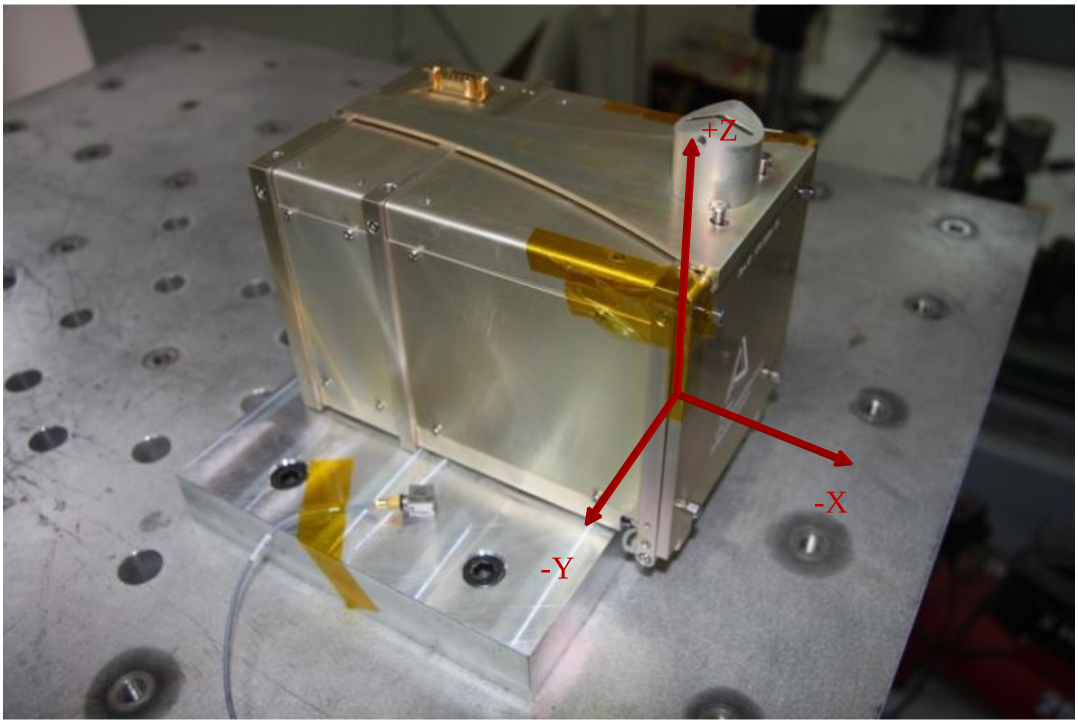

Figure 8.1: SPL is connected to the shake table via a compatible aluminum adaptor

In addition, it is important to determine the coordinate axes and define the load levels according to these axes. Figure 1 and 3 illustrates the reference axes used load direction definitions. Using the axes of ITUpSAT1 in Figure 4, we can describe the position of the satellite inside the SPL. ITUpSAT1 is located in the launch adaptor, so that their +Z directions coincide. The +Y direction (the side of antenna mechanism) will coincide with the +X of direction of SPL. We use accelerometers to observe the response of the ITUpSAT1 to applied load levels. For this purpose, we use a measurement sensor, a 3-axis accelerometer, located on the SPL. This measurement sensor is placed near the door of SPL, where the most severe loads are expected and provide us information about the responses of all axes. Another accelerometer is placed on the aluminum adaptor as a reference sensor, to provide feedback about the vibration levels acting on the adaptor plate. Finally, a third sensor, which is a one-axis accelerometer, is used to check the accuracy of the 3-axis sensor.

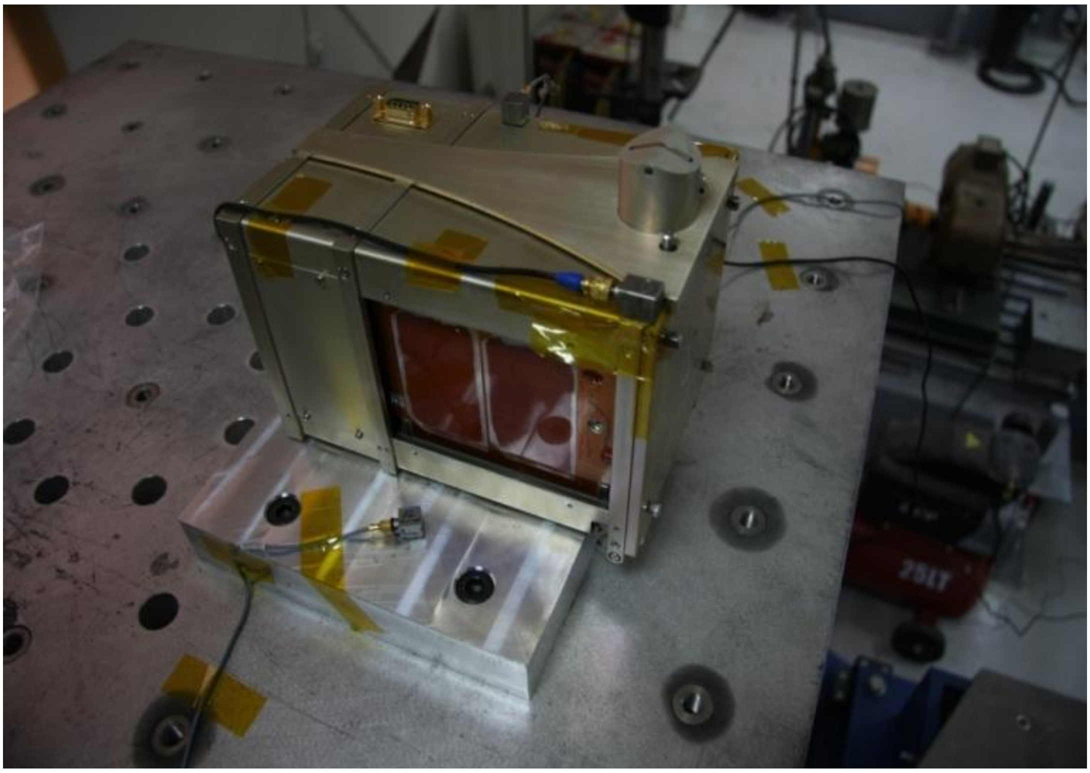

Figure 8.2: The accelerometers used for data acquisition during vibration tests

Vibration testing contains Survey Resonance, Sine Vibration and Random Vibration tests. The acceptance and qualification levels of the tests are given in the following subsections. Before and after all of the vibration test sequences, functional tests and physical attributes check is performed on the hardware components of the satellite. These are essential for the testing process, since it provides us information about a failure and its source.

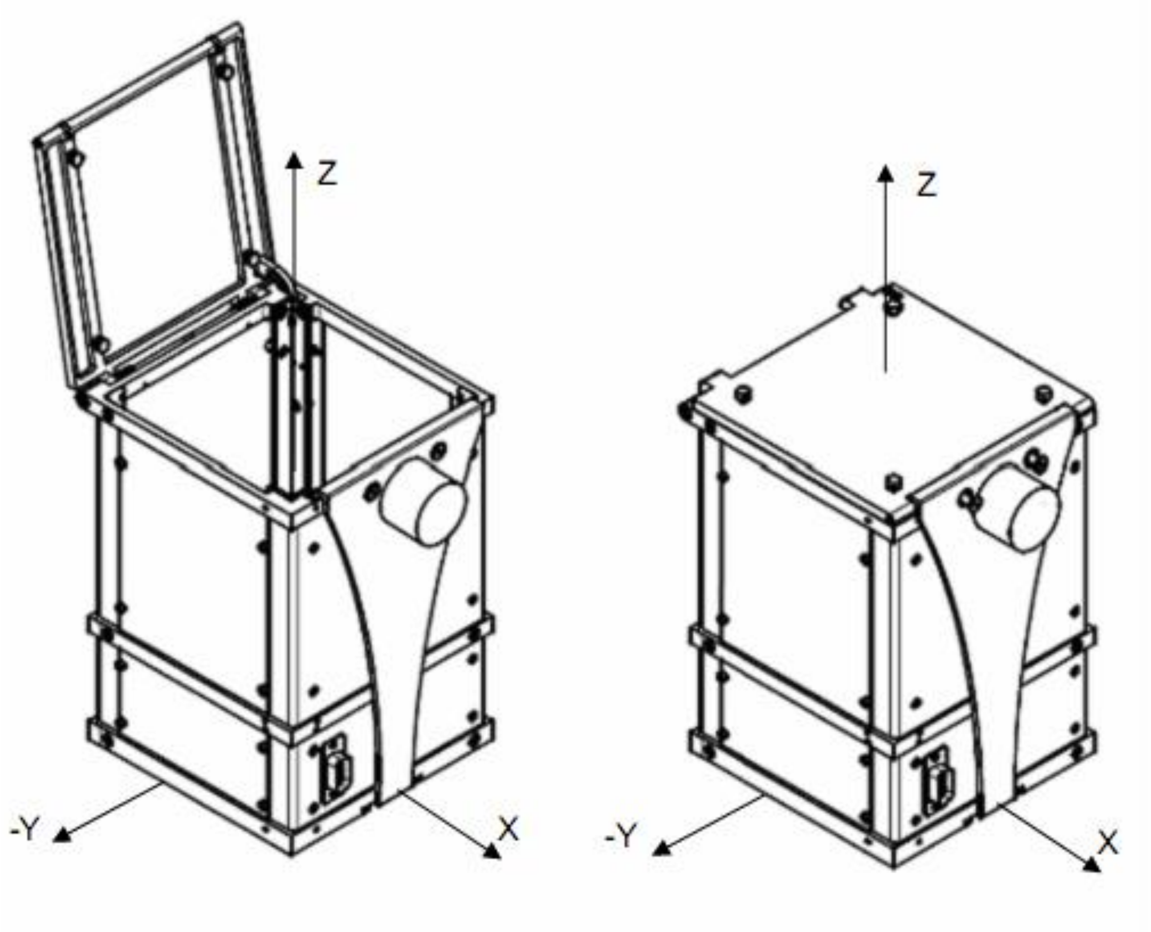

Figure 8.3: Axes of Single Picosatellite Launcher (SPL)

8.4 Shock Test

Shock Tests are essential, since it demonstrates the strength and durability of the satellite against the serious vibration load levels during boost and heat shield separation. Therefore, before and after the shock test of the SPL+ITUpSAT1 system, functional and structural checking must be strictly applied to the satellite to check its conformance.

The acceptance and qualification test levels for the shock test are: Acceptance level is 50 [g], for a duration of 10 [ms] (Half Sine Pulse).

8.5 Thermal Vacuum Testing

8.5.1 Thermal Vacuum Bake-out

Thermal vacuum bake-out is essential since it provides a reduction in the outgassing rates of organic materials by accelerating the molecular outgassing. Thermal vacuum bakeouts are carried out within a controlled vacuum environment. It is important that the temperature levels during the process should not exceed the allowable temperature levels of materials. The parts that will be baked out are exposing materials, subassemblies and higher level assemblies. These parts must be subjected to thermal vacuum bake-outs until the outgassing rate has become stable within a low level.

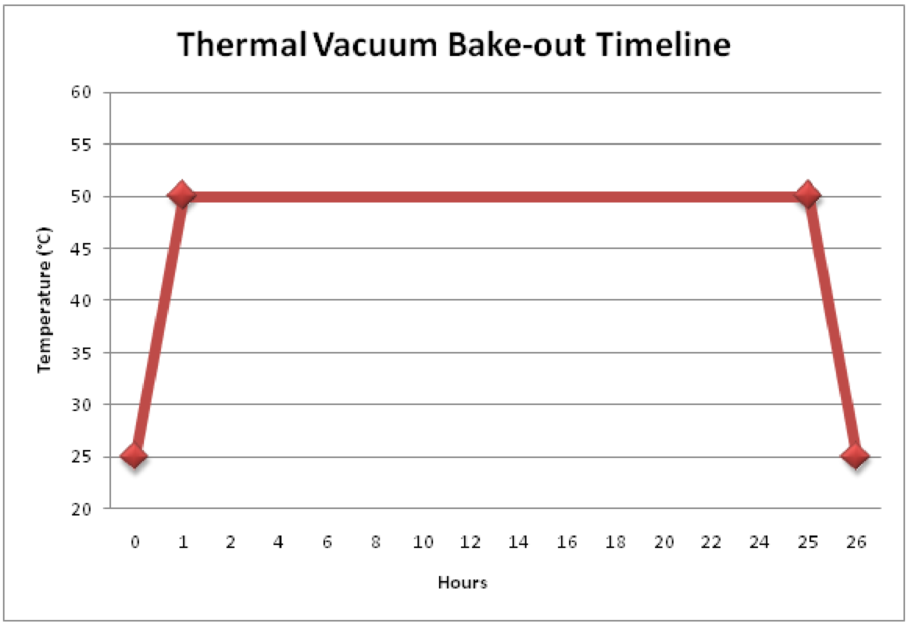

Figure 8.4: History of the Bake-out Procedure of ITUpSAT1

The bake-out process for the SPL+ITUpSAT1 system will last for 24 hours at 50°C. The process will be performed under the near vacuum environment, with a pressure of 10-6 Torr. Thermal vacuum bake-out procedure is presented in Appendix A in details.

8.5.2 Thermal Vacuum Cycle Test

After the bake-out, we will perform the thermal vacuum tests under the expected environmental conditions of the FM of ITUpSAT1. Under these circumstances, the conditions in space can be simulated; however it depends also on the orbital data too. According to the design information, ITUpSAT1 will encounter a testable condition.

The aim of this test is to demonstrate that ITUpSAT1 is able to survive under the thermal and pressure conditions experienced during its orbital life. According to the thermal analysis we performed previously, the test will consists of one hot and one cold cycle, shooting for a satellite temperature range of -35°C to +70°C. We will also perform the functionality check at -35°C, +25°C and +70°C during the cycling test.

Approximate test profile is estimated as below. The test is to be performed under the near vacuum environment, at a pressure of 10-6 Torr (1.33x10-4 Pa).

8.6 Flight Model (FM) Test Results

8.6.1 Vibration Test Results

The vibration tests applied on flight model of ITUpsAT1 consist of Sine and Random Vibration tests at the acceptance levels. Quasi-static loads and shock loads were not applied on the model to prevent satellite and its solar panels from unnecessary loading. Furthermore, the engineering model has passed the qualification level tests successfully with the same configuration of the flight model.

Vibration tests are performed on the satellite+SPL system to simulate the launch environment. SPL is mounted on the shaker with an aluminum adaptor as mentioned before. As a common practice, before and after all tests structural and functional checks and resonance survey test have been performed to observe and verify the endurance of the satellite.

As in the EQM tests, resonance survey tests on the satellite+SPL system provides information about what happens inside the satellite. Using acceleration spectrums through these tests, we have observed whether a structural change or failure occurs or not.

Below figures represent the first and last resonance survey test results in the Z (longitudinal direction). By inspecting these spectrums, we have concluded that the natural frequencies and the behavior of ITUpSAT1 are similar. There some modes below the 150 Hz, however these modes are originated by the characteristics of the shaker system. For the satellite+SPL system, we haven't observed any modes below 150 Hz.

Consequently, after the structural and functional tests, we have not observed any structural deformation on the satellite and any functionality loss on the components.

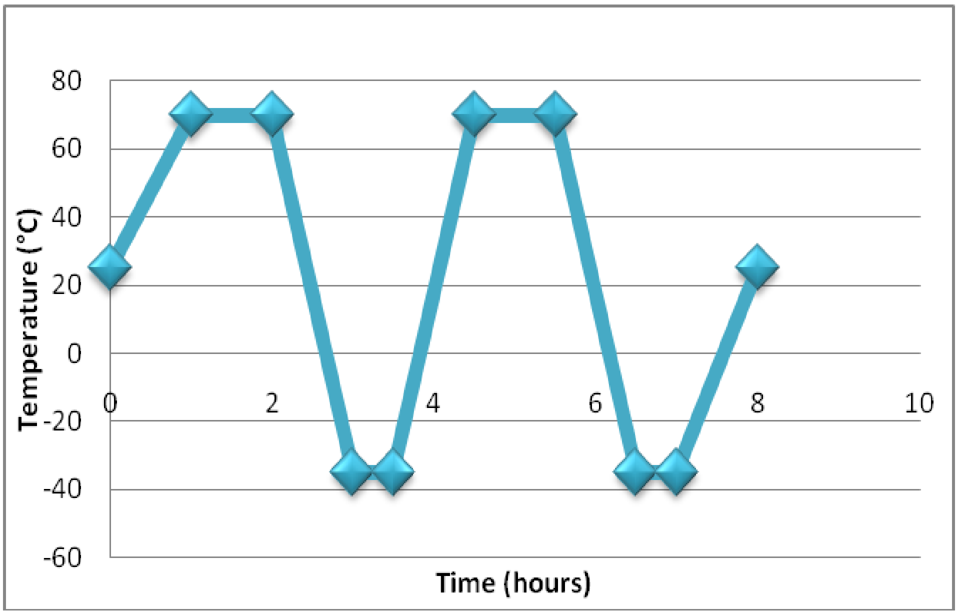

Figure 8.5: The TV Cycle Test of ITUpSAT1

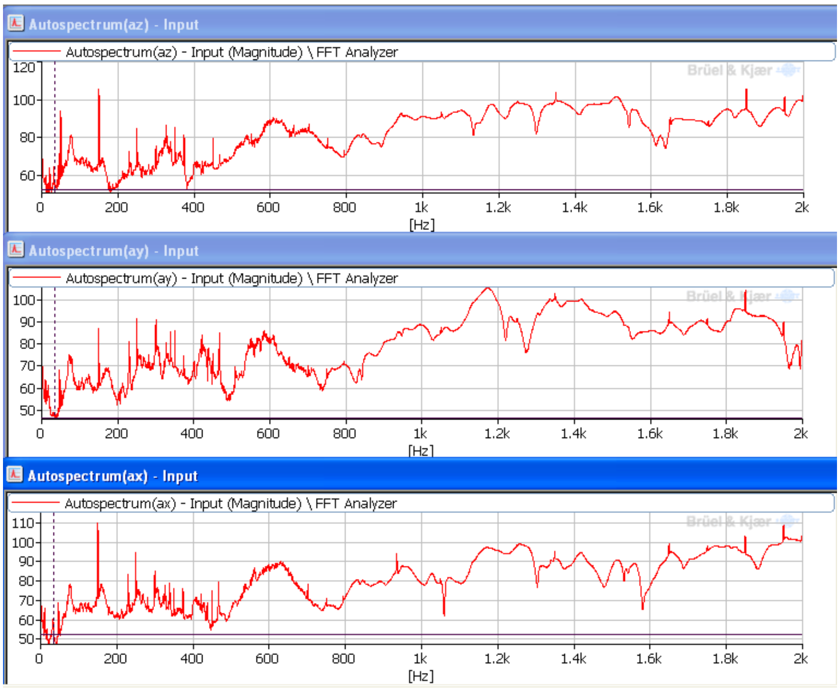

Figure 8.6: First resonance survey test results on the longitudinal direction of SPL

8.6.2 Thermal Vacuum Test Results

The flight model of the ITUpSAT1 is subjected to the thermal vacuum tests, namely thermal cycling and thermal forcing. The process for thermal forcing includes holding the satellite in the low temperatures down to -35°C and high temperatures up to 70°C in the vacuum environment for pre-determined periods.Nnote that the horizontal standby temperatures last for 60 minutes. Through the test the satellite was functional; we have received the telemetry and beacon signals and the photo continuously. During the test no functional problem observed.

The other test, thermal cycling is performed as in the qualification model cycling tests. The aim was to simulate orbital thermal loads that will act during the orbital period of the satellite. During the process ITUpSAT1 was also functional and no problem observed.

Lastly, the bake-out procedure is performed on the satellite as demanded by the launch service provider to prevent the satellite from undesired outgassing in launch and space. The satellite is hold in 10-6 Torr vacuum environment for 24 hours. After the process, functional and structural checks are performed and no problems observed. Below figure shows the bake-out process of the flight model.

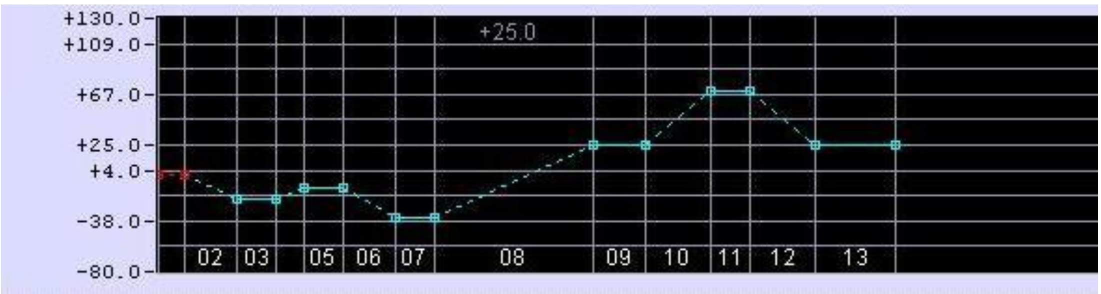

Figure 8.7: Example of a TV tests applied on ITUpSAT1

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APPENDICES

APPENDIX A: Thermal Vacuum Bakeout Procedure of ITUpSAT1

1. Clean the external surface of all hardware before inserting the SPL+Satellite into the thermal-vacuum chamber.
2. Clean the thermal vacuum chamber.
3. Place SPL+Satellite in to the thermal vacuum chamber.
4. Adjust the chamber pressure to 10^{-6} Torr with primary and high vacuum pumps respectively.
5. Starting at room temperature, approximately 25°C, raise the temperature of the shroud to 50°C.
6. Wait until the exterior surface of the hardware has reached 50°C (1 hour).
7. Let the system bake at 50°C for 24 hours.
8. Bring the chamber and hardware back to ambient temperature.
9. Thermal vacuum bake-out is complete.

APPENDIX B: Thermal Vacuum Cycle Test Procedure of ITUpSAT1

1. Clean the external surface of all hardware before inserting the satellite into the thermal-vacuum chamber.
2. Clean the thermal vacuum chamber.
3. Place the satellite into the thermal vacuum chamber (on the shelf).
4. Bring the chamber pressure to 10^{-6} Torr with primary and high vacuum pumps respectively.
5. Starting at room temperature approximately 25°C, bring the temperature of the shroud to 70°C and maintain 70°C temperature for 2 hours.
6. Bring the temperature of the chamber to -35°C and maintain -35°C for 2 hours.
7. Repeat step 5 and 6 for 7 times.
9. Bring the environment to the ambient temperature (25°C).
10. Thermal Vacuum Cycling Test is complete.

APPENDIX C: Vibration and Shock Test Levels of ITUpSAT1

Table C.1: Quasi-Static Test Levels

Longitudinal Axis (X)	+11 g
Lateral Axis (Y,Z)	+/- 6 g
Load Factor	1.25

Table C.2: Harmonic Test Levels

Type	**Harmonic**
Amplitude (g)	0.2
Frequency Range (Hz)	5-2000
Directions	X, Y, Z
Sweep Rate (oct/min)	2

Table C.3: Sine Vibration Test Levels

	Freq. Range (Hz)	**Acceptance Level**	**Qualification Level**
Longitudinal Axis	5-10	8 mm (zero to peak)	10 mm (zero to peak)
	10-100	3 g	4.5 g
Lateral Axis	5-8	8 mm (zero to peak)	10 mm (zero to peak)
	8-100	3 g	4.5 g
Sweep Rate		4 Oct/min	2 Oct/min

Table C.4: Random Vibration Test Levels

Frequency	**Acceptance PSD (g^2/Hz)**	**Qualification PSD (g^2/Hz)**
20	0.001	0.002
110	0.001	0.002
250	0.015	0.034
1000	0.015	0.034
2000	0.004	0.009
GRMS	4.47	6.7
Duration	1 min/axis	2 min/axis

CURRICULUM VITA

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