ISTNanosat-1 Quality Assurance, Risk Management and Assembly, Integration and Verification Planning

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''Live as if you were to die tomorrow. Learn as if you were to live forever.''— Mahatma Gandhi

Abstract

The ISTNanosat-1 is a 1U CubeSat developed in academic environment by students and teachers of IST with the primary objective of belonging to HUMSAT constellation, while developing technical and engineering skills in the participating students. Secondarily, it intends to take measurements on the Flyby phenomenon and serve as a technology demonstrator of a high speed ADC rad hard.

Inherent to the development of a project of this nature, the issues regarding quality, risk and AIV must be considered. Regardless of the academic environment in which the project is carried, it is required to ensure that the satellite is able to fulfill the intended mission and that the developed and testing is performed with the proper management. It was therefore developed within the frame of this work quality and risk management for the project, as well as the respective AIV plan, with particular focus on the test component.

The quality and risk management implemented focused on documentation and configuration management, risk management, selection and control of procurement, manufacturing, assembly and integration, as well as security and reliability considerations that guide the procedures and best practices to take into account in the project development.

Finally, the planned satellite AIV features high-level processes to be taken into consideration for the assembly and integration of ISTNanoSat-1, with the main focus on the planning of the test campaign for qualification and flight of the satellite. A complete test campaign was produced in agreement with the main standards of the space industry and taking into account the expected environment at launch or in orbit. Given the lack of definition if the launcher at this development stage, the test campaign was planned based on the utilization of the satellite in the worst case scenario.

The desirable implementation point was identified according to the requirements and space standards, being the achieved flexibility and lower restrictions compatible with the real needs of the development of a CubeSat. Moreover, the risks identified suggest continuous improvements to be implemented in the frame of the project. Regarding the presented test campaign, it is possible to be used to qualify the satellite for any possible scenario and plausible mission of a CubeSat. The work is therefore considered transverse and reusable for future projects of the same scope and can be seen as programmatic.

Keywords

istnanosat-1, cubesat, assembly, integration, testing, risk management, quality assurance

Resumo

O ISTNanosat-1 é um 1U CubeSat desenvolvido em âmbito académico por alunos e professores do IST com o objetivo primário de pertencer à constelação HUMSAT. Simultaneamente procura-se neste processo o desenvolvimento de competências técnicas e de engenharia nos alunos participantes. Secundariamente pretende-se efetuar medições relativas ao fenómeno Flyby Anomaly e utilizar o ISTNanosat-1 como demonstrador de tecnologia de um ADC rad hard de alta velocidade.

Inerente ao desenvolvimento de um projeto desta natureza, questão relativas a qualidade, risco e AIV devem ser consideradas. Independentemente do ambiente académico em que se realiza o projeto, é necessário garantir que o satélite é capaz de cumprir a missão pretendida e que o seu desenvolvido e teste é efetuado com a gestão devida. Foi, portanto, desenvolvida nesta dissertação a gestão de qualidade e risco para o projeto, assim como o respetivo plano de AIV com especial foco na componente de testes (AIT).

A gestão de qualidade e risco implementada focou desde gestão de documentação e configuração, gestão de risco, seleção e controlo de compras, manufatura, montagem e integração, assim como considerações de segurança e confiabilidade, que balizam os procedimentos e melhores práticas a ter em conta no desenvolvimento do projeto.

Por fim o planeamento da AIV do satélite apresenta a alto nível os processos a considerar para a montagem e integração do ISTNanoSat-1, tendo foco principal no planeamento da campanha de testes para qualificação e voo do satélite. Uma campanha de testes completa foi produzida e concordante com os principais standards da industria espaciais, tendo em conta os ambientes esperados, quer no lançamento quer em órbita. Dada a não definição do lançador neste estágio de desenvolvimento do satélite, a campanha de testes foi planeada na base de utilização no pior cenário possível.

Foi identificado o ponto de implementação desejável de acordo com os requisitos e standards espaciais, que apresenta a flexibilidade e menores restrições compatíveis com as reais necessidades do desenvolvimento de um CubeSat. Adicionalmente os riscos identificados sugerem melhorias a ser implementadas no âmbito do projeto. Relativamente à campanha de testes apresentada, figura-se capaz de qualificar o satélite para qualquer possível cenário plausível de missão de um CubeSat. O trabalho desenvolvido é, portanto, considerado transversal e reutilizável para um futuro projeto do mesmo âmbito e pode ser encarado como programático.

Palavras-chave

istnanosat-1, cubesat, montagem, integração, testes, gestão de risco, garantia de qualidade

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List of Acronyms

ADC	Analogue to Digital Converter
ADCS	Attitude Determination and Control System
AIT	Assembly Integration and Testing
AIV	Assembly, Integration and Verification
AMR	Anisotropic Magneto-Resistive
ARM	Advanced RISC Machines
ATCS	Active Thermal Control Systems
CDH	Command and Data Handling
CDS	CubeSat Design Specification
COM	Communications
COTS	Commercial Off-The-Shelf
CRM	Continuous Risk Management
CSLI	CubeSat Launch Initiative
CSP	CubeSat Space Protocol
DDS	Direct Digital Synthesizer
DET	Direct Energy Transfer
DM	Development Model
EEE	Electric, Electronic and Electro Mechanical
EEPROM	Electrically Erasable Programmable Read-Only Memory
EGSE	Electric Ground Support Equipment
EMC	Electromagnetic Compatibility
EQM	Engineering Qualification Model
ESA	European Space Agency
ESCC	European Space Components Coordination
ESD	Electro Static Discharge
ESTEC	European Space Research and Technology Centre
FEM	Finite Element Model
FM	Flight Model
FMEA	Failure Mode and Effect Analysis

FPGA	Field-programmable gate array
FTA	Fault Tree Analysis
GB	Gigabytes
GCR	Galactic Cosmic Rays
GPS	Global Positioning System
GS	Ground Station
HUMPL	HUMSAT Payload
IC	Integrated Circuit
IMU	Inertial Measurement Unit
IST	Instituto Superior Técnico
JAXA	Japanese Aerospace Exploration Agency
KTH	Royal Institute of Technology
LEO	Low Earth Orbit
LV	Launch Vehicle
MEMS	Micro Electro-Mechanical Systems
MEO	Medium Earth Orbit
MGSE	Mechanical Ground Support Equipment
MIT	Massachusetts Institute of Technology
MLI	Multi-Layer Insulation
MPPT	Maximum Power Point Tracking
MSP	Mixed Signal Processors
MSTL	Multidisciplinary Space Technology Laboratory
NASA	National Aeronautics and Space Administration
NRB	Non-conformance Review Boards
OBC	On-Board Computer
OSCAR	Orbiting Satellite Carrying Amateur Radio
PCB	Printed Circuit Board
PCDU	Power Conditioning and Distribution Unit
PFM	Proto-Flight Model
PIC	Peripheral Interface Controllers
P-POD	Poly Picosatellite Orbital Deployer
PRA	Probabilistic Risk Assessment
PROP	Propulsion
PTCS	Passive Thermal Control Systems
QM	Qualification Model

RISC	Reduced Instruction Set Computer
RF	Radio Frequency
RFC	Request for Change
RTG	Radioisotope thermoelectric generator
S&M	Structures and Mechanisms
S/C	Spacecraft
SEB	Single Event Burnout
SEE	Single Event Effects
SEFI	Single Event Functional Interrupts
SEGR	Single Event Gate Rupture
SEL	Single Even Latch Up
SET	Single Event Transients
SEU	Single Event Upsets
SHE	Single Event Hard Errors
SRLL	Simple Radio Link Layer
SSDL	Space Systems Development Laboratory
TC	Thermal Control
TID	Total Ionizing Dose
TNC	Terminal Node Control
TRL	Technical Readiness Level
UHF	Ultra-High Frequency
VHF	Very High Frequency

1 Introduction

When the CubeSat concept was proposed in 1999 by a joint effort between the California Polytechnic State University's Multidisciplinary Space Technology Laboratory (MSTL) and the Stanford's Space Systems Development Laboratory (SSDL) [1], few could predict the impact and applications this concept would have in the space industry, besides the academic field. At that time, Jordi Puig-Suari (MSTL) and Bob Twiggs (SSDL) were two professors looking to provide to their students means to acquire hands-on experience in the design and development of spacecrafts (S/C) within their one to two-year span in university and at no time they had no set intention to establish a new standard for satellite design [2]. Furthermore, the early reception by the Space community was anything but enthusiastic, with the concept reduced by critics to little more than a toy in space and therefore an additional possible issue in the increasing space debris problem [3].

Despite the early cold reception, the CubeSat concept soon became broadly accepted not only on the academic field but also among the aerospace industry, much due to key features as the standardization, very low cost and easiness of implementation [4]. Only in 2014, 82 CubeSats were launched among military, governmental, university and private (commercial) players, with increased complexity and mission objectives compared to their previous counterparts[1] [5]. This stretch in complexity and mission goals requires more attention to be put into the reliability issues of this type of S/C, without compromising the low cost and fast development spirit of the concept.

The present work incises precisely on the reliability aspects of an academic CubeSat, the ISTNanosat-1 developed by the Instituto Superior Técnico (IST) in Lisbon[2]. It aims to study and manage the risk component of the development project, as well as to develop and apply suitable assembly, integration and testing strategies and procedures for this S/C.

ISTNanosat-1 had its origins back in 2010 as the first CubeSat to be developed by IST. As mainly an educational project, the primary goal of this project is to provide students with the maximum amount of hands-on experience on space systems engineering, following the early spirit of the CubeSat concept, and therefore the use of Commercial Off the Shelf (COTS) subsystems was kept to a minimum. The

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[1] https://sites.google.com/a/slu.edu/swartwout/home/cubesat-database - 2016.
[2] http://istnanosat.ist.utl.pt - 2016

ISTNanosat-1 has the participation in the HUMSAT[3] project by contributing to enlarge its constellation as its main mission. Additionally, it aims to serve as technology demonstrator for mixed signal ICs developed by UNINOVA and to study the Flyby Anomaly [6]. ISTNanosat-1 has its launch predicted in the frame of the ESA initiative "Educational CubeSat Development" e "Fly Your Satellite!'. The S/C is being developed in a 1U form, according to the CubeSat specifications. The S/C consists of all the required S/C subsystems for the operation of the S/C, namely Electrical Power System (EPS), Attitude Determination and Control System (ADCS), Command and Data Handling (CDH), Thermal Control (TC) and Communications (COM) plus the Payload Modules. The Payload Modules will contain the HUMSAT specific board and aims as well to include as secondary payload the demonstrator for ADC board by UNINOVA.

1.1 Motivation and Objectives

In general, space missions are engineering processes largely driven by risk accountability and reduction, taking into consideration the costs and objectives associated. CubeSat missions are not typical space missions and represent a disruptive concept not only from the engineering point of view but also due to the intrinsically higher risk assumed in them. This being said, such missions are not exempt of risk reduction strategies and processes in their development though they correspond to reduced and/or tailored versions of practices used for large satellites.

Even if risk management plans are typically too complex and detailed for use in the development of a CubeSat, along the design and development of ISTNanosat-1, risks have to be identified, analyzed and mitigated through properly adjusted and coherent with the CubeSat concept risk management, making the S/C more risk tolerant.

Furthermore, it is necessary to implement quality, assembly, integration and testing procedures for the ISTNanosat-1. This serves to successfully qualify the ISTNanosat-1 by means of common standardized practices in the space sector which, besides ensuring the reliability of the S/C, make it suitable for piggyback rides as secondary payload. For the proper testing of the S/C and each individual subsystem proper test plans have to be developed and implemented alongside with the determination of the set-ups required for their execution.

1.2 Contributions

Most of engineering aspects related with the Assembly, Integration and Verification (AIV) of the ISTNanosat-1 besides the implementation of Risk Management for the ISTNanosat-1 project were covered in the work performed in the frame of this dissertation. In detail the current dissertation covers:

- Risk Management implementation

[3] http://www.humsat.org - 2016.

- Quality and Product assurance definition
- Launch and Space Environment identification and characterization
- Assembly, Integration and Verification aspects definition
- Test Planning and Procedures for Qualification and Flight

1.3 Structure

This dissertation is structured in 7 chapters. Chapter 1, the present chapter, is an introductory chapter covering motivation, contributions and structure of this dissertation.

The second chapter contains the state of the art of the CubeSat concept, where a technological overview is made alongside with an outline of the CubeSat missions to date and future trends. The work developed so far in the Quality Assurance and Risk Management of space missions is discussed as well, albeit the novelty of this thematic in the world of academic developed nanosatellites. Finally, this chapter contains the approaches for Assembly, Integration and Verification of CubeSats following the specifications of the CubeSat standard, in addition to tailoring and modifications applied by different space players such as national space agencies or commercial CubeSat suppliers.

In Chapter 3 the ISTNanosat-1 mission is overviewed, starting from the mission driving requirements, design and space environments. The launch vehicles possible to be used are also detailed, leading this to a full overview of the mission environment to be considered by the test planning.

In chapter 4 each of the subsystems of the ISTNanosat-1 is described and detailed, representing an overview of all the work previously performed in the scope of the ISTNanosat-1 project. Reliability key points or concerns for each subsystem are introduced in this chapter.

Chapter 4 addresses the Quality Assurance and Risk Analysis and Management implemented approaches for this project. A rationale for the choice of the quality assurance and risk management methodology is presented, followed by the detailed procedures and results of their implementation on this project.

Chapter 6 contains the definition of the AIV Plan, including the tests philosophy and the detailed test plan for the space qualification of the S/C.

Chapter 7 closes the work performed in this dissertation where the main outcomes and conclusions are stated.

2

State of the art

2.1 CubeSat concept and subsystems

CubeSats, as developed up to the present day, are defined as either Pico or Nanosatellites (Figure 1 refers), whose physical dimensions and mass comply with standardized units defined as multiples of 10 x 10 x 10 cm and 1.33 Kg, respectively [7][8]. These base units define the 1U CubeSat, whereas 2U units (10 x 10 x 20 cm and mass up to 2.66 Kg) and 3U units (10 x 10 x 30 cm and mass up to 4.00 Kg) are also common among the developed CubeSats in-orbit nowadays [9]. More recent approaches to CubeSats envision the design of 6U (10x20x30 cm) and even 12U (20x20x30 cm) S/Cs [10], although this approaches still lacks the standardization granted to the smaller CubeSats sizes ranging from 1U to 3U provided by the CubeSat Design Specification rev.13, by CalPoly (In the referred revision 3U+ units have been also covered by the CubeSat standard [8]). These are slightly larger but with the same mass as 3U units.

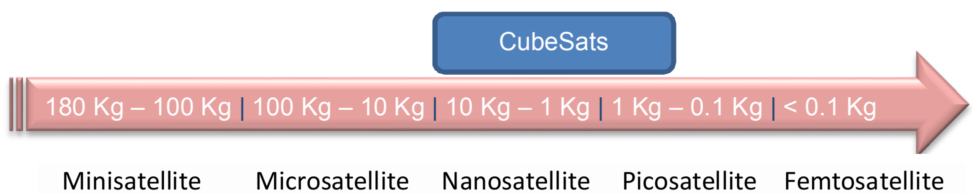
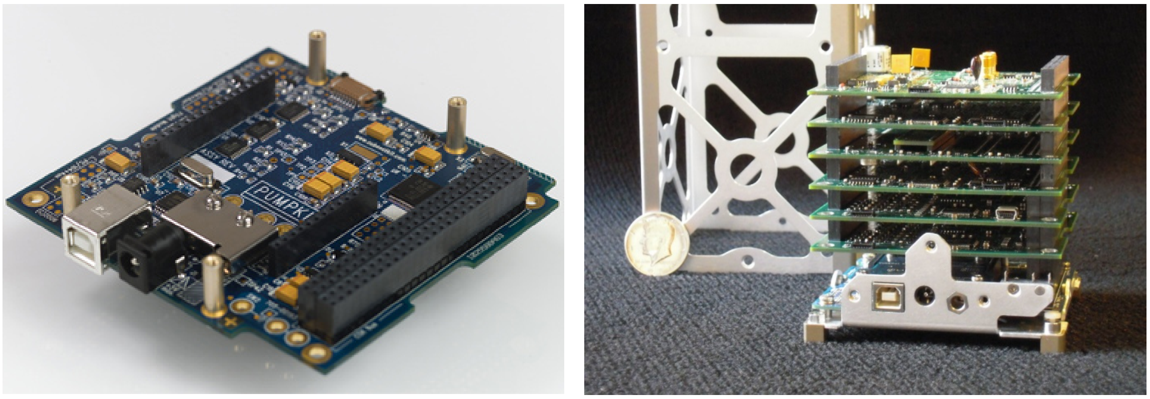


Figure 1 - Classifications of small spacecrafts

The architecture of these S/Cs is based on the stacking of electronics PCBs, using typically the PC/104 form factor (90 x 96 mm) or the 94 x 94 mm form factor [11], inter-connected with a power and data bus by means of common electrical interfaces and data protocols, which allow a high degree of integration and modularity in the conception of CubeSats (Figure 2 refers). Typically, these PCBs are populated with COTS electronic components which, though usually not space qualified, are a low cost alternative with acceptable risk for the kind of missions CubeSats are designed for [12]. Nowadays, COTS previously used in CubeSats missions can be taken as pre-qualified or with flight heritage increasing the degree of confidence in their usage. Overall, the typical subsystems in CubeSats are Structures and Mechanisms (S&M), Electrical Power System (EPS), Attitude Determination and

Control System (ADCS), Command and Data Handling (CDH), Communications (COM), Thermal Control (TC) and Propulsion (PROP), besides the mission Payload.



(a)[4] **(b)**[5]

Figure 2 - PC/104 PCB (a) and CubeSat internal stacks (b)

The clear definition and standardization of the form factor and external dimensions of CubeSats designs made it possible for the adoption of standard deployment containers, initiated by the development of the P-POD in CalPoly [13]. This deployment system utilizes a rail & spring-loaded pusher plate mechanism, actuated upon the opening of a hinged door and represents very low requirements for the interface and integration with the launcher vehicles, allowing CubeSats to be easily placed as secondary payloads at a very low launch cost [3]. Currently several other sources of CubeSat deployers are available, sharing in general the same working principles [14].

Other key factors for the low cost of CubeSats are their easiness of integration and testing, mainly due to their size, and the use of small ground stations with off-the-shelf equipment [15]. All together, these factors enable low cost space activities to smaller players like universities, research institutes, small companies or even nations. The low cost of space activities is coupled in CubeSat with fast satellite development periods making them even more attractive for entry-level players [16][10]. The full conception of a CubeSat takes between 1 and 3 years, depending on the system complexity and Design, Development, Test and Verification approaches [17].

Despite numerous advantages, CubeSats have intrinsic limitations on the achievable absolute performance, mainly due to physical constrains. For instance, Optical and RF payloads have their performance and capabilities closely related with their physical dimensions. Most antennas or high resolution optical instruments do not fit in the CubeSat dimensions. For this reason, CubeSats cannot replace larger S/C in many applications but can provided low cost access to space for educational purposes, in-orbit technology demonstration, specific earth observation and remote sensing, science missions and in the future even interplanetary exploration [10].

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[4] http://www.hbird.de/sites/hbird.de/files/ooStack-with-background_300.jpg - 2016

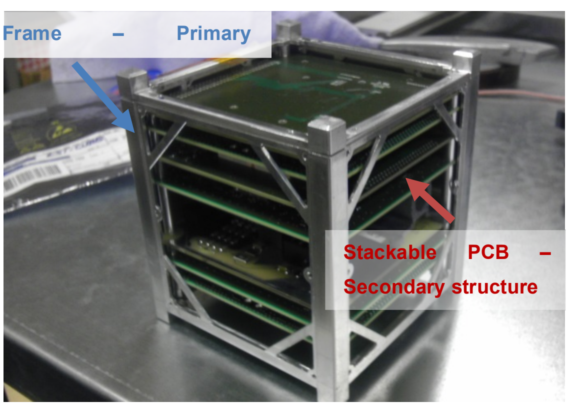
[5] http://www.cubesatkit.com/images/CSK_FM430_710-00252-C.jpg - 2016

2.1.1 Structures and Mechanisms (S&M)

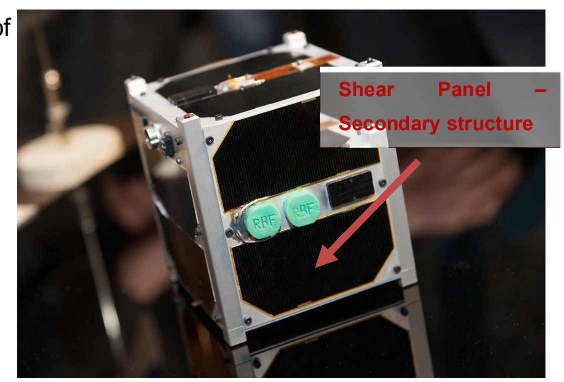
In order to keep the integrity of the S/C and a common interface for all the subsystem a structure capable of properly handling the mechanical loads of all the mission phases must be present in S/C. Auxiliary mechanisms for specific functions (e.g. solar array deployment) may also be required to allow full planed functionality of the S/C, according to different mission objectives [18][19].

CubeSats require very simple structures made typically of aluminum alloy (Aluminum 5005, 5052, 6061 and 7075 [8]) and are widely available through commercial suppliers like Pumpkin, ISIS and SSTL, with some exceptions laying on custom design approaches like the Swisscube, where an entire block of aluminum was machined or the PrintSat where additive manufacturing processes are being used for the first time in CubeSat.

Typical CubeSats structures consist of a load-carrying primary structure with elements such as rib, side frames, rail guides and a secondary structure consisting of shear panels and stackable PCB mounting elements, which grant some additional robustness to the structure as depicted in Figure 3. The mounted stackable PCBs alongside with other possible flight modules can be built up in the secondary structure at first and included in the primary structure at the end of the satellite integration process, ensuring therefore accessibility of the flight avionics [1]. Additionally, the use of the load carrying frame and detachable shear panels, which can be removed at any point of the assembly process, allows the access to all the mounted parts of



(a)[6]



(b)[7]

Figure 3 - CubeSat internal mounted structure (a) and external assembled structure (b)

The S/C structure must also contain Kill Switch mechanisms (typically two redundant spring-loaded mechanical switches) in order to guarantee that the S/C is turned off all the time inside the deploying

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[6] http://en.wikipedia.org/wiki/ESTCube-1 - 2016
[7] http://forte.delfi.ee/news/kosmos/estcube-1-paases-ka-teisest-kokkuporkest-kosmoseprugiga.d?id=66538674 - 2016

system and is turned on once released into its orbit. Thus, these mechanisms are located on the feet of the primary structure [8].

The external physical dimensions of the structure are strictly correlated with the deploying system used to carry the S/C into orbit, for that reason CubeSat must be in line with the aforementioned P-POD or similar deployment devices used as standard deployers for CubeSat S/Cs.

Due to the limited total volume of CubeSats, mechanisms are for now largely restricted to deployment and antenna pointing mechanisms and release mechanisms to deploy solar panels and tethers. Some other mechanical applications are being developed like a long deployable boom by KTH in Sweden to use with sensitive scientific magnetometers or reaction wheels for attitude control [20][21][22].

2.1.2 Electrical Power System (EPS)

All the S/C function relies on electrical power making the electrical power system a key subsystem of any S/C, since a failure necessarily results in the loss of the mission. Electrical Power systems mainly rely on a primary energy source, related with power generation, a secondary energy source, related with the storage of energy, and a Power Conditioning and Distribution Unit (PCDU), necessary to manage the power requirements and deliver the appropriate power levels to all S/C loads when required [10].

CubeSat EPS use Solar Cells/Solar Arrays as primary power system, as the use of other primary power systems (e.g. fuel cells, RTGs…) is naturally limited by the volume, weight and cost requirements as well as for the S/C power needs. Solar Cells/Solar Arrays makes use of solar radiant energy, to convert it via photovoltaic effect into electrical energy and the power available from these source can go up to a 10 to 15 W interval for a 1U CubeSat architecture [12][23]. The power obtained depends on the configuration and type of cell technology used. Configuration wise, CubeSats either have solar cells body mounted or, for a small percentage of them, have deployable solar panels. In terms of the technology used, Gallium Arsenide (GaAs) cells are the most used, mainly due to high efficiency (up to 30%) and limited size [24], while silicon cells are also used when cost is a major driver since their price is much lower than GaAs technology and a very limited number of CubeSats use Copper Indium Gallium (di)selenide (CIGS) [9]. There is a significant number of CubeSats that made no use of solar arrays as primary power source, running exclusively on non-rechargeable batteries for their typically short mission duration [9].

Batteries are typically used as secondary power sources providing power when the primary power system (Solar cells) is not available or is not capable to produce the required amount of power, namely during eclipses or peak power needs [17]. This implies that rechargeable batteries are recharged by energy provided by solar cells/arrays in the sunlight. A large number of batteries technologies is available nowadays (Lithium, Mercury, Cadmium-Nickel, Lithium-ion, Lithium-Polymer etc.) whereas for CubeSat applications the most common are Lithium Ion, Lithium Polymer technologies [20] due to their high energy density although some CubeSat missions use Nickel-

Cadmium or Lithium-Chloride batteries. An exceptional case is the one of Delfi-C3 [25] that presents no batteries whatsoever.

A proper power handling and distribution requires PCDUs. These feature for most CubeSats Direct Energy Transfer (DET) and Maximum Power Point Tracking (MPPT) power distribution architectures [10].

DET implements direct connection between the solar array and the battery, therefore the battery voltage, that varies with the battery's charge level, defines the operating point of the solar array [26]. Compass-1 and QuakeSat are two successful examples of DET implementation.

MPPT regulates the operating point of the solar array by controlling the operation of a switching converter between solar array and load, leading to operation at the optimal point [26]. This architecture adds intermediate components which dissipate excess power, having as possible consequence lower performances for CubeSats when compared to DET, besides all the extra system complexity added [9]. AAUSAT-1 is an example with an EPS system based on MPPT.

2.1.3 Attitude Determination and Control System (ADCS)

Attitude Determination and Control System involves generally a multitude of sensor and actuators so that the orientation of the S/C can be properly determined by the former and the vehicle can be re-oriented to the desired attitude by the latter. These are paramount to payload orientation, communications systems pointing and optimization of the power generation of the S/C.

Among all subsystems in a CubeSat, ADCS is most likely the one that presents the wider variety of configurations going from no ADCS at all to advanced 3-axis stabilization. Even orbit control with Nano-propulsion starts to be implemented as in ESTCube-1 [27], though propulsion application for CubeSats are merely at their dawn.

The typical sensors used in CubeSat missions for the determination of the attitude are described below:

Earth Sensors – use Earth as reference for attitude determination through the use of thermopile sensors or photodiodes to locate the curve of the Earth. Due to the temperature difference of the earth counter between equator and poles these sensors calculate de difference of temperature or infra-red radiation between Earth and space and the displacement to nadir can be obtained [28]. Such type of sensor has been successfully implemented on MIT's Micro-sized Microwave Atmospheric Satellite (MicroMAS).

Gyroscopes – are relative attitude sensors, reflecting a rate of change in attitude and not the absolute attitude of the S/C. An inertial reference sourced from other sensors is required. Most useful to determine faster attitude changes and without requiring any observation of external objects, thus no being limited in case of occlusion of these objects. Gyroscopes are wide spread for CubeSat applications as they benefit for low cost and reduced sized high availability of COTS solutions for this sensors.

GPS receivers – can use the GPS signals not only for orbit control (orbital position determination) but also for ADCS purposes, in particular to determine the direction of a ground target. GPS receivers can be used to their best potential in LEO orbit missions[8]. Radio Aurora Explorer II (RAX 2) makes use of GPS subsystem to fulfil its space weather study mission.

Magnetometers – sense the magnetic field strength or direction (when capable of three axis sensing). Used in conjunction with a map of the Earth's magnetic field stored in memory and knowing the S/C position its attitude can be determined [29]. This limits their use to near Earth orbits missions. Has for gyroscopes magnetometers solutions are widely available and used for CubeSats.

Star trackers – typically high accuracy devices, these sensors are optical devices that measure the position of stars making use of cameras of photocells. Star catalogs are in memory and used as reference for attitude determination. The size, weight and computational requirements of these devices are still limiting factors for a broader use in CubeSats [9]. ARMADILLO CubeSat makes use of such technology on his mission to detect sub millimeter level dust and debris in LEO.

Sun sensors – Indicate the orientation of the sun with respect to a reference coordinate system by detecting the intensity difference between radiation arriving from the solid angle determined by Sun's boundaries and that arriving from adjacent regions within the sensor's field of view. They are commonly integrated in the satellite solar panels and are the most widely used in CubeSat missions such as Delfi-C3 or the Cloud CubeSat [25]. These sensors are naturally limited by the availability of sun light thus do not provide attitude determination during eclipses.

Despite the initial trend for CubeSats to rely on passive stabilization mechanisms more recent approaches to attitude control in CubeSat contemplate the use of active technologies. The main actuators used in CubeSat missions for attitude control are described the following:

Magnetorquers – produce a magnetic field that interacts with the Earth's magnetic field in such a way that the counter-forces produced provide useful torque. Simple light and low power consumption devices, they rely on either air core coils or metal core coils technologies and are a preferred device for attitude control in CubeSats, despite their lower accuracy when compared to other actuators [30][31].

Reaction wheels – are heavy flywheels that work by creating a torque through changing their momentum. The attitude control on the S/C is implemented by spinning up or down the reaction wheel to create torque and therefore vehicle rotation [10].

Other actuators as momentum wheels (similar to reaction wheels in principle but mainly with stabilization purposes and where the wheel is always spinning at a very high speed), fluid dampers, gravity gradient boom, thrusters or spin stabilization strategies can be used in CubeSat missions for attitude control [9].

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[8] GPS system consists of a constellation of satellites orbiting in MEO orbits at approximately 20,200 Km pointed to the Earth. GPS receivers must therefore be within the limits of this orbit to operate normally.

Future of ADCS subsystem will focus on the enhancement of pointing accuracy so more demanding payloads such as optical instruments for remote sensing missions can be used in CubeSat missions.

2.1.4 Command and Data Handling (CDH)

Command and Data Handling subsystem is the center of decision of a S/C. It performs the gathering and processing of data from and to all the other subsystems of the S/C and inherently controlling these subsystems.

The stated functions are provided by the S/C flight computer, in principle a general purpose processor though other on-board processing may be used depending on the architecture of the S/C. In addition to the processing unit, memories, clocks, and interfaces to communicate with other subsystems (system bus) constitute the CDH.

Processing units for CubeSats focus on low power consumption while being self-contained systems. PICs and MSPs are highly common on CubeSats, while ARMs processors, with increased processing capabilities compared to the PICs and MSPs, gain mostly sharing low power consumption profiles [32]. On-Board Computer (OBC) solutions based on field programmable gate arrays FPGA technology, as used in the QuakeSat project [33], start to be increasingly used as FPGAs with space heritage become more widely available.

Non-volatile memories used in CubeSats are a vital part of the systems due to the limited transmission data rate and therefore need for data storage. They provide storage to the operating system, application programs and critical data. Non-volatile memories used tend to be Flash memories and with these, total memory can easily reach several GB at a very low cost [34] used for the data storage. The use of this technology is preferred to other such as EEPROM mainly to the enhanced capacity to sustain radiation environment at a very accessible cost.

In order to connect the various components of the system, a variety of buses can be used taking into account the specific requirements. Most CubeSats make use of the Inter-Integrated Circuit (I^2C) protocol to allow for distributed command and data handling due to high flexibility and re-configurability, low power consumption and the fact that most microcontrollers have it already integrated. This comes however with a compromise in terms of stability, and several CubeSat missions have experienced total bus stuck leading to a loss of data on a required bus reset, thus in the end compromising system availability [35]. Serial Peripheral Interface (SPI) and CANbus (a more robust solution with increased data rate) have been often used in CubeSat missions as well, among a multitude of other bus options available [17].

With the increment of the complexity and requirements of CubeSat missions, high performance on-board payload data processing hardware will be required, thus systems relying on high data rate bus like CANbus and processing units based on FPGAs and ARM will likely see its use increased [36].

2.1.5 Communications (COM)

Communication capabilities in a S/C are arguably one of the most important features of any satellite. Only by means of proper communication with the satellite the data acquired by the payload can be received and operational commands can be transmitted to the S/C. Though not so common for CubeSat missions, the communication subsystem can even correspond to the payload of the S/C itself (for telecommunications missions, which correspond to the biggest commercial satellite market) [37]. Mission success is therefore highly dependent on the reliability of the COM subsystem.

COM subsystem is organized in space and ground segment though only the space segment is treated in the current chapter. The ground segment of the communication system is described in the chapter 2.1.9.

COM subsystem (Space segment) is constituted in its core by transceivers (transmitter and receiver), antennas and the communication processor to handle the communication protocols (Figure 4 refers).

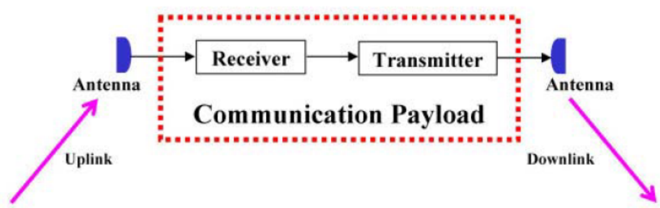


Figure 4 – Space segment COM Simplified Schematic Diagram

The goal of a COM system is to maximize the amount of data transmitted in the minimum amount of time possible while taking into consideration hardware, price and power requirements. The selection of the frequency spectrum used is reflection of this trade-off. For CubeSats frequency spectrums used are limited to VHF/UHF (30 to 300 MHz/ 300 MHz to 3 GHz) and S-band (2 to 4 GHz), though the last one only recently started to be implemented and used [38]. VHF/UHF transceivers work on radio amateur frequencies and protocols offering very limited data rates (between 1.2 to 9.6 kbps). S-band transmitter offers higher downlink data rates from 100 kbps to 1Mbps. The VHF/UHF are usually connected to dipole deployable antennas while S-band transceivers are matched with low-gain patch antennas [10]. The communication protocols used in CubeSats are radio amateur basic protocols. The most widely used is Amateur X.25 (AX.25) mostly due to its simplicity. Other protocols though not so common are used as well such as Simple Radio Link Layer (SRLL) and CubeSat Space Protocol (CSP), developed for the AAUSAT3 [39].

2.1.6 Thermal Control (TC)

The thermal behavior of a S/C has to be defined and controlled to make sure that the components and subsystems of the S/C remain in their optimum range of temperatures (generally between -10 and 40 ^{0}C) and gradients while withstanding the broad variation of temperatures in a space environment

[40]. By controlling the heat exchange with Space, the internal temperature distribution and the heat exchange between internal and external parts, it is possible to assure the proper thermal work environment for all the parts of the S/C. In the case of CubeSats, this subsystem is highly dependent of the configuration of the S/C and type of attitude control, since configuration determines the thermal behavior of the S/C and the type of attitude control can determine the environmental exposition during flight of the S/C external surface [21]. As for its implementation, the Thermal Control is realized in most cases through the use of Passive Thermal Control Systems (PTCS); low cost, low risk and high reliability as well as no power consuming solutions [41][40]. The use of thermal coating (paint or tape) and thermal transfer via radiators and structural elements (e.g. bolts, spacers) is common practice in the CubeSat framework with the advantage of these technologies being fully developed and demonstrated (TLR 9)[9] through numerous past missions. In Pharmasat CubeSat, for example, titanium bolts and Ultem washers were used in order to achieve a proper thermal control of the heat coming from the solar panels to the payload [42]. Multi-layer insulation (MLI) is used in less extent due to dimensions and compactness of the S/C. DelfiC-3 was equipped with MLI to achieve proper thermal balance in the S/C despite the low temperatures in eclipses [21].

On the other hand, Active Thermal Control Systems (ATCS) start to gain more possible applications. ATCS flew already in Compass 1 [40] and MASAT-1 with the purpose of heating up the S/Cs batteries' while in eclipse by use of Electrical Resistance Heaters. With the growth in size and power requirements of CubeSats (e.g. 6U CubeSats) there is a tendency for the heat generated in larger CubeSats to be more than the one that can be passively dissipated. The use of Thermo-electro coolers or miniaturized heat switches that can actively control the heat flow to radiators in order to enable an efficient thermal design are already envisioned for CubeSats thermal control.

2.1.7 Propulsion (PROP)

Propulsion systems provide to a S/C attitude control and orbital maneuvering capabilities often needed to fulfil the mission objectives, through the generation of thrust. An enormous variety of propulsion systems for S/Cs are available nowadays within chemical, electrical, nuclear propulsion as well as propellantless technologies, each with its own advantages and shortcomings [43][44]. Due to the CubeSat specifications, mainly weight and volume, not all these technologies are applicable and propulsion systems capabilities are currently limited. Propulsion systems for CubeSats consist so far in cold gas systems and electric propulsion [43][10]. Cold gas systems available are based on propane or nitrogen as propellant, providing thrust levels of the order of mN with a high thrust resolution and specific impulse performance in the order of 100s [45]. Electric propulsion for CubeSat is on its way to be used in orbit through a Pulsed Plasma Thruster developed by Mars Space Ltd and Clyde Space flying SAMSON mission from Technion, Israel in 2015. The low Delta-V and thrust and

[9] Technical Readiness Level (TRL) are a set of management metrics that enable the assessment of the maturity of a particular technology and the consistent comparison of maturity between different types of technology. TRL range from 1 (lower development level) to 9 (higher development level given to flight proven equipment after successful mission).

high thrust resolution make these systems suitable for limited orbit control applications where precise maneuvers are required (e.g. close proximity maneuvering around another space object, formation flying etc.). Solar sails, MEMS based cold gas propulsion and other variants of electrical propulsion (Field emission electrical propulsion, micro colloid thrusters) are currently in development or towards qualification [10].

2.1.8 Launch and Deployment Systems

Launch Systems

Launch vehicles are the means by which S/C can be put into orbits and their selection must have into consideration factors like the desired orbit, size and weight of the payload, launch date and costs [44]. Since dedicated small launchers are yet to be fully developed for CubeSats, these are limited for now to access space as secondary or piggyback payloads. This fact defines the main conditionals to the CubeSats missions, since the operating orbit is determined by the primary payload and the launch date by the readiness of the primary payload, launch vehicle and launch site [46]. Since the deployment systems allow flexibility on the choice of the launcher, rather than the selection of a launcher itself, CubeSat missions tend to look for launch opportunities with primary payloads that better suit their mission goals, both on timing and orbit.

The most common launch vehicles used for CubeSat missions to put the S/C in orbit are the following listed in Table 1[10]:

Commonly Launchers for CubeSats	
Antares – 110	Minotaur-1
Atlas-5	Minotaur-4
Delta II	PSLV-CA
Dnepr	Progress
Falcon-1	Rokot-KM
Falcon-9	Shuttle
H-2	Soyuz
Kosmos	Taurus
Long March 2D	Vega

Table 1 – Extended list of launchers used for CubeSats

The launch vehicles, environment and requirements for the different launchers hereby presented will be further discussed and detailed in chapter 3.

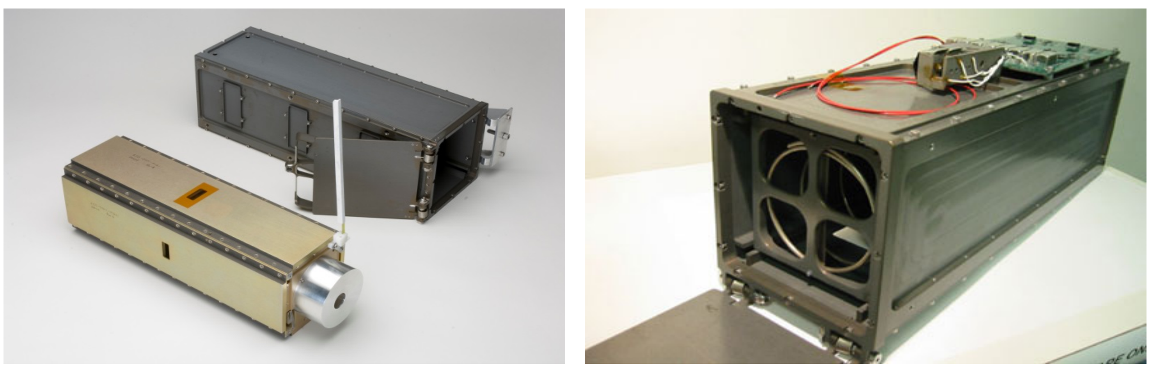
[10] https://sites.google.com/a/slu.edu/swartwout/home/cubesat-database - 2016.

Deployment systems

The deployment systems for CubeSats are one of the key parts of the entire concept and to many correspond to the true breakthrough of the CubeSat concept itself [1]. They allow CubeSats to be launched as secondary payloads without the concern of fully qualifying each satellite, since the deployment systems are already qualified technologies to fly along primary payloads. The safety of the primary payload is therefore guaranteed while a reliable standardized deployment system is provided for the CubeSat and flexibility in the launch vehicle options is maintained. The costs and time of development of CubeSats are greatly reduced as well thanks to this approach [47][1].

The standard deployer for CubeSats was created in CalPoly and named Poly Picosatellite Orbital Deployer or P-POD. The P-POD is a high strength Aluminum 7075-T73 container which can hold up to 10 x 10 x 34 cm of deployable CubeSats, in any combination of configurations up to 3U satellites. The deployer provides as well as Faraday cage effect so the hosted payloads comply with the EMC standards and do not interfere with the primary payload [8][48].

P-POD (Figure 5 refers) is designed to deploy CubeSats at 1.6 m/s with linear trajectories, though different exit velocities are possible. When multiple CubeSats are placed in one deployer additional spring plungers are placed between CubeSats to provide an initial separation between these payloads [49].



(a)[11] **(b)**[12]

Figure 5 – CubeSat P-PODs (a)(b)

A multitude of similar deployers very similar in structure and working with the same basic principles of the P-POD have been developed and successfully used to complement the offer in deployers. These are the cases of the ISIPOD, an up to 3U deployer from ISIS company, the EZPOD, a 6U and 12U version of the ISIPOD, the T-POD, a 1U deployer, the X-POD, a customizable approach capable to deploy up to 14 Kg, including CubeSat standard and the J-SSOD, the deployer present in the ISS with capabilities up to 6U CubeSats [50]. Besides these deployers, secondary payload adapter systems like Naval Postgraduate School's CubeSat Launcher (NPSCul) are available to host up to 24

[11] http://www.nasa.gov/centers/ames/images/content/152693main_genebox-015.jpg - 2016
[12] http://www.thespacereview.com/archive/1490b.jpg -2016

CubeSats, adding complexity on the integration with the launcher but allowing an elevated number of S/C to be deployed at once [51].

2.1.9 Ground Systems and Operations

Ground systems and operations consist on the means implemented to monitor telemetry received at the ground station and configuring on board equipment to execute the mission. This requires ground stations equipped to communicate with the S/C

. Being CubeSat missions so far only placed in LEO orbits, the communication windows are short relative to a specific ground stations and the propagation delays resultant from the distance result in difficulties in data retrieving. Ground stations are composed by receptor/emitter antenna systems, Antenna pointing and tracking systems (antenna control), frequency converters and amplification stages (low noise and high power). This is illustrated in Figure 6.

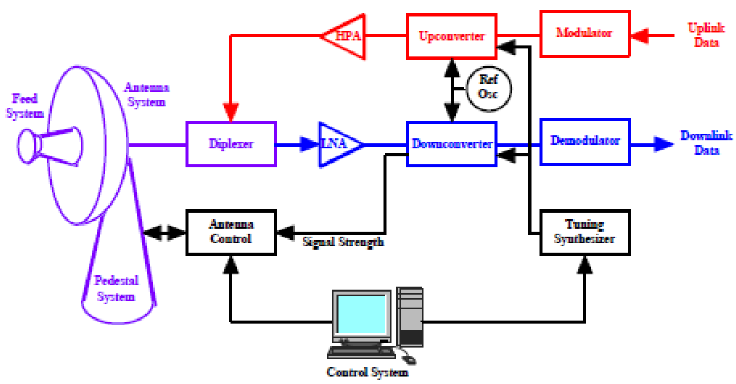


Figure 6 - Typical summarized ground station architecture [52]

Though communication with CubeSats can be performed through a small low-cost dedicated ground station, radio amateur ground station networks such as the Global Educational Network for Satellite Operations (GENSO) and SATNet enhance the capabilities for CubeSat mission and tackle the limitations imposed by the short windows imposed by LEO orbits [39]. The GENSO systems allows for CubeSat controllers to communicate with their S/C by the use of a remote ground station via internet within the satellite range at that time. Architectural illustration of the GENSO is depicted in Figure 7:

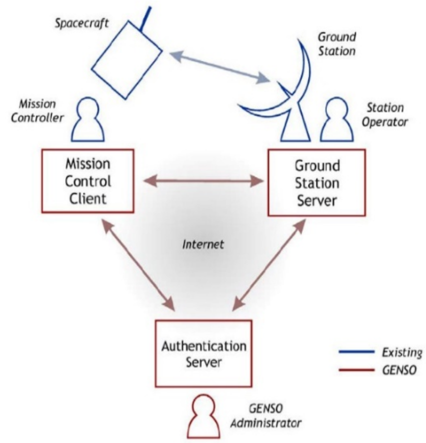


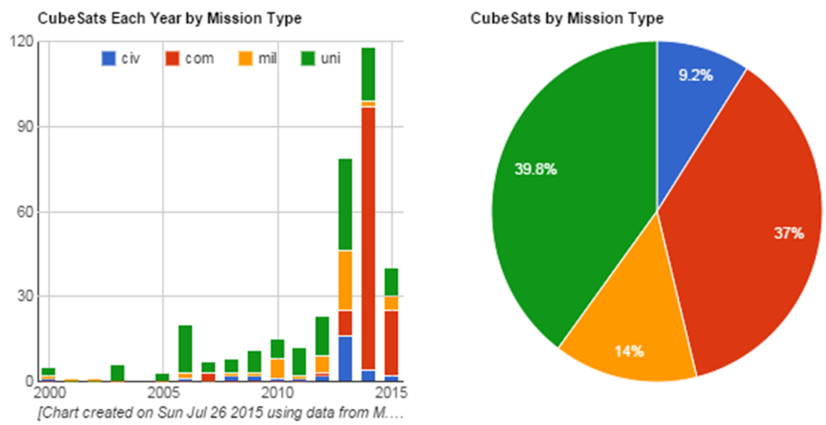
Figure 7 – GENSO general architecture[13]

2.1.10 Current Players, Applications and Future Trends

Current players and applications

The range of players working nowadays in CubeSat field goes from universities, research institutes industrial players, governmental agencies,

About 350 CubeSats have been launched so far[14]. According to their mission type these can classified as commercial, university, military and civil CubeSats (Figure 8 refers). It is possible to understand that though most of the CubeSats are still university, there has been in recent year a strong push on the commercial usage of CubeSats in particular due to the role of Planet Labs.



(a)[15] (b)[16]

Figure 8 – CubeSats each year by mission type (a) and total by mission type (b)

[13] http://www.esa.int/spaceinimages/Images/2008/03/How_GENSO_will_fit_into_the_existing_satellite_communication_framework - 2016
[14] https://sites.google.com/a/slu.edu/swartwout/home/cubesat-database - 2016.
[15] http://www.nasa.gov/centers/ames/images/content/152693main_genebox-015.jpg - 2016
[16] http://www.thespacereview.com/archive/1490b.jpg - 2016

Planet Labs[17] is an American company that makes use of 3U CubeSats to create the constellation with the best temporal resolution available nowadays. This constellation allows from Earth imaging on the areas within the 52 degrees of equator, where most human populations and agricultural areas are located. For this constellation to be built and maintained more than 100 CubeSats have been launched[18].

Beside the straight commercial applications, the provided images even have humanitarian and environmental applications, from monitoring deforestation and urbanization to improving natural disaster relief and agricultural yields in developing nations.

Other relevant players in the CubeSat commercial landscape worth referring are ISIS[19], Surrey Satellite Technologies[20], Aerospace Corporation[21], Spire[22] and Pumpkin[23] (as component suppliers). Though these do not have the same impact on numbers as Planet Labs, they represent the front line to further develop the use of CubeSats for commercial space application.

Beside the increasing role in commercial applications so CubeSat's start to be regarded within Space agencies across the world. For nations with small young space programs it is a low cost opportunity to gain space capabilities and know how, such is the case of the Colombian Libertad-1 [53]. Bigger agencies, namely ESA and NASA, develop their own CubeSats support program, aiming to improve the contact of university players with the space environment. ESA has provided launch opportunities in VEGA's launcher maiden flight in 2012 (where e-st@r, Goliat, Masat-1, Masat-1, Robusta, UniCubeSat-GG, XaTcobeo where launched) [3]. Currently a new ESA initiative called Fly your satellite! (FYS), an educational program with main focus on the support of the verification campaign and launching of CubeSats built in the academic sphere. (AAUSAT4, ConSat-1, e-st@r –II, OUFTI-I, POLITECH.1, Robusta-1B are part of this program).

CubeSat Launch initiative (CSLI) it's a NASA program to provide flight opportunities for small satellites. NASA's program is open for the participation of CubeSat missions that are in line with NASA's Strategic Plan and the Education Strategic Coordination Framework, opening doors for science, exploration, technology development, operations and education missions. Eleven EleNa (project ELaNa: Educational Launch of Nanosatellites) Missions have flown so far with, with a twelfth EleNa flight planned for early 2016[24].

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17 https://www.planet.com/ - 2016
18 https://www.planet.com/ - 2016
19 http://www.isispace.nl/cms/ - 2016
20 https://www.sstl.co.uk/ - 2016
21 http://www.aerospace.org/ - 2016
22 https://spire.com/ - 2016
23 http://www.cubesatkit.com/ - 2016
24 http://www.nasa.gov/directorates/heo/home/CubeSats_initiative - 2016

Trends

CubeSats tend in the near future to undergo a general size growth with the 6U missions already under development such as NASA's Dellingr[25] and plans for 12U (20x20x30 cm, 12-18 kg) applications are already in the pipeline. This growth in size aims to take the best of the CubeSat world in terms of standardization and achieve much higher capabilities that are limited by the reduced size of small CubeSats namely due to payload and instrumentation physical restrictions. This size CubeSats shall be able to accommodate as well on board propulsion for small maneuver capabilities.

Not only on the CubeSat bus itself progresses are expected as several small satellites/CubeSats dedicated launch systems are expected to come to life. Among current developments the S3 launcher[26], aims to deliver the first satellite in orbit in 2018. Making use of current and projected technology future CubeSat mission will aim for more complex objectives and applications. Constellations of CubeSats, interplanetary missions (namely lunar missions), higher grade Earth Observation or Science missions will make use of the CubeSats size growth and technological advances [4] [54].

[25] https://gsfctechnology.gsfc.nasa.gov/OuterBudge.html - 2016
[26] http://www.s-3.ch/en/home - 2016

2.2 Quality Assurance and Risk Management

Quality and Product Assurance as well as Risk Management are not project disciplines where the outmost concern and resources are typically allocated in CubeSat projects, both for the low resources available and for the complexity of the tools and philosophies typically used in this disciplines for larger S/C. Nevertheless, and with the on-going growth in complexity and applications of CubeSat projects either tailored or dedicated approach are being put in place in order to increase the rate of success, assess and reduce the project risks [55].

Quality Assurance is the segment of the quality management focused on providing confidence that the quality requirement will be fulfilled. Product Assurance on its side is a discipline devoted to the study, planning and implementation of the activities intended to assure that the design, controls, methods and techniques in a project result in a satisfactory degree of quality in a product of this project [56]. ESA’s ESCC-Q-ST-20C [57] is the main standard for Quality assurance in the European space industry.

“Risk Management is the process of risk identification, analysis, mitigation, planning and tracking of the root cause of problems and their ultimate consequences” [58]. Risk Management is implemented making use of risk analysis tools. The key and most widespread tools in space industry, but not only, can be summarized to Failure Mode and Effect Analysis (FMEA), Fault Tree Analysis (FTA), Probabilistic Risk Assessment (PRA).

FMEA had its origin in the military and aerospace American industry in the 1950s and it has since a reference for risk analysis in the space industry. In short it is a qualitative method but a quantitative basis can be added when mathematical failure rate models are added, put in action using historical data and inferential statistics to identify and define the failures [59]. The FMEA is in principle a full inductive (forward logic) analysis to systematically analyze subsystem, parts and component failures and identify the resultant effects on system operations. A successful development of an FMEA requires that the analyst include all significant failure modes for each contributing element or part in the system [60].

FTA is an analytical technique starting on an undesired state of the system and subsequent analysis in the context of operation to identify the possible ways the undesired state (top event) can occur. The fault tree itself is the graphic representation of the possible sequential combinations of faults that origin the undesired state identified [61]. The fault tree represents the Boolean logical interrelationships of the basic events that lead to the undesired state/event, as illustrated in Figure 9. Though not common practice in CubeSat missions, projects such as HERMES have implemented FTA as risk analysis and management tool [62].

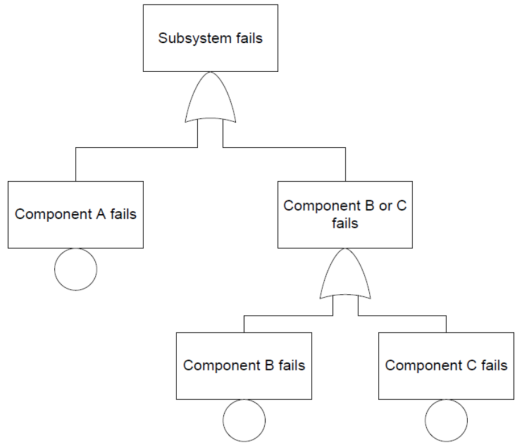


Figure 9 - Sample fault tree diagram [62]

PRA is an analytical method that quantifies risk metrics of higher complexity requiring the specification of the hazard, the identification of the initiating events and estimation the frequency of each of these initiating events. Upon the assumption of occurrence of each initiating event and given the response to that event, the combination of failures is identified. The calculation of the likelihood of all the consequences to the same outcome. The likelihood of the outcome is the sum of the sequence frequencies. Aforementioned tools like Fault tree analysis, event tree model and human reliability analysis are used in conjunction with modeling methods including monte carlo to make such assessment complete [63].

Due to their enhanced complexity, the tools aforementioned tend to be unfeasible for CubeSat missions. A low cost approach has been specifically developed for CubeSat, where low cost analytical methods are preferred over more detailed analysis such as PRA. This approach entails 3 major steps: risk identification, determination of mitigation techniques and monitor the progress risks [58]. The detailed sub steps are depicted in Table 2.

Main Step	Sub-steps
A. Risk identification	1 Review the mission concept of operations 2. Identify root causes 3. Classify priority of risk 4. Name responsible person 5. Rank likelihood (L) and consequence (C) of root cause 6. Describe rationale for ranking 7. Compute mission risk likelihood and consequence values 8. Plot mission risks on L-C chart
B. Determine mitigation techniques	Choices consist of: 1. Avoid the risk by eliminating root cause and/or consequence 2. Control the cause or consequence 3. Transfer the risk to a different person or project 4. Assume the risk and continue in development
C. Track progress	Plot the mission risk values on an L-C chart at key life cycle or design milestones to see progress.

Table 2 – Detailed Steps of Risk Management Plan as proposed by [58]

NASA on its side establishes Continuous Risk Management (CRM) [64] for small budget and schedule projects, namely nanosats, where larger sized CubeSats fit. Continuous Risk Management approach, states a repeated cycle of risk identification and handling in order to achieve acceptable risk levels at reduced project costs (~0.5%). The working principles of this approached are summarized in the following CRM diagram (Figure 10 refers). The tools to perform the risk assessment and management are among the ones aforementioned. The specific use of each tool is project dependent.

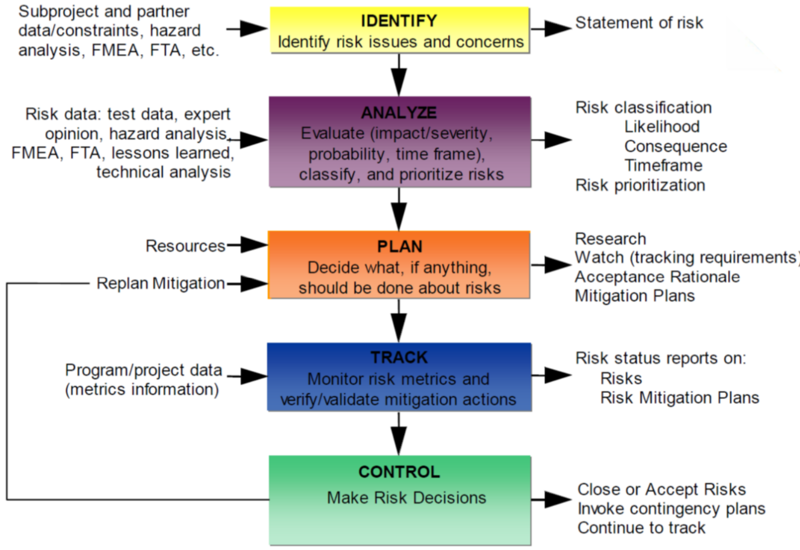


Figure 10 - NASA's CRM Diagram [64]

ESA's risk management tailored approach is mostly based on FMEA approach to risk management as exemplifying method and it is a spread approach in space European projects. Nevertheless, the applicable standard refers to the requirements for a risk management implementation rather than on how such implementation shall be performed [65]. Figure 11 illustrates the Risk Management steps as defined by ESA risk management guidelines.

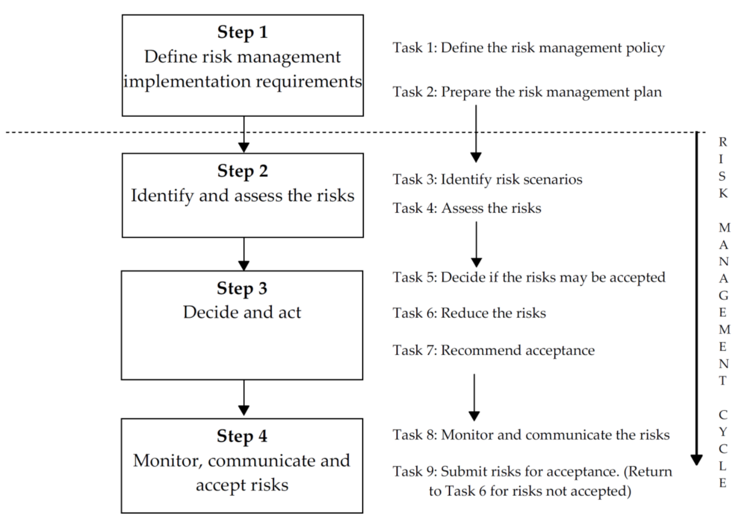


Figure 11 – Risk Management steps as defined by ESA's risk management guidelines [65]

2.3 Assembly, Integration and Verification

Assembly, Integration and Verification discipline (AIV) aims to ensure that the S/C will perform the expected mission with a high level of confidence and probability of success [44]. Verification can be performed using different methodologies being the main test, analysis, inspection and review of design though often used in combination. The Assembly, Integration and Testing (AIT) discipline refers to the major and most visible part of the AIV process that includes the assembly of the different satellite components, parts and subsystems, subsequent integration and testing.

All major space agencies (ESA[27], JAXA[28], NASA[29]…) have established AIV and AIT planning procedures with the primary intent to guarantee Qualification and Acceptance of the S/C. These define three different Model Philosophies for the qualification and acceptance: Prototype, Proto-flight and Hybrid approach. Prototype approach implies all qualification testing to be performed in one or several Qualification Models (QM) and that the Flight Model (FM) shall undergo a full acceptance test campaign [44]. Proto-flight approach implies for qualification and acceptance to be performed all at once on the same model to be flown, typically using qualification test levels and acceptance duration. This is called the Proto-flight Model (PFM) [66]. The Hybrid approach aims to combine the benefits of the two previous approaches. In this approach, specific Qualification tests can be carried in dedicated models for particular, and usually critical areas (e.g. Thermo-mechanical model). Acceptance testing is to be carried out on the PFM.

AIV approaches for CubeSats are growingly being implemented based on larger S/C typical standards and procedures though tailored to simpler implementations so that complexity/quality balance is not compromised [44]. To better illustrate this, the general development approach is depicted in Figure 12.

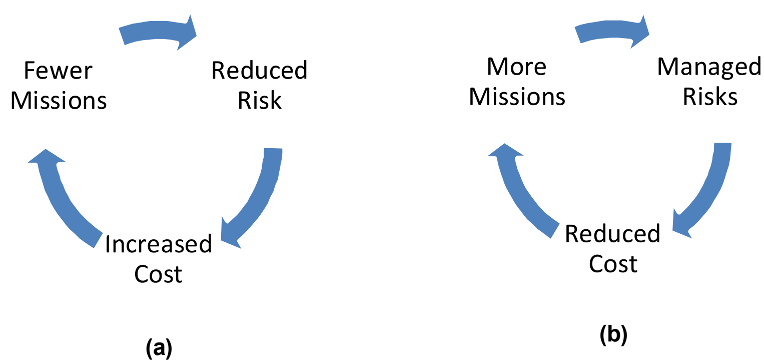


Figure 12 – General development approach for large S/C (a) and CubeSats (b)

Most Assembly and Integration for CubeSat's represent only small challenges when compared to its larger counterparts. Nevertheless, it is current practice to apply the proper standards as good

[27] http://www.esa.int/ESA - 2016
[28] http://global.jaxa.jp/ - 2016
[29] https://www.nasa.gov/ - 2016

practices for these aspect of the development program, being most of ESA ECSS project management and product assurance standards a practical example of such under use standards.

Verification is strongly based in testing of the S/C. Typical testing sequences for CubeSat’s are encompassed of a full functional assessment of the S/C and tailored environmental testing (e.g. as established per CubeSat Design Specification [8]). Testing is of the most relevance to guarantee that all the subsystems, developed by different teams/individuals, operate correctly and that the proper interfacing has been performed.

Functional tests of CubeSats are typically performed in ambient conditions and controlled environment. Nevertheless, end to end tests (Development to PFM Functional) of nanosatellites have been performed using high altitude balloons, being Balua[30] and Balloonsat [67] two examples of such approach. This approach allows to place the S/C in near LEO conditions, not easy to replicate in normal development lab, and validate via testing the sensors chosen for the scientific measurements, space segment subsystems, the operational software for the space and ground segment as well as the ground segment hardware [68]. This approach has the cost advantage over environmental testing in specific facilities, though costs are still relevant in the scope of a CubeSat project.

[30] http://balua.org/

3

Mission overview

3.1 Mission Design and Requirements

The ISTNanosat-1 main mission is to take part on the HUMSAT constellation and therefore the mission design and requirements are highly driven by this constellation needs.

Based on the need of the HUMSAT constellation to achieve global coverage in its operation the ISTNanosat-1 shall operate in a LEO circular polar orbit with inclinations very close to 98º. These orbits allow for the S/C to run over the complete Earth's surface in one single day [49].

Regarding the operational orbit lifetime of the ISTNanosat-1 it shall be, at least, 1 year. Therefore, the minimum altitude for the ISTNanosat-1 shall have an orbit with a lifetime of, at least, 2 years; leaving 1 year of safety margin. An expected orbit interval is between 600 Km and 800 Km.

In order to successfully complete any project, the establishment of requirements is a step to assure that the goals to complete are clear and drive the project. ISTNanosat-1 requirements have yet to be iterated based on the space industry requirements, project mission requirements and mostly upon HUMSAT requirements. HUMSAT requirements are available in [69] and [70] , for reference.

3.2 Launch Vehicles

Launch conditions and environment, as well as some of the related environmental requirements are intrinsically related with the launcher used to put ISTNanosat-1 in orbit. The list of launch possibilities is increasingly growing with more and more CubeSats being launched. A thorough analysis of the most common launchers conditions and requirements used in CubeSat launches is made. The majority of launchers for CubeSat missions[31] are hereby listed in Table 3. In the scope of this document the launchers highlighted in bold will be considered, namely the conditions and requirements of these launcher will be compiled and studied as they represent the most likely choices at this stage for the launch of the ISTNanosat-1. The choice of such launchers is based on the number

[31] https://sites.google.com/a/slu.edu/swartwout/home/cubesat-database - 2016. .

CubeSats already launched (so the most used launchers were selected) and the likelihood of use in the ISTNanoSat-1 project, this being particularly for the ESA launchers (Soyuz and Vega) and Dnepr.

Launch Vehicles	
Antares – 110	**Minotaur-1**
Atlas-5	Minotaur-4
Delta II	**PSLV-CA**
Dnepr	Progress
Falcon-1	Rokot-KM
Falcon-9	Shuttle
H-2	**Soyuz**
Kosmos	Taurus
Long March 2D	**Vega**
M-5	

Table 3 – Main Launch vehicles for CubeSats

3.3 Mission Environments

Environmental conditions refer to the launch and space environments conditions that the ISTNanoSat-1 is expected to encounter throughout the mission. Naturally this is highly correlated with the chosen launch vehicles. The typical maximum environments for the launch vehicles aforementioned have been compiled from the respective used guides ([71][72][73][74][75][76][77][78] refer) and resulted in the following graphics (Figure 13, Figure 14, Figure 15, Figure 16, Figure 17, Figure 18 refer). The environment conditions established will be used to specify the subsequent Environmental requirements specification and test levels for type of lead. The loads defined later as part of the test plan shall be used and applied for the system and subsystem design and verification of the S/C.

3.3.1 Mechanical Environment

(Quasi) Static loads

During flight, S/C is subjected to both static and dynamic loads from the launcher, boost motors and its own spin if applicable. This loads act both in the S/C and in the S/C/LV interfaces and shall be sustained by the load bearing structures of the S/C. A summary of the maximum (Quasi) Static accelerations the S/C is subject per LV is depicted in Table 4.

Launcher	Lateral Acceleration (g)	Longitudinal Acceleration (g)
Antares	-1±0.3	6 ±0.5
Atlas 5	0.4	5
Dnepr	-1±0.7 (ground)/0.5±0.5 (flight)	7.8±0.5
Falcon 9	±2	-2 to +6
Minotaur 1	0.4	3.8
PSLV	±6	-2.5 to +7
Soyuz	0.4	4.3
Vega	0.9	5.5
Worst Case Loads	±6	7.8±0.5

Table 4 - Quasi Static Loads of selected launchers

In conjunction with the aforementioned (quasi) static loads dynamic loads are induced in the S/C and the resulting environment is characterized by the random and sinusoidal vibration loads, acoustic loads and shock loads of the launch environment.

Sine-equivalent vibration

Sine equivalent dynamics reflect the physical vibration environment undergone by the S/C during the launch as a result of the pressure oscillations of the motors as well as Pogo oscillations (self-excited vibration caused by combustion instability [79]).

Sine-Equivalent Vibration Environmental levels for PSLV were not available from the launcher guide and acceptance test levels were assumed as baseline. As for Minotaur no data relative to sine equivalent vibration is available and this loads are encompassed in other mechanical loads. Finally, Falcon 9 sine equivalent vibration environment is derived specifically for each payload via coupled loads analysis. Detailed methodology for the sine vibration curve is depicted in [74].

Sine-equivalent vibration loads for axial and lateral directions are depicted in Figure 13 and Figure 14 respectively. Among them, the heaviest loads correspond to the axial loads of PSLV (lower frequencies) and the Vega launcher (higher frequencies). Lateral maximum loads do not go above 1g acceleration whilst PSLV axial maximum load reach 2.5g. Considering the uncertainty in the mounting direction of the S/C in the launcher at this stage, worst case, loads shall be expected in the S/C independently of the direction considered and therefore the envelope of loads aforementioned (PSLV and Vega loads) shall be considered for respective testing.

Random vibration

Random vibration loads on the S/C result primarily due to acoustic phenomenon with a slight portion being transmitted through the S/C interface. Most Launcher manufacturer state that these loads are encompassed by both the sine equivalent and acoustic vibration loads.

Random vibration environmental levels for PSLV were not available from the launcher guide and acceptance test levels were assumed as baseline. Vega random vibration environment is stated to be covered by acoustics. No reference is made in Falcon 9 launcher guide to random vibration environment. For Antares the random vibration environment at the payload interface is encompassed by acoustics and coupled load results [71]. Finally, Atlas launcher guide states that acoustic vibration environment covers more accurately the random vibrations present during flight.

Random vibration loads are depicted in Figure 15. Among them, the heaviest loads correspond to the Dnepr loads. This load envelop is therefore the worst case scenario from the data available and shall be considered for respective testing.

Acoustic vibration

It is at lift-off and transonic flight stages apply the most acoustic stress on the S/C and therefore the payload. At lift off this stress results from the engine operation, more specifically plume impingement on the pad. At the transonic flight phase the unsteady aerodynamic phenomena as turbulence inside the boundary layer and shock waves prevail. During the remaining stages of the flight acoustic loads are substantially lower and Figure 16 highlights the acoustic maximum envelope for the considered S/C. This envelope corresponds to a composed envelope of Minotaur-1 (up to 250 Hz) and Dnepr (from 250 Hz onwards loads).

Acoustic vibration environmental levels for PSLV were not available from the launcher guide and acceptance test levels were assumed as baseline.

Shock loads

Several shock loads are typical of a launch events. These loads happen at the vehicle hold down release at lift off, at the different stages separations, at the fairing separation and finally at the S/C release and separation. The two last events represent the maximum loads the S/C has to endure during the launch. Maximum shock loads for the different considered launchers is depicted in Figure 17. Worst case envelope loads correspond to an envelope of Antares loads (up to 300Hz) and Minotaur-1 loads (300Hz onwards) corresponding to a maximum acceleration of 3500g.

3.3.2 Electromagnetic Environment

The Electromagnetic environment inside the fairing is influenced by the LV emission and reception equipment as well as the range equipment, part of the ground support, such as radars and local communication networks. The Electromagnetic environment is in general similar among the different launchers due to the use of the same frequency ranges, mostly UHF, S-Band and C-Band and depicted in Figure 18.

Both Antares and Minotaur Launcher don't' have a full spectrum data available but the maximum values are highlighted. Antares has a maximum peak at C-band of 153 dBμV/m (inferior to the environment of Falcon 9) and Minotaur a peak in the same region of 148 dBμV/m, both inferior to the environment of Falcon 9. PSLV data regarding the electromagnetic environment is not available in the launcher user's guide.

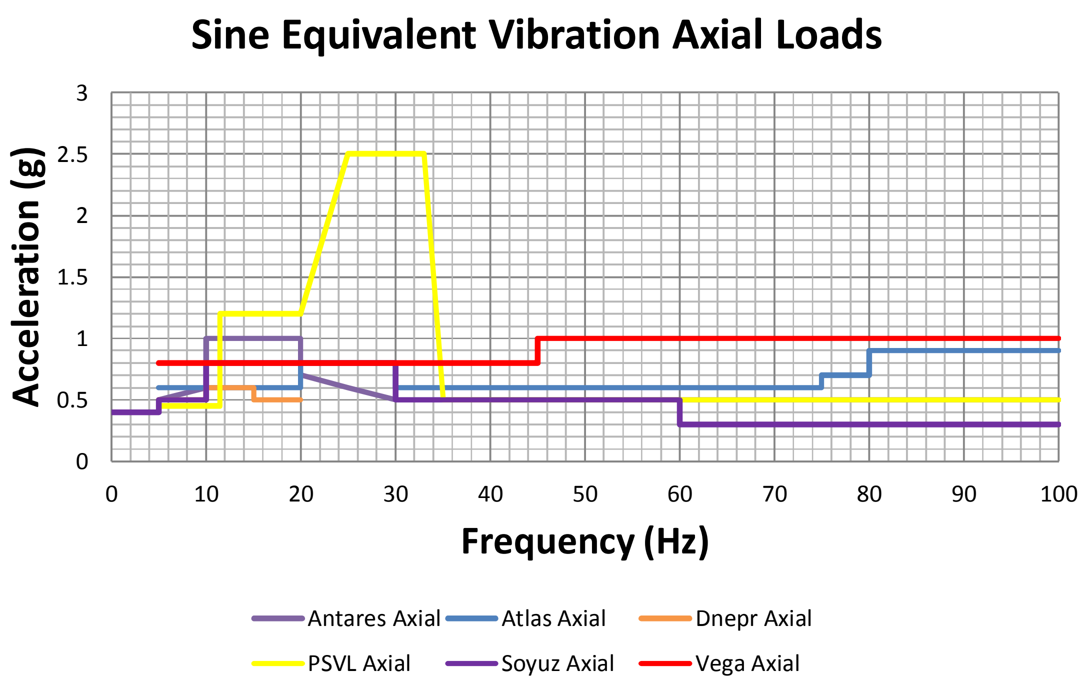


Figure 13 - Sine Equivalent Vibration Axial loads of selected launchers

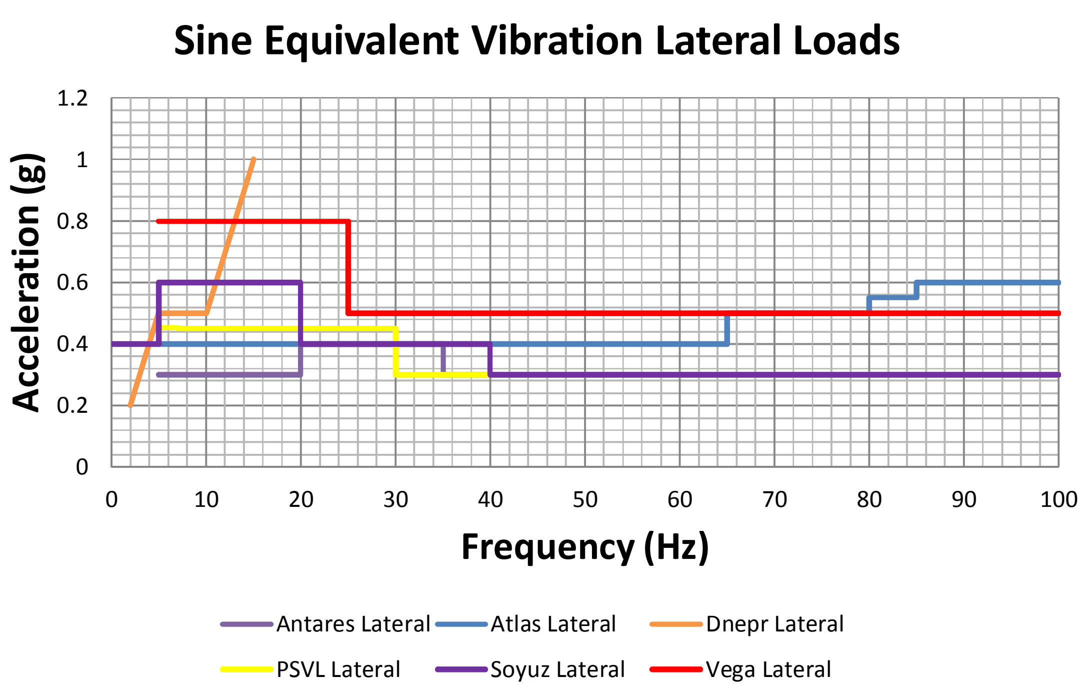


Figure 14 - Sine Equivalent Vibration Lateral loads of selected launchers

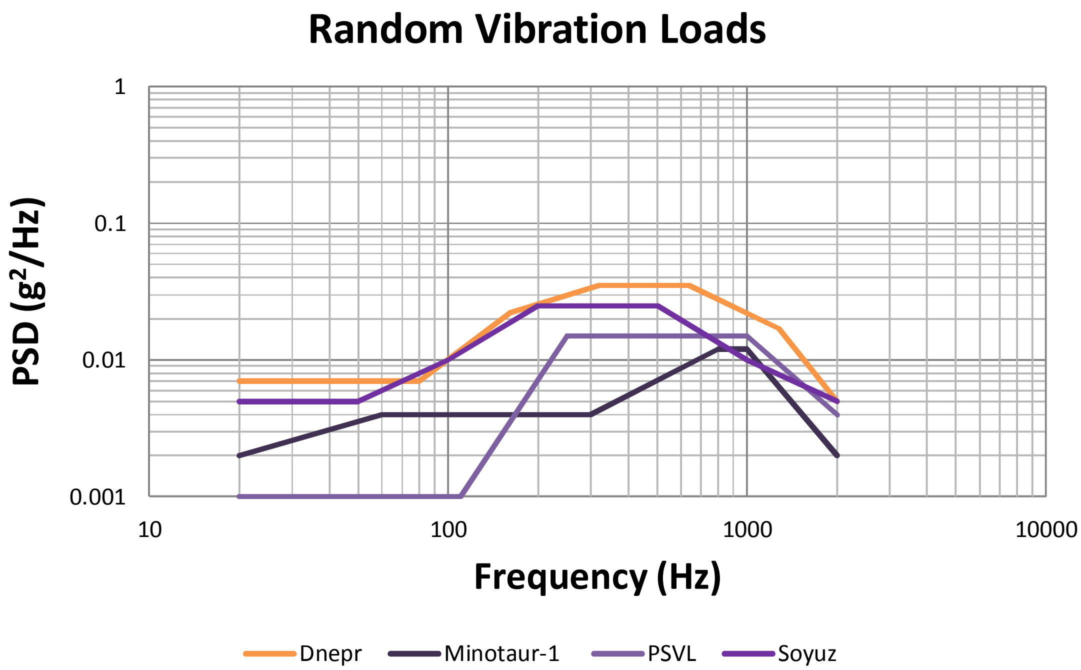


Figure 15 - Random Vibration loads of selected launchers

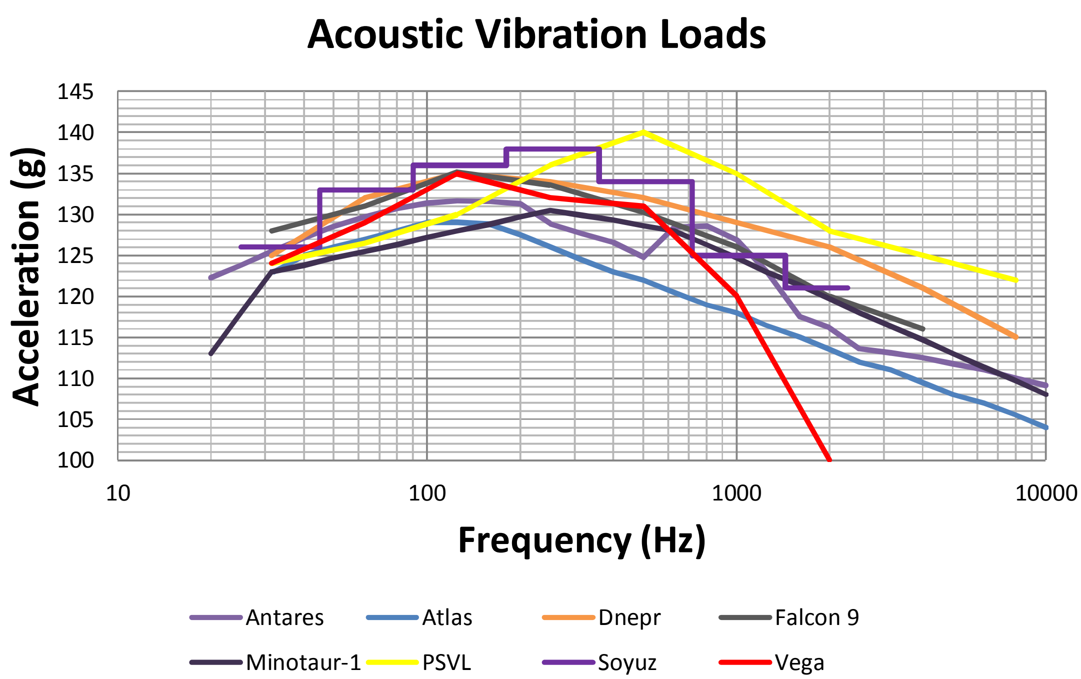


Figure 16 - Acoustic Vibration loads of selected launchers

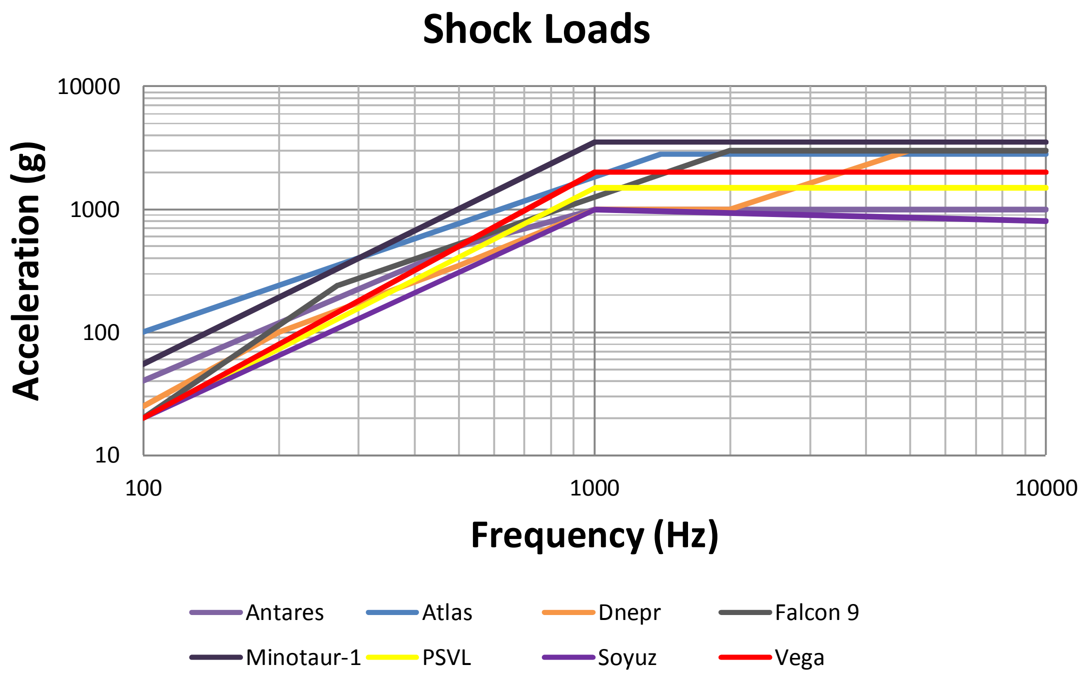


Figure 17 - Shock loads of selected launchers

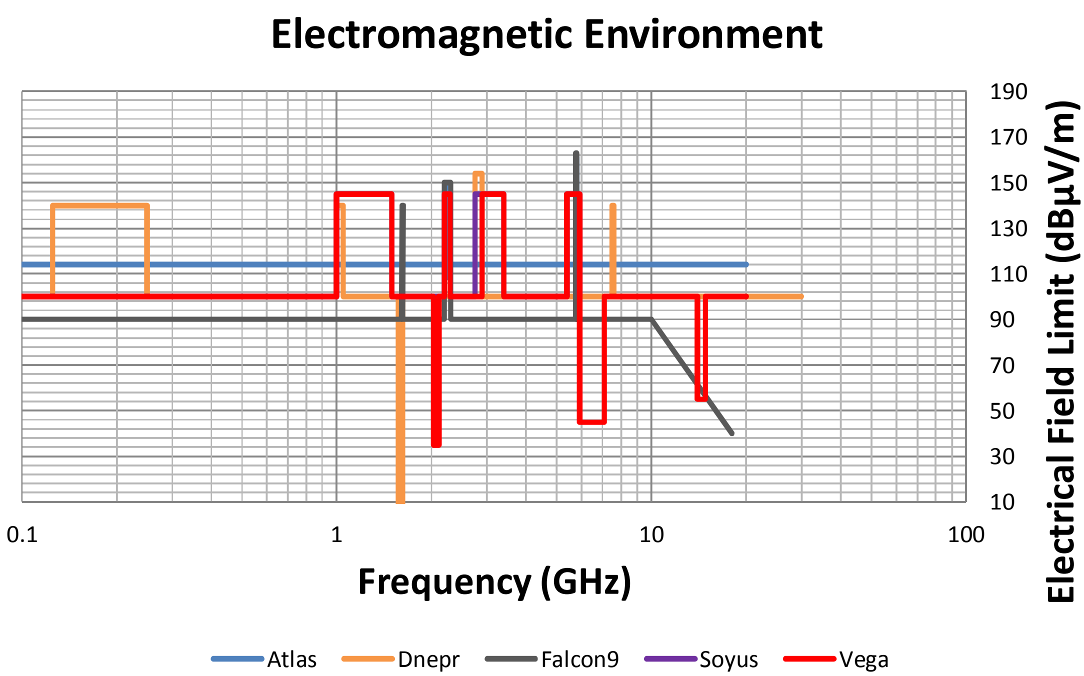


Figure 18 – Worst Case Electromagnetic Environment of selected launchers

3.3.3 Thermal Environment

Ground operations

Ground operations have a fairly known and controlled environment both at the S/C preparation stage as well as at the pre-launch under the fairing stage. Minimum possible temperature on the chosen launchers is 5 ºC and maximum is 35 ºC. As for humidity, it varies from 30% to possible 80%. These are summarized in Table 5.

Launcher	S/C preparation		Pre-launch at fairing	
	Temperature (ºC)	Humidity	Temperature (ºC)	Humidity
Antares	15.5 to 25.5	30% to 60%	15.5 to 25.5	30% to 60%
Atlas 5	21 to 27	50±5%	6 to 26	20% to 50%
Dnepr	21 to 27	<60%	5 to 35	<80%
Falcon 9	21±5	50±5%	10 to 27±5	20% to 50±5%
Minotaur 1	23±5	45±15%	13 to 29	45±15%
PSLV	-	-	10 to 15	45±5%
Soyuz	23±2	55±5%	10 to 25±2	55±5%
Vega	23±2	55±5	11 to 22±1	55+5%

Table 5 – Temperature and Humidity conditions on ground operations of selected launchers

Flight environment

Flight environment during launch is characterized in two different phases, before fairing jettisoning and after fairing jettisoning. Expected maximum temperature is 93ºC and maximum heat flux expected is of 1135 W/m^2. Table 6 summarizes the information for the selected launchers

Launcher	Before fairing jettisoning	After fairing jettisoning
Antares	Below 93 ºC	N/A
Atlas 5	Below 93 ºC	Heat flux decreases to 1135 W/m^2
Dnepr	Thermal flux acting on the S/C from the inner surface of gas-dynamic shield will not exceed 1000 W/m^2.	N/A
Falcon 9	Below 93 ºC	Below 93 ºC
Minotaur 1	Below 93 ºC	Below 93 ºC
PSLV	120 dynamic shield will not exceed 1000 W/m^2.	N/A
Soyuz	800 W/m^2	1135 W/m^2
Vega	1000 W/m^2	1135 W/m^2

Table 6 - Thermal conditions in flight environment of selected launchers

3.3.4 Space Radiation Environment

Unlike earth atmosphere protected environment, the space environment majorly stresses S/C components to several and hazardous types of cosmic radiation. Radiation in space has its origin in varied sources of emission in or beyond our solar system. The two main types of radiation are Galactic Cosmic Rays (GCR) and the radiation emanated from the Sun. Cosmic rays are mainly composed by energetic protons and heavier ions with about 2% of electrons and positrons, with a relative constant fluency over time. Sun originated radiation is composed mostly of low energy particles, mostly protons, also referred solar wind. Solar flares and solar coronal mass ejections, though not constant as the solar wind, these events create very high energy protons and heavy particles, having potential to even deform the Earth's magnetic field.

Earth's Magnetic field interaction with solar related events and cosmic rays aforementioned result in trapped particles known as the Van Allen radiation belts [80]. Van Allen radiation 2 main belts extend from an altitude of about 1,000 to 60,000 kilometers (Figure 19 refers). The inner belt (In average 1000 km to 6000 Km above Earth's surface) is formed mainly by protons and electrons whereas the outer belt (13,000 km to 60,000 km) is formed mainly by trapped electrons. Inner belt, which can go as low as 200 km above the earth's surface in the area of the South Atlantic Anomaly is the only directly affecting with satellites in the LEO orbits such as CubeSats.

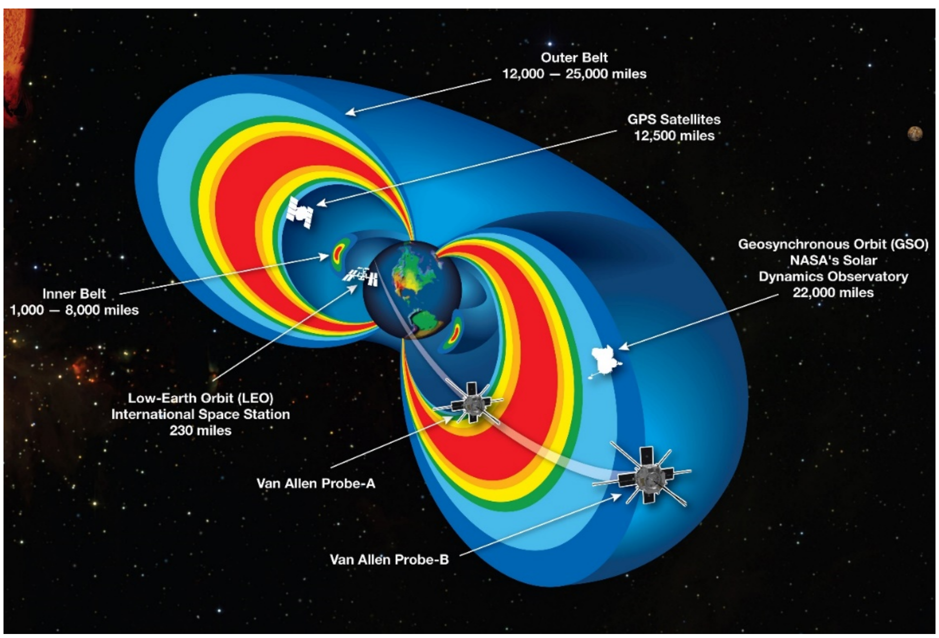


Figure 19 – Van Allen Radiation belts illustration[32]

Total Ionizing Dose

Total Ionizing Dose is measured in terms of the absorbed dose by the S/C mostly due to electrons and protons. TID can highly impact the devices, namely electronics on a S/C. Long term exposure to

[32] http://www.nasa.gov/sites/default/files/images/730056main_20130228-radiationbelts-orig_full.jpg - 2016

radiation can cause increased device leakage as well as power consumption, device threshold shifts, decreased or out of spec functionality, clock changes, among others.

A SPENVIS[33] analysis (ESA's open access radiation tool) has been performed in order to estimate the maximum levels of TID predicted for the ISTNanosat-1. For this a 55º equatorial circular orbit has been considered with different distance and mission time scenarios and for comparison purposes a polar 85º elliptic orbit has been analyzed. All simulations were done for solar maximum activity (worst case scenario). It is visible in these analysis (Figure 20, Figure 21, Figure 22, Figure 23 and Figure 24 refer) that for very thin aluminum shielding electron prevail as the main source of TID. At thicker levels of shielding, the trapped protons have higher influence in the TID and dictate its levels for the S/C. The summary of the analysis performed is depicted in Table 7.

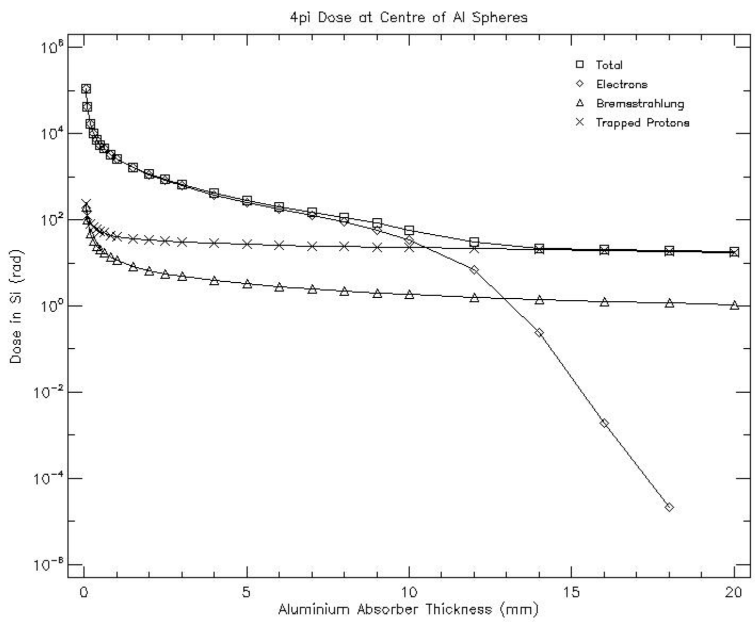


Figure 20 – Radiation levels for a 1-year circular orbit R=400 Km per Al shielding thickness

[33] https://www.spenvis.oma.be/ - 2016

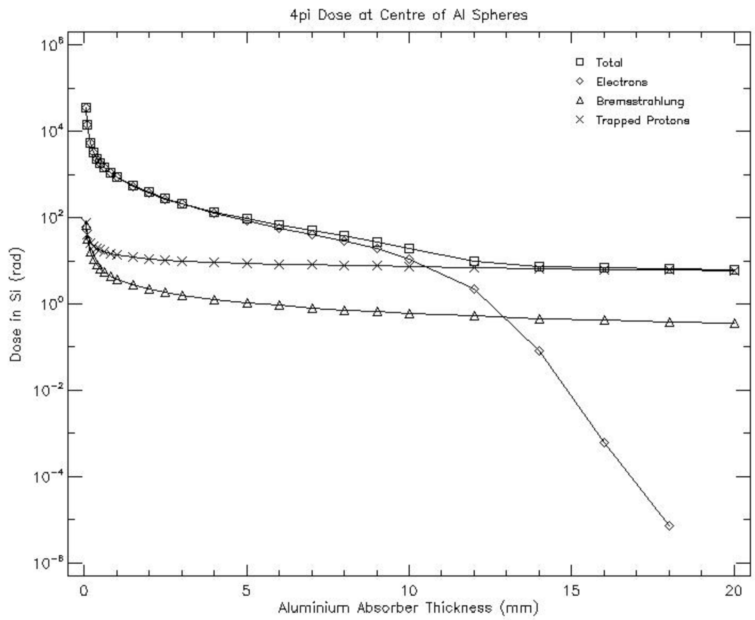


Figure 21 – Radiation levels for a 4 months circular orbit R= 400Km per Al shielding thickness

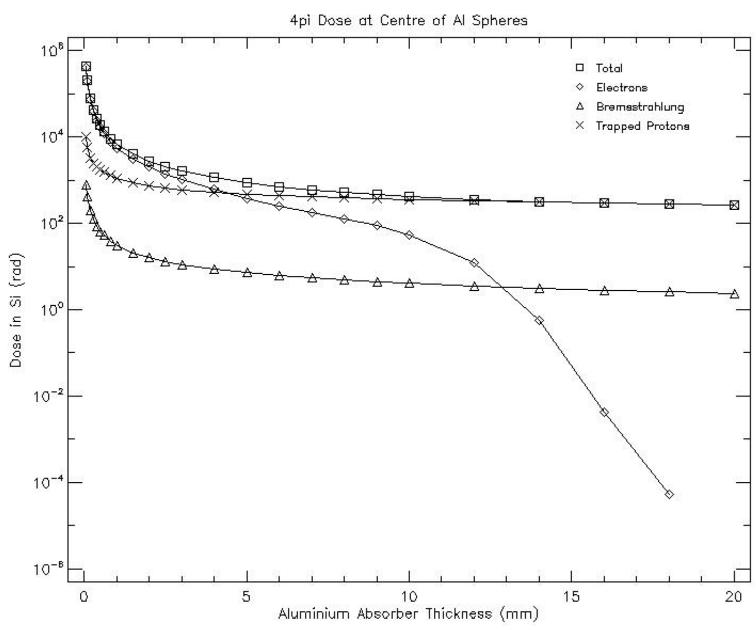


Figure 22 – Radiation levels for a 1-year circular orbit R=1000 Km per Al shielding thickness

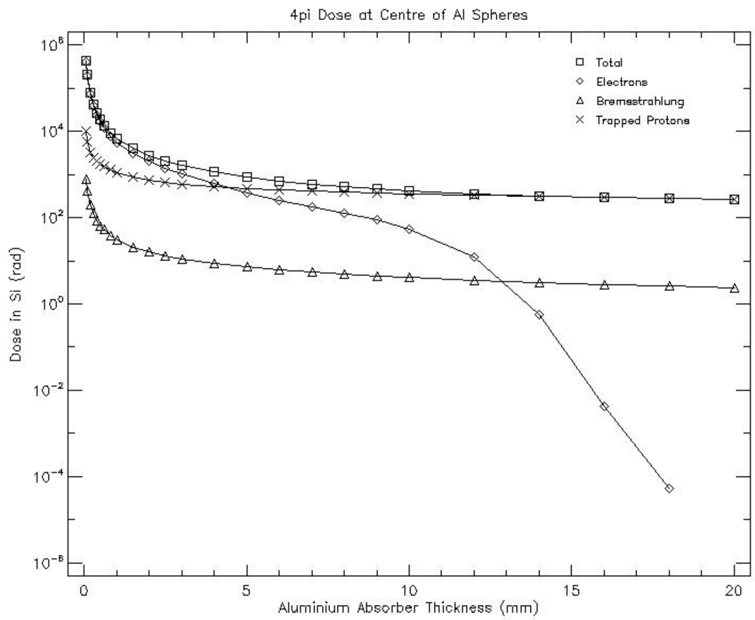


Figure 23 – Radiation levels for a 4 months circular orbit R=1000 Km per Al shielding thickness

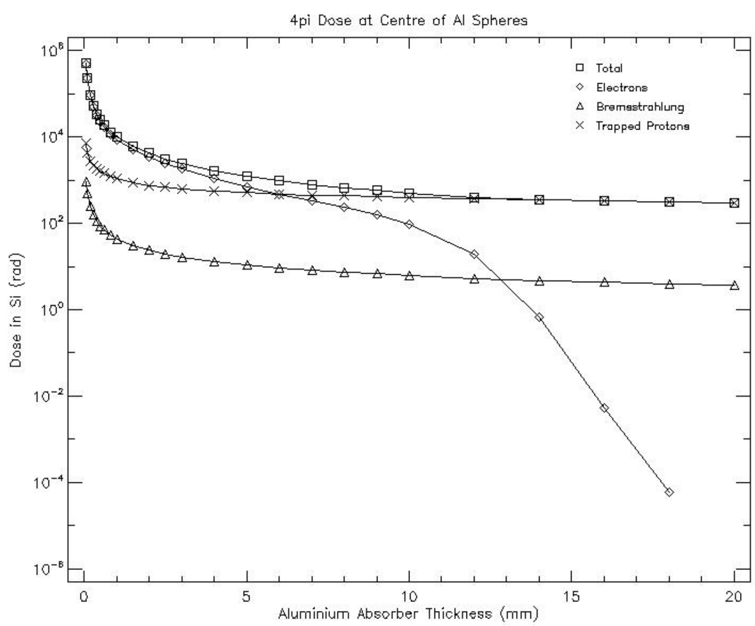


Figure 24 – Radiation levels for a 1-year polar orbit AP=10000 Km per Al shielding thickness

Mission time (years)	Orbit	Distance (Km)	TID (rad)(Si) with 1mm Al shielding
1	55 º Circular	400	$4x10^3$
0.33	55º Circular	400	$1x10^3$

1	55º Circular	1000	$2x10^4$
0.33	55º Circular	1000	$9x10^3$
1	85º Elliptical	Perigee 400 Apogee 1000	$1x10^4$

Table 7 – Total Doses for selected orbital scenarios

Most COTS with space heritage in use can withstand the radiation levels compiled above even for the very low shielding considered (1mm). Nevertheless, minimum time of mission requirement as to be considered in order to validate the radiation environment and the applicability of all the COTS used.

Single Event Effects

Single Event Effects (SEE) result from single high energy particles (cosmic rays and high energy protons) hitting electronic components of the S/C, with different possible consequences. These disturbances are a result of the ionization caused by the particles (Figure 25 refers). Single event upsets (SEU) are the most common non-destructive (soft errors) appearing normally as transient pulses in logic or support circuitry, or as bit flips in memory cells or registers. Other non-destructive SEE can be Single Event Functional Interrupts (SEFI) that lead to a temporal loss of the device's functionality. These can be recovered via reset or power cycle and often induced by SEU in control registers. Single Event Transients (SET) lead to transients on external signals in e.g. comparators or internal transients in e.g. CMOS, leading to erroneous data.

Destructive Single events (hard errors) are the severest consequences of high energy particles on components, since they interrupt the devices function and permanently damage the device. Single Even Latch Up (SEL), Single Event Burnout (SEB), Single Event Gate Rupture (SEGR) and Single Event Hard Errors (SHE) (this is a list of the most commons SEE. Exhaustive SEE are depicted in ECSS-E-ST-10-12C [81]).

SEL is triggered by heavy ions, protons and neutrons and can cause circuit lockup and/or catastrophic device failure. A component such as a CMOS can have its source or drain short circuited to ground due to the creation of a conductive "tunnel" by the hitting particle.

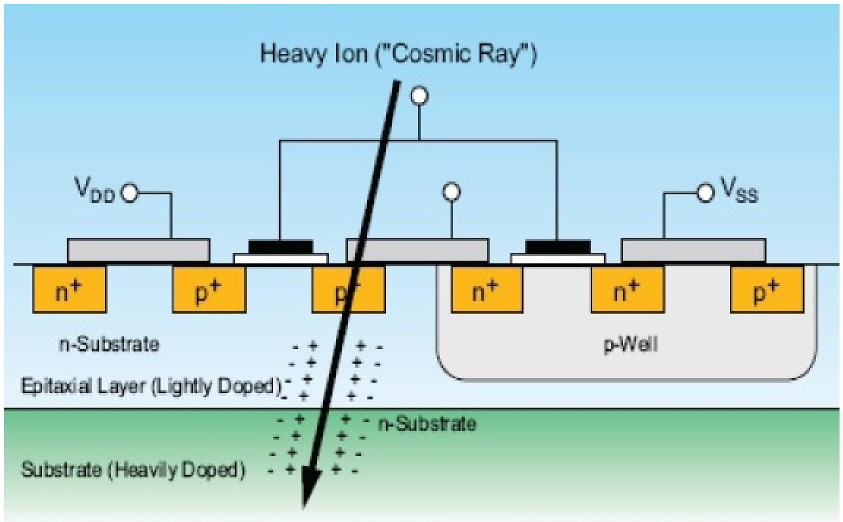


Figure 25 – Single Event Effect illustration on EEE[34]

SEB are always destructive events triggered by heavy ions and possibly protons and neutrons. Not turning power off devices after a latch up event may lead to a burn-out of the device.

SHE result on individual cells to be able to change state, can be described as stuck bit in memory. This can occur due to micro latch ups and power cycling is required to try to reverse the effects of the hard error. Finally, SEGR are a synergic event from TID and SEE, triggered by heavy ions and destructive for the devices.

[34] http://www.esa.int/var/esa/storage/images/esa_multimedia/images/2012/12/radiation-driven_single_event_effect/12469842-1-eng-GB/Radiation-driven_Single_Event_Effect.jpg - 2016

4

4 ISTNanosat-1 development status

ISTNanosat-1 subsystems and their known architectures and operational principles are hereby presented and resumed. For the sake of AIV planning, risk management and test loads it is of relevance to understand the system particularities and the critical components.

4.1 Structure and Mechanisms (S&M)

ISTNanosat-1's main frame structure is a custom designed 1U skeletonized structure as depicted in Figure 26. The structure will be made of lightweight rigid aluminum and all-stainless steel fasteners are to be used. Furthermore, the structure is fully alodyned for electrical conductivity and wear surfaces are hard ionized. The skeletonized structure will be complemented by covering elements, namely the solar panels and the internal structure.

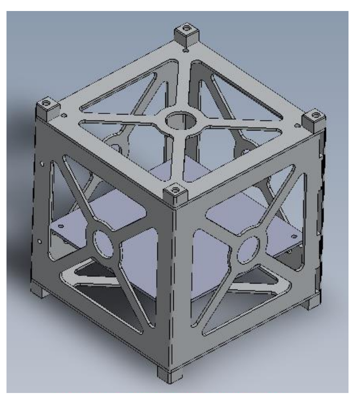


Figure 26 – ISTNanosat-1 structure 3D

The internal structure of ISTNanosat-1 (Figure 27 refers) will correspond to a typical CubeSat structure based on PC/104 boards. Both primary (HUMSAT board) and secondary payloads will be implemented in the form of PC/104 therefore no special conditions for this elements are expected. Note that ISTNanosat-1 is thought to contain both ADCS and CDH in the same PCB.

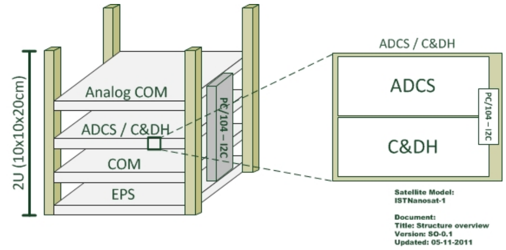


Figure 27 - ISTNanosat-1 internal structure overview

ISTNanosat-1 possesses no mechanisms besides the mandatory power switches. The switches will allow for the ISTNanosat-1 to be compliant with all the requirements demanding that CubeSats remain off during all the launch event and only after realize can be turned on. This is accomplished by the mechanism depicted in Figure 28.

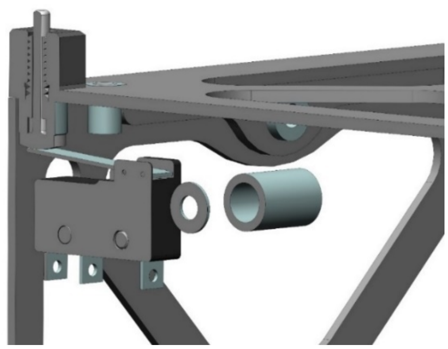


Figure 28 – ISTNanosat-1 power switch mechanism

No structural critical items for this subsystems are identified at this stage of development. Additionally, no major risk is predicted as long as load factors and material choice are kept within the known frame for CubeSat applications.

4.2 Electrical Power System (EPS)

The Electrical Power System of ISTNanosat-1 architecture is based on a Maximum Power Point Tracker controlled system. This architecture is depicted in Figure 29, and is composed by 4 sections: input block connected to the Solar panels, a regulated bus, a power distribution model and an independent microcontroller. The input block serves to regulate the voltage from the solar panels, the regulated bus is used for proper energy storage in the battery, the power distribution model is used to generate the regulated output voltages used to feed the other subsystems of the S/C and the dedicated microcontroller controls the operation of this subsystem whilst releases other systems from monitoring the EPS.

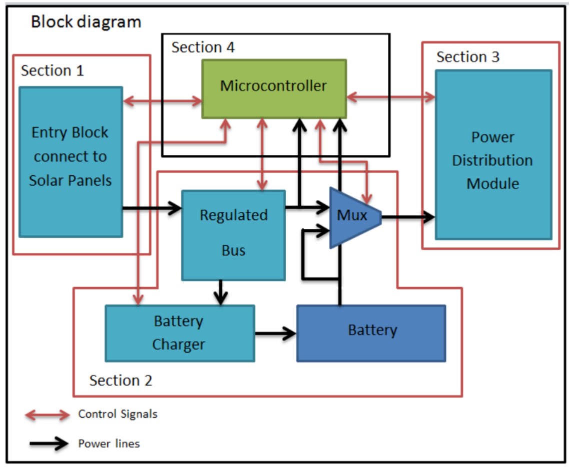


Figure 29 – EPS global architecture block [82]

Upon the writing of this dissertation only the first Engineering approach has been implemented and from the architecture depicted above several points remain to be assessed and tested. In what critical items is concerned, the microcontroller, the solar cells and the battery cells are regarded as critical and the respective consideration are depicted following.

Microcontroller added at EPS level is a late development of the EPS architecture, and though promising, remains to be validated. In what components are concerned the microcontroller chosen was a Texas Instruments MSP430F2112. The MSP430 series of components is rated to operate between -40ºC and +85ºC and it is one of the microcontrollers of choice for CubeSats with proven flight heritage. The choice of this microcontroller is mostly attached to the very low power consumption as it is not SEE or SEL proven and have been tested and found stable up to 10 Krad (Si) TID [83].

Solar Cells chosen were 30% Ultra Triple Junction (UTJ) Solar Cells from AzurSpace, flight proven and commercialized in the main CubeSat parts shops. Due to the specify of this component the choice of CubeSat commercial parts is regarded as a major risk minimizer.

Finally, the Battery Cells chosen was Varta's Pack Cell LPP 503562 DL. These batteries are lithium polymer batteries, a concept commonly used in CubeSats, though no space heritage was identified for the chosen model in particular. The operation temperature characteristics encompasses charge between 0ºC and 45ºC and discharge between -20ºC and 65ºC. These temperatures, more in specific the charge temperature high temperature limitations, determines a minor project risk that should be taken into consideration and further assessed via thermal simulation or actual testing. Since at charge temperatures within the S/C are expected at their highest and that space environment temperatures can superpose the 45ºC referred the risk of limitations in the charge mode are inherent. These thermal characteristics are nonetheless typically accepted and used in other CubeSats missions.

4.3 Attitude Determination and Control System (ADCS)

As previously referred in this dissertation, the ADCS is coupled with the CDH in one same board. Nevertheless, the two subsystems are hereby treated separately for they involve different and specific considerations.

The ADCS subsystem for the ISTNanosat-1 is composed by an IMU (Gyroscope, Accelerometer and Magnetometer), a 3 Axis magnetometer, a GPS receiver and Magnetorquers. An additional temperature sensor is hereby considered though not a functional part of the ADCS. The general architecture is depicted in Figure 30.

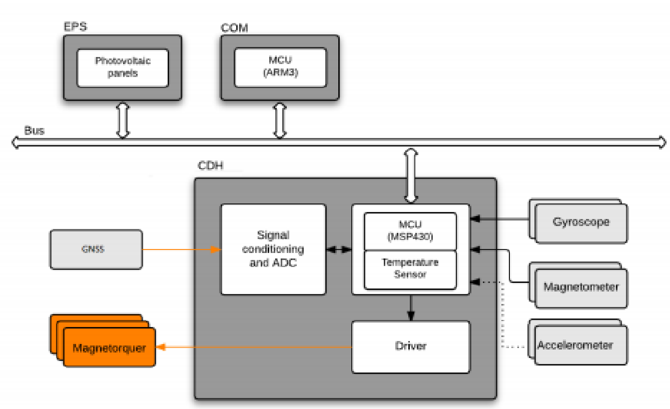


Figure 30 – Joint Board ESP+CDH architecture

The IMU chosen is MPU9250 from InvenSense, is composed of three different of 3-axis sensors: a gyroscope, an accelerometer and a magnetometer. Used for precision tracking of the S/C's attitude, this device has a gyroscope with full-scale range up to ±2000 °/sec (dps) (user-programmable), an accelerometer full with-scale range up to ±16 g (user-programmable) and a magnetometer with full-scale range of ±4800 μT. In what operation is concerned the device is rated to operate between -40ºC to 85ºC. The MPU9250 as no know space heritage. To tackle this constrain, a secondary redundant attitude sensor has been implemented in the design.

This sensor is a 3 axis magnetometer model HMC5983 from Honeywell. The main purpose of this sensor is the aforementioned redundancy to the IMU as well to enable different user programmable scales from a sensor range of ±8 gauss. This device is a temperature compensated three-axis integrated circuit with magnetoresistive sensors plus an ASIC containing amplification, automatic degaussing strap drivers, offset cancellation and a 12-bit ADC. The operational temperature is between -30ºC and 85ºC, while non-operational temperature ranges from -40ºC to 125ºc. These sensors have known space heritage with no failures to date on Perseus-M for over one year and are planned for Aalto-2. They make use of the same AMR technology with flight heritage on LusoSpace[35] ESA space qualified Magnetometer.

The GPS receiver is yet to be identified for the ISTNanosat-1. At this stage a Piksi[36] based module is being considered. This device is intended for several other CubeSat mission developments mostly due to its reduced size, low power consumption and centimeter accuracy.

In what actuators are concerned, in order to control de attitude of the S/C, Magnetorquers are planned to be developed in-house using air core concepts These actuators are yet to be designed but an architecture with copper coil windings on the inner faces of the ISTNanosat-1 is predicted.

Finally, and though not part as an ADCS instrument, the system is monitored with a digital temperature sensor from Analog Devices, ADT7320. This device has an operation temperature ranging from -40ºC to +150ºC encompassing all the expected mission temperatures range. Though no space heritage is known, this sensor is planned to fly in the Canadian ECOSat[37].

4.4 Command and Data Handling (CDH)

The Command and Data Handling Unit (CDH) is the main orchestrator of the ISTNanosat-1, managing all the other subsystems. Therefore, this unit must be designed and built both reliable and efficient. As previously referred this unit is coupled with the ADCS subsystem. A similar MCU MSP430 is used as the onboard computer. CDH interfaces with directly with the Beacon system, the aforementioned temperature sensor and ADCS sensors and actuators, GSMK modem and naturally the PC/104 bus. I2C is used as the primary way to interface between the different components of the subsystems. For reliability and redundancy reasons if some problem occurs with a I2C component two mechanisms are implemented: First a 2 redundant I2C bus approach was put in place; Second even if all the redundant I2C bus fail, the CDH cuts the bus communication with that component via an enable line. If the problem occurs with the CDH, the COM subsystem masters a SPI with 3 shared lines and enable lines for each slave. Additional measure regarding the interface with remaining components are hereby summarized and the general architecture of the subsystems is depicted in Figure 31:

The GPS receiver requires direct communication with the CDH or the ADCS MCUs, therefore the data transmit line is shared by both the CDH and ADCS systems. For data reception, it is guaranteed by one line by the CDH and another line by the COM as inputs of a Multiplexer whose output is connected to the data reception input of the GPS. The selection of the MUX is controlled by a bit of the CDH with a bit of the COM. With this configuration, if one subsystem fails, the other subsystem can revert the selection bit of the subsystem that failed and can control the GPS.

The GMSK modem provides one I2C interface and one SPI interface. The SPI interface will be used to allow the direct data transfer between the COM subsystem and the modem. As a redundancy

[35] www.lusospace.com - 2016
[36] https://www.swiftnav.com/piksi.html - 2016
[37] http://www.ecosat.ca/ - 2016

measure a direct connection between the modem and the CDH MCU is established via the internal CDH board I2C.

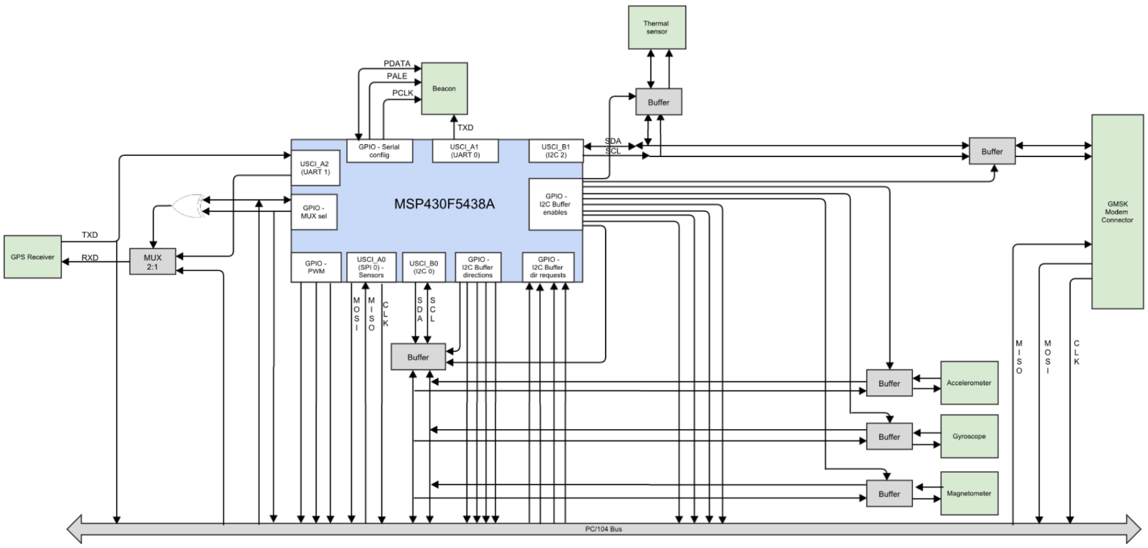


Figure 31 - CDH subsystem interface architecture

4.5 Communications (COM)

The COM subsystem is responsible for processing all in/output space-link digital data and to route information to other systems in the ISTNanosat-1. The COM handles the connections not only with ground stations but possibly other satellites. Through the links established the information regarding spacecraft internal conditions, tracking guides, commands and payload data is exchanged.

For the Ground Station (GS)/Satellite link, AX.25 protocol will be used. The type of frames used is Unnumbered Information (UI) which provide a simple implementation (e.g. no sequence number tracking is needed). Inside the Information field, which can have any size (as long as both receiver and transmitter support it – default is 256 bytes), CubeSat Space Protocol (CSP) packet will be transmitted, which permits layer 3 and 4 addressing, besides the layer 2 provided by AX.25 (Figure 32 refers).

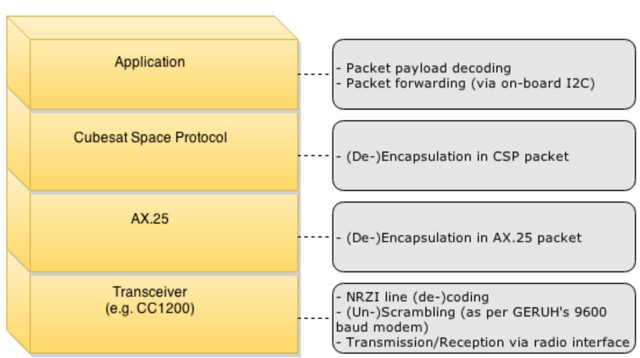


Figure 32 – Ground S/C link COM frame.

In the scenario of failure of the COM encoding and decoding capabilities, the CDH processor has the ability to perform redundant actions i.e. to do the packet encoding and decoding. However, and since this processor is both slower and more memory limited, CSP is not to be implemented in it; and only AX.25 shall be used.

LPC1833 ARM Microcontroller is manufactured by NXP semiconductors is considered to for the COM subsystem. This component is rated to operate up to 125 ºC and has a storage temp range from -65ºC to 150ºC. No space heritage is known to this component.

4.6 Payload

In order to perform its primary and secondary missions the ISTNanosat-1 will carry onboard two payloads. The primary payload will consist of the HUMSAT payload. This payload shall be composed of the sensor spacecraft communications radio (a PC/104 independent board) and the Payload independent antenna and shall perform the following tasks:

- The HUMPL payload shall be capable of receiving data from user sensors.
- The HUMPL payload shall transmit data to the sensors of the users.
- The HUMPL payload shall provide housekeeping data about its internal state to the OBDH subsystem.
- The HUMPL payload shall provide an estimation of the Doppler error in the frequency received from the data of the sensors.

Secondary payload inclusion is not yet certain at this stage of development, though it shall be constituted by an independent PC/104 PCB that includes the Rad-Hard ADC and the corresponding data processing unit. This unit shall be provided by UNINOVA though its inclusion is so far pending on the match between the data transmission capabilities of the ISTNanosat-1 and the throughput required for this secondary payload.

5

5. Quality Assurance and Risk Management

5.1 Configuration and Documentation Management

Documentation is essential in face of the staff turnover typical of academic environment projects and to assure proper tracking of all the phases, developments and updates in the project. In order to guarantee both compliance with the industry standards and know-how maintenance and project information tracking the following configuration and documentation management where implemented as follows.

5.1.1 Documents Requirements

The project outcomes shall be fully documented, as a minimum, making use of the following key documentation (Table 8 refers) as per industry standards and requirements, launcher requirements and CubeSat specific requirements (As agreed within project scope):

Document Title	Summary of Content
General description document	Provides a general description of the satellite's systems and ground segment.
Preliminary review report	Preliminary system concept + Confirm feasibility + Results of technical analysis
System requirements document	Complete system + Technical Studies + Results of pre-development activities + Subsystem design + Preliminary verification plan
Preliminary design report	Design + interface control + budgets of link and data analysis + verification plan + subsystem preliminary

	design + Preliminary operations plan
System design report	Final systems design + final subsystems design + final verification plan and operations plan + engineering models and critical components + engineering models tests
Quality Report	Qualification models + Results of qualification models tests + System-level tests + User manual + Results of the project
Acceptance Review Report	Flight models and test results + Result of system-levels test + User manuals + discussion results
Flight readiness review	Final version of all documentation + Mission operations plan
System performance report	Analysis of the mission
Disposal report	Information about system disposal.

Table 8 – Project Documentation definition

The tailoring of the documentation requirements was performed for ISTNanosat-1 project through the tradeoff between the complexity of implementation and actual value to the project results.

5.1.2 Document Management

In order to effectively manage the expected documentation of the project, even more when additional project documentation shall be performed on top of the thesis developed in the scope of the project, information control strategies were identified. The goal of this Information/Document Control strategy here presented is to ensure a full access to all the information of the project to each of its intervenient. Hence this allows for more productivity in the performance of each individual task, traceability of the work performed and more efficiency in problem detection and resolution at a system level.

Documentation Identification

ISTNanosat-1 document referencing scheme has been devised to follow quality norms that guaranty that each document is to be uniquely identified and easily traceable. The following reference for document naming scheme has been implemented:

File name: <Project (Text - 3 to 7 characters) > - <Subsystem or Field (Text – 4 characters max)> - <Type (Text - 3 characters max)> - <Number (Numeric – 2 characters) ><Issue (Text – 1 character)>

The text fields previously identified, *project*, *subsystem or field* and *type,* must use the project document identification assigned nomenclature which is defined as follows:

Project

NS1 ISTNanosat-1

Subsystems or Field

ADCS	Attitude Determination and Control System	CDH	Command and Data Handling
COM	Communications	EPS	Electrical Power Supply
PL	Payload	Q	Quality
SE	Systems Engineering and Project Management	SM	Structures and Mechanisms
TC	Thermal Control		

Table 9 – Subsystem/Field Acronyms

Type

NCR	Non-Conformance Report	RQ	Requirements
AIVP	Assembly integration and Verification Plan	SD	Specifications Document
PR	Procedure	SR	Status report
PRS	Presentation	TN	Technical Note
QAP	Quality Assurance Plan	TP	Test Procedure/Test Plan
RE	Reviews	TR	Test Report/Test Result
RP	Report		

Table 10 – Document Type Acronyms

An example of the detailed scheme is the Quality Assurance Plan, which detailed explanation follows:

<Project>	**-**	***<Subsystem or Field>***	**-**	***<Type>***	**-**	***<Number>***	**-**	***<Issue>***
NS1	**-**	**Q**	**-**	**QAP**	**-**	**01**	**-**	**A**

Record, Approval, Maintenance procedures

All the documentation of the ISTNanosat-1 project shall be, in the possible measure, keep in an online database. Natural exceptions are parts/boards logbooks, though an online transcription of these is highly recommended and to be considered in the scope of the project. This database shall be accessible to all project intervenient and it is of the responsibility of the Quality Assurance and Project Managers to guarantee the proper up to date status and accuracy of the documentation.

The documentation storage is constituted by two key elements: A version storage service and a web based wiki. The version storage service is the main documentation storage tool of the project and is to be used mandatorily by all project intervenient and it must contain all the project information and documentation. For this purpose, a SVN client has been implemented. Here working version of the documents, software, design (mechanical, PCBs, electronics) as well as final releases have their versions controlled at the different levels of their development.

The document version database has been implemented with the structure depicted in Figure 33.

Work in progress documents are kept in trunk folder. A review and approval definition procedure had been implemented to all the project related documents. The document under development shall be classified as 'in progress' and shall be stored in the subject folder within the authors subfolder until the release for review.

The release of a document for review shall be communicated by the author(s) and the documents released for review are then stored in tag folders. The reviewing process shall take place by the lead responsible (s) and all changes in the document shall be tracked in the word processor software to be later implemented by the author(s). Peer review might as well be put in place, though corrections and comments of peers must be accepted by the lead responsible(s) to be implemented in the document. Subsequently to the review, correction and approval of the document this shall be place in the subsystem area made available in the web based wiki to the remaining project intervenient. Finally, obsolete documentation is kept in for traceability.

The web based wiki serves as an easy access portal and shall contain key information of each subsystem or field of the project, such as the status of development or general reference documentation in the wikis intranet. It serves also as the platform for information sharing with the general public and project main web page. Here the final versions of project documentation are upload for easy access to all the team members.

After properly approved a document can only be altered through a Request for Change. The Request for Change must be approved by the document initiator and Project Management before change can be implemented. It is up to the Request for Change initiator to update the document after approval and to properly store the previous version of the document.

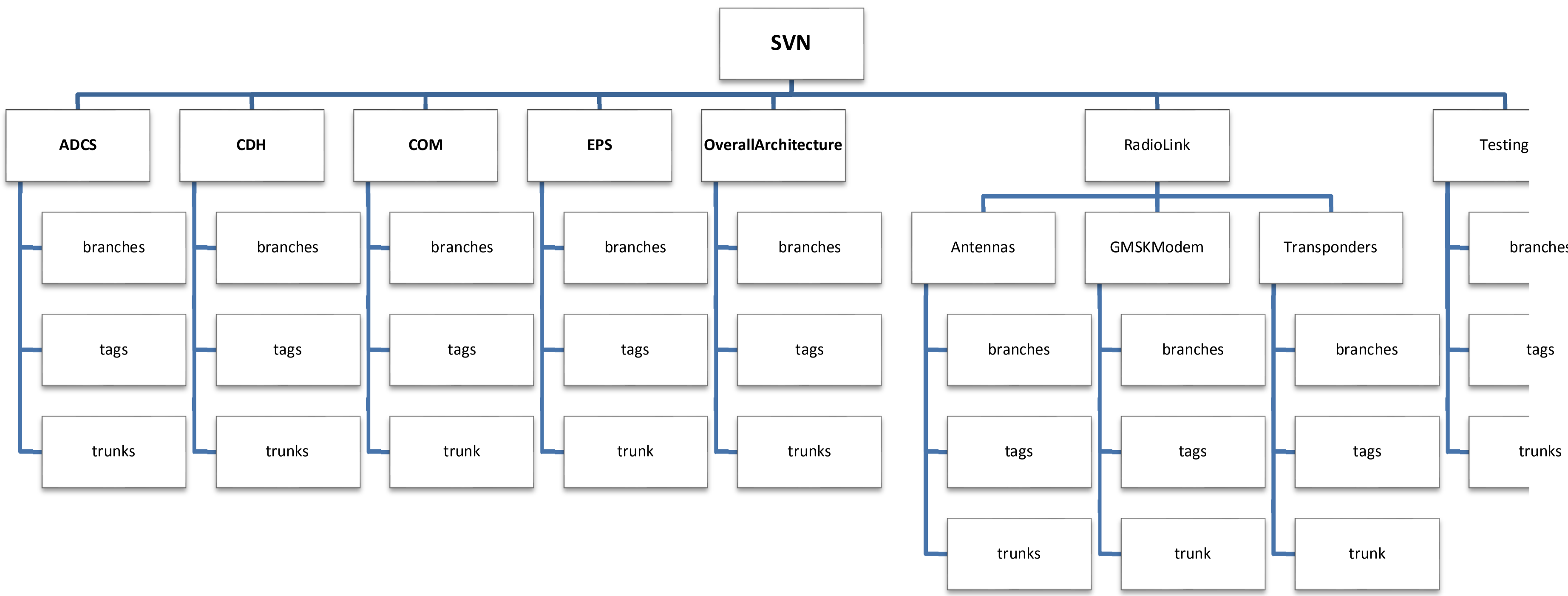


Figure 33 – Project SVN Structure

5.1.3 HW and SW Traceability

All HW item must be identified by a part number and, if necessary, a serial or lot number such that every single item has a unique identifier. The part number must be related to engineering drawings number. Configured SW must be identified by a unique code and version number. These number shall be used and referred in all the documentation related with the correspondent HW and SW.

5.1.4 Non-Conformance Reporting

Non-conformances refer to a departure from the established agreements (be it a work product standard, a predefined procedure, a work product specification, a document plan, an international standard…) that occur during the development process of the ISTNanosat-1. Non-conformances of the design or flight configuration shall be recorded in Non-Conformance Reports (NCR) and properly assessed in Non-conformance Review Boards (NRB).

Non-conformances shall be classified as critical, major or minor, with the following criteria, as per [55]:

- Critical Non-conformances are those, which may affect safety and occur during/after qualification/acceptance testing at any item level.
- Major Non-conformances are those which are not critical, but may have an impact on the defined requirements in the following areas:
  - Operation, functional or contractual requirements;
  - Reliability, maintainability;
  - Lifetime;
  - Interchangeability

Additionally, any non-conformances shall be classified as major in the following cases:

- Deviation from the qualification/acceptance test procedures and expected results at any level of integration;
- EEE component non-conformities after delivery from the manufacturer shall be classified as major, except the following non-conformances at incoming inspection, which may be classified as Minor:
  - Failures, where no risk for a lot related reliability or quality problem exists;
  - The form, fit or function are not affected;
  - Minor inconsistencies in the accompanying documentation
- Minor Non-conformances shall be considered all the remaining deviations from the requirements that have low to no impact on the project outcome.

Critical and Major non-conformances shall be notified by the raiser via the internal mail system after proper review of the facts. The following project progress meeting shall serve as Non-conformance Review Board where all relevant parts should be represented (if the NCR involves external parties, they shall be represented if possible).

Minor non-conformances shall be notified by the raiser after proper review of the facts. If considered relevant by PM, it shall be addressed the project progress meeting following the notification.

Non-conformances Review Boards shall identify the root causes of the non-conformances and implement the proper corrective actions: Rectification, request for deviation or waiver of the requirements, or use ‘as-is’ in the low risk cases.

5.2 Risk Management

As presented in Section 2.2, Risk management is the process of identification, analysis and mitigation of risks while tracking the root causes and their consequences for the project. Risk Management strategies implemented in this project are mainly based on [58]. The choice of this methodology was made on trade-off between process implementation complexity and project benefit. As a first university endeavor in Space projects and satellite developments, key project management and quality assurance procedures are to be implemented alongside with the development of the project itself. Therefore, a simple yet complete Risk management system has been chosen. Risk Management shall be implemented through the effectuation of 3 main steps:

- Risk identification
- Mitigation techniques
- Risk Monitoring

5.2.1 Project Risk Identification

Risk identification implies:

1. Creation of risk log per subsystem and general risk register
2. Identification of risks per subsystem/area
3. Name responsible person
4. Rank Likelihood (L) and Consequence (C) of risk
5. Describe rationale for ranking
6. Compute the risk rating with likelihood and consequence values
7. Average the risk per subsystem to achieve the mission risks
8. Plot mission risks (subsystem) on L-C Chart
9. Rank and classification of mission risk and subsystem risk

The risk identification is to be performed along the project according to risk definition aforementioned in cooperation with each individual key subsystem responsible. Inherently these subsystems responsible are the risk responsible person on risk concerning their subsystem.

Likelihood (L) and Consequence (C) of risk are ranked according to Table 11 and Table 12 respectively. A short rationale for the ranking is hereby provided. Computation of the risk rating is implemented multiplying the Likelihood (L) with the Consequence (C). L-C Charts shall be plotted in a risk matrix as per according to Figure 38 in Annex 2. For each subsystem risks shall be ranked as well as the mission rick for each of the subsystems.

Level	Likelihood	Probability of occurrence
1	Not Likely	~10%
2	Low Likelihood	~30%
3	Likely	~50%
4	Highly Likely	~70%
5	Near Certainty	~90%

Table 11 – Risk Likelihood Ranking

Level	Technical	Schedule	Cost
1	Minimal or no consequence to technical performance	Minimal or no impact	Minimal or no impact
2	Minor reduction in technical performance or supportability, can be tolerated with little or no impact on the program	Able to meet key dates	Budget increase or production cost increases (1% of the budget)
3	Moderate reduction in the technical performance or supportability with limited impact on the program	Minor schedule slip. Able to meet key milestones with no schedule float	Budget increase or production cost increases (5% of the budget)
4	Significant degradation in the technical performance or major shortfall in supportability; may jeopardize program success	Program critical path affected.	Budget increase or production cost increases (10% of the budget)
5	Severe degradation in technical performance; cannot meet key technical/supportability threshold; will jeopardize program success.	Cannot meet key program milestones.	Exceeds budget threshold. (>10% of budget)

Table 12 – Risk Consequence Ranking

Risk Register table is in continuous implementation according to Table 17. Risk ID shall be unique and composed by the subsystems acronym (as defined per Table 9) and an ordered number, considering the order of creation of the registry (e.g. EPS-01 shall correspond to the first risk identified for the EPS subsystem). Risk name is the name given to the risk. Shall be short and concise key identifiable information of the risk. Risk description shall contain detailed description of the risk itself. The information thereby shall be self-explanatory and fully characterize the risk. Likelihood (L) and Consequence (C), Risk rating and Rational are defined as per in section 5.2. Status shall contain the current status of the risk (open, closed, obsolete). Action shall include the actions taken to tackle the risk as well as the date of their implementation. The Risk register for ISTNanosat-1 is attached in Annex 1.

5.2.2 Project Risk Mitigation

Mitigation techniques shall consist of one of the following options

- Risk avoidance by the elimination of the root cause
- Control the cause or consequence
- Transfer the risk to different person or project
- Assume the risk in the project and continue development

Risk monitoring shall be performed by updating the mission L-C chart and registering its evolution along the different phases of the project. This shall be primary the responsibility of the Quality Manager and Project Manager.

5.3 Procurement Selection and Control

Procurement Control in the scope of the ISTNanoSat-1 project shall be performed based on best practices derived from industry standards. Though most of the subsystems of the project are in house developments, its key components are procured.

The procurements control shall incise over the selection of the procurement sources, the procurement documentation control and the inspection of incoming item.

No hard requirements for procurement sources are mandatory in the scope of this project. Nonetheless suppliers that make available the proper procurement documentation are preferred. Procurement sources with work experience on the CubeSat field should be preferred, more even for identified critical items. All procurement sources of critical items shall be listed and kept along the project. All the documentation related to the procurement of critical items shall be kept as well, either in physical format, in an exclusive folder, and/or in digital format, depending on how information is provided by the suppliers.

5.3.1 Incoming Inspections

Upon reception, all incoming supplies and related documentation shall be inspected and accepted prior to any use or permanent storage. Incoming inspection aims to verify the conformance of the procured items with the procurement documentation and shall include:

- Verification of the packaging conditions and the status of any environmental sensors on the package, if applicable.
- Visual inspection of the delivered items, with appropriated handling measures namely for critical and/or ESD sensitive parts.

- Verification of correct identification and, where appropriate, configuration identification for conformance with the ordering data.
- Performance of inspection and tests on selected characteristics of incoming supplies and/or test specimen submitted with the supplies. This might imply case by case functional testing.
- Identification of the shelf life of limited life items.
- Reference number attribution and proper storage of the parts according to Section 5.4.4.

5.3.2 EEE Component Selection Process

The selection of EEE components for the project has its focus on focus on COTS components, for both assess ability and cost budget. The selection of these COTS shall be in the scope of ISTNanosat-1 prioritized taking into account (and by order):

- Non prohibited components (as per section 5.3.3)
- Space heritage (Data of COTS from previously CubeSat flights shall be used)
- High reliability proven even if on non-space applications, fitting the purpose of the operational environment over the mission expected lifetime. For these a few critical points shall be taken into consideration namely:
    - Operating margins and de-rating of mission-critical components.
    - Radiation tolerance (both Total Ionizing Dose and Single Event Effects).
    - Outgassing properties in vacuum (both TML and CVCM values) in order to control any contamination of sensitive surfaces (e.g. optics or solar panels).
    - Avoidance of materials known to degrade significantly in the space environment (e.g. atomic oxygen, UV, vacuum), leading to failure.

Components that do not have known space heritage shall be therefore noted and its use justified according to their reliability expectation on the Components, Parts and Materials list.

5.3.3 EEE Prohibited components

For safety purposes the use of a range of components shall not be contemplated in the ISTNanosat-1 project according to the relevant industry standards. The use of components with the following characteristics shall be prohibited and any exception shall be clearly justified:

a) Limited life;

b) Known instability;

c) May cause a safety hazard;

d) May create a reliability risk.

For the purpose of this project such components have been complied in the following list:

- Wet slug tantalum capacitors (except for CLR79 construction using double seals and a tantalum case);

- Plastic encapsulated semiconductors (except when used on short-duration missions in a pressurized environment);
- Hollow core resistors;
- Wire-link fuses;
- Potentiometers;
- Non-metallurgically bonded diodes;
- Non-solid tantalum capacitors with silver case;
- Dice with no glassivation;
- Unpassivated power transistors;
- Any component whose internal construction uses metallurgic bonding with a melting temperature not compatible with the end-application mounting conditions;
- Components containing: cadmium, lithium, magnesium, mercury, radioactive material, pure tin (electroplated or fused), beryllium oxide (except if the health and safety hazards are identified in the specifications).

5.3.4 Materials, Mechanical Parts and Processes Selection

Material, mechanical parts and processes shall be selected based on the in house know-how and known best practices for CubeSat S/Cs. The demonstration of the suitability shall be made by analysis (CAD/FEM) or test, (e.g. mechanical and/or vibration tests). Both Mechanical (vibrations, accelerations, shocks), chemical (corrosion, contamination, monatomic oxygen...) and combined effects shall be taken into consideration in the choice of ISTNanosat-1 Materials.

5.4 Manufacturing, Assembly and Integration

5.4.1 Critical Items Control

Critical Items are defined as any item functionally critical to the operation, safety and reliability of the ISTNanosat-1. Single point of failure related items, as well as hardware with not proven space heritage are considered critical items. The control of these parts shall be performed via:

- Serialization and unique identification number.
- Identification and marking of all related documentation (procurement, design, manufacture, testing).
- Maintenance of the results of inspection and tests.

5.4.2 Workmanship Standards

Industry standard (mainly ECSS) workmanship standards for established processes shall be used as reference for the work performed and the subsequent inspections. It must be highlighted that these standards are not normative or binding in the scope of the project and shall be implemented on a best effort approach.

For processes where no standards are established good quality workmanship shall be based on the expertise of the project staff. General acceptance of workmanship quality is of the responsibility of the Project Manager.

5.4.3 Cleanliness and Contamination Control

Cleanliness shall be kept along the project for all the components, assembled parts, subsystems, models and CubeSat. All works (inspections, assemblies, soldering) shall be performed ideally on dedicated clean work areas. Handling, packaging and storage shall keep as well in mind the contamination control of the mission items. The Project dedicated storage shall be kept clean and all project item kept under contamination control.

The final assembly of the CubeSat shall be performed in a clean room environment. Upon assembly protective measures such as dust bags shall be implemented.

5.4.4 Handling, Storage, and Transportation of Hardware

In the handling of all H/W it must take into consideration the susceptibility and the future use in space of these parts. Therefore, ESD sensitivity and cleanliness considerations must be made and proper preventive action taken. All hardware meant to fly (including possible flight spare parts) shall be handled in the following conditions:

- In a clean environment that avoids particle contamination to the parts.
- Parts must be handled with powder free gloves.
- The personnel handling flight parts must use a lab vest (The use of lab cap and glasses is recommended).
- Parts must be handled with the use of proper ESD protections.

All parts must be stored with proper part identification and all ESD susceptible electrical parts shall be stored in ESD safe containers. A locked cabinet must be used to store the parts. Parts shall be separate as Non-Flight Hardware, Flight Hardware and Miscellaneous Flight Supplies. Each subsystem board must have its unique logbook and both board and correspondent logbook must be stored together in a unique container. The container and the logbook must be clearly marked with the identification of the respective board, so a univocal correspondence can be easily made.

During transportation all hardware shall be properly covered (in ESD bags if the hardware is ESD sensitive) and boxed in such way its integrity is not compromised by the transport, namely fix within its container.

5.4.5 Logbooks

Logbooks are a major part of the traceability of the work performed on H/W on the project. The Logbook structure devised for this project must contain all Historical information, which is significant for the operation of the item. This implies a dated record of:

- All the handling activates of the boards/models.
- Temporary installation and removals (For the FM, temporary installed item must be accounted and possibly tagged to prevent them from being incorporated in the final flight configuration).
- Non-conformances.
- Deviations.
- Tasks to be completed.

Logbooks must be kept at board level and an AIV logbook for the different models (EM, PFM…) must be kept.

5.5 Other Considerations

5.5.1 Space Debris Mitigation

Space Debris Mitigation will be guaranteed via orbital placement and according to international regulation. ISTNanosat-1 shall be place in an orbit between 400 Km and 800 Km (with an expected operational lifetime in LEO orbit of 1 year) and due to the orbit placement it shall not have a lifetime orbiting Earth longer than 25 years.

5.5.2 Safety

ISTNanoSat-1 shall comply with all the safety standards namely the ones directly imposed and connected with the launch service. As launch service is yet to be defined at the time of this work detailed safety concerns shall be defined upon launcher selection.

5.5.3 Reliability and Maintainability

ISTNanosat-1 shall limit its single point of failures to a minimum by using according design strategies. If at all unavoidable, single points of failure shall be tested to guarantee no critical or catastrophic failure occurs in the CubeSat. Implemented measures shall prevent failure propagation via the implementation for an effective failure isolation. For this Single point failures shall not:

- Affect the behavior of the power bus.
- Affect communication with the satellite.

Maintainability is a process for assuring the ease by which a system can be restored to operation following a failure.

Therefore, ground commands shall be implemented in order to recover the satellite in case of a failure (when the satellite is in Safe mode). Furthermore, the satellite shall have fault tolerance occurring in the CDH subsystem due to Single Event Effects (SEEs). The CDH shall be able to recover from these faults and this capability shall be verified as well during ground testing. Finally, the on-board software and mission parameters stored in on-board memory shall be able to be patched during the mission.

6

AIV Plan

6.1 Assembly and Integration Plan

6.1.1 Integration Sequence and constrains

The Integration procedure of the S/C, independently of the model, shall guarantee the interface connection and the proper operation at integrated subsystems and S/C level. This procedure is depicted and detailed below.

- Pre-integration Visual Inspection
- Interface Verification (mechanical and electrical)
- Electrical and Mechanical Integration
- Integration Test and Functional Verification
- Closing Elements Integration (leads, solar arrays and MLI)
- Full Functional Test

Pre-integration Visual Inspection – All the S/C flight material shall be checked before integration to guarantee their suitability for assembly. Mechanical conditions of the material shall be checked and inspected and conformance with the post fabrication and test condition is to be checked.

Interface Verification – Interfaces shall be verified, first via the verification of the proper interface documentation. If applicable, electrical interfaces shall be tested and compared to their expected values.

Electrical and Mechanical integration – The different components (mechanical, system boards, payload) shall be integrated mechanically and electrically. Detailed integration sequence is not possible at this stage of the development, though a specific integration sequence procedure shall be put in place.

Integration Test and Functional Verification – All links and interfaces shall be verified via functional checks. The Output from the S/C shall be already be within the expected range of operation at this stage and all subsystems shall be confirmed as operational at this stage of integration.

Closing Elements Integration – After functionality is verified the S/C shall be closed with the final assembly elements namely leads, solar arrays and/or MLIs. This integration implies the conclusion of the integration process and FM shall be considered closed from this stage on-boards and any opening of the S/C is strictly forbidden.

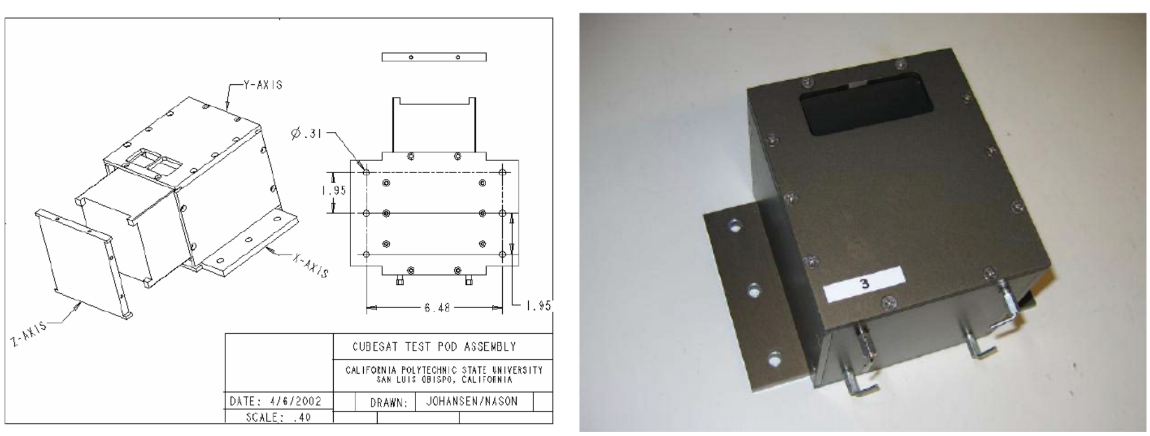
Full Functional Test – The S/C main functions and subroutines shall be tested to guarantee that all the subsystems and the S/C on it all are performing according to the expected functional parameters. A Dataset shall be recorded regarding the main functional outputs to be assessed by test. This dataset shall be used as baseline for all the space craft testing/characterization.

No major constrains in what integration is concerned are foreseen for the IST-Nanosat 1 in particular since no highly sensitive optical payload (which would require particular attention to the mounting environment and cleanliness of the environment) is to be incorporated in the S/C. Nevertheless, the proper environment requirements should be considered for the integration as per chapter 5.

6.1.2 MGSE/EGSE

Mechanical Ground Support Equipment (MGSE) and Electrical Ground Support Equipment (EGSE) shall support both the assembly and test of the S/C and ground segment of the ISTNanoSat-1. As primordial and mandatory condition the MGSE/EGSE and any other support system shall not interfere in any way in the outcome of the tests to space qualify the S/C, being in particular immune to the test conditions and compliant with safety requirements.

The proposed possible solution for the MGSE architecture is hereby depicted in Figure 34. This MGSE was developed by CalPoly and should be used as baseline for the development of a project MGSE. It shall be highlighted the external adjustment to properly secure and fit the S/S inside the Test POD, emulating the conditions of assembly of the S/C inside the launch POD. Additionally, the opening in the POD shall be put in place so that the test measurement accelerometer can measure the accelerations on the S/C, and not on the test POD.



(a) (b)

Figure 34 – CubeSat Test POD schematics (a) and Test POD (b) [84]

EGSE shall be developed to assess the functionality of the S/C, with the possibility to perform measurements during test, namely in during the Thermal Vacuum test. EGSE shall be composed PC's and laptops, miscellaneous laboratory, cabling (compatible with the test temperatures), instrumentation (oscilloscopes, spectrum analyzer), radio equipment (Radio FM/single side band, Terminal Node Controller (TNC)), pin savers, a dedicated programming, debug and communication interface board with the S/C equipment. A 2 segment architecture is depicted. It encompasses a Remote segment to acquire the communications made by the S/C, to instruct and send commands remotely to the S/C (these two segments together are comparable to the Ground Station) and a Command and Test segment to directly interface and diagnose all the different and available interface points of the S/C, particularly built to endure the test conditions. This architecture is depicted in Figure 35.

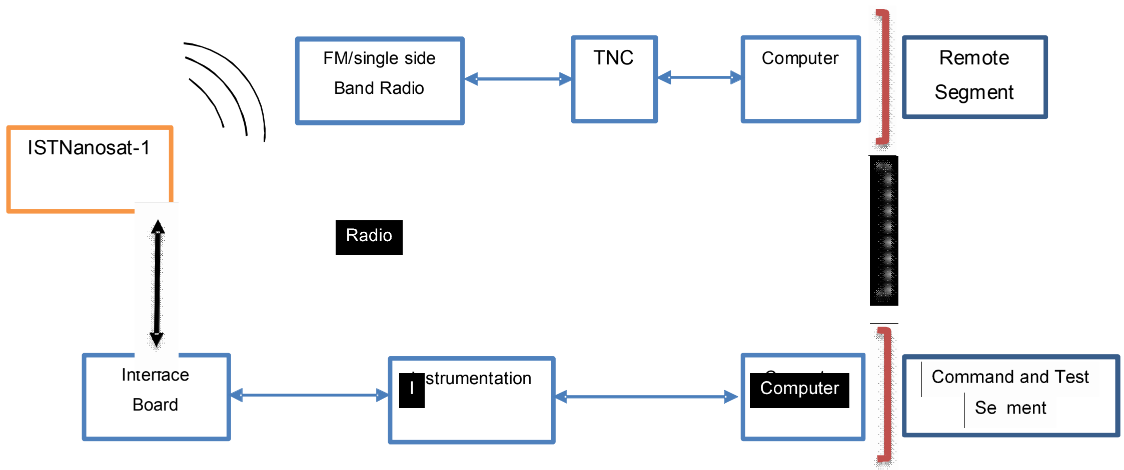


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Figure 35 - EGSE High Level Architecture

6.1.3 Payload Calibration

Considering its missions, the Payload chosen for the ISTNanosat-1 does not require specific calibration before flight. The only expected element of the S/C expected to require calibration is an DDS function generator, though is considered outside of the scope of this chapter.

6.2 Testing/Model Philosophy

Three different Model Philosophies are defined by the ECSS and used in the scope of ESA projects: Prototype, Protoflight and Hybrid approach.

Prototype approach implies all qualification testing to be performed in one or several Qualification Models (QM) and that the Flight Model (FM) shall undergo a full acceptance test campaign.

Protoflight approach implies for qualification and acceptance to be performed all at once on the same model to be flown, typically using qualification test levels and acceptance duration. This is called the Protoflight Model (PFM).

Hybrid approach aims to combine the benefits of the two previous approaches. In this approach specific Qualification tests can be carried in dedicated models for particular, and usually critical, areas (e.g. Thermo-mechanical model). Acceptance testing is to be carried out on the PFM.

Due to its inherit flexibility and good compromise between costs and reliability validation the hybrid approach has been selected for the ISTNanosat-1.

The adopted philosophy results in the implementation of 4 model levels:

1. Development Models (DM) – subsystems level
2. Engineering Mechanical Model (EM) – S/C level
3. Engineering Qualification Model (EQM) – S/C level
4. Protoflight Model (PFM) – S/C level

Development Models – These shall be composed by the development boards constituting each of the S/C subsystems.

Engineering Mechanical Model – Shall be a representative model of the fundamental structural elements of the S/C. Mechanical dummies of the Subsystems shall be used and the Model tested for mechanical stress.

Engineering Qualification Model – Complete and functional model of the S/C that is fully representative of the components and manufacturing processes to be used in the flight model.

Protoflight Model – Flight Model based on the Engineering Qualification Model. Same manufacturing processes and components must be used.

6.3 Test Program

The main verification methodology implemented in S/C projects is testing, due to the assurance given by tests results when compared to other verification methodologies in order to guarantee the compliance with the project requirements.

The approach selected for the ISTNanosat-1 aims to assess reliability and functionality has early as possible in the development process. In that sense, development models (i.e. each subsystem) are to be bench tested and tested making use of atmospheric balloons, namely the BALUA platform, for functionality. Mechanical models shall be tested for qualification level mechanical loads to guarantee the integrity of the elements bearing the major mechanical loads before these tests are applied in the costlier EQM. EQM model shall undergo the full qualification test campaign (Figure 37 refers) to guarantee the compliance of the S/C engineered. EQM model shall be kept after testing and its parts used for spare parts for the PFM. PFM shall undergo the acceptance test campaign prior to flight (Figure 36 refers). The general test program is depicted in Table 13.

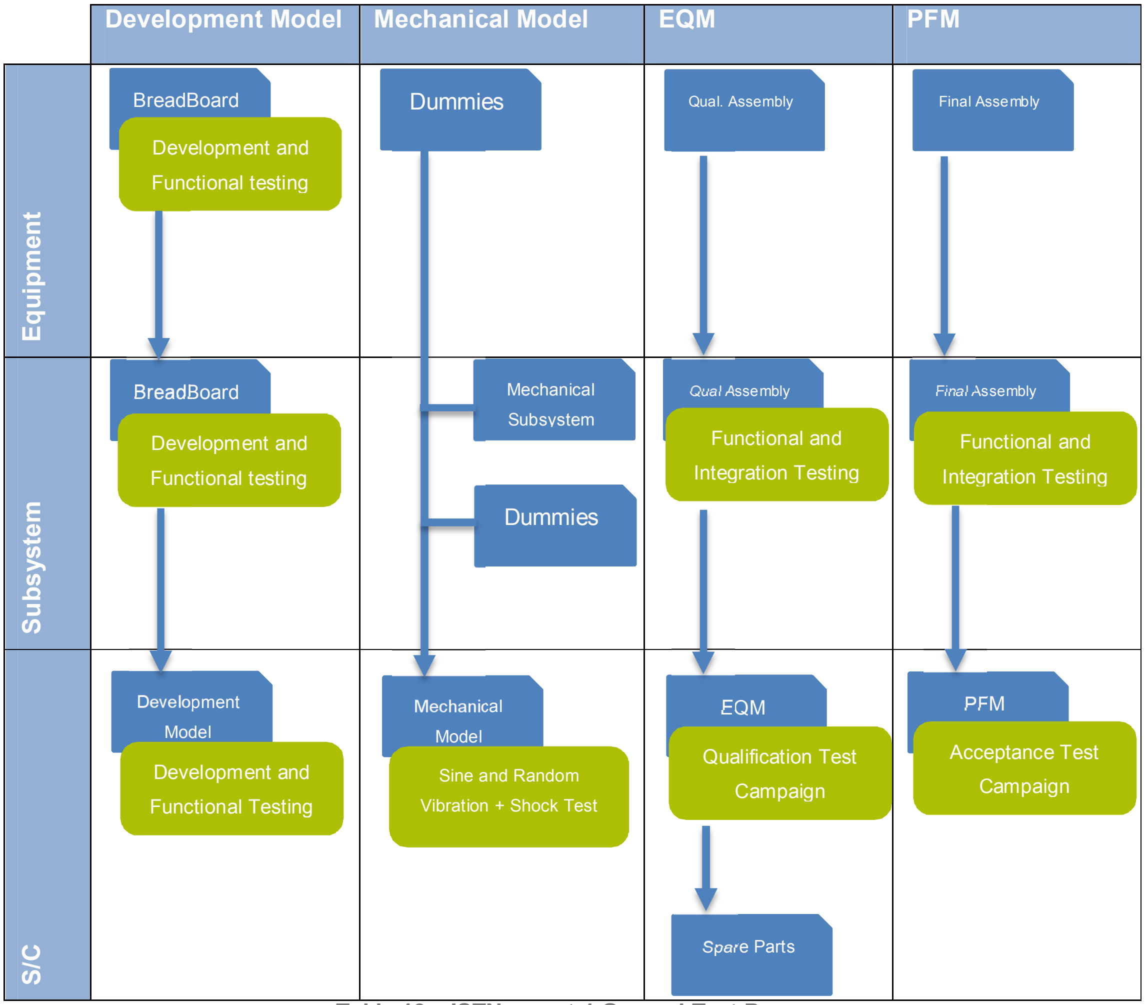

Table 13 – ISTNanosat-1 General Test Program

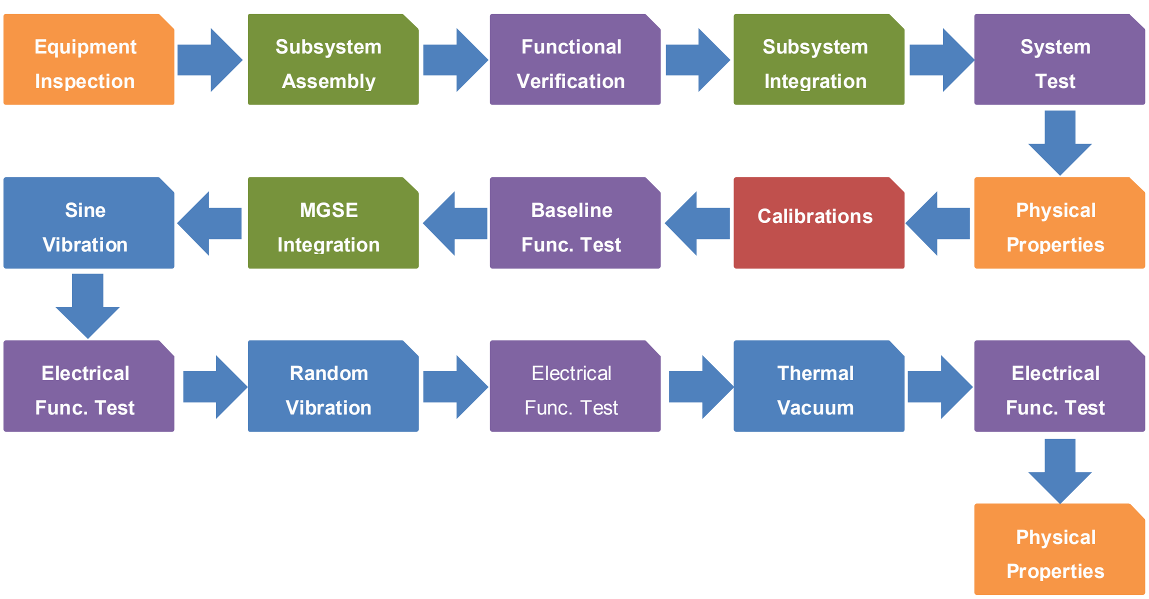

Figure 36 - PFM Detailed Test Flow

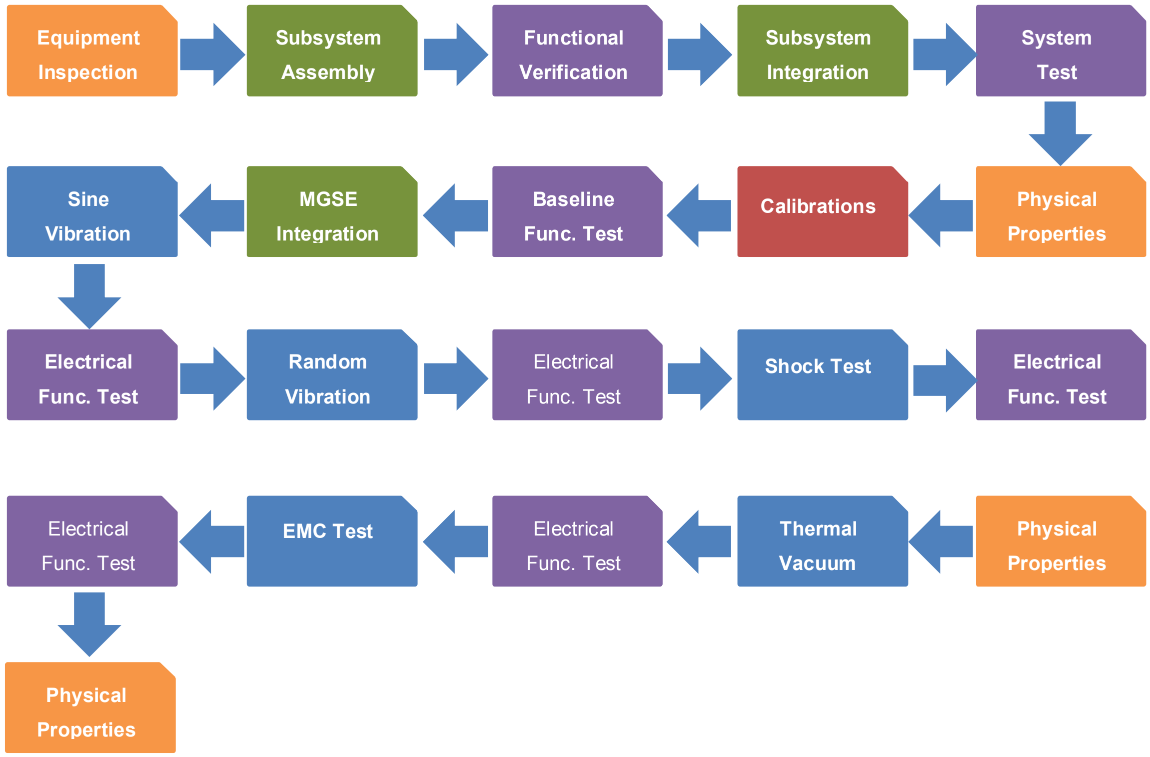

Figure 37 – EQM Detailed Test Flow

6.3.1 High Altitude Balloon tests

High altitude balloons provide the possibility of near space environment being capable to flight into the stratosphere (generally between 18 to 37 Km). Though primarily used as weather balloons, the use of high altitude balloons as platform for experiments in near space conditions has been increasingly growing. In this context a partnership between ISTNanoSat-1 and the BALUA Project[38] has been created so that Development Models of the created subsystems and Engineering Qualification Models can be tested for functionality in near space conditions.

6.4 Testing Conditions

6.4.1 Ambient Test Conditions

Unless otherwise specified, all measurements and tests shall be made at room ambient atmospheric pressure, temperature and relative humidity conditions as stated in Table 14. Whenever, these conditions must be closely controlled in order to obtain reproducible results, a reference temperature of 22°C, a relative humidity of 50% and an atmospheric pressure of 101 KPa shall be used together

[38] http://balua.org - 2016

with the tolerances required to obtain the desired precision of measurement. Actual ambient test conditions should be recorded periodically during the test period.

	Temperature (ºC)	Pressure (KPa)	Relative Humidity (%)
Ambient	22 ± 3	101 ± 5%	50 ± 10

Table 14 – Ambient Test Conditions

6.4.2 Measurements Tolerances

The accuracy of instrument and test equipment used to control or measure the test parameters shall be in general, one order of magnitude better than the tolerance for the variable to be measured. Exceptions shall be specified in the relevant specifications.

6.4.3 Test Tolerances

The test tolerances shall be applied to the specified nominal test values. Unless specified in each test procedure in section 6.6, the maximum allowable tolerances on test conditions or measurements shall conform to acceptance levels as stated in ECSS-E-ST-10-03 depicted in Table 15. The tolerance on test parameters specifies the maximum range allowed within which the specified test level (input level) or measurement (output) may vary and excludes instrument accuracy.

Test parameters	Tolerances
Temperature (°C)	T_{min}= 0/-4 T_{max}= 0/+4
Relative Humidity	±10%

Table 15 – Test Tolerances for temperature and humidity

6.5 Functional and performance verification

6.5.1 Planned Test Facilities and Set-Up

Functional and Performance verification test will make use of the EGSE to be produced based on the architecture depicter in Figure 35. Besides the proposed EGSE no additional facilities or set-ups are required.

6.5.2 Test conditions and procedure

The final test conditions and procedures for the functional verification are not yet determined due to the incomplete design at the time of this thesis. Nevertheless, a limited sequential procedure for functional and performance verification is proposed. The work presented in the following Table 16 is the compilation of undocumented work from several internal project parts and it is hereby depicted for the sake of presenting a complete test campaign.

Step	Description	Short Procedure	Fail Criteria
1	**Solar Panel test**	1. Expose the S/C to an energy source - Solar or artificial illumination 2. Measure the 9V power bus current	Current equals 0
2	**Battery Test**	1. Alternate feeding area 3 (exit converters), using either the bus or the battery. 2. Read battery levels, its tension and current, in each of the charging states.	TBC
3	**Test Main Power Supply (12V, 5V and 3.3V)**	1. Read consumption values at tension regulator exit 2. Read analog control signal	TBC
4	**Test Reserve Power Supply (5V and 3.3V)**	1. Turn off corresponding main SEPIC converter (if active) 2. Turn on the backup converter under testing 3. Read consumption values at Tension regulator exits 4. Read analog control signals	TBC
5	**AX25 Protocol test**	1. Send a specially crafted frame to the AX25 loopback interface. (By special one means a frame with an arbitrary but static payload field). 2. Wait and receive the frame.	Sent and received frames are not equal
6	**CSP Protocol test**	1. Send a specially crafted frame to the CSP loopback interface. (By special one means a frame with an arbitrary but static payload field). 2. Wait and receive the frame.	Sent and received frames are not equal
7	**Data Bus Test**	1. Write 0x01, 0x02, 0x04, 0x08, 0x10 … 0x80 into RAM. 2. Read back those values	Read back values differ from written values
8	**Persistent Memory test**	1. Perform a checksum/hash of each stored file.	Checksum differs from stored value
9	**Gyroscope Test**	1. Turn the MPU9250 sensor on; 2. Measure the current consumption; 3. Test the communication with the sensor using the WHO_AM_I field; 4. Measure the current rotational speed; 5. Activate the self-test function and measure	>4mA; Response to WHO_AM_I field is other than 0x71.

		again; 6. Repeat the items 4 to 6 10 times and calculate the average for each axis.	
10	**Accelerometer Test**	1. Turn the MPU9250 sensor on; 2. Measure the current consumption; 3. Test the communication with the sensor using the WHO_AM_I field; 4. Measure the current acceleration; 5. Activate the self-test function and measure again; 6. Repeat the items 4 to 6 10 times and calculate the average for each axis	>4mA; Response to WHO_AM_I field is other than 0x71.
11	**Magnetometer**	1. Turn the MPU9250 sensor on; 2. Measure the current consumption; 3. Test the communication with the sensor using the WHO_AM_I field; 4. Measure the current magnetic field; 5. Activate the self-test function and measure again; 6. Repeat the items 4 and 5 10 times and calculate the average for each axis.	>4mA; Response to WHO_AM_I field is other than 0x48.
11	**Magnetometer (HMC5983) Test**	1. Turn the HMC5983 sensor on 2. Measure the sensor's current consumption 3. Test the communication with the sensor using the identification fields: Register A, Register B and Register C; 4. Test the communication with the sensor using the identification fields: Register A, Register B and Register C; 5. Change the sensor to continuous-measurement mode with gain of 5. 6. Measure the current magnetic field; 7. Activate self-test function and measure again; 8. Repeat the items 4 to 6 10 times and calculate the average for each axis	>1mA; Response to 3 is different of 0x48, 0x34 and 0x33, respectively
12	**Magnetorquers test**	1. Turn the magnetorquer on using a PWM signal of 10%; 2. Measure the current consumption (if implemented); 3. Measure the value of the generated magnetic field; 4. Change the value of the PWM signal to	>5% deviation of nominal current consumption

		50% and repeat the measurements 2 and 3; 5. Repeat steps 1 through 4 for each available magnetorquer (0 to 5).	
13	**Repeat step 12 and for each available magnetorquer**		

Table 16 – Functional and Performance preliminary test procedure

6.6 Test Plan, Criteria and Methods

Physical properties test, Sine vibration test, random vibration test, mechanical shock test, thermal vacuum cycling test and EMC test shall be carried out according to specific plans and procedures so that the s/c can be properly qualified and space flight worthy. Each of these test intents to guarantee the suitability of design and construction of the s/c for the mission. Physical properties test aims to guarantee that the s/c can be properly placed inside a P-POD (or similar) so that it can be launched. Sine Vibration, random vibration and mechanical vibration aim to test the s/c for the launch environment compiled in chapter 3 of this dissertation. EMC test aims to determine the compatibility of the s/c with its self-induced electromagnetic environment. Finally, thermal vacuum cycling test will reproduce vacuum and temperature extremes of the space environment stressing the materials and components of the s/c. For all of these tests detailed procedures were created, according to the applicable criteria and considering the environmental stresses expected. These detailed procedures and test levels of the tests aforementioned are presented in Annex 3.

7

Conclusions

The work performed in the scope of this dissertation intended to implement quality and product assurance, risk management and AIV procedures for a university CubeSat ISTnanosat-1. The implementation and definition of these procedures was based on the space industry standards and tailored according to CubeSat typical and best practices so a compromise between relevance and complexity was achieved.

The quality and product assurance procedures hereby depicted are general and applicable to any CubeSat development, with the potential to passed on to future missions besides IST-NanoSat-1. Nevertheless, they are not static definitions within the project and shall be updated when so deemed reasonable. Regarding risk management, both S/C and program related were analyzed to achieve a complete risk assessment and initiate the appropriate risk management.

Finally, upon the analysis of the typical environments for CubeSat missions, the AIV considerations for the ISTNanosat-1 were defined. In the scope of the current dissertation a tangible assembly, integration and verification strategy and plan was implement. High level assembly and integration procedures were implemented and more detailed integration sequence shall be put in place once the full systems definition is made. A wide range, complete test campaign for ISTNanosat-1 is proposed based on the worst case environment for the typical launchers used for CubeSat satellites. The implementation of such test campaign will guarantee the compliance of ISTNanosat-1 with any of the highlighted launchers which result in an almost limitless use of this S/C, independently of the launcher chosen to be used in the future.

As future work remains the continuation of the quality and risk management procedures established. Quality and product assurance has to be enforced throughout the full development of the ISTNanoSat-1 project to guarantee minimization and control of possible failures. In the same sense risk management is a continuous work to be developed until the end of project so that the risks identified can be further mitigated and corrected.

Regarding the complete test plan implemented, EGSE and MGSE equipment shall be put in place upon further definition of the total system. Instrument and payload calibration shall be defined and put in place. Launcher selection will determine the final flight conditions and environment and therefore, if deemed necessary from the implementation point of view, some of the requirements and tests depicted in this dissertation can be adapted to less demanding levels. This can have a particular impact in the development trade off, most likely easing the design engineering practices.

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ANNEXES

Annex 1 - Risk Register

Risk ID and name	Description	Likeli. [1-5]	Cons. [1-5]	Risk Rating	Rational	Actions
SE-1	Inability to find the desired components	1	2	2	Due to non-defined project deadline, likelihood of delays as to be considered minimum	Components chosen have at first hand availability verification
SE-2	Mechanical design delays	1	2	2	Due to non-defined project deadline, likelihood of delays as to be considered minimum	TBD
SE-3	Software design delays	1	2	2	Due to non-defined project deadline, likelihood of delays as to be considered minimum	TBD
SE-4	Delay due to issues with the payload provider	1	3	3	Due to non-defined project deadline, likelihood of delays as to be considered minimum	Close contact with HUMSAT constellation responsible is kept during the development phase of the S/C
SE-5	Delay due to inadequate project documentation	1	3	3	Due to non-defined project deadline, likelihood of delays as to be considered minimum	Documentation Management and requirements established as part of this dissertation according to space standards
SE-6	Loss of project information and documentation	2	3	6	Not all the documents are at this stage in the proper online tools and therefore local information can be lost	SVN, wiki and Cloud drive are implemented to be used to store all project data

SE-7	Loss of project hardware	3	3	9	No project specific and exclusive area is created for the hardware and both missing or misplaced parts are likely to occur.	Create a project storage as planned by the quality management procedures
SE-8	Lack of sufficient training of the team members performing the flight qualification tasks.	2	4	8	Qualification mistakes would have a big impact in the project costs, deliveries and worked performed	Qualification task shall be performed in qualified facilities with either the support or supervision of local trained staff
SE-9	Attrition or turnover of team members	4	2	8	Due to the academic environment of this project students tend to leave the project each semester after graduation	A complete documentation and configuration management is to be put in place as well as the availability of later communication with the leaving staff is to be guaranteed
SE-10	Sudden loss of crucial team members	1	4	4	No sudden leave is expected since for students have their leave planned and the staff professors are unlikely to move	Documentation and configuration management
SE-11	COTS components prices increase	4	1	4	Though the price change in COTS is likely to occur its impact is rather small due to their general low cost	Create buffer budget for unpredictable situations
SE-12	Inability to obtain funding	1	5	5	The non-funding of the project puts the project at risk since it is still in development phase	TBD
SE-13	Delay of receiving promised funding	1	2	2	Low impact due to no schedule constrains to date	Funding based is allocated via academic funds
SE-14	Launch opportunity cannot be guaranteed	2	5	10	Mission cannot be executed	Project is part of the ESA Flight you satellite initiative
SE-15	Deviation from desired orbit upon launched	1	4	4	HUMSAT mission objectives can be seriously compromised. Secondary mission will unlikely suffer an impact.	Aim for reliable launch providers, namely ESA through the Fly your Satellite Program.
PL-1	Software interface issues between payload and spacecraft bus	2	4	8	Impossibility to perform the mission is at stake	Thorough software Testing
PL-2	Hardware/electrical interface issues between payload and spacecraft bus	1	4	4	Mission compromising risk but less likely to occur if S/C passes Qualification	Thorough Testing and Qualification

PL-3	Payload malfunction due to electronic issues	1	4	4	Mission compromising risk but less likely to occur if S/C passes PFM tests	Main Payload with space heritage. PFM testing
COM-1	No frequency on which to communicate with spacecraft due to delay in receiving frequency allocation	1	2	2	Due to non-defined project deadline, likelihood of delays as to be considered minimum	Allocation of the COM band to be performed as soon as possible
COM-2	Failure of spacecraft radios	2	5	10	Mission compromising risk	Test and qualification
COM-3	Failure of spacecraft antennas due to improper deployment or activation	1	4	4	Effective Communication might be compromised	Deployment test and simplicity in the deployment mechanisms design must be implemented
COM-4	Failure of ground station	1	2	2	Very unlikely event	Established ground station with large background and adequate facilities is available
CDH-1	Failure of flight computer	4	2	8	Space environment will eventually lead to such an event. Redundancy and short mission lifetime reduce the impact	Redundant components and connection have been implemented and operation of the S/C is guaranteed. No single point of failure
ADCS-1	Failure of sensors	4	2	8	Space environment will eventually lead to such an event. Redundancy and short mission lifetime reduce the impact	Redundant components implemented
ADCS-2	Failure of actuators causing unstable spacecraft motion (due to either hardware or software issues)	1	4	4	Impact may degrade the communication capabilities of the S/C	TBD
EPS-1	Failure of power regulation/battery system	2	5	10	Mission compromising failure	EPS measures implemented shown in Chapter 4.
EPS-2	Failure of solar panels to generate power	1	4	4	Not mission critical due to the large number of solar panels	Power budget margin implementation. Full coverage of the S/C with Solar panels increasing the number and therefore redundancy
TC-1	Unexpected thermal environment	2	2	4	Severity is controlled by the extensive and demanding test campaign	Worst case scenario Qualification test campaign
SM-1	Unexpected vibration environment	2	2	4	Severity is controlled by the extensive and demanding test campaign	Worst case scenario Qualification test campaign

Q-1	Spacecraft will not deorbit within 25 years after end-of-life	1	1	1	No mission impact though not compliant with space requirements. Likelihood is very low as S/C is planned to be deployed below 800 Km	Aim for reliable launch providers, that can guarantee deployment at low orbits
Q-2	Spacecraft does not meet requirements (i.e. dimension, mass limits, structural/thermal analyses)	3	4	12	Project requirements yet to be frozen imply a serious risk	Complete definition of mission requirements and Project control over the known requirements
Q-3	Selected key component with no known space heritage do not survive test conditions	3	3	9	Relatively likely event, that if not detected early in the development process can have a relevant impact in costs likely due to redesign needs.	Alternatives must be identified in the design phase and components without space heritage shall be selected based on their range of operation

Table 17 - ISTNanosat-1 Risk Register

Annex 2 - Risk Matrix

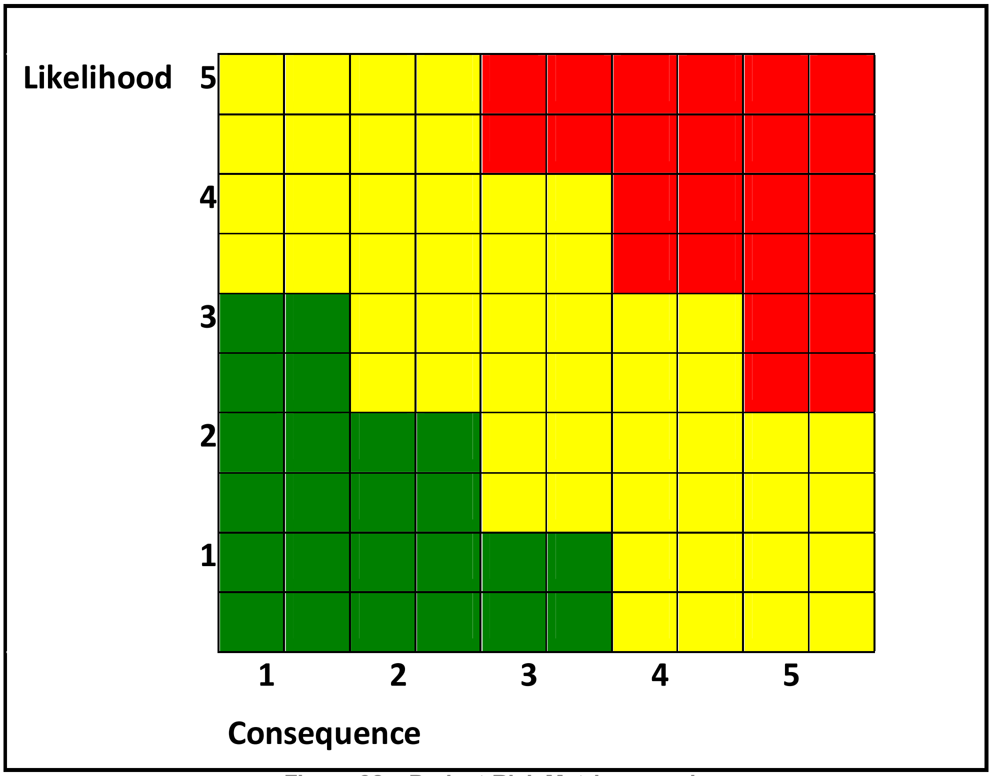

Figure 38 – Project Risk Matrix example

Annex 3 – Test Plan procedures

Physical Properties Test

Mass, Dimensions, and Center of Gravity and Moments of Inertia of the S/C shall be measured to validate their compliance with the launch interface and mechanical requirements for CubeSats.

Requirements

Mass: The maximum mass of a 1U CubeSat shall be 1.33 kg,

Dimensions: 100x100x100 mm.

Center of Gravity with respect to geometric Center: The 1U CubeSat center of gravity shall be located within 4.5 cm from its geometric center in the Z direction (S/C longest dimension).

Planned Test Facilities and Set-Up

For the mass measurement a calibrated scale with resolution down to 10^{-3} Kg and surface area capable of containing the S/C's longest face. For Center of Mass measurement dedicated Center of Mass scales shall be used. Ideally Mass and Center of Mass measurement shall be performed on the same equipment. For Dimensions check a measuring tape shall be used.

Test Conditions and Procedure

Step	Description	Pass-Fail Criteria	Measured Result
	Mass measurement		
1	Measure S/C Mass with calibrated weight and resolution down to 10^-3 Kg. Register the results.	< 1.33 Kg	
2	Make 2 additional measurement repetitions and register the results.		
	Dimensions Measurement		
3	Measure the X dimension of the S/C with a measuring tape. Register the results	<100 mm	
4	Measure the Y dimension of the S/C with a measuring tape. Register the results	<100 mm	
5	Measure the Z dimension of the S/C with a	<100 mm	

	measuring tape. Register the results		
	Center of Mass measurement		
6	Measure/Calculate the Geometric Center of the Space Craft		
7	Measure the S/C's Center of Mass		
8	Measure the moments of Inertia in each axis.		
9	Calculate the Difference for the 3 axis between Center of Mass and Geometric Center	Center of gravity shall be located within 4.5 cm from its geometric center in the Z direction	

Table 18 – Physical properties test procedure

Sine Vibration Test

Requirements

Sine Vibration Qualification Requirement are based on ECSS-Q-ST-10-03.

Load levels are defined as KQ x Limit Load Spectrum sweep with 2 Oct/min duration and ranging from 0 Hz to 100 Hz. The qualification factor KQ is given in ECSS-E-ST-32-10 clause 4.3.1 and KQ=1.25. Limit Load Spectrum is considered as the worst case envelop from the launch environment depicted in Figure 13 and Figure 14. Since Axial loads envelope is of greater value then Lateral loads those are considered as baseline values for the Qualification test levels.

Planned Test Facilities and Set-Up

The required equipment for Sine Vibration test must be capable to perform accelerations from 0.5g to 3.75g according to Figure 40 levels at a sweep rate of 2 Oct/min. Vibration loads are tested in shaker devices. Figure 39 portraits an shaker example. An example of possible test house with such capabilities is the Instituto de Soldadura e Qualidade (ISQ).

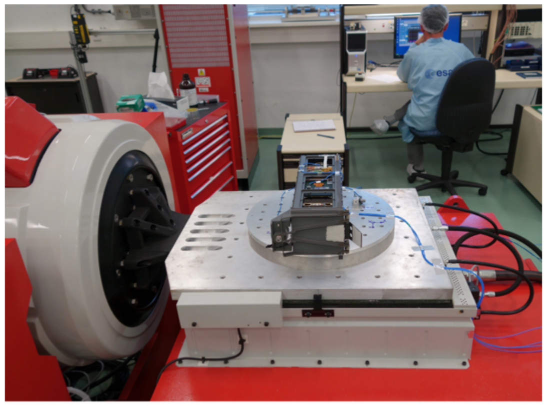

Figure 39 – Vibration Shaker example[39]

Test conditions and Test Procedure

	Qualification Model Level	Protoflight Model Level
Duration	Sweep at 2 Oct/min, 5 Hz – 100 Hz	Sweep at 4 Oct/min, 5 Hz – 100 Hz
Load	(KQ = 1.25 x Limit Load Spectrum)	(KQ = 1.25 x Limit Load Spectrum)
Tolerances	Frequency: ± 2 % (or ±1 Hz whichever is greater) Amplitude: ± 10 % Sweep rate (Oct/min): ± 5 %	
Number of applications	1 on each of 3 orthogonal axes	

Table 19 – Sine Vibration test condition

Frequency (HZ)	5	5	10	10	100
Acceleration	0.5	1.0	1.0	3.75	3.75

Table 20 – Sine Vibration Load

[39] http://www.arxterra.com/wp-content/uploads/2015/05/CubeSatTesting.jpg - 2016

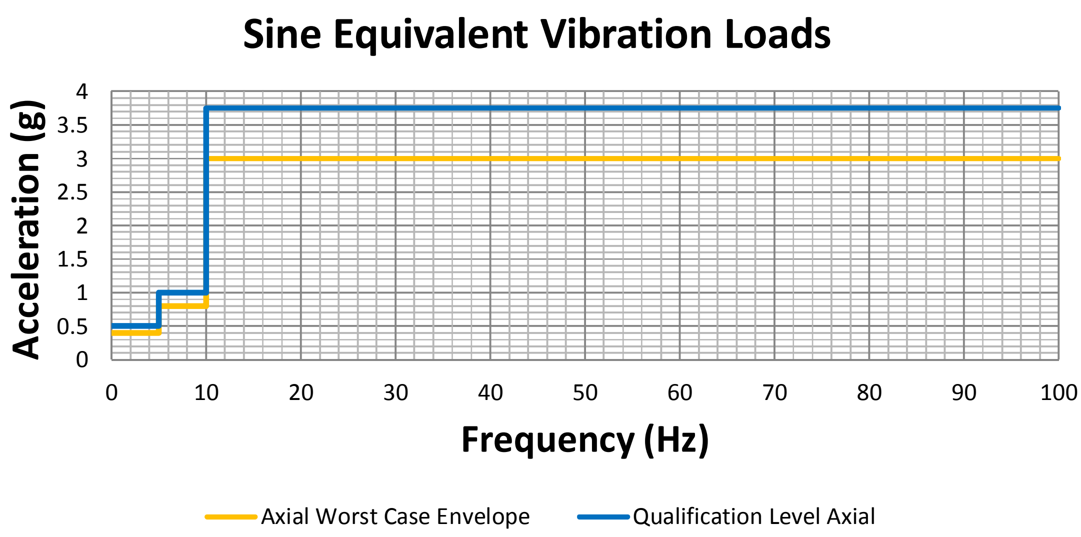

Figure 40 – Sine vibration Qualification Test Level

Step	Description	Pass-Fail Criteria	Measured Result
1	Electrical Baseline Functional	-	
2	Attachment to the MGSE and placement in the test equipment	-	
3	Resonance search (5 to 100 Hz at 0.15g amplitude)		
4	Test performance in X-axis		
5	Visual Inspection	Constructional defects	
6	Electrical Post-test Functional	No response deviation	
7	Test performance in Y-axis		
8	Visual Inspection	Constructional defects	
9	Electrical Post-test Functional	No response deviation	
10	Test performance in Z-axis		
11	Resonance search (5 to 100 Hz at 0.15g amplitude)	Severe resonance response variation from 3	
12	Visual Inspection	Constructional defects	
13	Electrical Post-test Functional	No response deviation	

Table 21 – Sine Vibration test procedure

Random Vibration Test

Requirements

Random Vibration Qualification Requirement are based on ECSS-Q-ST-10-03.

Load levels are defined as Maximum spectrum expected + 3 dB on PSD Values. Limit Load Spectrum is considered as the worst case envelop from the launch environment depicted in Figure 15. Many of the highlighted launchers had their random vibration environment not depicted or defined by acoustics, which is of most relevance for large payloads and not CubeSats. Therefore, it was considered of relevance to further enhance the load requirements to the NASA GEVS requirements for systems with mass below 25 Kg. Both worst case envelope related qualification level and GEVS are depicted in Figure 41.

Planned Test Facilities and Set-Up

Random Vibration can make use of the same equipment as Sine Vibration test.

Test conditions and Test Procedure

	Qualification Model Level	Protoflight Model Level
Duration	2 minutes	1 minute
Load	Maximum spectrum expected + 3 dB on PSD	Maximum spectrum expected + 3 dB on PSD
Tolerances	Amplitude (PSD, frequency resolution better than 10Hz): 20 Hz - 1000 Hz - 1 dB / +3 dB; 1000 Hz - 2000 Hz ± 3 dB Random overall g r.m.s. ± 10 %	
Number of applications	1 on each of 3 orthogonal axes	

Table 22 – Random Vibration test conditions

Frequency (Hz)	20	50	800	2000
PSD (g^2/Hz)	0.026	0.16	0.16	0.026
G (RMS)	14.2			

Table 23 – Random Vibration GEVS Load

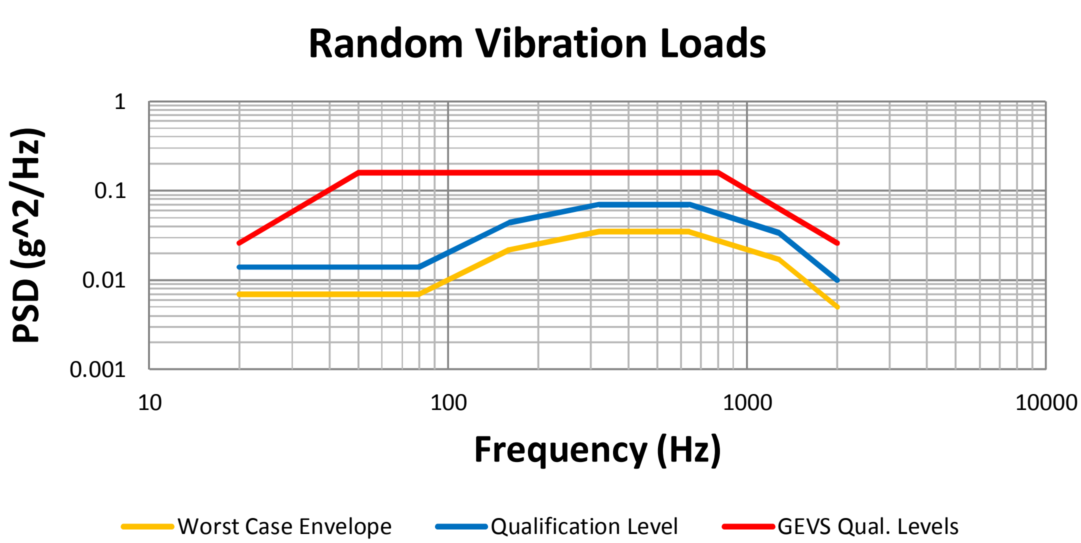

Figure 41 - Random Vibration Qualification Test Level

Step	Description	Pass-Fail Criteria	Measured Result
1	Electrical Baseline Functional	-	
2	Attachment to the MGSE and placement in the test equipment	-	
3	Resonance search (5 to 100 Hz at 0.15g amplitude)	-	
4	Test performance in X-axis		
5	Visual Inspection	Constructional defects	
6	Electrical Post-test Functional	No response deviation	
7	Test performance in Y-axis		
8	Visual Inspection	Constructional defects	
9	Electrical Post-test Functional	No response deviation	
10	Test performance in Z-axis		
11	Resonance search (5 to 100 Hz at 0.15g amplitude)	-	
12	Visual Inspection	Constructional defects	
13	Electrical Post-test Functional	No response deviation	

Table 24 – Random Vibration test procedure

Mechanical Shock Test

Requirements

Random Vibration Qualification Requirement are based on ECSS-Q-ST-10-03.

Load levels are defined as Maximum expected shock spectrum +3 dB. Limit Load Spectrum is considered as the worst case envelop from the launch environment depicted in Figure 43. The non-performance of shock test at Protoflight models is assumed since this are highly stressful tests for the parts, and such, Protoflight shock test would represent an overstress to the S/C.

Planned Test Facilities and Set-Up

Test equipment for shock test shall be constituted by either a drop shock tests as illustrated in Figure 42 or any other equipment capable to securely apply the required loads (Shock table, pyro shock tester…).

Figure 42 – Drop Shock test example[40]

Test conditions and Test Procedure

	Qualification Model Level	Protoflight Model Level
Duration	Duration representative of the expected environment (Typical duration is between 20ms and 30ms.)	N/A
Load	Maximum expected shock spectrum +3 dB qualification margin	N/A
Tolerances	Shock level - 3 dB/ + 6 dB and 50 % of the SRS amplitude above 0 dB	N/A
Q-factor	10	N/A
Number of applications	2 on each of 3 orthogonal axes	N/A

Table 25 – Mechanical Shock test conditions

Frequency (HZ)	100	300	1000	10000
Acceleration	135	540	4725	4725

Table 26 – Mechanical Shock Load

[40] http://www.lansmont.com/wp-content/uploads/2014/05/Standard-shock-test.jpg - 2016

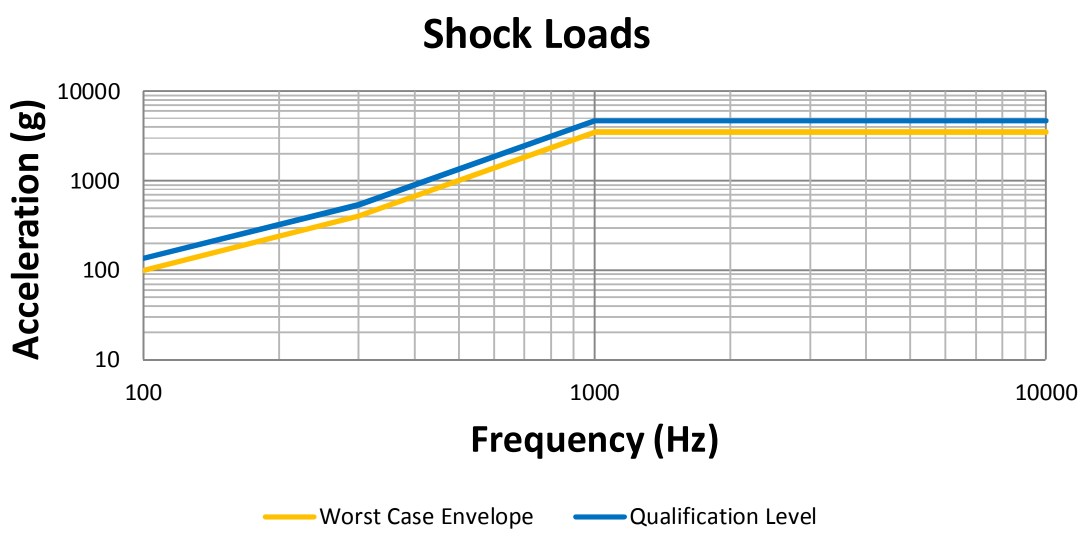

Figure 43 – Mechanical Shock Qualification Test Level

Step	Description	Pass-Fail Criteria	Measured Result
1	Electrical Baseline Functional	-	
2	Attachment to the MGSE and placement in the test equipment	-	
3	Test performance in X-axis		
4	Visual Inspection	Constructional defects	
5	Electrical Post-test Functional	No response deviation	
6	Test performance in Y-axis		
7	Visual Inspection	Constructional defects	
8	Electrical Post-test Functional	No response deviation	
9	Test performance in Z-axis		
10	Visual Inspection	Constructional defects	
11	Electrical Post-test Functional	No response deviation	

Table 27 – Mechanical Shock test procedures

Thermal Vacuum Cycling (TVC) Test

Thermal Vacuum Cycling Test will place the S/C in close to space conditions while in operational mode. Both the functionality in vacuum and at the range of temperatures is verified. Additionally, this test is strategically performed after mechanical tests so that any failure or cracks created by these tests can be stressed, as per real operation conditions (mechanical stresses during launch and subsequent space environment conditions).

Requirements

Thermal Vacuum Cycling Qualification Requirement are based on ECSS-Q-ST-10-03.

Load levels are defined as $T_{Qual\ max}=T_{max}$ + 5ºC and $T_{Qual\ min}=T_{min}$ - 5ºC, for both operation and non-operational range. In order to embark the worst case scenario non-operational temperatures are considered for the TVC to be performed in ISTNanosat-1 at this stage.

Planned Test Facilities and Set-Up

A thermal vacuum chamber (Figure 44 refers) is required to perform TVC test. This chamber shall be capable of reaching the intended pressure levels while maintaining a controlled temperature environment during all the temperature cycles. Additionally, a feedthrough shall exist so that direct physical link with the S/C is possible during testing, for functional assessment. ISQ and Lusospace are two Portuguese companies with the facilities to perform such tests.

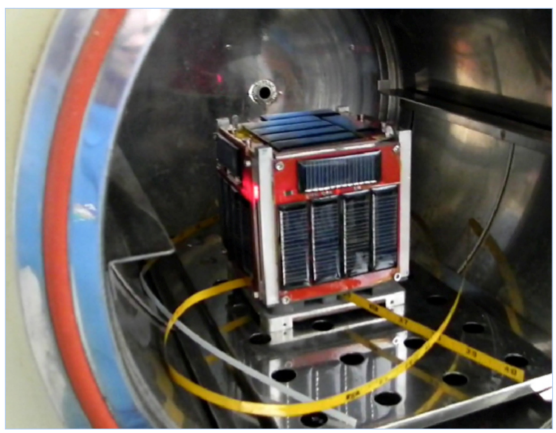

Figure 44 – Thermal Vacuum Chamber example[41]

Test conditions and procedures

The test levels for TVC, based on [85] and according to a generic mission profile at this stage, to be applied to ISTNanoSat-1 EQM and PFM are summarized in Table 28 and depicted in Figure 45:

	Qualification Model Level	Protoflight Model Level
Number of Cycles	8	4
Min.Temperature*	-55 (Mission T_{min} -10 C)	-55 (Mission T_{min} -10 C)

[41] https://upload.wikimedia.org/wikipedia/commons/7/73/F-1_CubeSat_thermal_vacuum_test.jpg - 2016

Max. Temperature*	70 (Mission T_{max} +10 C)	70 (Mission T_{max} +10 C)
Dwell time at Tmax	1 hour	1 hour
Dwell time at Tmin	1 hour	1 hour
Temperature rate	<5 ºC/min	
Pressure	$<10^{-5}$ Pa	
Stabilization criterion	1/10 ºC/min	

Table 28 – Thermal Vacuum Cycling test conditions

*Temperature at the S/C.

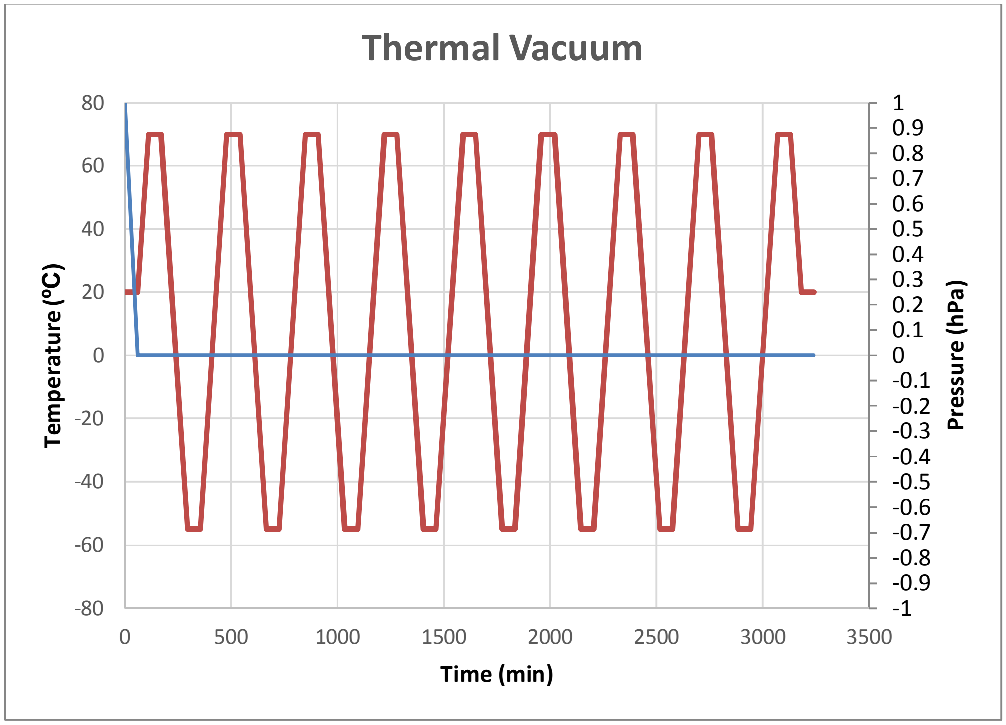

Figure 45 - Thermal Vacuum test levels

Step	Description	Pass-Fail Criteria	Measured Result
1	Electrical Baseline Functional	-	
2	Placement in the test thermal Vacuum chamber	-	
3	External Interface Connection and short Run functional test	-	
3	Test Performance with online data acquisition	-	
4	Visual Inspection	Constructional defects	
5	Electrical Post-test Functional	No response deviation	

Table 29 - Thermal Vacuum test procedure

EMC Test

EMC test is typically composed by emission and susceptibility testing. The first refer to the level of electromagnetic field generated by the S/C whilst the second refers to the tolerance of the S/C to the electromagnetic environment it will encounter, including its self-induced electromagnetic environment.

Due to the mandatory no operation of the CubeSats from launch until 30min after separation only auto-compatibility is off concern for this type of S/C.

Requirements

Intrasystem EMC does not contain specific requirements or test conditions besides the necessary full operation of each subsystem of the S/C.

Planned Test Facilities and Set-Up

An anechoic chamber (Figure 46 refers) is necessary to eliminate the ambient and background electromagnetic emissions so that only the electromagnetic emission from the spacecraft are measured. In Portugal, Anacom and ISQ are potential test houses to be used for this test.

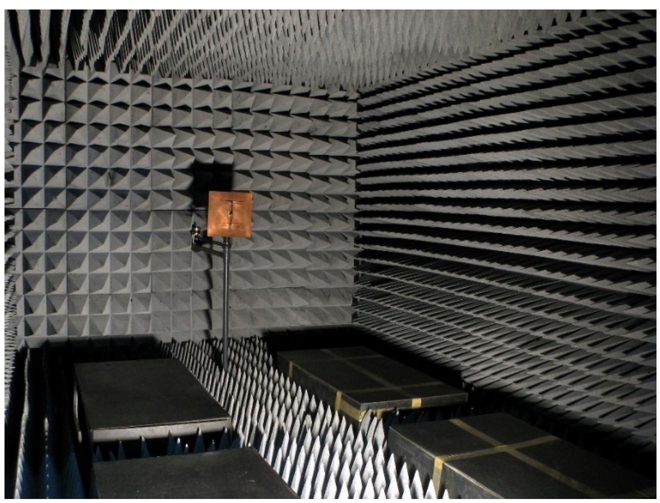

Figure 46 – Anechoic chamber example for EMC testing[42]

Test conditions and procedures

The test for intrasystem EMC shall be performed in an anechoic chamber and monitoring of each of the subsystems shall be possible during the test while the emissions of the S/C are recorded. The obtained data shall be cross related to understand the origin of any non-conform behavior.

[42] https://upload.wikimedia.org/wikipedia/commons/0/0a/Radio-frequency-anechoic-chamber-HDR-0a.jpg - 2016