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TEST CAMPAIGNS OF NANOSATELLITES

TESTOVACÍ KAMPANĚ NANOSATELITŮ

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INSTRUCTION:

CubeSat missions are increasingly used in academia and industry thanks to rapid, low-cost access to space. However, the spacecraft must survive harsh conditions during launch and operate reliably in the demanding space environment. Therefore, systematic verification and testing are essential to reduce mission risks. The aim of this thesis is to design, execute, and evaluate a structured verification and test process for a selected CubeSat system in accordance with relevant standards.

1. Review relevant CubeSat standards and define a reference CubeSat architecture and verification baseline for selected spacecraft.
2. Analyze CubeSat test types, objectives, and classifications and map them to the verification needs of the selected system.
3. Design a comprehensive verification and test plan for the selected spacecraft, including test sequence, required resources, and pass/fail criteria.
4. Develop detailed verification and test procedures for the planned tests, including measurement methods, and reporting templates.
5. Execute the defined tests according to the developed procedures.
6. Evaluate the test results, discuss compliance against requirements, and propose improvements or corrective actions for the testing process.

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REFERENCE:

- [1] CAPPELLETTI, Chantal, Simone BATTISTINI a Benjamin K. MALPHRUS. CubeSat handbook: from mission design to operations. San Diego, CA, United States: Academic Press is an imprint of Elsevier, [2021]. ISBN 978-0-12-817884-3.
- [2] California Polytechnic State University, CubeSat Design Specification Revision 14, NASA, 2022. [Online]. Available: https://www.nasa.gov/wp-content/uploads/2018/01/cubesatdesignspecificationrev14_12022-02-09.pdf
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ABSTRACT

This thesis focuses on the verification and testing activities carried out during the development of CubeSat missions. The work provides an overview of testing approaches commonly applied to nanosatellite projects and discusses their role throughout the overall verification campaign. Particular attention is then devoted to environmental testing. Furthermore, the work addresses the verification philosophy, test sequence, documentation, and practical aspects for planning and executing a test campaign. The practical section demonstrates selected verification activities using examples from the KOSTKA CubeSat mission developed by the YSpace student team. The overview and practical experience presented may then serve as a reference for future CubeSat development and student satellite projects preparing environmental verification campaigns.

KEYWORDS

CubeSat, Verification, TVAC testing, Electromagnetic compatibility (EMC), Radiation testing, Vibration testing, Spacecraft qualification

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Author's Declaration

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Topic: Test campaigns of nanosatellites

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For the purposes of proofreading, source searching, and translation, AI tools were utilized in the creation of this work.

Contents

Introduction	12
1 CubeSat platform and Design standards	14
1.1 Development of the CubeSat standard	14
1.2 Mechanical and Structural requirements	15
1.2.1 Unit and External Dimensions	15
1.2.2 Frame and Internal Volume Constraints	16
1.2.3 Rail Interface and Surface Requirements	17
1.3 CubeSat System Architecture	18
1.3.1 Typical CubeSat Subsystems	18
1.3.2 PC/104 Standard	19
2 Selected CubeSat System - KOSTKA	21
2.1 Mission Technical Baseline	22
2.2 Subsystem Overview	22
2.2.1 On-Board Computer	22
2.2.2 Electrical Power System	23
2.2.3 Communications	25
2.2.4 Payload - Experimental UHF receiver	25
2.3 Architecture	26
2.3.1 Internal Architecture	27
2.3.2 Mechanical Interfaces	28
2.3.3 External Components	29
3 Overview of CubeSat Testing methods	31
3.1 Classification of Tests	31
3.1.1 Environmental Tests	32
3.1.2 Functional and Performance Tests	33
3.1.3 Integration and System-level Tests	34
3.2 Environmental Testing methods	35
3.2.1 Vibration Testing	35
3.2.2 Thermal Vacuum testing	41
3.2.3 Electromagnetic Compatibility	46
3.2.4 Radiation Testing	50
3.3 Test Campaign Strategy	53
3.3.1 General Principles	54
3.3.2 Test Sequence	55

3.3.3	Test Levels and Model Philosophy	56
3.3.4	Documentation	56
4	Test Campaign Planing	58
4.1	Campaign Overview	58
4.2	Testing Structure	59
4.3	Documentation	59
5	Integration Tests	61
5.1	Solar Panels Verification	61
5.2	Kill Switch Verification	63
5.3	Satellite Assembly	64
5.4	Antenna Deployment Test	65
5.5	Final Dimensional Verification	66
6	Vibration Campaign	68
6.1	Procedure Preparation	68
6.2	Test Configuration	70
6.3	Test Execution	72
6.4	Test Evaluation and Data Processing	73
6.5	Summary	75
7	TVAC Campaign	76
7.1	Test Procedure	76
7.2	Campaign Readiness and Limitations	79
7.3	Summary	80
	Conclusion	81
	Bibliography	83
	Symbols and abbreviations	86

List of Figures

1.1	Representation of different CubeSat configurations [4]	15
1.2	Examples of CubeSat structure	16
1.3	Example of deployer and rail mechanism [5]	17
1.4	Block diagram of CubeSat subsystems	19
1.5	Showcase of PC/104 PCB [8]	20
2.1	Render of KOSTKA CubeSat	21
2.2	KOSTKA On-Board Computer	23
2.3	AMUN EPS [9]	24
2.4	Solar panels	24
2.5	Communications board	25
2.6	Payload PCB	25
2.7	KOSTKA coordinates reference frame	26
2.8	KOSTKA stack configuration	27
2.9	UHF (showcase) [10] and TunaCan antennas	29
2.10	Camera placement	30
4.1	ExoPod Nova 16U [25]	58
5.1	Closeup of the unwanted connection	62
6.1	Defined vibration profile [26]	68
6.2	Final placement of accelerometers	69
6.3	Actual placement of Accelerometers	70
6.4	Mounting of KOSTKA during vibration tests	71
6.5	Sine sweep comparison in Z axis	74
6.6	PSD curve and control signal (Z axis)	75
7.1	Illustration of Thermocouples locations	77
7.2	Functional test of TunaCan antenna in vacuum conditions	78

List of Tables

2.1	Baseline parameters of KOSTKA	22
2.2	Placement of individual subsystems in the stack	28
3.1	Sine vibration quick-fact table	36
3.2	Random vibration quick-fact table	38
3.3	Shock test quick-fact table	39
3.4	Quasi-static test quick-fact table	40
3.5	Key parameters of Thermal Vacuum testing	44
3.6	Key parameters - Outgassing / Bake-out	45
3.7	Key parameters of EMC	49
3.8	Key parameters of Radiation testing	53
5.1	Final dimensional verification of KOSTKA	67
6.1	Acceleration at defined frequencies [26]	69
7.1	Thermocouples positions	77

Introduction

The rapid development of small satellite technologies in recent years has significantly eased access to space missions for universities, research institutions, and commercial organizations. The standardized CubeSat platform has enabled the development of relatively inexpensive satellites with much shorter development cycles compared to traditional spacecraft missions. Originally introduced primarily for educational purposes, CubeSats are now increasingly used for technology demonstrations, scientific experiments, Earth observation, communication systems, and many other purposes. At the same time, the increasing complexity of these missions places growing demands on their reliability and mission assurance.

Despite their relatively small size and simplified architecture, CubeSats are exposed to a wide range of environmental conditions throughout their lifecycle. Mechanical stresses during launch, thermal cycling, vacuum conditions, electromagnetic compatibility requirements, and radiation effects represent significant challenges that must be considered during the development process. The use of commercial off-the-shelf components, limited redundancy, and tight development budgets further increase the importance of properly planned verification and testing activities. Environmental testing is therefore a critical part of the satellite development process and serves to reduce the risk of mission failure caused by errors in design, manufacturing, or integration.

Verification campaigns for CubeSats are typically based on established aerospace standards and launch provider requirements. Standards such as ECSS or NASA GEVS provide a common foundation and recommended procedures for satellite verification, but their implementation is often not straightforward for small satellite missions. Many requirements are intentionally defined in a generalized manner so that they can be applied across a wide range of missions and satellite types. As a result, the final verification campaign typically requires tailoring to the specifics of the mission, as well as engineering judgment. In practice, the selection of individual tests, their levels, durations, and acceptance criteria are guided not only by technical requirements but also by available resources, scheduling constraints, acceptable mission risks, and the availability of test facilities. In particular, the launch provider's requirements often define the minimum mandatory environmental verification activities, specifically in the case of vibration testing, where specific launch vehicle data is necessary for the proper definition of test conditions.

This thesis thus focuses on the verification and testing of CubeSat missions, with an emphasis on the overall structure and practical implementation of a verification campaign. Rather than providing a detailed analysis of a single testing method, it will focus on a broader overview of testing approaches relevant to CubeSat missions, including environmental testing, verification philosophy, test sequences, documentation, and practical aspects of preparing and conducting a test campaign. This theoretical discussion is then complemented by actual examples and experiences gained during the development, integration, and testing of the KOSTKA CubeSat mission.

1 CubeSat platform and Design standards

As a platform, CubeSat represents a standardized approach to nanosatellite design, thereby facilitating rapid development and enabling access to space projects for both academic and commercial purposes. Standardization plays a significant role in simplifying the development process and helps to eliminate the risks associated with transport and launch into orbit. It also ensures compatibility with proven launch systems, which further helps mitigate risks and reduce launch costs. This chapter introduces the CubeSat platform and associated design standards that define the fundamental requirements for these systems. These requirements then form the basis for the testing activities described in the following chapters.

1.1 Development of the CubeSat standard

The initial concept for CubeSat originated in the 1990s at California Polytechnic University (Cal Poly). The goal was to develop a standardized nanosatellite platform suitable for academic purposes, with focus on a simple, affordable approach to enable hands-on education in spacecraft design and subsequent operation. The basic element of this concept was the introduction of 1U, a cubic structure with external dimensions of 10x10x10 centimeters. This modular approach allows easy scalability while maintaining geometric consistency and compatibility with launch systems. [1] [2]

With increasing applications of CubeSat platform, it became necessary to clearly and precisely define its individual parameters. A formal set of requirements was created to ensure safety during integration with launch systems and compatibility between different launch providers. This gave rise to the CubeSat Design Specification document, which defines the fundamental external interfaces and limits for the internal configuration of the CubeSats themselves. [1] [2]

With expansion beyond the academic sphere and into commercial industry, the complexity of these systems and the demands on their reliability began to grow rapidly. As a result, adherence to standardized design rules became essential and formed the basis for structured development, integration, and verification processes. [2]

1.2 Mechanical and Structural requirements

This section describes the requirements for the mechanical and structural components of the CubeSat platform. As mentioned in previous sections, these requirements are mainly defined in the CubeSat Design Specification document. Therefore, the following definitions of individual components will be based on this standard.

1.2.1 Unit and External Dimensions

The basic element of this platform is the 1U (1 Unit). In theoretical terms, it is a cubic structure with external dimensions of 10 x 10 x 10 centimeters, which is more accurately referred to as an external envelope.

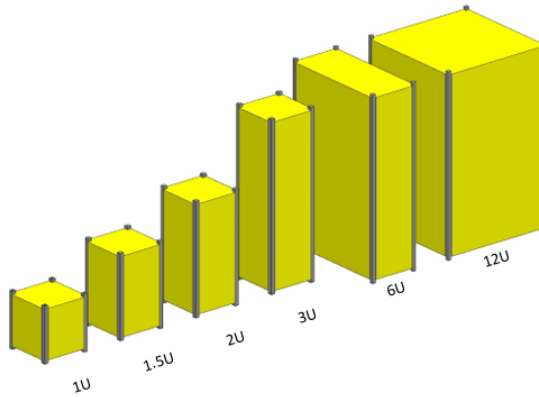


Fig. 1.1: Representation of different CubeSat configurations [4]

These units can then be used to build larger satellites by stacking basic modules in defined directions. Not all configurations are entirely valid, so we define a so-called family of CubeSats, which specifies the individual configurations options. Currently, these combinations range from 1 to 16U. Some of these configurations may also have a "+" (Plus) variant, where the satellite is extended in the -Z direction by a cylindrical volume, often nicknamed a Tuna Can. The size of this Tuna Can is not strictly defined and therefore depends on the launcher itself, or on an agreement with the launch provider. [1]

Individual configurations do not have all dimensions in multiples of 10. In certain directions, mainly along the Z axis, the structure is often extended by standoffs. These ensure that the outer elements of the satellite does not come into contact with either the deployer or another satellite, if there are more than one in a given slot. These standoffs extend the satellite by approximately 0.5 to 0.7 cm in both the positive and negative directions of the given axis. For a basic 1U, this means that its nominal length in the Z-axis is 11.3 cm. [1]

The standard also allows parts to protrude 6.5 mm from other sides, outside the rail system. These volumes are most often used for the placement of solar panels or antennas. It should also be noted that the value of 6.5 mm is specified by the CDS, but it may often vary among different launch providers and must be subject to an individual review. [1]

1.2.2 Frame and Internal Volume Constraints

The entire CubeSat consists of several main parts. One of the most important is the supporting structure itself. Its external dimensions should meet the outlined requirements of the external envelope. Theoretically, we can divide it into two parts, a structural skeleton and rails. Structural skeleton ensures the mechanical strength of the entire structure. Rails serve as the mechanical interface with the launcher itself. The entire structure then forms a mechanical support for other subsystems. [1]

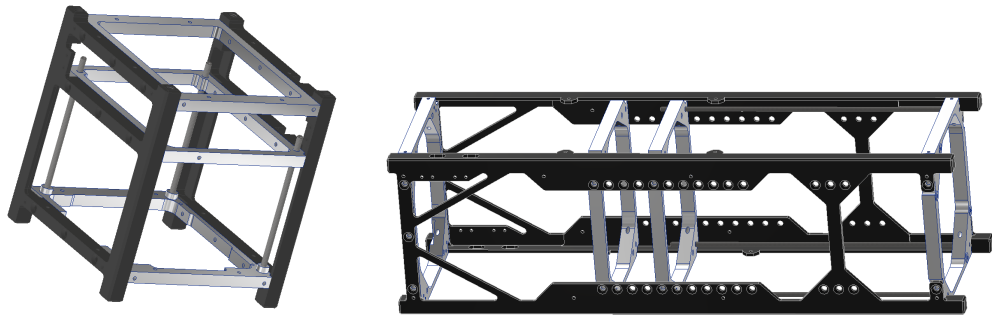


Fig. 1.2: Examples of CubeSat structure

If we consider the individual elements of the structure in a 1U configuration, we come to the conclusion that the internal space of the satellite is considerably limited. Therefore, it is necessary to ensure the effective arrangement of individual components and subsystems. This is why the concept of a stack was introduced. It refers to the style of mounting PCBs in the satellite interior and their subsequent fastening. Individual boards have limited space and are connected using board-to-board connectors. [1]

Overall structure must ensure sufficient mechanical integrity to withstand exposure to launch loads and subsequent deployment conditions. These loads mainly refer to the effects of the launch vehicle, such as vibrations or shock events. Therefore, the structure must be modeled to be sufficiently strong to protect individual subsystems.

1.2.3 Rail Interface and Surface Requirements

The rail system mentioned in previous sections is the main interface between the CubeSat itself and the deployment mechanism. During integration and subsequent deployment, the satellite is guided along rails inside the deployer, ensuring controlled movement and proper alignment during the actual deployment process. This interface plays a critical role in preventing mechanical interference and helps to ensure that a stable load is applied to the CubeSat or dispenser throughout the ejection sequence. For these reasons, the CDS again specifies and strictly defines the requirements to be met by this interface. [1]

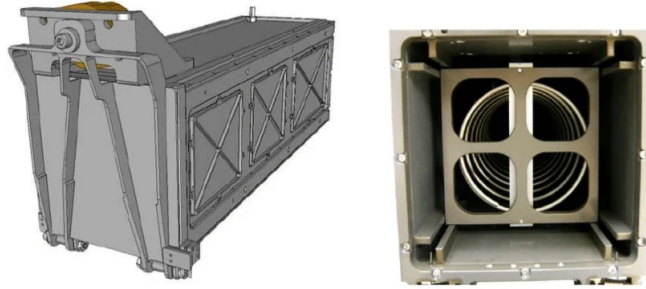


Fig. 1.3: Example of deployer and rail mechanism [5]

CubeSat design specifications define several geometric constraints for rail interfaces to ensure compatibility with the standardized deployer model. The main idea is that the deployer guide rail should provide sufficient contact with the satellite. The document specifies that the width of the satellite's rail contact surface should be at least 8.5 mm wide. This distance is always measured from the edge of rail surface to the first outer protrusion. It is further defined that there must be more than 75% physical contact along the length of this area. The resting 25% of this area can be recessed, for example, to accommodate deployment switches. [1]

In addition to the geometric restrictions the standard specifies requirements for the surface quality of these rails. The roughness of the rail surface should not exceed 1.6 μm . This reduces friction at this interface and helps to mitigate possible satellite jamming. It is also recommended that the edges of these rails to be rounded in order to achieve no mechanical interference at the corners of the guide rails. For external aluminum surfaces that come into contact with the deployer, a surface treatment, typically hard anodizing, is required to prevent so-called cold welding and to further smooth the contact between satellite and deployer rail surfaces. [1]

The mechanical requirements outlined in the foregoing subchapters define the fundamental constraints that determine the direction in which the structural configuration and external interfaces should be designed. Compliance with these require-

ments is the cornerstone for ensuring compatibility with standardized launchers and the safe satellite deployment into orbit. These parameters therefore represent key aspects that must be review during the subsequent verification and testing process.

1.3 CubeSat System Architecture

A CubeSat is a coherent system composed of several functional subsystems that work together to ensure the successful completion of a given mission. Each subsystem has its own responsibilities and a specific set of tasks, such as generating and distributing electrical power throughout the satellite or communicating with ground-based stations. These subsystems must then take into account the satellite's limited volume and weight constraints while maintaining reliability and enabling interaction between them. The overall system architecture plays a key role to ensure the satellite's operability and cohesion.

1.3.1 Typical CubeSat Subsystems

As mentioned above, CubeSats typically consist of several functional elements that together ensure the proper operation and performance of the satellite during a given mission. The individual subsystems are responsible for specific functions and also cooperate with other subsystems to form a fully functional system. Conceptually, we can divide them into the following categories.

The structure mentioned earlier is also seen as a subsystem. Its purpose is to provide a mechanical platform that offers sufficient mechanical strength and durability to protect the sensitive components inside the satellite. The main source of energy for satellites is typically solar panels, and this energy must also be stored in some way, most commonly in batteries. This is handled by the Electrical Power System, also known as EPS. The EPS is further responsible for distributing this energy throughout the entire spacecraft. Another critical component is the communications subsystem. As the name implies, its primary purpose is to ensure contact with the mission control center. Radio transceivers and associated antennas are the most commonly used means of achieving this. [1][6]

In addition, a satellite may also be equipped with an Attitude Determination and Control System or ADCS. Its primary function is to control the satellite's orientation in orbit, using a combination of sensors and actuators. As a matter of course, this is the only subsystem that is not required to be present on a satellite. [1][6]

Overall control of the satellite is handled by the On-Board Computer, or OBC (sometimes also referred to as Command and Data Handling - C&DH). This subsystem serves as the satellite's central processing unit, managing on-board operations, processing data, and coordinating the internal communications of the satellite. [1][6]

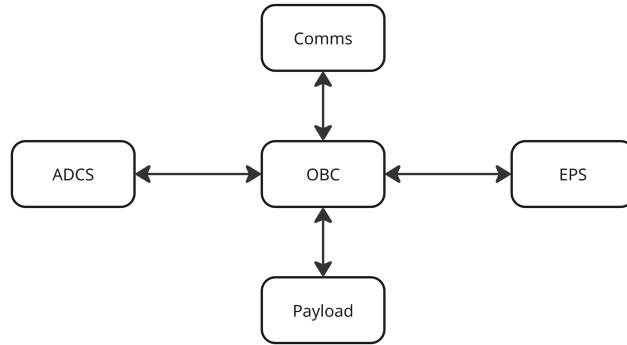


Fig. 1.4: Block diagram of CubeSat subsystems

The figure 1.4 above illustrates the theoretical flow of commands and information via satellite in autonomous mode. If the OBC itself issues a command instructing otherwise, or if the operator does so via a communication link, the individual systems can also communicate directly with each other without OBC intervention.

Whether the last of the subsystems can also be considered a payload. This is not a specific device with a precise purpose, but rather the reason for which the mission is being carried out. It can therefore involve various categories of devices, which may subsequently fall under other subsystems as well. An example would be that, in the case of communications satellites, the transceiver is both a communications subsystem and a payload. [1][6]

1.3.2 PC/104 Standard

PC/104 is a widely used standard for integrating electronic subsystems or other embedded systems into the CubeSat platform. This standard defines not only the electrical and mechanical properties of PCBs but also the associated interfaces for a compact, modular design. This standard was created to provide a space-efficient and robust architecture for applications with strictly constrained size and power consumption. [7]

A key feature of the PC/104 architecture is its self-stacking capability; more specifically, individual modules can be connected directly via board-to-board connectors. This approach allows multiple subsystem boards to be stacked vertically

on top of each other, creating a compact yet mechanically stable assembly. The standard further specifies that PCBs should not exceed 90x96 mm in size. Its name is derived from the fact that it utilizes a 104-pin interstack connector. [7]



Fig. 1.5: Showcase of PC/104 PCB [8]

The modular nature of the PC/104 standard allows for flexible configuration, with each board implementing specific subsystems, as mentioned in the preceding sections. This further simplifies the integration of the final system and the testing process, since the subsystems can be developed and verified independently even before they are integrated into the final satellite. [7]

2 Selected CubeSat System - KOSTKA

The satellite ultimately selected for this thesis was the project named KOSTKA. This is a mission led by a student team called YSpace, based at the Brno University of Technology (BUT). It serves as a technology demonstrator for in-house developed components, and the launch of this mission is scheduled for the end of the second quarter and beginning of the third quarter of 2026.

The original primary objective of the mission was to test the functionality of Miniaturized Heat Switch (MHS) and establish its flight heritage. However, due to delays in the development of the MHS, the primary objective of this mission was changed during development. At this stage of the project, the secondary objective of the mission was the measurement of background noise in the UHF spectrum using a dedicated onboard payload. Following the cancellation of the MHS payload, this became the primary objective of the mission.

The dedicated payload for the measurement of interferences is a custom-developed radio and antenna system based on TunaCan format. These modules are tuned to operate effectively at UHF frequency band. The main goal is to collect data about the noise and also to verify the performance of these module for their potential use in future missions.

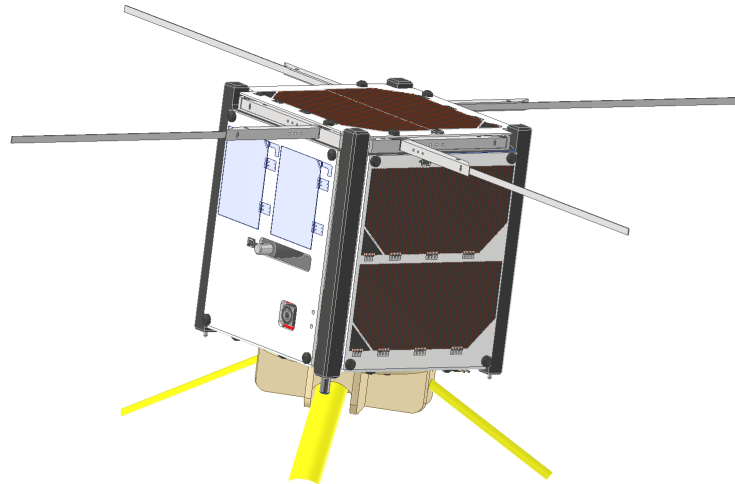


Fig. 2.1: Render of KOSTKA CubeSat

In addition to the mission's primary objective, the satellite carries an OBC developed in-house. The secondary objective is to verify the functionality of this module and provide it with flight heritage.

Beyond its technical objectives, the satellite is also equipped with a camera that will periodically take images and transmit them back to Earth via radio communication, where this data should be accessible to the general public-specifically, amateur radio enthusiasts.

2.1 Mission Technical Baseline

This section defines the mission technical baseline for the selected satellite. The baseline represents a set of key system-level parameters that characterize the KOSTKA from a structural, operational, and environmental perspective. The purpose of this overview is not to provide a detailed description, but rather to establish the fundamental constraint and boundary conditions under which the satellite operates.

Table 2.1: Baseline parameters of KOSTKA

Parameter	Value
Form factor	1U+ CubeSat
Dimensions	10 x 10 x 11,3 cm
Weight	~1 kg
Orbit	LEO; SSO; ~550 km
Communications	UHF; 436.870 MHz
Temperature range	-40 °C - 70 °C
Payload	experimental UHF receiver
1. resonance (From STM Vibe)	674 MHz
Deployables	TunaCan antenna; UHF antenna
ADCS	PMAGCS

2.2 Subsystem Overview

This section provides a high-level overview of the main subsystems of the selected spacecraft. Each subsystem is briefly described in terms of its main functions and overall role in the spacecraft as a whole.

2.2.1 On-Board Computer

The on-board computer for the KOSTKA mission is an in-house developed module based on the PC/104 standard, whose functionality is built around the STM32H723 32-bit Arm Cortex-M7 microcontroller. This chip provides the main computing

power for the entire module, and one of its key advantages is its compatibility with many RTOS systems. This processor is complemented by a secondary STM32U75 microcontroller, which is also a 32-bit Arm processor, whose primary intended purpose is to support the operation of a predefined payload. Although the module is developed specifically for this mission, its design also takes general applicability into account, so that the module can be used in other CubeSats as well.

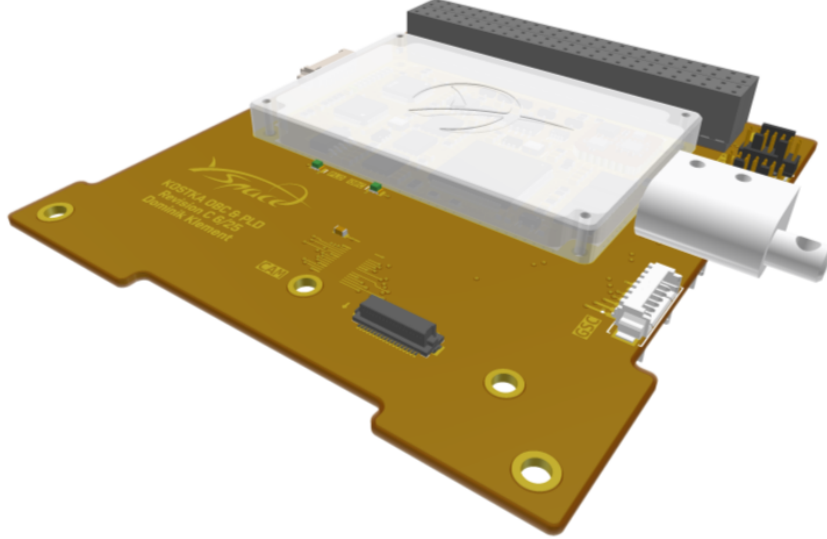


Fig. 2.2: KOSTKA On-Board Computer

The main objective of this module, in the scope of the KOSTKA mission itself, is the overall operation of the satellite. The H7 processor is responsible for managing communication between the other subsystems and storing the acquired data, i.e., on-board data handling in general. In this case, the U5 microcontroller manages the thermometers distributed throughout the structure of the satellite. It can also serve as a partial replacement for the H7 unit in the event of its failure.

2.2.2 Electrical Power System

The electrical power system is responsible for the production, distribution, and storage of energy on the satellite. Since this is a highly critical component for the success of the entire mission, a commercial off-the-shelf (COTS) solution was chosen. Consequently, the AMUN EPS from SpaceManic was selected. This subsystem regulates and distributes power to individual subsystems via dedicated power buses. The EPS architecture is based on centralized power management, which includes voltage regulators and protective mechanisms to ensure the safe and reliable operation of the entire satellite.

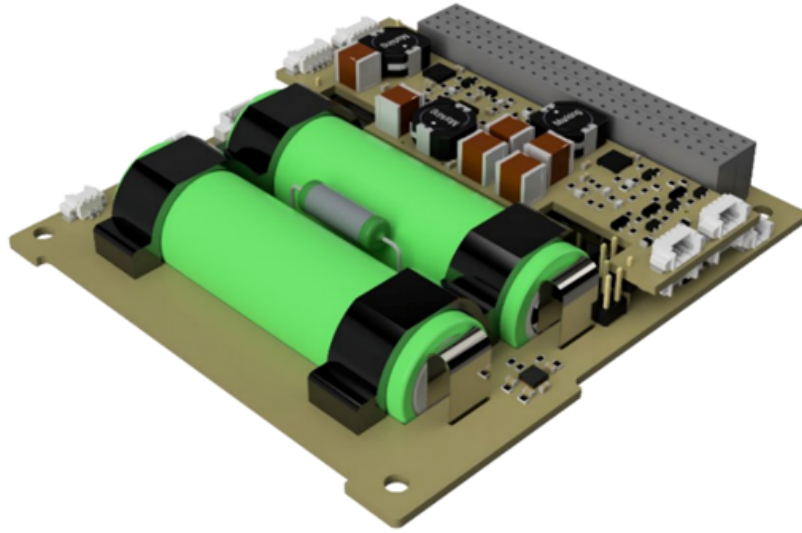


Fig. 2.3: AMUN EPS [9]

Solar Arrays

As mentioned earlier, the EPS also manages photovoltaic panels. These are manufactured in-house and use GaAs cells for maximum efficiency. The EPS has three inputs with MPPT regulators, so the individual solar panels are connected in such a way that the opposite side (i.e., X+/X-, etc.) is always connected to the same input. The individual panels are also equipped with measurement electronics, such as photodiodes to determine the satellite's orientation or thermometers to monitor their temperature.

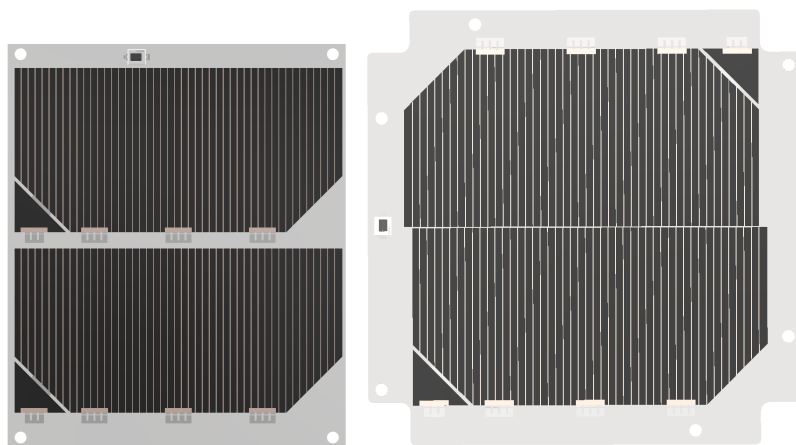


Fig. 2.4: Solar panels

2.2.3 Communications

The communication unit consists of two parts: the comms stack module and the antenna unit. Its main purpose is to transmit acquired data to the ground station as well as to receive operator commands. The communication PCB is based on the MURGAS unit, which is also manufactured by SpaceManic. This is a radio transceiver operating in the UHF band. It is then complemented by an antenna from 2NDSpace, which is tuned to the given frequency band. Here again, both components are commercially available solutions, and for a very similar reason as with the EPS: they are critical devices for the success of the entire mission, so a proven, functional solution was chosen.

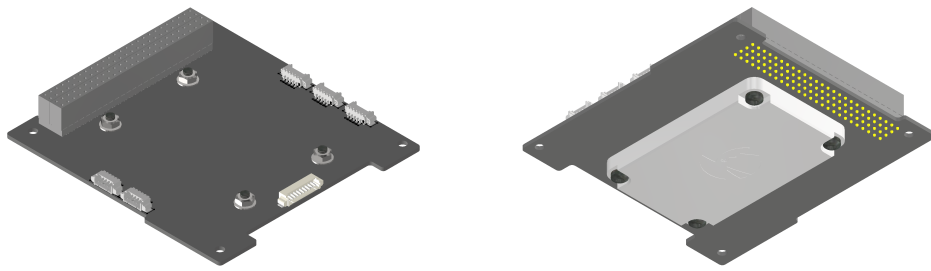


Fig. 2.5: Communications board

2.2.4 Payload - Experimental UHF receiver

As mentioned above, main payload of this mission is an experimental radio. Like the communication subsystem, the payload consists of a radio board and an antenna. Both of these components were developed and built in-house. The radio itself operates in the UHF frequency band. Though in this mission it functions solely as a receiver. The antenna is based on a so-called TunaCan design.

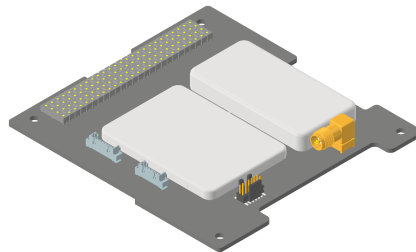


Fig. 2.6: Payload PCB

2.3 Architecture

This section describes the overall physical configuration of the selected satellite. KOSTKA is constructed on the CubeSat platform, which defines the external dimensions, structural layout, and integration constraints.

The satellite adheres to the Standard Plus form factor, with all subsystems, including the payload, integrated into a single common internal stack. This approach allows efficient use of interior space while also simplifying the final integration and interconnection between the individual subsystems.

The internal layout is limited by the structural requirements of the CubeSat itself and the need to comply with the deployer's interface specifications. The result is a design that emphasises compact stacking of modules and the efficient allocation of the available space inside the spacecraft.

To determine the satellite's orientation, a Cartesian coordinate system is used, with the origin of this system located at the geometric center of the considered satellite. The principal axes of the coordinate system are then aligned with the CubeSat body. This convention is used throughout the following subsections when explaining the placement of individual subsystems and external components.

The overall configuration reflects a balance between functional requirements, mechanical constraints, and considerations regarding the resulting system integration, thus forming the basis for the internal architecture described in the following sections.

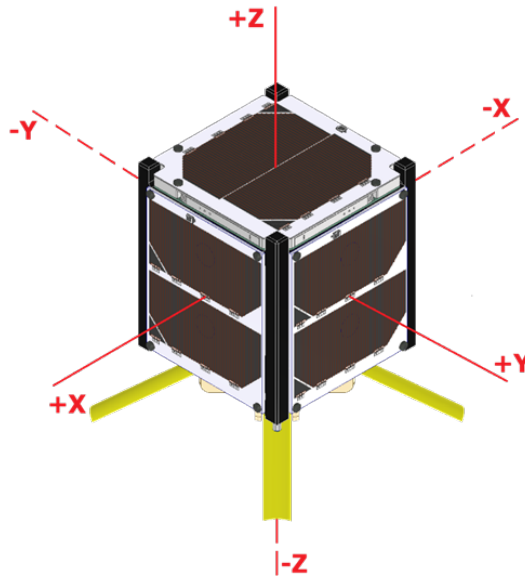


Fig. 2.7: KOSTKA coordinates reference frame

2.3.1 Internal Architecture

This subsection presents the internal architecture of the KOSTKA satellite, focusing on the layout and integration of individual subsystems throughout the satellite's usable volume. It focuses primarily on the arrangement and configuration of the stack of onboard electronics and other internal components.

The internal architecture of this satellite is based on a stacked modular approach that utilizes the PC/104 standard. As mentioned in previous chapters, this approach uses board-to-board connectors to interconnect individual boards, which allows for vertical interconnection and thus creating a compact, self-supporting structure that does not require an interconnect unit or board.

The main guiding principle in configuring the stack was to simplify as much as possible the cabling that connects the individual external components to their respected control units located inside the satellite. For these reasons, both communication boards, namely the experimental receiver and the Communication PCB, are located at opposite ends of the internal stack so that the external cabling leading to the respective antennas is as short as possible. The Communication PCB is therefore located on the Z+ side, and the experimental receiver on the Z- side. Connected to the experimental receiver is the onboard computer, to which the EPS is subsequently interfaced. This configuration was chosen because a solar panel will also be located on the Z+ side on the antenna module, and thus this layout remains consistent with the goal of shortening the harness as much as possible.

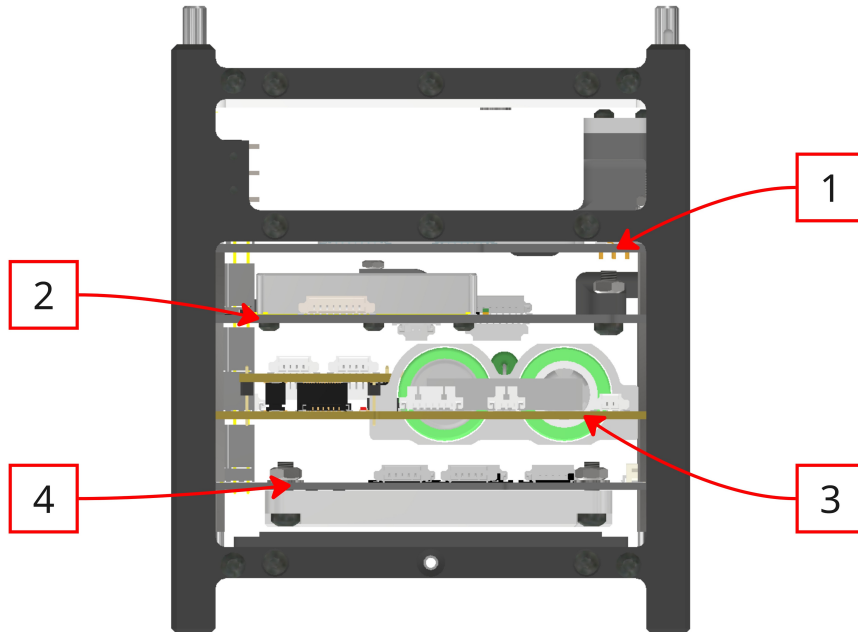


Fig. 2.8: KOSTKA stack configuration

Table 2.2: Placement of individual subsystems in the stack

Position	Subsystem
1	Experimental UHF Receiver
2	OBC
3	EPS
4	Communications Board

The PC/104 interface provides a unified connection for power distribution across the individual modules in the stack, while also enabling bus and data connections between them, thereby again minimizing the amount of cabling required.

The stack itself is secured to threaded rods protruding through a certain part of the satellite’s interior. The individual spacing is defined by spacer studs screwed onto these threaded rods. This ensures a fixed spacing between the individual boards and contributes to the structural integrity of the entire spacecraft.

Overall, the chosen stack architecture enables a compact and well-integrated system while minimizing the complexity of the internal harness. At the same time, it presents limitations in terms of accessibility and thermal management, which must be further subsequently considered.

2.3.2 Mechanical Interfaces

The interface between the KOSTKA satellite and the deployer is covered via guide rails located along the longitudinal edges of the structure. These are designed to meet all the requirements specified by CDS while minimizing the impact on the satellite’s internal space. The rails thus serve as the main contact surfaces during integration and deployment.

In addition, the structure is equipped with functional elements at the interface between the satellite and the deployer, which also ensure reliable and safe deployment of the satellite. KOSTKA is equipped with so-called separation springs, which are located in opposite corners on the side Z- of the structure. Here, they come into contact with the deployer wagon itself, helping to separate it from the structure. During integration, this mechanism is compressed into a position where it is flush with the structure.

Furthermore, kill switches or, more specifically, deployment switches, are integrated into the structure. These are positioned diagonally in the second set of opposite corners on the side of Z-. The switches remain engaged for the duration of the satellite’s presence in the deployer. After deployment, disengaging this mechanism ensures that the satellite’s electronics are activated, thereby allowing the satellite to enter operational state.

2.3.3 External Components

This subsection covers the external components of the KOSTKA satellite, such as solar panels, antennas, and similar elements. It focuses on their configuration in terms of how they influence the functionality of the system and their closely linked connection with the internal architecture.

Most of the external components can be divided into two categories. The first category is communication. It consists of two antennas, a UHF antenna and a TunaCan antenna, which are located on opposite sides of the Z axis. The UHF antenna is located on the Z+ side. It is a typical low-profile CubeSat antenna design that, together with the transceiver on the communication board, serves as the main channel for communication with the satellite itself. The TunaCan antenna is then located on the opposite side, i.e., on the Z- side. It serves for the purposes of an experimental receiver. As the name implies, it is based on the TunaCan principle, meaning it is a cylindrical protrusion. Both of these antennas can also be classified as deployables, as they are folded during integration and launch. After the satellite is launched, they subsequently unfold; both antennas feature a 4-element configuration, where two opposite elements are deployed sequentially, meaning the antennas open in two stages.

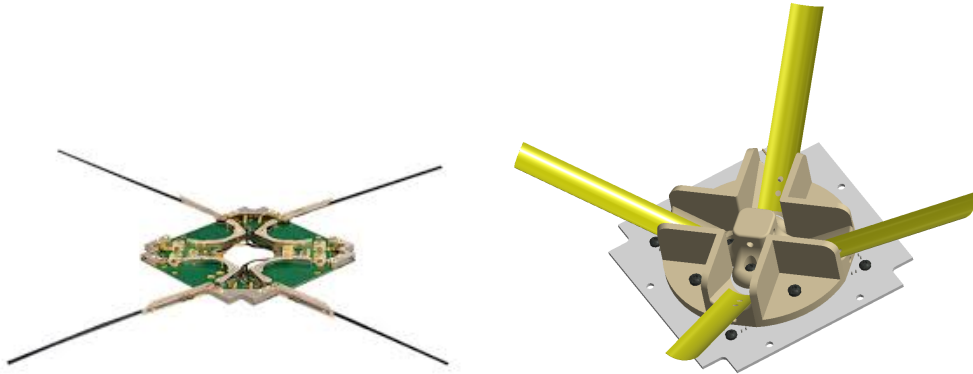


Fig. 2.9: UHF (showcase) [10] and TunaCan antennas

The second category consists of solar panels. These are distributed across five of the satellite's six possible sides. There are three different types of solar panels. The first type features two large cells designed to meet the dimensional constraints specified by the CDS while also fitting into the designated position. This type is located on the X+, Y+, and Y- sides. On the Y+ side, there is a solar panel of the same shape, but it features only two small solar cells at the bottom of the panel. Its upper part is utilized for openings serving the RBF, GSE connector, and the camera lens. The fifth solar panel is then located on the Z+ side, i.e., on top of the UHF

antenna. Here, the electrical configuration is similar to that of the side solar panels, i.e., two large cells. However, the shape and thickness of the panel are modified to ensure compliance with all standards.

The final external component is the secondary payload, namely the camera. As mentioned in the previous paragraph, it is mounted on the X- side. It is positioned that most of the module is hidden behind the solar panel, while the camera lens protrudes to the outer edge of the solar panel through the previously mentioned opening in it.

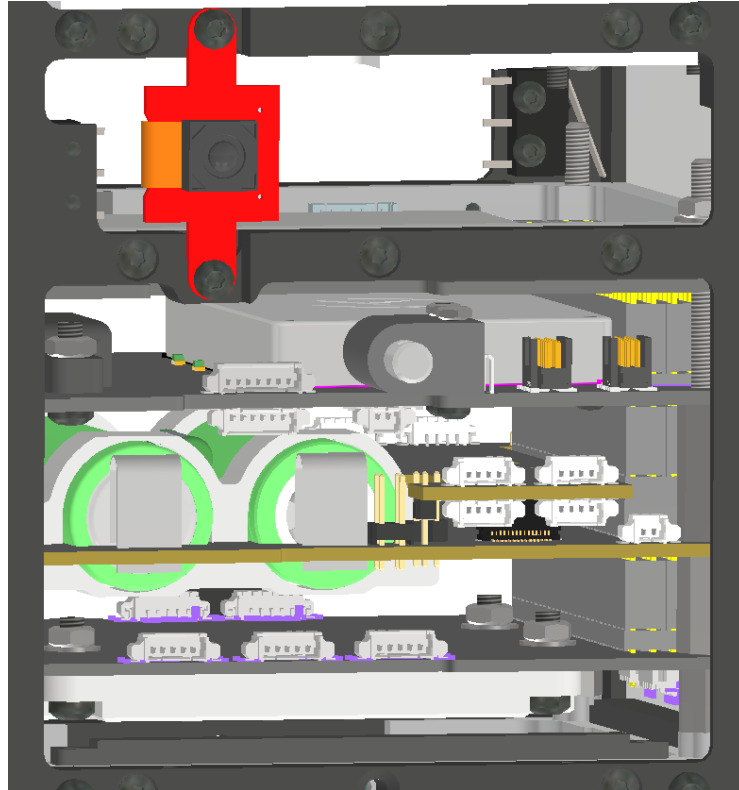


Fig. 2.10: Camera placement

The described architecture of the KOSTKA satellite forms the basis for subsequent integration and testing activities. The chosen compact stacked configuration, together with the placement of external components and mechanical interfaces, directly influences maintainability and testability, and thus the resulting definition of verification procedures. In particular, the high level of subsystem integration and the minimization of internal wiring require careful consideration during testing, as individual components cannot be easily accessed or isolated. The presented configuration therefore establishes the framework conditions for the verification approach described in the following chapters.

3 Overview of CubeSat Testing methods

Testing is an integral part of the design and verification process for space systems. In the case of satellites, particularly small satellites such as CubeSats, it plays a key role in ensuring reliability and functionality under conditions where post-launch repairs or maintenance are not possible. Unlike ground-based systems, satellites must operate autonomously throughout the entire planned mission duration, which significantly increases the demands on the quality of development and the resulting performance verification.

In the context of systems engineering, testing is one of the primary methods of verification, aimed at demonstrating compliance with defined requirements at the component, subsystem, and system levels. This process begins in the early stages of development and continues throughout the entire project until the final pre-launch testing. Insufficient or poorly designed testing can lead to late detection of design or manufacturing flaws, which significantly increases the risk of a mission failure. [11]

From a practical standpoint, testing can be divided into several basic categories. Environmental tests simulate the conditions during launch and the subsequent environment in which the satellite will operate, while functional and performance tests verify that the system is working properly. Together, these tests provide a sufficient degree of confidence that the satellite will be able to fulfill the mission goals throughout its operational life.

3.1 Classification of Tests

In order to effectively verify the performance and reliability of the CubeSat, testing activities are divided into several categories based on their purpose and the conditions they are intended to simulate. This classification provides a structured framework for organizing verification processes and ensures that all relevant aspects of the system are addressed.

The classification of tests also reflects the various phases of system development and the level of integration, range from individual components to the entire satellite itself. The subsequent selection of specific tests then depends on mission requirements, system complexity, chosen modeling philosophy, or the requirements of the launch provider.

3.1.1 Environmental Tests

Environmental testing is a crucial part of the verification process for space systems. It is used to demonstrate that a satellite can withstand the conditions to which it is exposed to during launch and subsequent operation in orbit throughout its lifecycle. These conditions include mechanical stresses during launch, thermal and vacuum conditions in orbit, and the electromagnetic and radiation environment throughout the satellite lifecycle.

According to ECSS standards, environmental tests are defined as tests that simulate the environmental conditions to which a satellite will be exposed to during its operational life. Their purpose is not only to verify compliance with specified requirements, but also to provide confidence in the satellite's design and craftsmanship. [11]

Environmental tests are conducted within the context of a structured test program, which is defined as part of the overall verification process and is implemented across various levels of system decomposition, such as the equipment and system levels. The selected tests, their sequence, and the levels at which they are applied depend on the chosen modeling philosophy, meaning the use of a qualification, flight, or protoflight model. [11]

Various testing approaches are used in this context. Qualification tests are conducted at elevated levels to demonstrate the design margins. From this perspective, so-called acceptance testing is then used for the final flight model, where testing is performed at reduced load levels to avoid overloading the flight systems while verifying their integrity. The Protoflight approach combines these two methods by applying higher load levels over a shorter period of time directly to the flight model. Thus, the need to produce multiple units is eliminated. [11]

The main categories of environmental tests relevant to CubeSats are mechanical stress tests, thermal-vacuum tests, electromagnetic compatibility tests, and radiation resistance tests. These categories correspond to the primary conditions that stress and degrade the satellite during its operational life in orbit. These individual categories are then divided into specific types of tests, each performed according to different methodology.

3.1.2 Functional and Performance Tests

Functional and performance tests are intended to verify that the satellite operates correctly and meets all the requirements defined by the mission. While environmental tests focus on the satellite's ability to withstand the external conditions of space, these tests aim to verify the satellite's proper behavior and evaluate its performance under the nominal conditions in which it will operate.

The primary objective of functional tests is to verify that individual subsystems, and subsequently the system as a whole, respond correctly to individual commands and operate as defined by the requirements. This includes verifying the proper functionality of the onboard software, communication interfaces, and telemetry acquisition systems, as well as verifying the proper transition between the satellite's different modes. These tests ensure that all system functions are implemented correctly and that the interaction between individual subsystems, or with the ground station, is consistent and reliable. [11][13]

On the other hand, performance tests are designed to demonstrate that the system meets the specified performance requirements. This includes metrics such as positioning accuracy, power consumption, data channel throughput, and the quality of the communication link between the satellite and the ground station. The goal is not only to demonstrate that the satellite is operating correctly but also that it meets the required levels that define the criteria for a successful mission.[12]

According to established verification practices, these tests are conducted at various phases of the satellite's composition, ranging from individual components to the fully integrated satellite. Functional and performance tests are routinely conducted throughout the entire development process. They are most commonly used, for example, in environmental testing, where they are performed before and after the environmental tests are completed. This allows verifying that during the simulation of the conditions in which the satellite will operate, there is no degradation of its performance that could occur due to these environmental stresses. [11]

A typical use of functional and performance tests is their inclusion in end-to-end testing. End-to-end testing represents a complete simulation of a mission scenario in which all relevant parameters are tested under the conditions expected to be present in the space where the satellite will operate. These tests are typically performed using GSE or simulators of these systems so that the given conditions are replicated as accurately as possible and the results are valid.[12][14]

The evaluation criteria for these tests are based on the system's correct response, operational stability, and compliance with the defined performance requirements for the spacecraft. Any deviations or unexpected behavior of the device are then classified as anomalies and must be further analyzed during the verification process.

3.1.3 Integration and System-level Tests

Integration and system-level tests are designed to verify proper interaction between individual subsystems and validate the functionality of the entire satellite. These tests represent a critical step in the verification process, as they ensure that the system as a whole operates just as coherently as the individual modules do on their own. [12]

Integration tests focus on the interfaces between individual subsystems and on identifying potential incompatibilities that may be encountered during the integration process. As individual components are integrated into more complex assemblies, verification processes are performed at these interfaces to ensure proper data exchange, electrical compatibility, and that everything fits together mechanically. This procedure is essential for detecting flaws that cannot be identified during testing of the individual components of each subsystem. [12][14]

System-level tests are then performed on the fully integrated satellite, and their primary objective is to verify compliance with system-level requirements. These tests typically include verification of the system's overall functionality, simulation of individual scenarios expected to occur during the mission, and end-to-end testing, which is then performed on both the space and ground segments. An example of part of this process could be, verifying the proper operation of LEOP and the subsequent ability to establish communication. The goal is to demonstrate that the system is capable of functioning as required by the mission under representative operating conditions. [12][14]

Similar to other types of testing, integration and system-level tests are performed incrementally across individual system levels in accordance with established practices. These activities should, in turn, be defined in the overall assembly, integration, and test program, which specifies the individual steps of the procedures, the associated tests, and the expected verification materials. [15]

The evaluation criteria for integration and system-level tests are based on the correct operation of individual interfaces, the successful execution of system functions, and the absence of any anomalies during system operation. If such anomalies do occur, they must be addressed during the verification process itself so that the system is fully functional at the end and it is ensured that the final configuration meets all specified requirements.

3.2 Environmental Testing methods

Following the previous general description of environmental tests, this section focuses on specific test methods used to simulate the environmental conditions a satellite may encounter. Each individual test is further divided into specific variants that differ in their purpose, test setup, and the specific conditions they are intended to reproduce.

For each category of environmental testing, a structured approach is applied, which clearly outlines the objective of each test, how to conduct it, and how to properly evaluate its results. Emphasis is placed on the relationship between test conditions and the corresponding actual mission environment, as well as on the criteria subsequently used to assess system integrity and performance.

The methods presented reflect standard practice used for the verification of space systems, where tests are designed to demonstrate compliance with defined requirements and to identify potential weaknesses in design and implementation. The subsequent interpretation of test results is a crucial part of the verification process, providing the basis for confirming the satellite's readiness or identifying the need for further design corrections.

The following chapters therefore describe the details of environmental testing methods, such as vibration, thermal vacuum, electromagnetic, and radiation tests.

3.2.1 Vibration Testing

The purpose of vibration tests is to simulate the mechanical stresses applied to the satellite during launch. These stresses are typically caused by several factors, including the rocket's propulsion system, aerodynamic effects, and the structural coupling between the payload and the launcher. The primary objective of vibration tests is to verify that the satellite can withstand these dynamic conditions and that no structural damage or functional degradation would occur.

In addition to verifying structural integrity, vibration tests are also used to identify potential weaknesses in the mechanical design, such as insufficient stiffness of the used materials, loose components, or poor mounting of subsystems. These tests also serve to validate analytical models and ensure that the satellite maintains its functionality and performance after being exposed to launch conditions.

Vibration tests are also commonly divided into several categories, each of which represents a different type of mechanical loading that may be encountered.

Sine Vibration

Sine vibration tests are primarily used to identify structural resonances and verify the dynamic behavior of a given satellite under controlled harmonic loads. This type of testing provides valuable insight into the natural frequencies of the structure itself, which makes it possible to verify the analytical models used during the design process. [16]

This test is typically performed as a swept sine excitation, in which the frequency of the load signal is continuously increased while the acceleration remains constant. According to ECSS standards, these tests are usually conducted over a frequency range of 5 to 2000 Hz. The exact frequency spectrum is often variable, and its range is defined according to the requirements of the launch vehicle and the mission. These excitations are typically applied sequentially along all three principal axes of the satellite. Typical acceleration levels are primarily defined by the launch vehicle itself, to which a margin is subsequently applied based on the selected verification approach. [11][16]

One of the key outputs of sine vibration tests is the identification of resonance frequencies. These are represented as peaks in the measured spectrum of the structure's response. These frequencies are compared with the assumptions established prior to the tests, with any significant deviations indicating that a change in structural stiffness, loosening of mechanical connections, or potential damage may have occurred. A commonly used evaluation criterion is to compare the movement of the aforementioned resonance peaks before and after testing. [11][16]

Table 3.1: Sine vibration quick-fact table

Parameter	Typical Value / Range	Description
Frequency range	~5-2000 Hz	Defined by launcher
Sweep rate	1-2 oct/min	Rate of frequency change during test
Acceleration levels	0,5-2 g	Depends on test approach and launcher
Test axes	3 axes (X, Y, Z)	Sequential excitation in principal axes
Test type	Sweep sine	Harmonic excitation with varying frequency
Evaluation criteria	Resonance shift; No structural damage	Comparison before/after test

From a practical standpoint, sine tests are useful for detecting issues such as loose components, insufficient joint strength, or structural couplings between subsystems. Due to their controlled nature, they are often used as a diagnostic tool before and after tests that place greater stress on the system, such as random vibration tests, allowing us to subsequently determine whether structural degradation has occurred.

Random Vibration

Random vibration tests are used to simulate the broadband dynamic environment generated by the launch vehicle itself. Unlike sine vibration tests, which apply deterministic harmonic excitation to the satellite, random vibration tests use a stochastic input defined by a Power Spectral Density (PSD) profile. This approach provides more realistic representation of the complex vibrating environment present during the launch.

These tests are conducted across a specific frequency range, which is most often defined directly by the requirements of the launcher itself or by applicable standards. This excitation is applied simultaneously across a wide frequency spectrum, with the energy applied to the structure distributed according to a defined PSD curve. Such tests are typically performed sequentially along all principal axes of the satellite, as indicated by standard testing practices. [11]

In practical applications, random vibration tests are one of the primary methods for verifying the structural integrity of a spacecraft in the conditions of launch. Environmental verification programmes, as defined in standards, require that tested or analysed hardware demonstrate the ability to withstand the expected mission conditions. Random vibration tests are therefore commonly included in both qualification and acceptance campaigns. [16]

Qualification testing is conducted under more severe conditions than those expected during actual flight, in order to demonstrate sufficient design margins. Acceptance testing, on the other hand, is conducted at reduced levels to verify the integrity and quality of workmanship without unnecessarily stressing the flight unit. This distinction forms the foundation of the overall verification philosophy used in aerospace engineering and is also reflected in CubeSat development practices. [16][11][14]

As indicated in the preceding text, the PSD profile is a key parameter for random vibration tests and is typically expressed in units of g^2/Hz . The PSD integral over the entire frequency spectrum defines the total Root Mean Square (RMS) acceleration levels applied during the test. This value represents the total energy input into the structure and is commonly used as an indicator of the test's severity. [16]

Random vibration tests are particularly effective in detecting structural flaws that may not be apparent in sine tests, such as issues related to material fatigue, insufficient structural damping, or failures at mechanical interfaces. The evaluation criteria for this type of vibration testing include indicators such as the absence of structural damage, a stable dynamic response, and proper satellite behavior before and after the test. Functional tests are typically performed before and after the satellite is subjected to vibrations, which is a standard verification principle applied throughout the entire test phase. [12]

Table 3.2: Random vibration quick-fact table

Parameter	Typical Value / Range	Description
Frequency range	~20-2000 Hz (typically)	Frequency band over which the random vibration is applied, defined by launcher environment
PSD level	Mission-specific; typically 0.01-0.1 g ² /Hz	Power Spectral Density defining energy distribution across frequencies
RMS acceleration	~6-15 g RMS	Integrated value of PSD representing total vibration energy applied to the structure
Test axes	3 axes (X, Y, Z)	Sequential excitation in principal axes
Test duration	60-180 s per axis	Duration of vibration exposure for each axis
Qualification level	typically +3 dB to nominal flight level	Used to demonstrate design margins
Evaluation criteria	No structural damage; No degradation in performance	Comparison before/after test

Shock

Shock tests are conducted to simulate high-frequency transient loads generated during discrete events in the launch or deployment sequence, such as rocket stage separation, payload deployment, or activation of the deployment mechanism. These

events generate short-duration, very intense mechanical shocks that can affect sensitive components of the spacecraft.

A shock environment is typically interpreted by Shock Response Spectrum (SRS), which represents the maximum response of a single-degree-of-freedom system to which a shock input is applied. This allows for the description of highly complex transient events and their application in a standardized and comparable form across various test setups and mission scenarios. [16][17]

In practice, shock tests are rarely applied to CubeSats, mainly because the shock environment associated with the standardized deployers of these types of satellites is typically very mild. However, if a mission involves a specific deployment mechanism or if the test is specifically required by the launch provider, these tests remain relevant, as the standards for environmental verification define them as part of the overall mechanical testing campaign. [11]

Shock tests are usually performed using specialized equipment capable of generating high-frequency transient responses, such as mechanical impact devices or pyroshock simulators. The resulting shock input is then measured using high-frequency accelerometers, and the recorded response is compared to the required SRS envelope. [11][18]

Table 3.3: Shock test quick-fact table

Parameter	Typical Value / Range	Description
Frequency range (SRS)	~100 Hz - 10 kHz	Frequency range over which the shock spectrum response is evaluated
Peak acceleration	~2000 g	Maximum acceleration level during shock event
Pulse duration	~0.1 - 10 ms	Duration of shock event
Test type	Pyroshock / mechanical shock	Type of shock depending on source
Test axis	3 axis (X, Y, Z)	Shock is typically applied in all principal axis
Evaluation criteria	SRS envelope compliance; No structural or functional failure	Comparison of measured SRS with required limits and post-test functionality check

The evaluation criteria for shock tests, similar to those for other mechanical tests, include the survival of all components, the absence of mechanical damage, and the maintenance of performance after the test. Again, as with other environmental tests,

functional verifications are typically performed before and after the test to ensure that no degradation has occurred. [12]

Quasi-static

The quasi-static tests are used to verify that the satellite can withstand low-frequency loads corresponding to the maximum acceleration experienced during launch. These loads represent static or slowly changing conditions, such as thrust buildup and the overall acceleration of the rocket. [16]

For small satellites, such as many CubeSats, the verification of quasi-static loads is usually performed by analysis rather than through dedicated tests. However, if tests are required for some reason, the most commonly used method to perform them is the sine burst test.

A sine burst test consists of several short sinusoidal excitations at a low frequency, typically applied over several periods (generally 5) with a predefined amplitude. This method allows for the simulation of quasi-static loading even when using standard vibration test equipment and while maintaining controlled conditions. [11]

The evaluation criteria for sine burst tests are based on a comparison of the input excitation signal and the measured structural response. For quasi-static behavior, it is assumed that the response will closely match the input, without any significant amplification. Any noticeable increase in amplitude then indicates the presence of dynamic effects, such as resonance, which may point to insufficient structural stiffness. A response equal to or smaller than the input is then generally considered acceptable and confirms that the structure behaves in a quasi-static manner. [16]

Table 3.4: Quasi-static test quick-fact table

Parameter	Typical Value / Range	Description
Frequency	~5-20 Hz	Typically $< 2/3$ of 1. resonance
Number of cycles	3-10 cycles	-
Acceleration level	Up to 10 g	Launcher-specific
Test type	Sine burst	Short sinusoidal excitation used instead of true static loading
Test axis	3 axis (X, Y, Z)	-
Evaluation criteria	No amplification of response; No structural failure	Verification of structural integrity under steady launch loads

Additional Verification During Mechanical Testing

During the environmental tests described above, there are multiple functional verifications performed, typically as part of the mechanical testing campaign. One of the most important of tests is the power inhibit test, which ensures compliance with launch safety requirements.

The purpose of the power inhibit test is to verify that the satellite remains electrically inactive during handling and throughout the launch. This is ensured by certain inhibit mechanisms, such as the remove-before-flight pin or the deployment switch. These mechanisms ensure that the satellite's onboard systems do not activate while the satellite is still located inside the launch vehicle.

Power inhibit verification is not an environmental test on its own. However, it is commonly performed during vibration tests to ensure that all mechanisms serving these purposes function correctly even under the mechanical stress conditions that arise during launch. This approach reflects standard practices, where functional checks are performed in parallel with environmental tests to validate the functionality of system components under representative conditions. [12][16]

The evaluation criteria for this test are, logically, the absence of any unintended activation of any of the systems under consideration. If these criteria are not met, it is considered a very critical failure, as it can directly and negatively impact launch safety. [16]

3.2.2 Thermal Vacuum testing

The purpose of thermal-vacuum tests is to simulate the combined effects of vacuum and temperature extremes that a satellite experiences during its operation in orbit. In the space environment, there is virtually no atmosphere, which means that no heat can be transferred by convection. Therefore, thermal conditions differ significantly from those on Earth. Consequently the satellite's thermal behavior is governed by conduction and radiation.

The purpose of thermal vacuum testing is to verify that the satellite can operate properly within the specified temperature range, including when exposed to vacuum levels. In addition to functional tests, this test is used to identify potential issues related to material properties, assembly quality, or thermal design. Potential issues may include, for example, improper heat dissipation, thermal gradients, or material degradation under conditions of repeated thermal cycling.

The thermal vacuum test is a key component of environmental verification, as it provides confidence that the satellite itself will be able to maintain its structural integrity and functional performance throughout its operational life under mission-defined conditions.

Test Principles

Thermal vacuum tests are based on the principle of simulating heat transfer conditions representative of the cosmic environment. The absence of molecules in a vacuum eliminates convective heat transfer, which under normal conditions accounts for a significant portion of total heat transfer. As a result, heat transfer is primarily driven by conduction through the structure and radiation between surfaces. This gives rise to a significantly different heat distribution compared to terrestrial conditions. [16][19]

To simulate these conditions, the satellite is placed in a vacuum chamber, where the pressure is reduced to levels representative of the orbit's altitude or the mission's target environment. Thermal control is then provided by shrouds, which surround the device under test and simulate a radiant environment. By adjusting the temperatures of these shrouds, it is possible to simulate both hot and cold orbital conditions. [16][11]

An important parameter of thermal vacuum testing is the establishment of thermal equilibrium. At each temperature extreme, the system is given time to stabilize, and functional tests are performed only after this stabilization has taken place. This stabilization is primarily carried out to ensure that the observed behavior is associated with a specific steady state and not with transient effects. [16]

Thermal Vacuum Test Variants

Thermal vacuum tests are typically divided into two categories, which differ in their objectives and evaluation methods. These categories are thermal vacuum cycling and thermal balance testing. Although both categories are performed using the exact same equipment, they serve two distinct roles within the narrower scope of the verification process.

Thermal Vacuum Cycling

Thermal vacuum cycling is used to verify that a satellite is capable of operating properly when exposed to the anticipated temperature extremes it will experience over the course of its lifetime in a defined orbit. This test is performed in the form of cyclic changes between predefined hot and cold temperature limits, commonly referred to as the hot and cold cases, while the satellite is immersed in conditions of vacuum. [11]

During this test, the satellite is subjected to several temperature cycles, with each cycle consisting of a transition between specified temperature extremes and a certain dwell time at each extreme. Following this dwell time, functional tests

are conducted to verify that the satellite is operating correctly. The most common procedure is to verify, during the first cycles, whether the satellite can be powered on at the defined temperature extremes. It is then left powered on, and monitored to ensure that no functional anomalies occur across the entire temperature spectrum. [11][16]

The main objective is to demonstrate both the functionality of the satellite itself and its structural integrity when subjected to thermal stress. Repeated cycling can also reveal defects related to the properties of the materials used, the quality of the assembly, or incompatibilities resulting from thermal expansion. [11][16]

Thermal Balance Test

The thermal balance test is, on the other hand, used to validate the satellite's thermal model in order to verify the accuracy of the thermal design predictions. Unlike thermal cycling, this test focuses on achieving a steady state of thermal conditions rather than on repeated transitions between extremes. [11][16]

During this test, the satellite is subjected to controlled thermal conditions within narrow limits that are intended to represent a specific operational scenario. The system is then allowed to reach thermal equilibrium, after which measurements of heat distribution throughout the satellite are taken. These are then compared with the predicted results from the analytical model, and the model is adjusted as necessary to match reality. [16]

The primary objective of the thermal balance test is to correlate the thermal model with the measured results, which ensures that the analytical model can subsequently be used to estimate the satellite's thermal behavior in orbit. This is a critical verification step during satellite development, as an accurate thermal model significantly contributes to the mission's success. [11][16]

Table 3.5: Key parameters of Thermal Vacuum testing

Parameter	Typical Value / Range	Description
Pressure level	$\sim 10^{-5}$ to 10^{-6} mbar	Vacuum level used to simulate orbital conditions; ensures negligible convection
Number of cycles	4-8 cycles	Number of cycles applied during cycling test
Temperature range	$\sim -20\text{ }^{\circ}\text{C}$ - $60\text{ }^{\circ}\text{C}$	Values typical for CubeSats
Temperature transition rate	$\sim 1\text{-}3\text{ }^{\circ}\text{C}/\text{min}$	Controlled ramp to avoid unrealistic thermal gradient and stress
Thermal balance points	2-4 cases	Steady-state conditions; hot/cold + mission specific
Number of sensors	10-50 sensors	Typical value for CubeSats; Placed on the critical components and surfaces
Evaluation criteria	No functional failure; Stable temperatures; Model corelation within $\pm 3 - 5\text{ }^{\circ}\text{C}$	Acceptance based on performance and agreement with thermal model

Outgassing

Outgassing is a process in which volatile compounds are released from materials during exposure to a vacuum. In the absence of atmospheric pressure, trapped gases and volatile substances begin to be released from the materials, which can lead to contamination of nearby surfaces and so on.

This effect is particularly critical for satellite components such as optical systems, sensors, and solar panels, as the layer of substances that can accumulate on them drastically reduces their efficiency and the quality of the resulting data.

During thermal-vacuum testing, outgassing is typically addressed through controlled exposure to heat, often referred to as a bake-out. Exposing the satellite, or its components, to elevated temperatures, or even a vacuum, allows volatile substances to escape from the materials in a controlled environment before launch. This reduces the risk of subsequent contamination while the satellite is in orbit.

In standard practice, components expected to be more susceptible to contamination by volatile substances, such as PCBs or parts made of various polymers, are subjected to elevated temperatures at normal atmospheric pressure. If the satellite is equipped, for example, with a camera system that is particularly susceptible to the accumulation of contaminants, a vacuum is also used for the outgassing process. In terms of standards, the acceptable rate of mass loss due to the release of volatile substances from the overall structure is defined as 1 percent of the satellite's total mass. [11]

Environmental verification standards define outgassing and contamination control as critical aspects of satellite verification, particularly when the satellite is equipped with a sensitive payload. The proper selection of materials, cleanliness control, and pre-deployment preparation processes are therefore key steps in the overall verification strategy. [16]

Table 3.6: Key parameters - Outgassing / Bake-out

Parameter	Typical Value / Range	Description
Bake-out temperature	~50-125 °C	Material dependent; Elevated temperature to accelerate release of volatile compounds
Bake-out duration	24-72 hours	Time required to sufficiently reduce outgassing effects
Total Mass Loss (TML)	$\leq 1\%$	Maximum allowable mass loss of a material due to outgassing
Collected Volatile Condensable Materials (CVCN)	$\leq 0.1\%$	Fraction of released material that can condense on sensitive surfaces
Evaluation criteria	Compliance with TML/CVCN limits	Acceptance based on material suitability and contamination control

3.2.3 Electromagnetic Compatibility

Electromagnetic compatibility tests are conducted to ensure that the satellite will operate properly in an electromagnetic environment without causing or being affected by unwanted electromagnetic interference. In a complex electronic system such as a CubeSat, several subsystems operate in close proximity, which increases the risk of electromagnetic coupling and interference.

The EMC tests have two objectives: first, to verify that the satellite itself does not emit electromagnetic interference that exceeds the specified limits; and second, to verify that the satellite can operate when exposed to external electromagnetic disturbances. These requirements are essential to ensuring reliable communication with the satellite and the proper functioning of its individual electronic systems. [20]

Unlike other environmental tests, which primarily simulate external influences such as mechanical or thermal effects, EMC focuses on the interaction between individual subsystems and their electromagnetic environment. This includes both internally generated interferences and external interferences, which can affect the performance of the system itself if not properly addressed. [20]

EMC verification is an important part of the overall verification process, ensuring that all subsystems can operate collectively without interfering with one another, while also verifying that the satellite meets general electromagnetic compatibility requirements.

Basic Principles

Electromagnetic compatibility is based on the principle of controlling electromagnetic interference inside and around a spacecraft. Electromagnetic interference (EMI) refers to any unwanted electromagnetic disturbance that may affect the performance of individual electronic systems. EMC, on the other hand, describes a system's ability to function reliably in the presence of such interference, while the system itself does not generate excessive interference. [20]

Electromagnetic interference can be transmitted primarily through two mechanisms: conduction or radiation. Conductive interference is transmitted along electrical connections, such as power lines or signal cables. Radiated interference, on the other hand, is transmitted via electromagnetic fields in free space. Both of these transmission mechanisms are relevant to space systems, as all subsystems are integrated in close proximity and share electrical interfaces that can create multiple coupling paths. [20][21]

From a testing perspective, EMC verification can also be divided into two separate categories: emissions and susceptibility. Emissions testing involves measuring the levels of electromagnetic interference generated by the satellite itself and determining whether they are below the specified limits. Susceptibility, also known as immunity, then tests the ability of the system to operate correctly when exposed to external interference. [20][22]

These fundamental principles then form the basis for all EMC testing methods. By combining emission measurements with susceptibility testing, it is possible to determine both the satellite's impact on its electromagnetic environment and its robustness against external interference.

EMC Test Categories

Electromagnetic compatibility tests are typically divided into several categories based on the nature of the electromagnetic interference and the method used to evaluate it. These categories reflect the fundamental distinctions between different coupling mechanisms and system responses, thereby forming the basis for structured EMC verification.

Generally, EMC tests are classified based on two primary aspects: the type of interference propagation and the type of system response. From the perspective of propagation, electromagnetic interference can be transmitted either through physical conductors or via electromagnetic fields. This brings up the previously mentioned distinction between conducted and radiated phenomena. From the perspective of system response, testing focuses either on emissions generated by the unit itself or on its susceptibility to external interference [20]

The combination of these two perspectives defines the main categories of EMC testing, which include conducted emissions, conducted susceptibility, radiated emissions, and radiated susceptibility. Each of these categories addresses a specific aspect of electromagnetic behavior and requires a specific test setup, measurement techniques, and assessment criteria.

Through CubeSat verification, these test categories are applied at various levels of integration, ranging from individual components to the fully assembled satellite. This hierarchical approach ensures that potential electromagnetic interference issues are identified early in the development process and can therefore be resolved before the project reaches the system-level testing phase. [12]

Conducted Emissions

Conducted emissions arise from electrical noise generated within systems, which is then transmitted via power and signal lines. These disturbances are most commonly produced by electronic switches, such as power converters or digital units.

The purpose of conducted emission tests is to measure the levels of noise present on these interfaces and verify that they remain below the specified limits. A so-called Line Impedance Stabilization Network (LISN) is commonly used for these measurements, which provides a network with a defined, stable impedance, thereby enabling repeatable measurements of voltage interference across specified frequencies. Typical method to measure interference outside the power leads is to make the cables pass through an inductive probe. [20][22]

Conducted Susceptibility

The purpose of conducted susceptibility tests is to verify the robustness of the system by injecting controlled interference into the electrical interfaces and then observing the response. This makes it possible to verify that the system is capable of withstanding expected levels of interference without degradation or loss of functionality.

Testing is typically performed by feeding interference signals into the cables using special signal injection devices, such as injection networks or current injection probes. The resulting response of the DUT is monitored across a defined frequency spectrum, and its behavior is compared against the specified acceptance criteria defined by the applicable standards. [20][22]

Radiated Emissions

The purpose of radiated emissions testing is to measure the levels of electromagnetic fields generated by a system and to verify that they comply with specified limits. As with conducted emissions, excessive emissions can interfere with communication systems, sensors, and other sensitive equipment.

Measurements are performed using special antennas, primarily in anechoic chambers, to minimize the impact of external interference on the measurements. The strength of the emitted fields is again measured across a defined spectrum and compared with defined limits derived from standards. [20][22]

Radiated Susceptibility

The principle behind the radiated susceptibility test is similar to the conducted susceptibility test; it is used to verify vulnerability to external interference that is transmitted not via physical interfaces but through electromagnetic fields. During the test, the DUT is monitored to check for any anomalies in its functionality.

Testing is typically conducted in specialized laboratories using transmitting antennas and RF amplifiers to generate the required field. The applied fields are then defined by standards or by other subsystems within the satellite, or by the launch vehicle itself. [20][22]

Table 3.7: Key parameters of EMC

Parameter	Typical Value / Range	Description
Conducted emissions range	~30 Hz - 100 MHz	-
Radiated emissions range	~30 Hz - 18 GHz	-
Conducted susceptibility range	~30 Hz - 100 MHz	-
Radiated susceptibility range	~2 MHz - 40 GHz	-
Conducted disturbance level	~1-10 V	-
Radiated field strength	~1-50 V/m	-
Measurement equipment	LISN; probes; antennas	Equipment used depending on test category
Test environment	Shielded / anechoic chamber	Controlled environment to ensure repeatability and accuracy
Evaluation criteria	Compliance with limits; no malfunctions	System must meet emissions limit and remain functional during susceptibility test.

3.2.4 Radiation Testing

The space environment is a domain in which it is challenging to operate spacecraft systems, as exposure to various environmental factors can significantly affect their performance, reliability, and lifespan. Among these factors, ionizing radiation plays a significant role, as it directly influences the behavior and degradation of the electronic components and materials used for spacecraft manufacturing.

Radiation effects are addressed within the framework of radiation hardness assurance, which focuses on evaluating and mitigating the impact of radiation on electronic components. According to industry standards, examples of these categories include Total Ionizing Dose (TID) and Single Event Effects (SEE), where each category affects components through a different physical mechanism. [24]

To ensure that a satellite can withstand the expected radiation conditions, it is necessary to define the radiation environment, assess the impact on the systems, and verify that the design meets the requirements. This process combines analytical methods with experimental validation, following established procedures such as those described in ECSS standards, which provide a methodology for evaluating radiation exposure and defining sufficient design margins.[23]

This chapter focuses on characterizing the space radiation environment and the effects on satellite systems associated with this environment, with an emphasis on their implications for verification and testing.

Radiation Effects and Test Methods

Radiation effects in satellite systems are typically classified according to their physical mechanism and impact on system performance. These effects arise from the interaction of energetic particles with electronic components and materials, leading either to gradual degradation or to sudden transient disturbances.

From a verification perspective, each category of radiation effects requires a specific approach that reflects its particular physical behavior. Some effects accumulate over the course of the mission's lifetime and are therefore evaluated using long-term tests, while others involve individual particle interactions and are demonstrated using particle bursts.

The main categories considered in satellite design with regard to radiation effects are Total Ionizing Dose, Displacement Damage (DD), and Single Event Effects. Each of these categories is associated with different failure mechanisms and requires specific verification methods.[24]

Total Ionizing Dose (TID)

The Total Ionizing Dose represents the cumulative effect of ionizing radiation on electronic components over time. This is primarily caused by the satellite's long-term exposure to charged particles, such as electrons or protons, which deposit energy into the material, leading to a gradual degradation of its electrical properties.

The main effect of TID is the accumulation of electrical charge in the insulating layers of semiconductor devices, which can lead to a shift in their parameters, an increase in leakage current, or possibly to a loss of functionality. These effects are primarily critical for analog circuits, power devices, and sensors, where even a small shift in parameters can significantly affect their performance. [23]

From a testing perspective, TID is evaluated by exposing components or the system to a controlled radiation source, most commonly gamma radiation, in order to simulate the cumulative dose that can be expected over the mission's lifetime. The total absorbed dose is typically expressed in units of rad or gray, with test levels defined based on mission-specific radiation analyses and design margin requirements.

During testing, key electronic parameters of the device are monitored, and the relationship between these parameters and the total accumulated dose is tracked, which makes it possible to determine the degradation trend and identify the failure threshold. According to standards, TID is a key part of radiation hardness assurance and allows verification that components can withstand the anticipated radiation environment throughout the entire duration of the mission. [24]

Displacement Damage (DD)

Displacement damage, also known as Total Non-Ionizing Dose (TNID), is caused by the interaction of energetic particles with the material's crystal lattice, leading to a permanent structural defect. Unlike ionizing radiation, which primarily affects electrical parameters due to the accumulation of parasitic charge, displacement damage results in the physical displacement of atoms within the material.

These defects degrade the performance of semiconductor devices by reducing carrier lifetime and mobility. Displacement damage is particularly significant in optoelectronic devices such as image sensors, photodiodes, or solar cells, where it results in increased noise, reduced sensitivity, or a general degradation in performance. [23]

From a testing perspective, DD effects are evaluated using proton and neutron emitters, which more accurately represent the particles responsible for displacement damage in the space environment. Damage levels are typically expressed in terms of equivalent particle fluence, which allows for a correlation between test conditions and the anticipated mission environment.

According to ECSS standards, displacement damage, together with the ionizing dose, is considered as part of the overall radiation hardness assurance process, particularly for components that are expected to be sensitive to non-ionizing energy loss mechanisms. [24]

Single Event Effects (SEE)

Single-event effects are the result of a single energetic particle interacting with a sensitive region of an electronic device, causing a localized ionization event. Unlike cumulative radiation effects, SEE occurs instantaneously and can result in temporary or permanent disruption of system operation.

The impact of a SEE depends on the type of interaction between the particle and the component, as well as the component's sensitivity to these effects. Common types of single-event effects include, for example, Single Event Upset (SEU), which causes a temporary change in logic states, or Single Event Latch-Up (SEL), which can lead to damage due to high current if not properly and promptly addressed. Other, less common effects include Single Event Transients (SET) or Single Event Burnout (SEB), each of which has its own typical impact and implications for system performance and reliability. [24]

Particle beams are commonly used for SEE testing, most often employing heavy ions or protons to simulate the space radiation environment. The device's response is then characterized as a function of particle energy and Linear Energy Transfer (LET), which describes how energy is transferred to the material as the particle travels through it. [24]

Testing focuses on determining the threshold levels and cross-sections required to trigger a specific effect, which makes it possible to estimate the frequency of events under the anticipated mission conditions. [24]

Table 3.8: Key parameters of Radiation testing

Parameter	Typical Value / Range	Description
Total Ionizing Dose (TID)	$\sim 1 - 100$ krad	Cumulative ionizing dose over mission life time
Displacement Damage (DD)	$\sim 10^9 - 10^{12}$ particles/ cm^2	Particle fluence used to represent non-ionizing damage in materials and devices
Linear Energy Transfer (LET)	$\sim 1 - 100$ meV $\cdot cm^2/mg$	Parameter describing energy deposited by a particle along its path
Particle types	Electrons; protons; heavy ions	Types of particles used to simulate different radiation effects
Dose rate	$\sim 0.01 - 100$ rad/s	Rate at which ionizing dose is delivered during TID testing
Test environment	Particle accelerator / irradiation facility	-
Evaluation criteria	Parameter drift; functional failure	Assessment based on degradation trends and occurrence of failure

3.3 Test Campaign Strategy

While the previous subsections focused on individual testing methods and their objectives, an effective satellite verification process requires not only the selection of the appropriate tests, but also their proper organization and execution through a structured test campaign.

The test campaign strategy defines the overall approach to verification, including test sequences, the selection of levels at which the spacecraft will be tested, the use of various hardware models, and the management of associated documentation. Its purpose is to ensure that all relevant environmental and functional requirements are verified in a consistent and effective manner, while minimizing the risks associated with flight hardware.

This section describes the key parameters and considerations involved in planning and executing a satellite test campaign.

3.3.1 General Principles

The verification of space systems is guided by several fundamental principles that ensure the reliability and functional correctness of hardware subjected to assumed mission conditions. These principles define not only what individual tests should look like, but also how the overall verification process should be structured.

One of the key principles is the concept commonly referred to as "test as you fly, fly as you test". This principle states that test conditions and system configurations should replicate as closely as possible the environment in which the satellite will operate. Environmental verification standards emphasize that testing should simulate realistic mechanical, thermal, and electrical conditions to demonstrate the satellite's representative behavior and performance during the mission. This approach minimizes risks and discrepancies between ground testing and in-orbit operations. [16]

Another closely related concept is the requirement for a representative test configuration. The hardware under test should include all relevant interfaces, structural elements, and electrical connections in a configuration that closely reflects the final flight model. The correct configuration ensures that interactions between individual subsystems are captured during testing, which is a critical step for identifying integration-related issues. Verification activities therefore consider not only individual components but also their interactions within the context of the overall system. [6]

Another important principle is the concept of a progressive approach to testing, which progresses hierarchically through the system. Testing is performed incrementally at various levels of the satellite's composition, starting with individual components and continuing through subsystems to the verification of the entire system. This approach allows for the early detection of errors at lower levels, where troubleshooting is less complex and costly. The structured progression of integration and verification activities is a fundamental part of the systems engineering process and supports effective risk reduction throughout the entire project lifecycle. [6]

Furthermore, verification programs are typically tailored to the specifics of the mission and the hardware characteristics. The standards emphasize that testing requirements, levels, and durations can be adapted based on factors such as mission profile, risk classification, or design heritage. This approach ensures that the verification process remains both technically adequate and efficient in terms of resource allocation. [16]

Together, these principles establish a systematic framework for verification that strikes a balance between realism, efficiency, and risk mitigation. This framework also serves as a foundation to develop the testing methods and procedures that will be defined in the following chapters.

3.3.2 Test Sequence

The sequence of verification tests is closely linked to the principle of "test as you fly, fly as you test." In this context, the principle refers to exposing the satellite to environmental conditions in the same order as they are experienced during the mission. This means that the satellite is progressively exposed to mechanical and thermal environments in a manner that replicates the course of the mission.

A typical test campaign begins with functional tests under ambient conditions, which verify that the satellite is operating as intended and establish a baseline before the actual environmental testing campaign begins. This is followed by vibration testing, which simulates the mechanical stresses experienced during launch. In some cases, a shock test may also be performed, which simulates the impacts experienced during separation events.

After mechanical testing is complete, the satellite is then subjected to thermal vacuum tests designed to simulate its subsequent operation in orbit. This sequence reflects the transition from launch conditions to the space environment and ensures that the test campaign remains consistent with the anticipated mission profile.

Functional checks are repeated between individual tests to ensure that no degradation or loss of performance has occurred. This step is essential for identifying failures caused by previous environmental stress.

An exception to the described sequence applies to tests such as electromagnetic compatibility or radiation testing, whose position in the test campaign is generally considered less constrained. Unlike mechanical or thermal tests, these tests are not strictly tied to a specific phase of the mission timeline.

EMC testing is typically performed once a system is sufficiently complex to accurately represent its actual electrical behavior, whereas radiated testing is more often conducted at lower levels, such as on individual components or subsystems. As a result, the exact positioning of these tests within the overall test campaign is more flexible and is determined primarily based on project-specific constraints, test availability, and the verification strategy.

Finally, it should also be noted that the sequence presented here is a recommended approach that can be adapted based on specific mission requirements or project constraints. However, maintaining a logical progression from the launch

environment to orbital conditions remains the fundamental baseline for verification planning.

3.3.3 Test Levels and Model Philosophy

The definition of a test campaign is closely linked to the selection of hardware models and corresponding test cases. Different model philosophies are used for space projects to strike a balance between verification confidence, cost, and risk.

The commonly used approach is based on the use of a Qualification Model (QM) and a Flight Model (FM). In this case, the qualification model is subjected to environmental tests with qualification levels, which exceed the anticipated conditions under which the satellite will operate. This approach demonstrates sufficient design margins and ensures that the flight hardware is not unnecessarily over-tested. The flight model is then tested at reduced levels that more closely resemble the actual conditions, in order to verify that no errors occurred during integration, without risking any damage to the flight hardware.

An alternative approach is to use the Protoflight Model (PFM), where a single model is used for both qualification and subsequent flight. In this case, the hardware is tested at qualification levels, with test margins or durations typically reduced to limit the risk of damage while still providing sufficient confidence in the design. This approach is commonly applied for small satellites, as they are expected to have significantly limited budgets.

The choice of design philosophy has a direct impact on the overall test campaign and the associated risks. The QM/FM approach offers greater reliability and a lower risk of flight hardware failure, but at the cost of a higher budget and greater development effort. In contrast, the PFM philosophy simplifies the overall campaign and reduces costs, but requires more consideration of the testing levels and risk mitigation strategies employed.

3.3.4 Documentation

The proper execution of a verification campaign is supported by a structured set of documents that ensure consistency, traceability, and proper evaluation of test results. Documentation plays a key role in managing the verification process and demonstrating compliance with system requirements.

At the highest level, the test plan defines the overall strategy for the verification campaign, including the scope of testing, the selected methods, the test sequence, and the required levels to be applied. It thus serves as a reference for planning and coordinating all verification activities.

The individual tests are described in detail in test procedures, which specify the exact steps to be followed, the required equipment, the test conditions, and the criteria that must be met during the test. These procedures ensure reproducibility and reduce the risk of ambiguity during the verification activity.

The resulting data is then recorded in test reports, which should include the configuration used during the test, the measured data, the observed behavior, and a final assessment of compliance with the requirements. These reports provide the necessary evidence to demonstrate that the system meets all defined requirements.

Finally, to ensure clear traceability between requirements and verification activities, a so-called verification matrix, sometimes also referred to as a Verification Control Document (VCD), is used. This document establishes the link between individual requirements and the corresponding verification methods, which helps ensure that all requirements are properly addressed.

Together, these documents form a coherent framework that supports the planning, execution, and validation of the test campaign, and are essential for ensuring the transparency and accountability of the overall verification process.

4 Test Campaign Planing

This chapter describes the test campaign prepared for the KOSTKA CubeSat, which is described in one of the previous chapters. Its main objective is to define the approaches used to verify compliance with mission and launcher requirements. It outlines the test strategy, the test plan, the sequence of tests, and the associated documentation.

4.1 Campaign Overview

The test campaign for the KOSTKA satellite was defined based on the conditions of a mission in low Earth orbit, specifically SSO at an altitude of between 500 and 600 kilometers. Furthermore, the verification process was derived from requirements based on the environmental conditions created by the Falcon 9 launch vehicle. Additional constraints were imposed to ensure the satellite's compliance with the CDS, which defined the interface between the satellite and the ExoPod Nova deployer.



Fig. 4.1: ExoPod Nova 16U [25]

A protoflight approach was chosen for the campaign, using a single flight model that underwent a full test campaign. This choice influenced both the selection of test levels and the overall conduct of the tests.

The campaign is then be divided into three distinct phases. Prior to the start of integration, functional tests were performed on selected subsystems; examples include tests of the radiation pattern of the experimental antenna or the OBC functional testing. This phase was followed by the integration phase, during which

mechanical and functional checks were primarily performed, focusing on structural dimensions, interfaces between subsystems, and the functionality of the solar panels. The integration phase was followed by environmental testing. Here, vibration tests were scheduled to verify compatibility with the launch vehicle's vibration environment, followed by thermal vacuum tests.

4.2 Testing Structure

The structure of the test campaign was based on a combination of mission requirements, hardware limitations, and the adopted Protoflight Model approach.

A key factor in defining the test sequence is the need to account for the anticipated environmental conditions throughout the mission. Mechanical loads associated with the launch vehicle are addressed before tests simulating orbital conditions. As mentioned in previous chapters, this approach ensures consistency between test conditions and the subsequent actual mission conditions.

Another key aspect was the progressively increasing level of system integration. Individual subsystems were functionally verified on Flatsat before integration began to minimize the risk of errors carrying over into system-level testing. System-level tests were then primarily designed to verify the overall functionality and structural integrity of the fully assembled satellite.

The structure of the campaign was also partly influenced by practical considerations, such as the availability of testing capacity and the need to maintain continuity between the individual testing phases.

4.3 Documentation

The testing campaign for the KOSTKA mission was supported by a set of project-specific documents that defined the planning, execution, and evaluation of the results of individual tests.

A simplified test plan in the form of a structured table was created for reference and orientation. This document provided an overview of planned testing activities, including test types, models to be tested, and the approximate dates when the tests were to be conducted. As such, this document was used primarily to coordinate the campaign rather than as a detailed technical document.

Detailed procedures were prepared for the defined environmental tests, specifically for vibration and thermal vacuum tests. For the functional tests of the supplied COTS components, the test procedures were defined by the manufacturer and as such were adopted accordingly. Each procedure included a description of the test

objective, the satellite configuration in which the test would to be performed, and a definition of the test conditions, including test levels, profiles, and sensor placement. The procedures also defined test sequences, required power-off conditions, duration, and acceptance criteria.

The test results are documented in a set of test reports. These documents include a vibration test report and other reports required by the launch provider, such as verification of the satellite's external dimensions or mass. The structure of each report reflects the specific measurements and compliance with specified constraints.

The testing process was documented using a combination of measured data and visual records. Data from the sensors was recorded in tabular format and further processed for the evaluation. In addition, photo documentation was taken during the process to make it possible to determine retrospectively how the test was carried out. This documentation primarily aimed to capture the test setup, the placement of individual sensors, and the satellite's configuration during testing.

Although there was no formal verification matrix document, traceability between individual requirements and testing activities was ensured by a comparable section located directly within the table listing the individual requirements.

5 Integration Tests

This chapter describes the integration process for the CubeSat KOSTKA and the associated tests and measurements conducted during this process. While the satellite's general architecture was presented in previous chapters, this chapter focuses on the practical implementation of the system and the verification activities carried out during the integration process.

The integration itself was carried out in a controlled cleanroom environment to minimize the risks associated with contamination and to ensure the proper conditions for handling sensitive components. Although the project was developed by students, the integration process attempted to follow established engineering practices. Predefined procedures were used for key operations to guide the assembly sequence, ensuring correct execution and repeatability in case it became necessary to investigate the causes of any issues that occur.

Rather than being conducted as a separate phase, testing is performed continuously throughout the integration process. Individual components and subsystems were verified prior to integration, followed by functional tests at higher levels of assembly. As mentioned earlier, this approach facilitates early detection of faults, thereby preventing them from propagating to higher levels of assembly and ensuring that each integration step is verified and in a functional configuration.

5.1 Solar Panels Verification

The solar panels used on the KOSTKA satellite were partially manufactured in-house. The PCB design for the panels was developed by the team, while the production of these PCBs and the procurement of solar cells were subsequently outsourced. The final assembly, including the mounting of the solar cells onto these PCBs, was carried out internally.

The integration process itself involved attaching individual solar cells to the PCB using a special adhesive layer. This adhesive was first applied to the PCB, and care had to be taken to ensure proper alignment and the absence of any air bubbles, as these could negatively affect mechanical stability and heat transfer. After the protective layer was removed from the adhesive, the solar cell was placed on it. The electrical connection was established by soldering the side leads of the negative terminal to the conductive path on the PCB. The positive terminal, located across the entire bottom surface of the cell, was also connected using a soldered wire through a hole in the PCB. The cells on individual boards were then connected in series.

In addition to the solar cells, the individual PCBs also contained other components such as photodiodes and temperature sensors. These were used for monitoring and orientation purposes and were functionally verified prior to the actual testing of the solar cells.

Functional testing of the solar panels was always carried out essentially immediately after the completion of each unit. Since we were not equipped with a representative solar simulator, the tests were aimed more at demonstrating the basic functionality of the solar panels than at determining their maximum power output. A simplified verification approach was used to demonstrate correct electrical behavior. The panels were illuminated using a halogen lamp, and their output was measured using a variable resistive load and a multimeter. The main verification criterion was the voltage at the panel's output, which should correctly range from approximately 5 to 5.5 V under the given conditions.

However, during the testing phase, a problem was discovered. Some panels reported a much lower voltage than expected during measurement, approximately 2.6 V, which is about half the expected voltage. This behavior was traced to an unintended short circuit of one cell on the given panel. The main cause was identified as solder spilling between the negative terminal and the conductive back of the cell, which effectively short-circuited the positive and negative terminals. As a result, the solar cell behaved as a conductor rather than an active element, but fortunately, no damage occurred.

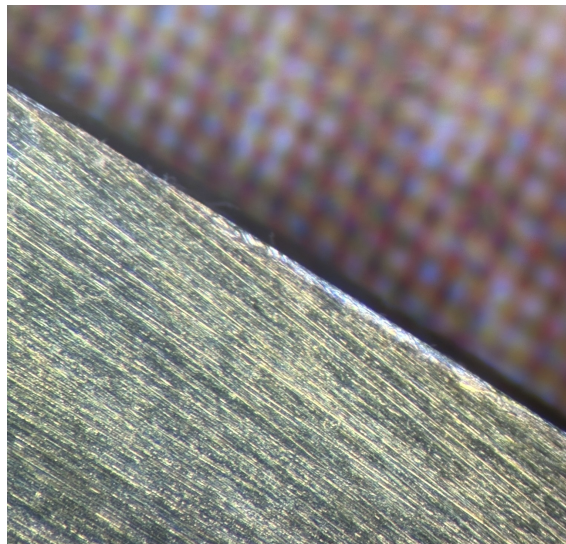


Fig. 5.1: Closeup of the unwanted connection

This defect was corrected by adjusting the placement of the solar cells on the PCB, primarily by attaching the cell to the board as far away from the conductive path as the footprint tolerances allowed. Assembling the remaining panels using

this approach successfully resolved the issue, and all panels ultimately functioned properly.

Testing, performed through voltage measurement, has confirmed that the solar panels are functioning properly overall and has highlighted critical aspects of the assembly process that require special attention to ensure reliable operation.

5.2 Kill Switch Verification

This section focuses on testing the satellite's safety mechanisms, specifically the kill switches, which are critical for ensuring safe operation during launch and subsequent deployment. The kill switches are designed to keep the satellite powered off while it is housed in the deployer and then allow it to be activated after it has been safely deployed.

The entire mechanism consists of two kill switches mounted on standoffs on the -Z side. Each switch consists of a special screw threaded through the structure into a cylindrical protrusion, which then extends beyond the structure. This screw passes through the standoff into the interior of the structure, where its head mechanically presses against a limit switch. When the satellite is placed in the deployer, the protruding cylindrical elements are forced into the structure, causing the screw head to press the limit switch and hold it in the closed position.

Testing was performed on the complete mechanism rather than on individual components. Initial tests were conducted on a separate structural component that included the kill switch, prior to the satellite's full integration. This approach allowed for better observation of the mechanism's functionality and easier access for any necessary adjustments.

During the early testing, a defect was discovered whereby, in some cases, the friction between the screw head and the contact surface of the limit switch was too high, which caused the mechanism to jam and prevented smooth actuation. To resolve this issue, a piece of Kapton tape was applied to the contact surface of the switch, thereby reducing friction between the screw and this surface. This modification significantly improved the reliability of this mechanism.

Subsequent tests were conducted to verify the durability of this solution and focused primarily on potential wear of the Kapton tape. These tests were repeated to determine the number of activation cycles the mechanism could withstand in this configuration without a loss of performance. These tests showed that, for reliable operation, the mechanism must not undergo more than approximately 150 cycles, which was then determined to be a sufficient number of cycles for the given purpose, even when accounting for a significant margin of safety.

The primary evaluation criteria were the correct electrical and mechanical behavior of the individual kill switches. Specifically, they were required to close reliably when pressed and then open again without obstruction upon release. These tests were conducted prior to the satellite's final assembly, as access to the mechanism itself would be very difficult once the satellite was fully assembled. Additionally, the critical nature of the kill switch system had to be taken into account, as a failure to release would prevent the satellite from booting up after deployment.

5.3 Satellite Assembly

The assembly of the satellite structure and the integration of the internal PC104 stack were carried out simultaneously as part of a continuous integration process. The procedure began with the installation of threaded rods into the upper section of the structure, followed by the progressive assembly of the stack itself, which, as mentioned earlier, consisted of the main radio, EPS, OBC, and subsequently the experimental radio. After the stack was installed, the remaining structural elements were mounted, completing the satellite's primary structure.

After the structural assembly, the internal harness was connected and routed through the satellite to the designated positions. External components, such as solar panels and antennas, were then progressively installed and connected to the prepared cabling.

During the integration process, functional verification was performed to confirm that the individual integrated subsystems were operating correctly. The primary focus of these tests was to verify communication between the individual subsystems and sensors. Testing was primarily conducted after the structure had been assembled but before the external components were permanently installed, in order to ensure that access to the interior of the satellite remained available.

During these tests, a problem was identified with the cabling used for the solar panel sensor interface. It was found that the connector pinout was reversed, which resulted in the sensors not being connected correctly. This problem was resolved by reconfiguring the individual wires in the connector so that they matched the required pinout.

To improve repeatability and reduce the likelihood of errors during assembly, the integration process was guided by a dedicated integration procedure consisting of a step-by-step checklist and a visual assembly guide that displayed the placement and orientation of individual components.

5.4 Antenna Deployment Test

The satellite uses a deployable double-dipole antenna consisting of four flexible antenna elements. The deployment mechanism relies on the elastic properties of the antenna rods, which are stored in a coiled configuration inside the antenna module and automatically extend outward upon release.

The release mechanism is based on the burn-wire system principle, in which individual antenna pairs are released sequentially. The antenna module itself contains its own internal control logic, which allows the release sequence to be initiated via a communication command sent over the system bus.

The launch test was performed according to the procedure provided by the antenna manufacturer. A development kit was then used to transmit the command to activate the deployment sequence. To prevent potential damage to the antenna elements during testing, the manufacturer recommends attaching the antenna to the satellite structure before conducting the tests. This recommendation has been followed during testing.

As mentioned, the test was conducted by initializing the sequence using the development kit, after which the antenna deployed correctly, as expected, with no anomalies or interruptions observed. The antenna elements were fully released and the sequential deployment process was completed successfully.

The deployment test was performed only once. This decision was based on the fact that the antenna module is commercially available and previously validated component. The primary objective of the test was not to qualify the design itself, but to verify that the delivered unit had not been damaged during transport and handling and could be integrated into the system.

Verification of the antenna's proper deployment was considered very important, as it represents a critical part of the communication chain between the satellite and the ground station. A failure during deployment would result in the loss of the primary communication capability, resulting in a single-point mission failure.

5.5 Final Dimensional Verification

After the satellite integration was completed, a final verification of the individual dimensions was performed to confirm compliance with the deployer's requirements. The measurements were taken using a sliding caliper, with both ends and the center section measured on each side to verify the consistency and straightness of the rails.

For the X and Y axes, the required dimensions were defined as 100 ± 0.1 mm. These measurements were critical for verification of the guide rails to ensure sufficient flatness and a proper fit within the deployer. Maintaining these tolerances minimizes the risk of the satellite getting jammed during deployment and, at the same time, ensures that it is not excessively loose when clamped by the rail mechanism of the deployer. Confirming the correct dimensions is essential for both reliable deployment and a stable positioning inside the deployer compartment.

On the Z-axis, the required dimensions were 113 ± 0.16 mm. Compared to the lateral dimensions, these tolerances are less restrictive, as the Z-axis dimensions are primarily related to ensuring that the satellite fits into the allocated deployer slot, rather than being related to deployment guidance or friction constraints. All measured dimensions then can be seen in the table 5.1 below.

Additionally, the dimensions of the TunaCan module were measured to verify that its external dimensions fit within the spatial constraints of the cutout in the deployer carriage.

Finally, the total mass of the satellite was measured using a laboratory scale. The resulting mass of the fully assembled satellite was 1042 g, which met the weight limit for given CubeSat configuration.

Overall, the measurements confirmed the satellite's mechanical compatibility with the deployment system. All measured dimensions and parameters were found to be within the required tolerances, thereby confirming the satellite's readiness for integration into the final deployment hardware.

Table 5.1: Final dimensional verification of KOSTKA

Rail	Position	Required dimension	Tolerance	Measured value
$X-$	Top	100 mm	± 0.1 mm	100.09 mm
	Center			100.1 mm
	Bottom			100.1 mm
$X+$	Top	100 mm	± 0.1 mm	100 mm
	Center			100.04 mm
	Bottom			100.01 mm
$Y-$	Top	100 mm	± 0.1 mm	100 mm
	Center			99.94 mm
	Bottom			99.94 mm
$Y+$	Top	100 mm	± 0.1 mm	99.98 mm
	Center			100 mm
	Bottom			99.97 mm
Z	$X-$	113.5 mm	± 0.16 mm	113.62 mm
	$X+$			113.6 mm
	$Y-$			113.62 mm
	$Y+$			113.62 mm

6 Vibration Campaign

This chapter describes the practical implementation of a vibration test campaign conducted on the KOSTKA satellite. It focuses on the preparation of vibration test procedures, the execution of these tests, and the subsequent evaluation and processing of the measured data. Furthermore, this chapter describes the preparation of the final test documentation and the deliverables required to verify the satellite's readiness for launch.

6.1 Procedure Preparation

The vibration test procedure was prepared primarily according to the requirements defined by the launch provider. These requirements specified the necessary environmental tests, along with their profiles and defined levels. Additional requirements not explicitly stated in this documentation were derived from ECSS standards.

The hardware configuration was consistent with the design philosophy of the Proto-Fight satellite model. These tests were conducted at qualification vibration levels but with the duration of acceptance tests. The test campaign consisted of low-level sine sweeps to identify resonance frequencies and subsequent monitoring of shifts in these frequencies, random vibration tests, and power inhibit tests. The random vibration and power inhibit tests were the only environmental tests explicitly required by the launch provider, while the others were merely recommended. The vibration tests were performed along all three orthogonal axes of the satellite.

The complete vibration profile used for this campaign is shown in the figure 6.1, along with the accompanying table 6.1 of frequency limits. The vibration procedure itself can be found in the appendices.

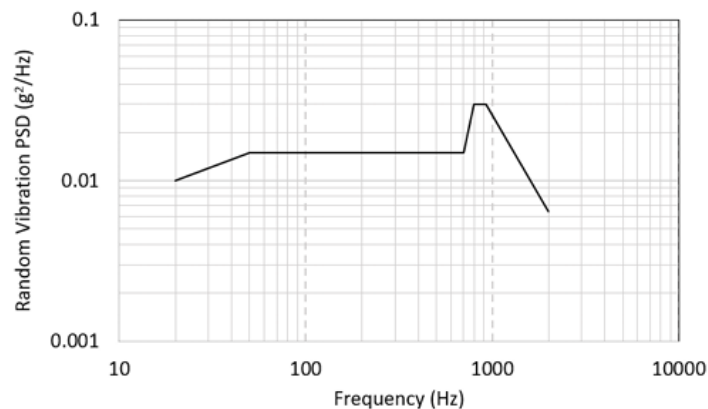


Fig. 6.1: Defined vibration profile [26]

Table 6.1: Acceleration at defined frequencies [26]

Frequency [Hz]	Random vibration [g^2/Hz]
20	0.01
50	0.015
700	0.015
800	0.03
925	0.03
2000	0.00644
GRMS	5.57

In the early stages of preparing the procedure, it was planned to use four accelerometers to measure the satellite's vibration response, with one of them to be located in the satellite's interior. However, the launch provider's requirements subsequently indicated that removing certain sizes of screws after testing was not possible, or under conditions of enhanced mechanical locking mechanisms, otherwise the satellite would need to be retested. Due to this limitation, the internal accelerometer was omitted and the final solution used only three external accelerometers.

One of the accelerometers was attached to the TunaCan antenna structure, the second was placed on the side solar panel near the screw so that the response would capture the structure's resonance, and the third and final one was placed on the solar panel board on the Z+ side. Here, the solar panel PCB is significantly thinned because the main antenna is located underneath it and it is only secured in two diagonal corners, so there was a risk that the unrestrained corners could oscillate significantly. The final placement of the accelerometers then can be seen in Figure 6.2.

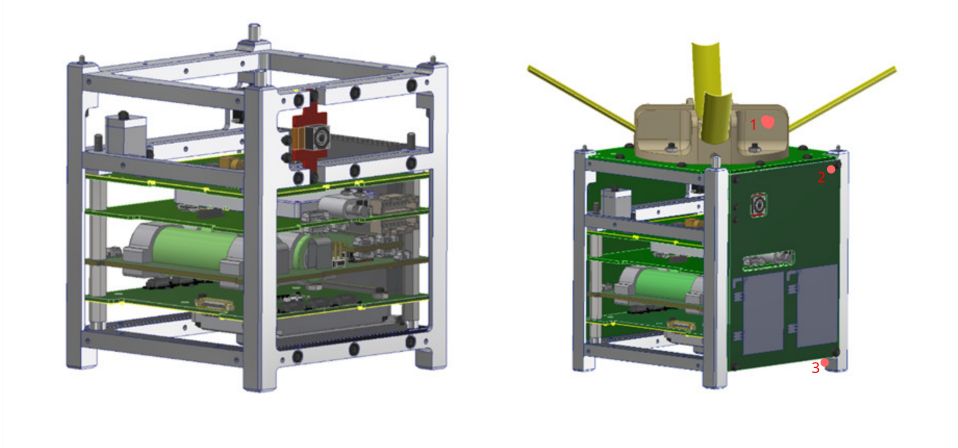


Fig. 6.2: Final placement of accelerometers

The individual limits for changes and sine sweep ramps were not defined in the requirements provided by the launch provider. As already mentioned, the values recommended by ECSS standards were used for reference. The limits for amplitude shift before the start and after the end of the test in a given axis were not allowed to exceed a change of more than 20%, and the frequency was not allowed to change by more than 5%. The ramp-up rate for resonance search was then defined as 2 oct/min.

6.2 Test Configuration

The vibration test was conducted at an external vibration testing laboratory using an electrodynamic shaker system. The satellite was mounted using a dedicated fixture that simulated the deployer rail interface. The fixture was provided directly by the testing laboratory and mounted on the vibration table via a baseplate interface.

The satellite was tested sequentially along all three axes, in the order Z, Y, and X. In all axes, the satellite was secured in the required orientation while maintaining the configuration of the mechanical interfaces.

The instrumentation setup consisted of three response accelerometers mounted directly on the satellite and two reference accelerometers used to control the shaker and monitor its response. One of the reference accelerometers was mounted directly on the table, and the other on the baseplate of the test fixture.

The accelerometers were attached to the satellite using Kapton tape and super-glue. A piece of Kapton tape was first adhered to the satellite's outer surface to prevent direct contact between the adhesive and the structure. A small layer of adhesive was then applied to the accelerometer, which was subsequently placed on the tape. The resulting actual placement of the sensors can be seen in the image below.

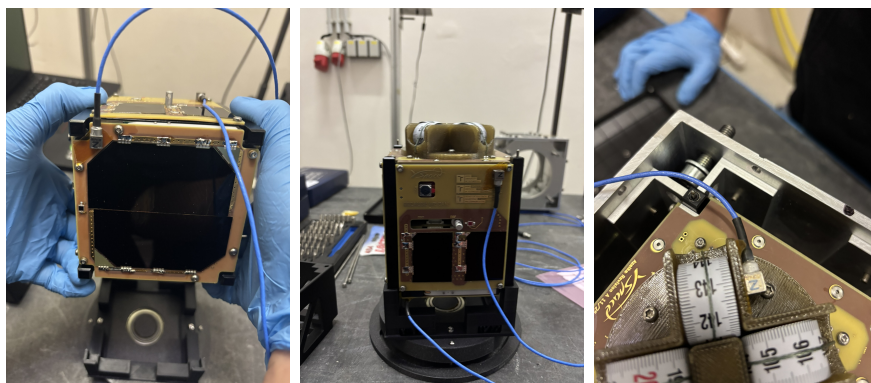


Fig. 6.3: Actual placement of Accelerometers

The power inhibit verification was performed during random vibration tests. Before starting the test on a given axis, the RBF was removed from the satellite, and the satellite was kept powered off using only the kill switches to simulate the launch configuration. The satellite was not actively monitored during the test, but verification was performed using onboard boot logs before and after the vibration campaign, thereby confirming that the satellite had not been activated if no changes were detected. Furthermore, if an unexpected activation had occurred, it would have been immediately detectable visually via LED, which would have lit up if the satellite had powered on.

Functional tests were conducted before and after the vibration campaign on the university premises. During the tests themselves, only a visual inspection of the satellite, its exterior, and visible areas was performed to identify any potential structural damage or loose interfaces. The overall mounting configuration used during the vibration campaign can be seen in Figure 6.4.

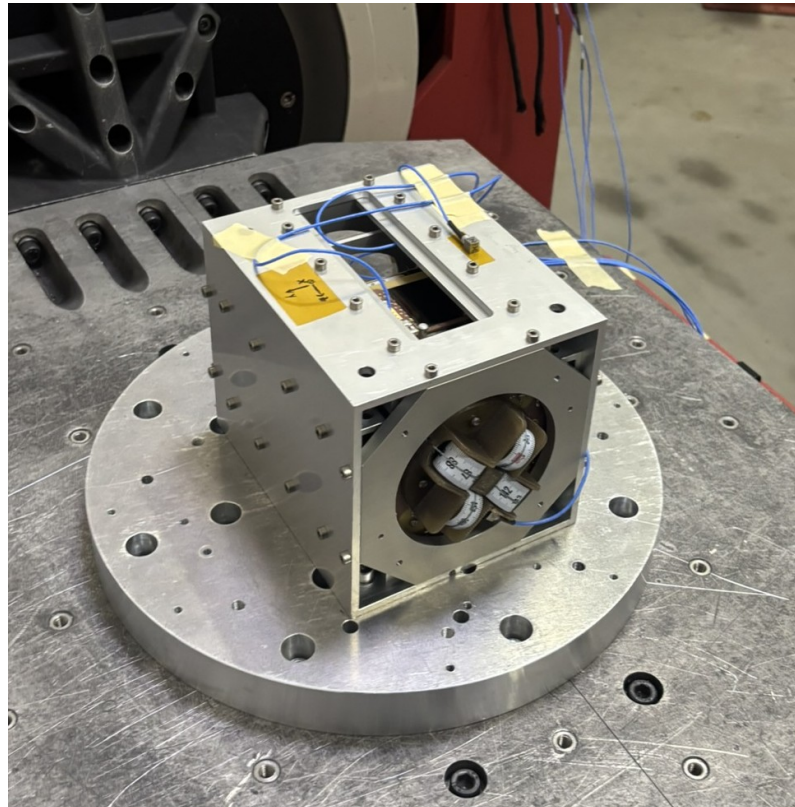


Fig. 6.4: Mounting of KOSTKA during vibration tests

6.3 Test Execution

After installation on the vibration table, the vibration test sequence started. Tests in each axis began with a low-level sine sweep in the frequency range from 5 to 2000 Hz, in order to identify the structure's resonance frequencies before initiating the random vibration load. The sine sweep was followed by low-level and intermediate-level random vibration testing, which, as the name suggests, were reduced levels by -3 dB and -9 dB, respectively, relative to the final qualification levels. Following this test, another sine sweep was performed to identify any structural changes or developing mechanical issues before exposing the satellite to the full level of random vibration test. This was followed by the actual main full random vibration test, after which another sine sweep was performed to compare the initial and final values and verify that the resonance had not shifted beyond the defined maximum described in the previous chapter. This sequence was repeated for the other axes as well.

Random vibration tests were performed without notching. Each random vibration exposure, including low-level, intermediate-level, and full-level tests, were performed for the duration of one minute. In this case, the one-minute duration is counted from the moment the test equipment reaches the specified test levels, not from the moment it is turned on.

The power inhibit verification was performed during the random vibration tests, as described in the previous section. No unexpected activations occurred at any point during the test campaign.

All vibration tests were completed successfully without any repetitions or interruptions. No structural damage, loose connections, or unexpected behavior was detected during the visual inspections conducted throughout the campaign.

From a practical standpoint, the most time-consuming part of the campaign involved changing the shaker's configuration between the Y and X axes. In this case, it was necessary to disconnect the vibration exciter from the slip table and then mount the entire test fixture, with the satellite, directly onto the shaker in the specified configuration.

The entire vibration test campaign was carried out over the course of a single workday. Excluding the initial preparations such as placing the satellite in the fixture and attaching the sensors, the tests in a single axis took approximately 1.5 hours to perform.

6.4 Test Evaluation and Data Processing

After the vibration testing campaign was completed, the testing facility delivered a vibration test report containing a basic evaluation of the measured data and the corresponding raw dataset. The report included a comparison of sine sweeps, identification of resonance frequencies, and a calculation of shifts in amplitude and frequency between individual measurements performed through sine sweeps.

For internal purposes, the measured sine sweeps were compared to identify possible structural changes caused by exposure to vibrations. The largest measured change was in amplitude and was approximately 19.8 percent, which was close to the defined acceptance limit. However, this shift was measured during intermediate verification, i.e., on a sweep performed between intermediate-level and full-level random vibration tests. After completing the entire vibration sequence for this axis, the response returned to a difference of approximately 5 percent relative to the initially measured values, indicating that no permanent structural change had occurred.

Considerable attention was given to the response of the solar panel on the Z+ side, which, from a mechanical standpoint, represented the most critical part of the satellite's structure. Previous structural simulations predicted that the first resonance frequency of this panel would be around 200 Hz, which corresponds to a frequency range that closely approaching the dominant vibration spectrum of the launch vehicle. However, the actual measured resonance frequency was around 650 Hz, which represents not only a significantly stiffer structural response than anticipated but also the lowest measured resonance frequency of the entire satellite. Overall, the satellite structure exhibited considerably higher stiffness than initially predicted by the simulations.

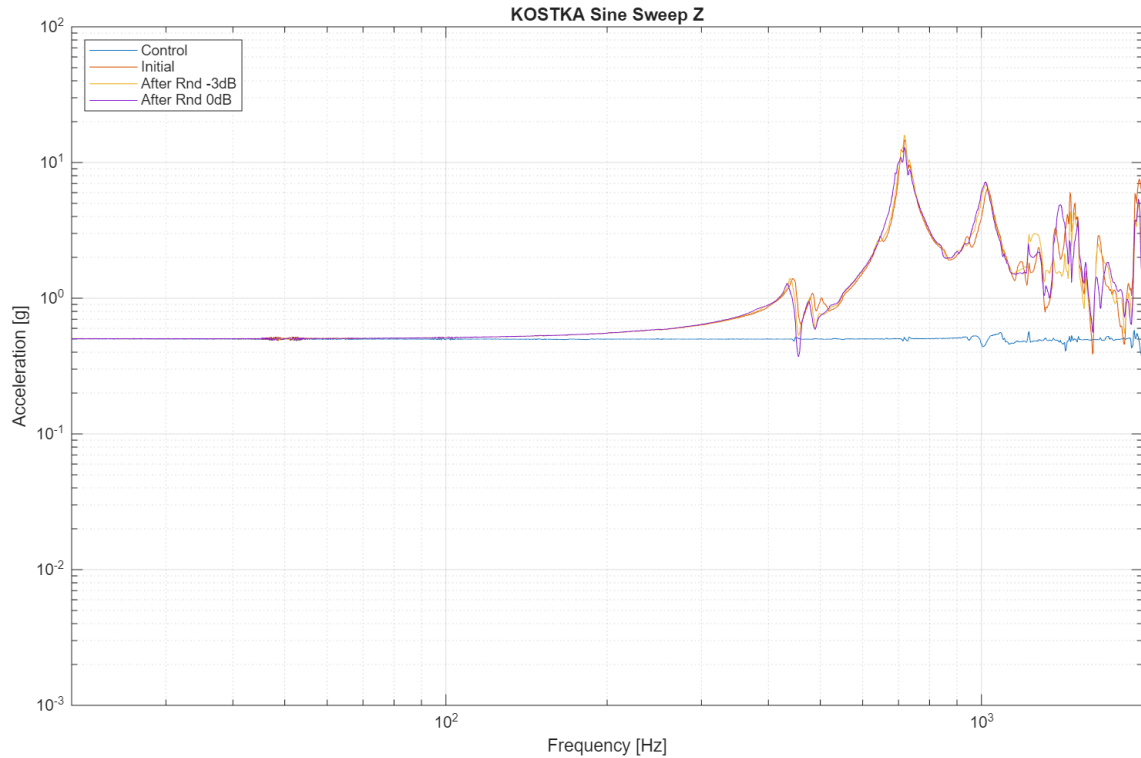


Fig. 6.5: Sine sweep comparison in Z axis

In addition to the report provided by the testing laboratory, the launch provider requested a separate vibration verification document. The original laboratory report did not meet the formatting and reporting structure requirements defined by the launch provider. Consequently, the measured dataset had to be reprocessed and reformatted according to the specifications of the submission template. The launch provider primarily required a PSD response curve for all tested axes with an overlay corresponding to the control signals.

Power inhibit verification was evaluated separately using onboard EPS boot logs. The number of recorded satellite power-ups before and after the vibration tests was checked to verify that no unexpected boot-ups had occurred.

Following the completion of the test, a detailed inspection of the entire satellite was conducted. All externally accessible areas of the satellite were visually inspected for structural damage, loose interfaces, damage to the solar panels, or wear on the guide rails. Additionally, particular attention was paid to the deployable mechanisms, where the burn-wire systems were checked to verify that there was no excessive wear or fraying of the wires used to keep the antennas closed. Nevertheless, no structural damage or abnormal behavior was detected, and the satellite remained fully functional after the completion of the vibration campaign.

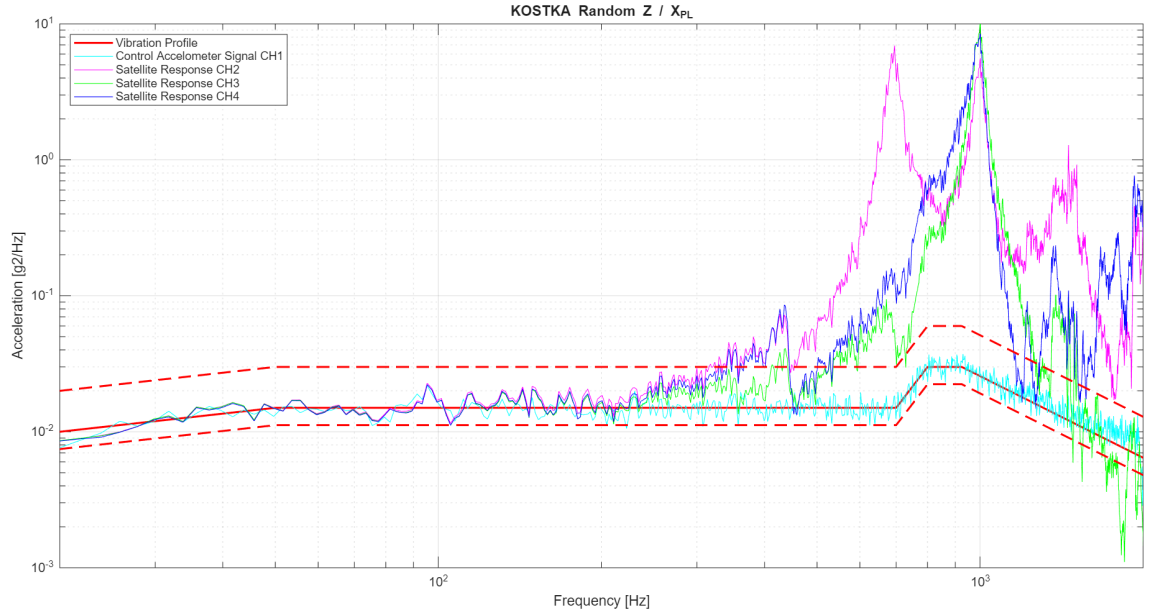


Fig. 6.6: PSD curve and control signal (Z axis)

6.5 Summary

The vibration test campaign successfully verified the satellite's structural integrity and launch readiness. All required vibration tests were completed along all three orthogonal axes of the satellite without any interruptions or the need to repeat the tests. No structural damage or unintended activation of the satellite was observed during the test campaign.

The measured vibration response confirmed that the spacecraft structure exhibits greater stiffness than initially predicted by simulations, with all measured resonance frequencies remaining within acceptance limits. The prepared vibration documentation and processed measurement data subsequently met the verification requirements defined by the launch provider and represented an important milestone toward the satellite's final flight acceptance.

7 TVAC Campaign

This chapter describes the preparation of the thermal vacuum test campaign planned for the KOSTKA satellite. Unlike the vibration tests described in the previous chapter, TVAC tests were not directly required by the launch provider, and their primary purpose was therefore to serve as an additional verification activity focused on the satellite's operational functionality during thermal cycling under vacuum conditions. The campaign also aimed to provide outgassing measurement data for the fully integrated satellite.

Although the final TVAC campaign was not carried out due to delays in the delivery of test equipment, and conducting the tests at other laboratories would have been too costly, a complete test procedure, thermal profiles, and a sequence of operational tests were prepared before the planned campaign was to take place.

7.1 Test Procedure

The thermal vacuum testing procedure was developed primarily in accordance with ECSS standards and recommendations. Unlike the vibration testing campaign, the limits for thermal testing were not directly specified by the launch provider, but were derived from the operating temperature limits of the individual satellite components instead, which were specified by their manufacturers.

The planned TVAC campaign was designed for a temperature range of -40 to 70 degrees Celsius and targeted a pressure of approximately 10^{-5} Pa. The complete thermal sequence consisted of four thermal cycles. The goal of the first cycle was to verify the satellite's ability to power up in both hot and cold conditions. The remaining 3 cycles were planned as operational cycles during which the satellite remains powered on, and throughout these transitions, the satellite's functionality and telemetry are monitored to ensure there are no potential anomalies in its operation.

The dwell time for both temperature extremes was set conservatively at 2 hours, in accordance with ECSS recommendations and available testing guidelines. The thermal transition rate was set at 1 °C/min to prevent excessive thermal stress on the satellite structure and its internal components.

The overall structure of the TVAC procedure is similar to the vibration procedure described earlier. The document includes references to test objectives, thermal limits, monitoring methods, and acceptance criteria. In addition, the procedure also contains a detailed step-by-step sequence defining the individual actions to be performed during the planned test campaign, along with associated functional checks and telemetry monitoring activities.

The planned acceptance criteria were primarily based on the satellite's successful operation during the thermal exposure. During the first thermal cycle, the satellite was required to boot up and initialize successfully under both hot and cold conditions. In the subsequent operational cycles, it was largely expected that the satellite would operate continuously without any anomalies, unexpected resets, or communication interruptions. Furthermore, it was also specified that no structural or mechanical degradation would occur during the campaign.

Satellite monitoring was designed to utilize both internal telemetry and external sensors. Continuous communication with the satellite is planned throughout the entire test duration in order to monitor operational telemetry and the status of its subsystems. In addition, a total of 12 external thermocouples were planned to be distributed around the satellite structure. The individual planned locations of the thermocouples are summarized in Table 7.1 and illustrated in Figure 7.1 below.

Table 7.1: Thermocouples positions

Sensor Number	Position
1	Experimental UHF PCB
2	Experimental UHF PCB under shield
3	TunaCan
4	OBC PCB
5	OBC PCB under shield
6	Inside solar X
7	Outside solar X
8	Space pin
9	Rail side X
10	Frame
11	EPS PCB
12	EPS Battery

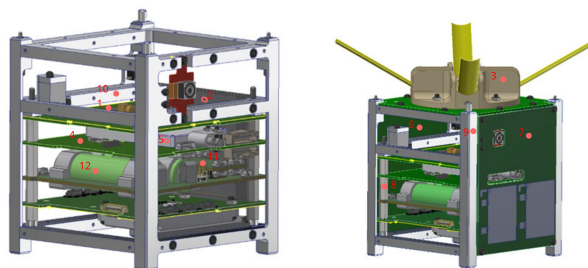


Fig. 7.1: Illustration of Thermocouples locations

In addition to operational verification, the campaign aimed to provide approximate data on the outgassing of the satellite. This measurement was planned to be performed utilizing a precision laboratory scale by comparing the satellite's mass prior to and immediately after the completion of the thermal vacuum campaign.

In addition to the planned TVAC campaign, a separate functional verification of the antenna deployment mechanism was conducted under vacuum conditions using a dedicated vacuum chamber. During the experiment, a pressure of approximately 10^{-5} Pa was achieved, allowing the conclusion that the antenna deployment mechanism functions correctly even under vacuum conditions. The antenna deployment was successful, and no anomalies were detected. Since the experiment represented only a standalone functional verification, no dedicated formal test procedure was created for its purpose.

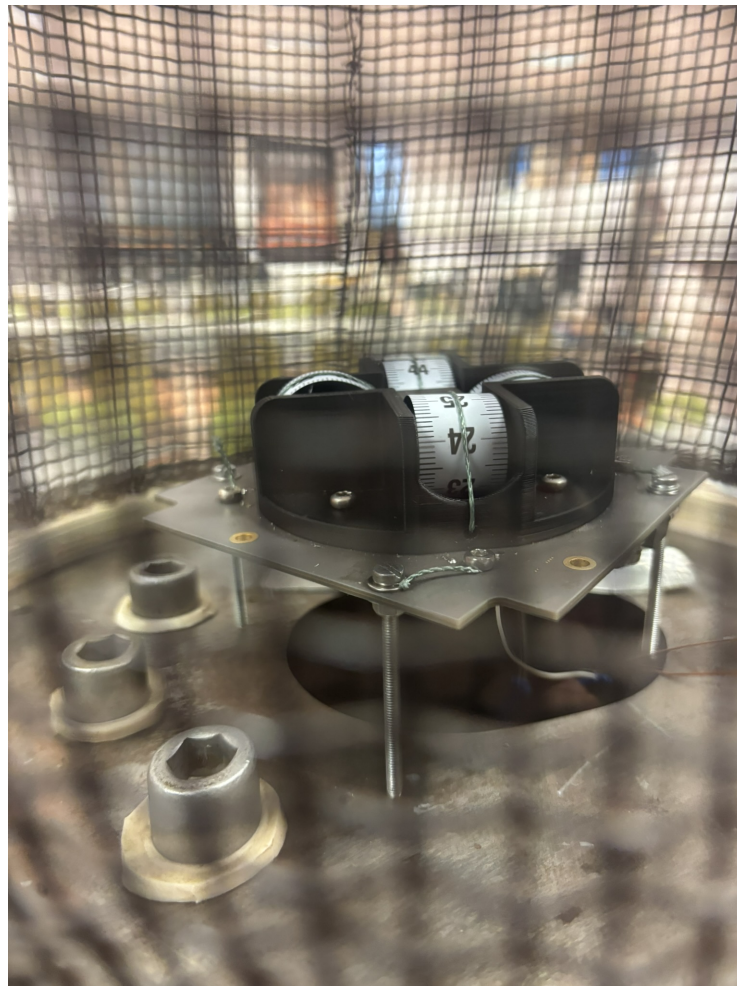


Fig. 7.2: Functional test of TunaCan antenna in vacuum conditions

7.2 Campaign Readiness and Limitations

At the time the thermal vacuum campaign was being prepared, it was assumed that a newly acquired thermal vacuum chamber would be available at the university by the required project timeframe. The entire TVAC campaign was planned around the assumption that this testing equipment would be available during the environmental testing campaign.

During the satellite integration phase, it became clear that the delivery and commissioning of this chamber would be delayed beyond the originally planned schedule. At that stage of the project, the remaining campaign activities had already been tightly scheduled, and significant modifications to the overall project timeline were no longer feasible.

An alternative that was considered was to conduct the TVAC campaign at an external testing laboratory. However, this option would have entailed significantly higher testing costs, along with additional scheduling constraints related to the availability of the external laboratory. Considering the remainder of the project schedule and the fact that the thermal vacuum campaign was not directly required by the launch provider, the final decision was made not to conduct the planned TVAC tests and to accept the associated risks.

Although the final execution phase was canceled, the TVAC campaign itself was fully prepared from both a procedural and technical standpoint. Before the campaign was canceled, the complete test procedure, thermal profiles, thermocouple layout, and operational test sequences had been finalized. From a project perspective, the satellite and documentation were considered ready for the planned tests.

The absence of the TVAC campaign did not directly affect KOSTKA's eligibility for launch, since, as previously mentioned, the launch provider's verification requirements did not mandate the submission of these test results. However, the cancellation of the campaign increased the mission's risks due to the lack of experimental verification of the satellite's operation under combined thermal and vacuum conditions.

7.3 Summary

The planned thermal vacuum campaign represents another environmental verification activity focused primarily on verifying the satellite's operational functionality during thermal cycling in a vacuum environment. Although the final execution of the campaign was not carried out due to delays in the availability of the planned test equipment, the procedural and technical preparation for the campaign was successfully completed.

The developed TVAC procedure defined the complete thermal profile, instrumentation layout, and acceptance criteria for conducting the planned tests. The prepared campaign demonstrated the overall readiness of the satellite and associated documentation for thermal vacuum verification activities, while also highlighting the importance of facility availability and scheduling constraints during the planning of environmental testing for small satellite missions.

Conclusion

This thesis focused on the verification and testing of CubeSat missions, with an emphasis on the practical implementation and organization of verification campaigns for nanosatellite projects. The thesis provided an overview of testing approaches commonly used in CubeSat development, including environmental testing methods, verification philosophies, documentation, and general test campaign planning. Considerable attention was devoted to the relationship between standardized verification approaches and practical constraints typically associated with CubeSat missions, such as limited resources, tight schedules, and restricted access to specialized testing facilities.

A significant portion of this thesis focused on environmental testing methods relevant to CubeSat missions, including vibration testing, thermal vacuum testing, electromagnetic compatibility, and radiation testing, along with associated verification approaches. The thesis discussed not only the purpose and principles of individual tests but also their roles within the overall satellite verification process. Special emphasis was placed on vibration testing due to its critical importance for launch survivability and the strong dependence of test conditions on the launch provider's requirements. The thesis also addresses the importance of verification documentation, including test plans, procedures, reports, and verification matrices, which form the core of a structured verification campaign.

The practical aspects of CubeSat verification were demonstrated using selected activities carried out during the development and testing phases of the KOSTKA mission. These practical examples provide insight into the preparation of test procedures, the implementation of environmental test activities, the interpretation of test requirements, and data processing and acquisition. The work also demonstrates that the preparation of a verification campaign often extends beyond the actual execution of tests, as significant engineering effort is required to correctly interpret generalized standards and transform them into mission-specific test procedures and acceptance criteria.

The thesis further emphasized that verification campaigns for CubeSat missions cannot be approached as a universally defined process. Although aerospace standards and recommended practices provide an important foundation for spacecraft verification, the final implementation must always reflect the specific requirements, objectives, and constraints of the given mission. In practice, the resulting verification strategy represents a compromise between the desired level of reliability, acceptance of mission risks, available financial and technical resources, planning constraints, and available testing infrastructure. This is particularly relevant for university and student teams, where verification levels often need to be balanced against the overall

complexity and feasibility of the project.

The experience gained during the preparation and execution of the required verification activities also underscores the importance of early planning and allowing for sufficient schedule buffers during mission development. Environmental test campaigns require significant coordination between satellite development, documentation preparation, test laboratory availability, and launch provider requirements. Insufficient time reserves can significantly complicate both the preparation and execution of verification activities, especially in the later phases of mission development.

Overall, this thesis provides a practical overview of verification and testing approaches applicable to CubeSat missions, while demonstrating selected real-world implementation aspects through practical experience gained from the KOSTKA project. The presented information may then serve as a useful reference for future CubeSat projects and student teams preparing environmental verification campaigns for nanosatellite missions.

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Symbols and abbreviations

ADCS	Attitude Determination and Control System
C&DH	Command and Data Handling
CDS	CubeSat Design Specification
COTS	Commercial Off-The-Shelf
ECSS	European Cooperation for Space Standardization
EMC	Electromagnetic Compatibility
EPS	Electrical Power System
GEVS	General Environmental Verification Standard
GSE	Ground Support Equipment
LEOP	Launch and Early Orbit Phase
LEO	Low Earth Orbit
MHS	Miniaturized Heat Switch
MPPT	Maximum Power Point Tracking
NASA	National Aeronautics and Space Administration
OBC	On-Board Computer
PCB	Printed Circuit Board
PMAGCS	Permanent Magnet Attitude Control System
PSD	Power Spectral Density
RBF	Remove Before Flight
RMS	Root Mean Square
RTOS	Real-Time Operating System
SSO	Sun-Synchronous Orbit
TVAC	Thermal Vacuum
UHF	Ultra High Frequency