FLY YOUR SATELLITE!

FYS Design Booster
CubeSat Proposal

CubeSat Name	AlbaSat
Form factor (1U, 2U, 3U, 6U)	2U

Proposing university/ institute

Name	Università degli Studi di Padova
Department If applicable	Department of Industrial Engineering (DII)
City	Padova
Country	Italy

The team leader declares that all the information provided in this proposal is true and correct to the best of his/her knowledge and undertakes to inform ESA of any changes immediately.

Federico Basana, PhD student	23/11/2022	Federico Basana
Name Job Title (professional) or Level of studies (student)	Date	Signature

Checklist

The CubeSat team is from a university based in an eligible state	☒
The CubeSat is 1U, 2U, 3U, or 6U	☒
The prime objective of the CubeSat project is educational and with no pecuniary interest	☒
A core team of at least 6 university student team members are participating in the project and comply with the eligibility criteria	☒
In addition to the 6, a minimum of 2 supervisors are covering the following three functions for the CubeSat team: (1) endorsing professor, (2) team leader, (3) system engineer	☒
The CubeSat team will rely on their own sponsors	☒
Compliance to the CubeSat Design Specification rev.13 (for 1U, 2U or 3U CubeSats) or the 6.0 CubeSat Design Specification rev1.0 (for 6U CubeSats) is reported in the dedicated annex of the proposal	☒
The CubeSat project is endorsed by the university with the signed endorsement letter attached to the online submission form	☒
No more than one proposal from the same university faculty/department has been submitted to the FYS Design Booster programme	☒
If the proposed project is a joint undertaking of multiple parties (e.g. two universities collaborating on one mission), one single university shall act as the leading university and the key point of contact vis-à-vis ESA. The collaborations with other organisations or institutions shall be clarified in the proposal, including a description of the mutual agreements.	☒

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ABBREVIATIONS AND ACRONYMS

ADCS	Attitude Determination Control System
CCR	Corner Cube Reflector
CDR	Critical Design Review
COTS	Commercial Off The Shelf
DM	Development Model
EM	Engineering Model
EPS	Electric Power System
FOV	Field Of View
FS	Flight Software
FTR	Functional Test Review
GS	Ground Station
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Phase
MDR	Mission Definition Review
MEMS	Micro Electro-Mechanical System
MRR	Modulating Retro Reflector
NP	Nadir Pointing
OBC	On-Board Computer
P/L	PayLoad
PAT	Position Acquisition and Tracking
PCB	Printed Circuit Board
PDR	Preliminary Design Review
PFM	ProtoFlight Model
QC	Quantum Communication
QKD	Quantum Key Distribution
RSA	Rivest-Shamir-Adleman
SLR	Satellite laser Ranging
SQC	Satellite Quantum Communication
SSO	Sun Synchronous Orbit
TBC	To Be Confirmed
TBD	To Be Defined
TLE	Two Lines Element
TRL	Test Readiness Level
TT&C	Telemetry, Tracking and Control

STATUS OVERVIEW

The team has started its activities in December 2019. Firstly, the mission objectives have been defined, then a state of the art analysis has been performed. From the state of the art analysis results, the team executed the feasibility study of the mission and defined preliminary mission requirements. The team performed preliminary analysis to retrieve preliminary design of the subsystems. Currently, the project is at the level of the Preliminary Design Review (PDR). Future activities will include the Baseline Design Review, component level tests for verifying and consolidating the satellite design and the consolidation of the design at the level of the Critical Design Review (CDR).

Activity	Started		Concluded		Comments
	[Y/N]	Date (mm/yy)	[Y/N]	Date (mm/yy)	
Mission objectives definition & needs identification	Y	12/19	Y	03/20	
Concept definition & Feasibility study	Y	03/20	Y	12/21	
Preliminary design phase	Y	12/21	Y	09/22	
Detailed design	N	/	/	/	
Thermal analysis	Y	03/20	N		- Preliminary FEA analysis conducted
Structural analysis	Y	03/20	N		- Preliminary FEA analysis conducted
AOCS analysis	Y	03/20	N		
Other analysis	Y	03/20	N		- Preliminary risk assessment analysis
In-house-developed prototypes manufactured and tested	Y	06/22	N	/	- Impact sensor prototyping ongoing, other payloads are COTS
In-house-developed engineering/flight models manufactured and tested	N	/	/	/	
CubeSat subsystems or payload procured or manufactured	N	/	/	/	
Subsystems, payload or ground station tests performed (if yes, specify which systems, tests and test levels)	N	/	/	/	
FlatSat Integration & Testing (specify) what kind of tests)	N	/	/	/	
CubeSat stack testing	N	/	/	/	
CubeSat fully integrated	N	/		/	
Ground station installed	N	/	/	/	

Activity	Started		Concluded		Comments
	[Y/N]	**Date** (mm/yy)	**[Y/N]**	**Date** (mm/yy)	
Ground station operational	N	/	/	/	
System level tests performed	N	/	/	/	
Flight ready	N	/	/	/	
Launch opportunity secured	N	/	/	/	
Other (please specify)	/	/	/	/	

ABSTRACT

Alba CubeSat UniPD is a student team of University of Padova. Its objective is to launch for the first time at our university a CubeSat made by students thus allowing the student to make a hands-on activity during their studies. Since 2009, students from the University of Padova have been involved in several ESA hands-on activities (Rexus-Bexus, Drop Your Thesis, Fly Your Thesis), and this will be the first time we apply for the Fly Your Satellite! Programme.

The proposed mission has four independent objectives: (1) to collect in-situ measurements of the sub-mm space debris environment in LEO, (2) to study the micro-vibration environment on the satellite throughout different mission phases, (3) to do orbit and attitude determination through laser ranging; (4) to investigate alternative systems for possible Satellite Quantum Communication applications on nanosatellites. The proposed technological experiments aim to obtain data that will enrich the current knowledge of the space environment and will provide precious information useful for the further development of some research projects currently performed at the University of Padova.

In order to reach the said objectives, in these years the activities of the team aimed at developing a 2U CubeSat equipped with four payloads. The first payload is an impact sensor that will be placed on one of the outer faces of the satellite and will be able to count the number of debris impacting the spacecraft. It will be based on the strong background vested in our university in this field, with several impact sensor prototypes/breadboards already evaluated and calibrated in the CISAS hypervelocity impact laboratory (mostly in the frame of research contracts with ASI). The second one is a Commercial Off The Shelf (COTS) sensor that measures the micro-vibrations experienced by payloads in a CubeSat in different mission phases. The third one consists in a number of COTS Corner Cube Retroreflectors that are placed onboard the satellite. Thanks to these CCRs, Satellite Laser Ranging (SLR) will be performed to collect data on the satellite position and attitude using a facility recently developed at the University. The fourth consists of a Modulating-Retro Reflector (MRR) where one of the mirrors is a MEMS Grating Modulator. By deforming one of the mirrors, it is possible to change the angle of reflection of a laser beam coming from ground thus obtaining an intensity modulator. The MEMS Grating modulator is a COTS component produced by Boston Micromachines Corporation. The architecture of the standard 2U platform is configured to include the needed subsystems, i.e., Telemetry, Tracking and Control System (TT&C), Electric Power System (EPS), Attitude Determination Control System (ADCS), On-Board Computer (OBC). No propulsion is requested. COTS components are extensively used for building the satellite.

1 MISSION DESCRIPTION

1.1 Mission Objectives

The proposed mission has four independent objectives:

- to collect in-situ measurements of the sub-mm space debris environment in LEO;
- to study the micro-vibration environment on the satellite throughout different mission phases;
- to do precise orbit and attitude determination through laser ranging;
- to investigate alternative systems for possible Satellite Quantum Communication applications on nanosatellites

Besides these technological objectives, the main purpose of the project is to develop a unique educational experience: since 2009, students from the University of Padova have been involved in several ESA hands-on activities (Rexus-Bexus, Drop Your Thesis, Fly Your Thesis), and this will be the first time we apply for the Fly Your Satellite! programme.

The proposed technological experiments aim at obtaining data that will enrich the current knowledge of the space environment and will provide precious information useful for the further development of some research projects currently performed at our university. These projects are in the fields of small satellites technologies, with focus on (1) space debris, (2) highly-stable pointing mechanisms, and (3) pointing, acquisition and tracking of small satellites with laser payloads, (4) satellite Quantum Communication. The mission objectives are presented in some details in the following paragraphs.

Collect in-situ measures of the space debris environment

The first objective is to count the number of impacts with small debris (sub-mm size). The small debris population cannot be observed from ground and available models require validation through in-situ measurements [1,2]. To this aim several missions and payloads have been proposed, e.g. the ESA “Debris inOrbit Evaluator” (DEBIE I, DEBIE II) [3], the NASA “Space Debris Sensor” onboard the ISS [4], the “in-situ micro-debris measurement system” from JAXA [2], the piezoelectric sensor developed by University of Texas at Austin onboard the Armadillo CubeSat [5], and the solar panel-based impact detector SOLID [6]. To reach this first objective we are developing an impact sensor that will be placed on the face of the satellite along the velocity direction (not covered by solar panels).

Study the micro-vibration environment on the satellite

The second objective is to collect information on micro-vibrations experienced by payloads on a CubeSat in different mission phases. Micro-vibrations on spacecraft represent an issue for payloads requiring high pointing accuracy and/or stability over time [7], and they might represent a particular concern for CubeSats and small satellites (that normally are not equipped with very-high performance attitude control systems). Furthermore, simulating real orbital disturbances is difficult on ground [8, 9], and hence collecting reliable measures of the possible vibrations spectra in realistic operational scenarios is a significant research activity. Micro-vibrations in space have been measured in few missions such as SAMS-II and AMAMS by NASA [10, 11]. However, to our best knowledge there is a lack of information about micro-vibrations on small satellites and there is also the need of correlating such vibrations with typical mission events (e.g., activation of actuators, thermal cycling, debris impacts, etc.) [12]. This will help to improve the pointing accuracy and stability of future high-performance payloads for small satellites. To reach this second objective, we aim to measure vibrations with COTS sensors, such as accelerometers, and correlate the measurements with scheduled as well as unpredicted mission events.

Perform orbit and attitude determination through laser ranging

The third objective is to perform precise orbit determination using laser ranging. Satellite Laser Ranging (SLR) has been used in many important scientific missions for precise orbit determination (e.g., GOCE [13] and Galileo [14]). Several examples of CubeSat using this technique can be found such as Nice Cube [15], BeoCube [16] and CUBETH [17]. In addition, attitude determination based on the returned signal from the laser will be implemented. Example of this technologies can be seen used on non-cooperative target such as ENVISAT [18] and proposed for a series of small satellites [19]. In order to reach this objective, a number of COTS Corner Cube Retroreflectors will be placed onboard the satellite and SLR will be done to collect data on the satellite range and range rate. To this aim, we will use a new facility recently developed at our university for testing laser communication with LEO satellites. This facility is based on a telescope with automated azimuth/elevation control, a beacon laser and a receiver section with active focusing of the laser spot. We are also considering the development of an attitude determination algorithm.

Investigate alternative systems for possible Satellite Quantum Communication applications on nanosatellites

The security of telecommunication systems is typically ensured by cryptographic protocols that are based on computational assumptions (computational security). For instance, the complexity of factorizing large prime numbers is at the basis of the Rivest-Shamir-Adleman (RSA) public-key cryptosystem. However, this technique will be irremediably compromised with the arrival of quantum computers, which can provide an exponential speed-up in performing complex computational tasks, as factorizing prime numbers. Quantum Key Distribution (QKD), once paired with the symmetric cryptosystem called One Time Pad (OTP), offers a solution to this issue since the security of the key generation process is guaranteed by physical laws such as monogamy and no-cloning theorem. This type of security is called unconditional security and it does not require additional assumptions as in the case of computational security.

Quantum Communications (QC) typically requires weak optical signal that reach the quantum level. These signals suffer from the high attenuation of fibers or ground-to-ground free-space link (no QKD is possible over some hundreds of kilometers). For this reason, Satellite Quantum Communication (SQC) can be exploited to overcome such limit as losses on a satellite-to-ground link are far lower than a ground-to-ground. As a matter of fact, SQC had become the center of several studies and research to reach global coverage with an SQC satellite network, bringing QKD everywhere.

Within this framework, the CubeSat technology will represent a pivotal point in the implementation of the global SQC network. Moreover, proper ground stations, with the capability to correctly detect polarized single-photon (the most effective qubit encoding for a free-space QKD link), shall be designed and deployed to fully exploit such satellite network.

The Quantum Future payload aims at providing a reference system that can be used by any ground station to promote itself to a “quantum ground station”, able to interface with the future SQC network. Furthermore, as the role of “space quantum engineer” is yet to be totally defined, the possibility to perform experiments with a dedicated quantum payload and ground station can be a key point for the new generation of students aiming at acquiring new skills.

1.1.1 Sustainable Development Goals

The educational objective of the project is not limited to academic studies. By participating in the project, students will also learn how to work in a team and to respect their peers. The members of the team will follow the United Nations Sustainable Development Goals (SDG) inside and outside the project. In particular, the team has identified four SDG to be pursued during the project.

Quality education

The Alba CubeSat project was founded for educational purposes. Since its creation, particular attention has been placed into ensuring inclusive and equitable quality education: the team gives to every student the possibility to freely participate (Target 4.4). The project started its activities in December 2019, right before the pandemic outbreak. During the pandemic, students have found in the team a way to continue cultivating their passions. The project encouraged the students to continue studying during the pandemic. In this fashion, the project has been supporting and has been enhancing the skills that the students acquire in the academic courses. In order to solve project challenges, students deepen what they have learnt in academic courses: students participating in the project develop technical and vocational skills (Target 4.3).

Gender equality

The participation in the project is open to every student who wants to apply. No discriminations are made during the recruiting process (Target 5.1). All the members of the team strongly believe in gender equality and we all work to ensure that this is respected inside and outside the team. Every member of the team has the same opportunities for leadership at all levels of decision-making (Target 5.5).

Industry, Innovation and Infrastructure

The Alba CubeSat team has proposed four different mission objectives. In these objectives several innovative aspects can be identified (Target 9.5). For instance, to our best knowledge, there is a lack of information about micro-vibrations on CubeSats, so our mission has the objective to measure them for the first time. In addition, the collaboration with QuantumFuture group will permit identifying new method for satellite Quantum Communication systems.

During the development of the CubeSat, the team is going to contact small and medium-sized enterprises in the area. This will foster collaboration between the university and small industrial enterprises, promoting the growth of companies and the consequent creation of jobs (Target 9.1, 9.3).

Climate action

All the team members are well aware about the effects of human activities on the environment. However, the space environment is also endangered by them. By collecting in-situ measurements of the sub-mm space debris environment in LEO it will be possible to better understand the effects of the human activities in space. The team will not only work to combat climate change on Earth but will also help study the debris population in space caused by human activities. The team commits to improve the awareness on the space debris problem and their mitigation. (Target 13.3).

1.1.2 ESA roadmaps

The proposed mission aligns with two ESA strategic goals. They are in the topic of AOCS/Positioning and Telecom. The following table reports the strategic goal ID and the motivation of why the proposed mission is compliant with the identified strategic goals.

ID	Detail	Motivation
68	Position sensors	Corner Cube Reflectors will be mounted on the satellite. They will be used to do precise orbit determination through laser ranging. In addition, they can be used for attitude reconstruction of tumbling/uncontrolled objects.
86	Optical Communications for Space	By employing a MRR coupled with a MEMS Grating Modulator, the proposed mission will demonstrate new technology for Quantum Communication. As a matter of fact, the future Satellite Quantum Communication network will require both compact satellites and effective ground stations, able to detect polarized single-photon. By exploiting the MRR, a given GS will be able to calibrate the required setup to correctly interface with an SQC payload. The outcome of this project will be fundamental in the developing and testing of future MRR with also polarization capabilities, for an effective application to real Quantum Communication scenarios.

1.2 Mission Phases and Mission Timeline

In this section, the different mission phases will be presented and analysed. Three mission phases have been identified: Launch and Early Orbit Phase (LEOP), operations and disposal phase. Figure 1 shows a predicted timeline for the 500 km SSO orbit, highlighting how the phases are planned across the mission lifespan.

LEOP (~ 2 months)

During the LEOP phase, the spacecraft will be commissioned into orbit and it will be prepared for nominal operations. The phase is further subdivided in sub-phases, tracing the development of the mission. These are not considered proper mission modes given their sequential nature.

1. **Launch and orbit insertion:** The launch phase is expected to last for some hours (TBC); all sub-systems shall be turned off. After the in-orbit delivery, the OBC, EPS and ADCS subsystems are activated and the next sub-phase is initiated.
2. **Detumbling manoeuvre:** The detumble shall reduce the rotational rate to a threshold of TBD deg/s right after deployment. The TTC sub-system and the payloads are kept not active. A robust ADCS control is used for the manoeuvre.
3. **Activation:** Antennas will be deployed after detumbling by triggering a disposable retention mechanism. Power control will be also crucial: batteries will be already charged

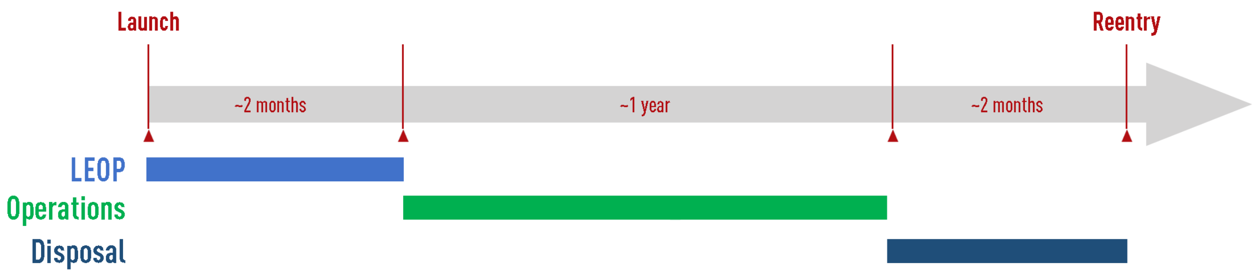


Figure 1: Proposed mission phases timeline for the 500 km SSO orbit.

at deployment and they will be connected at the spacecraft release, ensuring proper power generation and distribution to the first vital subsystems and ultimately to the mission payloads.

4. **Coarse NP and first contact:** The spacecraft will achieve radio ground contact. The ADCS shall provide a coarse pointing of TBD deg. Two Lines Element (TLE) data will allow the first acquisition from the ground station antenna. Furthermore, attitude control will be employed to properly deploy and control the orientation of the antennas.
5. **Commissioning:** The remaining sub-systems will be switched on, checked and enabled in a specific sequence. Payloads will be tested and calibrated, reaching operation readiness.

Operations (~ 2 year)

Once the LEOP phase will be successfully completed, the Operations phase will be initiated. During this phase the spacecraft will perform its main mission tasks, collecting scientific data, activating and running the payloads and downlinking the data to the ground station facility. All non-essential systems are turned off in case of unexpected or hazardous events.

Disposal (~ 2 months)

For a 500 km SSO-like orbit the phase will be activated ~2 months prior to the expected re-entry. Alternatively, the phase will be activated when the batteries capacity drops below a TBD Ah threshold. When the pre-disposal operations will be completed, the satellite will passively de-orbit. From the simulated altitude evolution the re-entry phase (without considering the passivation and waiting phase) will last ~2 weeks. Nonetheless, a significantly longer window has been chosen to extensively study and predict the re-entry phase (by evaluating the atmosphere effects on the orbit evolution) to ensure a safe disposal.

- **Passivation and waiting:** During this phase all switches will be opened and the batteries will be drained.
- **Re-entry:** The orbit will keep lowering due to aerodynamic drag. Under 80 km, the spacecraft will completely burn-up in the atmosphere.

Furthermore, the disposal phase can be directly accessed before the 1-year mark set by payload operations if some critical failure occurs: the phase bypass can be enforced from ground control from the safe mode or if specific conditions are met on-board. It should also be noted that even if the initial orbit is changed, the spacecraft must meet the 25-year maximum post-operational deorbiting requirement, as stated by the European Code of Conduct on Space Debris Mitigation.

1.3 Concept of Operations

1.3.1 Operations phase

During the LEOP phase the system will perform a series of tasks to prepare the satellite for nominal operations. These tasks consist in: (1) detumbling, (2) reaching coarse Nadir Pointing, (3) antenna deployment and (4) first contact with ground station. Once these tasks will be successfully executed the system will enter the operations phase. Several modes have been defined for this phase, namely: idle mode, communication mode, P/L micro vibration mode, P/L impact sensor mode, P/L Quantum Future mode and safe mode. The Flight Software will be organized into these modes in order to efficiently operates all the subsystems during the operational phase. A mode change is the consequence of a pre-scheduled event or a change in the environment and can occur both via ground communication or autonomous

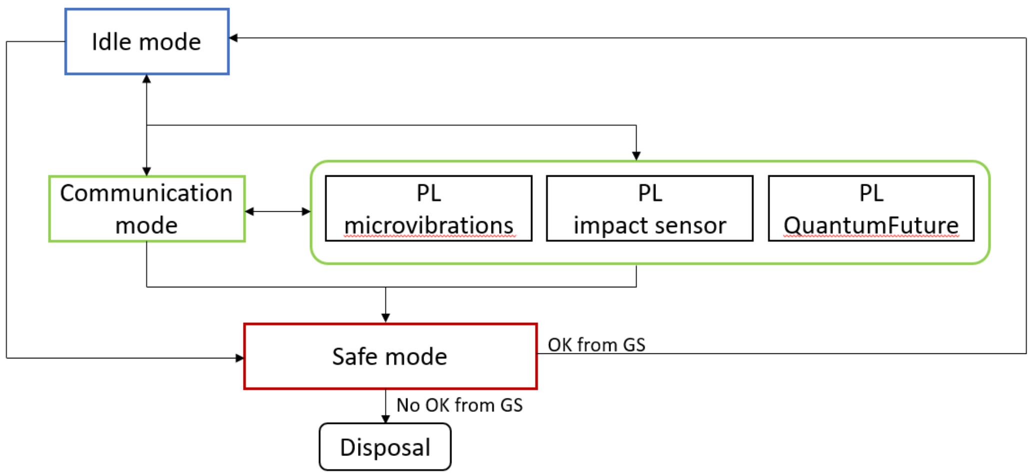


Figure 2: Scheme of the operations phase modes

on-board control. Figure 2 shows the operations modes scheme. The modes are shortly presented in the following.

Idle mode

In this mode no operations will be scheduled, the satellite shall recharge the batteries and stay alive. No payload equipment will be activated. From this mode, the spacecraft can transition into the operational modes described below. When the tasks will be completed the spacecraft shall return to idle mode.

Communication mode

This mode is designed for communication with the ground, ADCS shall provide a target pointing accuracy to the GS antenna of less than ±20 deg (TBC). This mode will be activated with pre-scheduled time-tag queue or in case of direct command from the ground when the spacecraft is in safe mode. Telemetry and payload data will be transmitted to the ground and the spacecraft will be able receive commands from the GS.

P/L micro-vibrations mode

In this mode the micro-vibrations payload will be sampling the micro-vibrations on board. In addition, the impact sensor will be operative. This mode will be activated with pre-scheduled time-tag queue. Micro-vibrations will be sampled and the data will be stored onboard. The data will be transmitted to ground when the downlink is available.

P/L impact sensor mode

In this mode the impact sensor shall be turned on. The sensor will sample at a specific sample rate. If an impact is detected, data from the payload will be transmitted to the OBC and stored onboard. They will be transmitted to ground when the downlink is available.

P/L Quantum Future mode

The MRR, i.e., the payload of Quantum Future, and the impact sensor shall be turned on. This mode is activated with pre-scheduled time-tag queue. The mode is activated if the payload is visible by a ground laser station.

Safe mode - detumble

The satellite shall enter the safe mode if one or more of the following conditions are reached:

- No ground contact for TBD days;
- Off-nominal telemetry TBD;

- Battery voltage drops below TBD;
- Rotational rate exceeding a TBD threshold;
- Command from the ground.
- …

In this mode all the non-essential system shall be turned off. The spacecraft shall autonomously execute a sequential list of TBD actions in order to maximize the possibilities of recovery, providing a safe state where the power supply is ensured using the least number of sensors and actuators. If the detected rotational rate will exceed the nominal value, a detumble mode shall be activated in order to reduce the rotational rate using only magnetorquers as actuators. Once the spacecraft will reach a safe state, it shall try to achieve a contact with ground. The satellite will exit the safe mode only if it receives specific commands from the ground. If the spacecraft will not achieve a ground contact in more than TBD days, the spacecraft shall autonomously enter the disposal phase. The spacecraft will regularly output "heartbeats" to alert any listeners of its status. The heartbeats only contain the telemetry that is critical for assessing the instantaneous state of the spacecraft without having to downlink a full telemetry log. Heartbeats include information about the satellite identification and status.

1.3.2 On-Board Autonomy

In nominal operational modes, direct commands from the ground are used only to update or override the on-board commands scheduling. The spacecraft will perform its activity autonomously using pre-scheduled events and mode changes associated with time-tag queues. It will operate without ground commands for up to TBD days. The highest level of autonomy of the spacecraft will be achieved in safe mode.

1.3.3 Mission Planning

The process of mission planning starts from the high-level constraints of the platform, payloads and ground station. First, the newest available orbital parameters will be determined using standard TLE or laser ranging data, depending on the availability. Then the pass times and relevant events during the orbit will be determined, along with updates to the specific mission goals. Considering the platform constraints (such as max power, memory availability and temperature limits) and the payloads constraints, a first schedule will be produced. The output of this process is a scheduled command list for the spacecraft (time-tag queue) and real time procedures for the ground station.

1.3.4 Scheduling

The on-board software will execute the commands that are saved in its memory as time-tag queues when its on-board clock will match one or more time-tag associated with the commands.

1.4 Mission Analysis

Orbital design is a crucial step in the mission development, both to guarantee suitable conditions for the payloads to operate and ensure compliance with the restrictions for LEO spacecrafts. Different orbits have been considered by employing the standard ESA Mission Analysis tools. MASTER and SPENVIS have been employed for the environmental analysis, DRAMA for the impact analysis and GMAT, STELA and AIRBUS SYSTEMA for precise long-term integration and multiple-subsystems simulation.

Table 1: Mission lifetime in function of the orbit

Orbit ID	Altitude [km]	Inclination [deg]	Life time
1	400	SSO	~ 4 months
2	408 (ISS)	51.6 (ISS)	~ 4 months
3	470	SSO	~ 1 year 2 months
4	500	SSO	~ 2 years 2 months
5	635	SSO	~ 25 years

The expected mission lifetime is constrained by ESA directives for LEO occupation to a maximum of 25 years [20]. This sets a constraint to the orbit selection. A set of simulations has been run in DRAMA (through the OSCAR module) to evaluate the lifetime for different orbit designs. In the analysis an ISS-like orbit has been considered together with SSOs with different altitude. The main results are summarized in Table 1, assuming a spacecraft mass of 2.2 kg and a surface cross section of 0.025 m^2. The drag and reflectivity coefficients have been set respectively to 2.2 and 1.8 and the departure date is set to 2024-01-01 12:00:00.

It can be seen that an ISS delivery would not meet the requirements of a 1-year operation window set by the payloads. The required range in altitude between 470 km and 635 km. In terms of inclination, a lower bound of INC > 45.4 deg is set to grant passages over the expected ground station location (Padova). Hence, the CubeSat mission is compatible with any orbit within the altitude range approximately 470 – 635 km and with INC> 45.4 deg. The mission orbit does not pose any requirement on eccentricity, RAAN/LTAN and launch window except from the ESA directives for LEO occupation to a maximum of 25 years.

A 500 km SSO is a good candidate to meet both payloads and ESA requirements in terms of lifetime. SSOs have raised popularity when designing EO and weather missions. This has resulted in "crowded" regions in inclination and altitude, leading to an increased conjunction probability. Catalogs such as SpaceTrack provide orbital data for objects larger than 10 cm in diameter, while sub-mm debris cannot be tracked (see Appendix D) [21].

2 DESIGN DEFINITION

2.1 System Description

To reach the set objectives, we propose to use a 2U CubeSat platform. The main functional requirements of the satellite bus are:

- To give support to payloads in terms of structural rigidity, power and data handling;
- To provide power supply to the satellite subsystems (through solar panels and batteries);
- To perform attitude determination and control in order to point the space debris sensors in the direction of the velocity vector and the corner cube retroreflector to ground;
- To transmit telemetry and payload data to the ground control centre.

Commercial Off The Shelf (COTS) components are extensively used for building the satellite. Only the space debris impact sensor is developed and tested by the team (in this respect, several space debris sensor breadboards have been already developed and tested in the University of Padova’s CISAS Hypervelocity Impact Facility in the last decade).

Table 2: Budget of the expected envelope of the subsystems.

Subsystem	Expected envelope (U)	Expected Height (mm)
TTC	0.21 U	21
EPS	0.35 U	35
ADCS	0.43 U	43
Impact sensor (motherboard)	0.14 U	14
Micro-vibration sensor	0.11 U	11
MRR + CCR	0.38 U	38
OBC	0.18 U	18
Total envelope	1.8 U	180
Envelope + 10% margin	1.98 U	198

The architecture of the standard 2U platform is configured to include the needed subsystems, i.e.:

1. TT&C: Telemetry, Tracking and Control System
2. EPS: Electric Power System
3. ADCS: Attitude Determination Control System
4. OBC: On-Board Computer

The subsystems will comply the PC-104 standard and will be mounted in a vertical orientation. Table 2 shows the expected envelope and the height of each subsystem. The impact sensor stripes will be mounted on the outer face of the satellite, hence only its motherboard is considered in the budget. For the time being, all the components will interface with the OBC through I2C or UART protocol (TBC). However, in future development of the Data-Handling a CAN Bus approach will be considered. Figure 3 shows the functional architecture of the satellite. No propulsion is requested. The payload and the subsystems are presented in detail in Sec. 2.3. The product breakdown of the system is reported in Appendix D with a table showing the commercial names, their sellers and the Technology Readiness

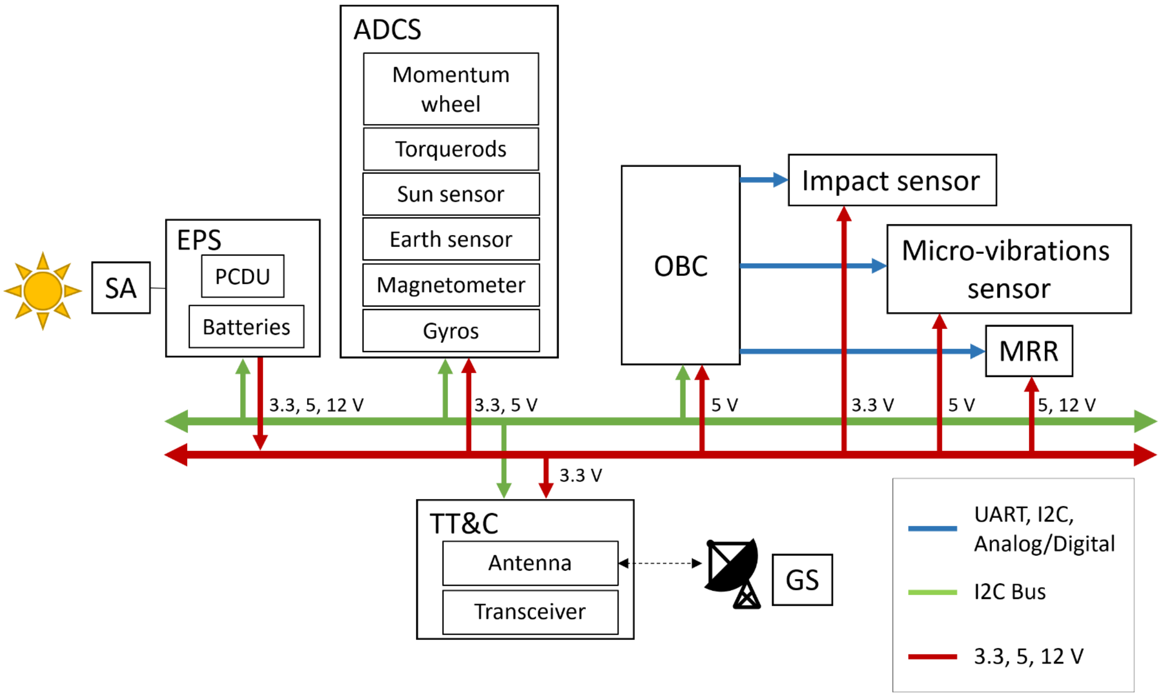


Figure 3: Functional architecture of the satellite.

Level (TRL) of the components. The team is currently aware of the issues related to the procurement of the components from outside EU vendors and the integration problems that may arise from getting them from different manufacturers. Hence, in the next iteration of the design particular attention will be posed on finding equivalent components from a single EU provider to ensure that the project timeline and milestones are respected.

Figure 4 shows the internal layout of the CubeSat. Figure 5 shows the satellite in a stowed configuration. The envelope dimensions are 110.1x110.3x227.8 mm.

Table 3 shows how much the components protrude from the plane of the rail; the reported measures are normal to the surface.

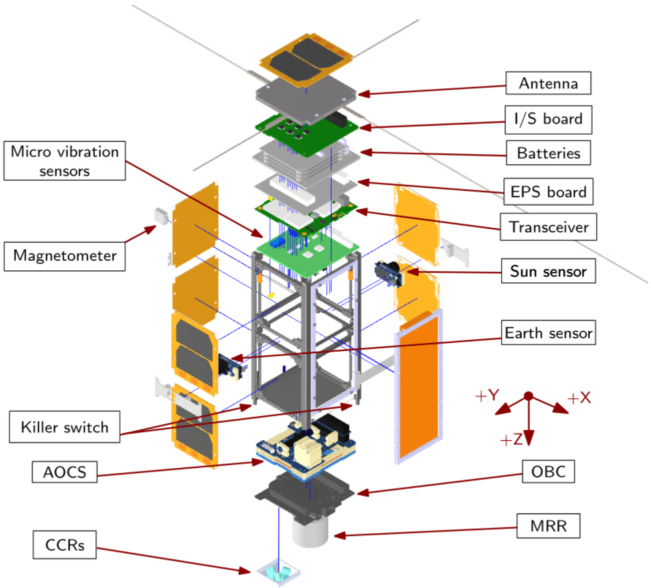


Figure 4: CubeSat exploded view.

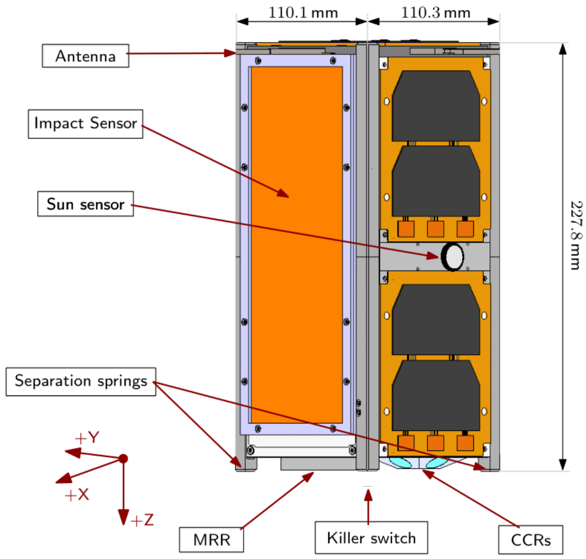


Figure 5: Representation of the CubeSat in the stowed configuration.

Table 3: Components protrusion from the plane of the rail (normal direction).

Component	Face	Protrusion [mm]
Sun sensor	- Y	4
Earth sensor	+ Y	4
Impact sensor frame	+ X	4.1
Magnetometer	- X	5.67
MRR	+ Z	0
CCR	+ Z	0
Antenna UHF (deployed)	± Y	170
Antenna VHF (deployed)	± X	550
Magnetometer (deployed)	- X	83.6

Figure 6 shows the satellite in the deployed configuration. The envelope dimensions are 1200x440.3x236.6mm. Table 3 reports the dimensions of the deployable components.

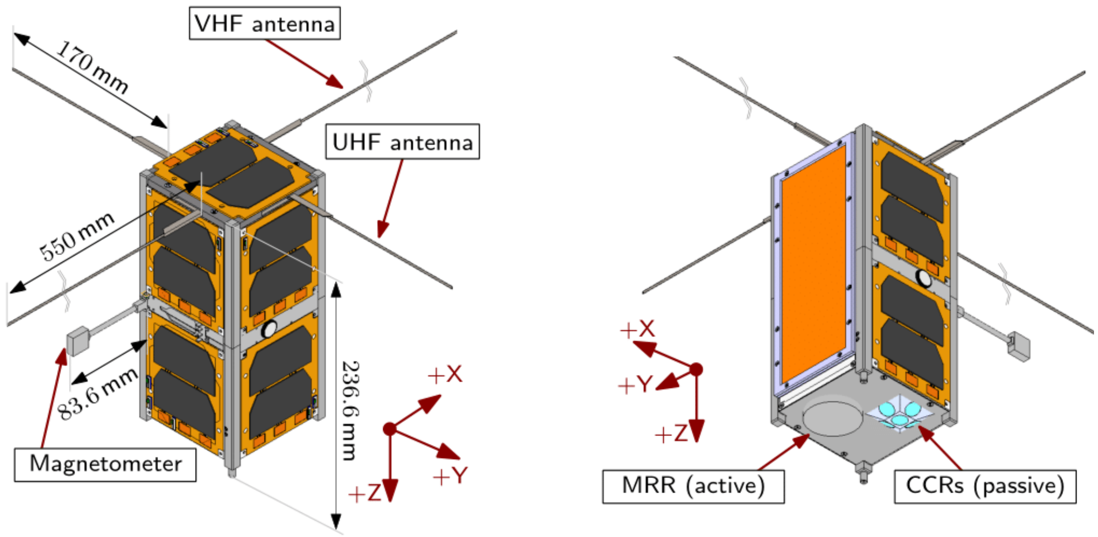


Figure 6: Representation of the CubeSat in the deployed configuration.

2.2 System budgets

For the time being, the team has completed the preliminary budget of mass, power and pointing. All the reported values are TBC.

Mass Budget

In the mass budget analysis, a margin of 5% has been considered for the COTS components while a margin of 20% has been considered for the in-house-developed components. In addition, a margin of 20% has been applied at system level. The expected mass of the system considering margins is 2579 g (Table 4).

Table 4: CubeSat mass budget.

	Mass [g]	Qnt.	Total [g]	Margin [%]	Total with Margin [g]
TTC					
ISIS deployable antenna system	98	1	98	5	103
Transceiver	75	1	75	5	79
TOT			173		182
ADCS					
Integrated ADCS	300	1	300	5	315
Sun Sensor	30	1	30	5	31.5
Earth sensor	30	1	30	5	31.5
TOT			360		378
OBC					
BeagleBone Black	40	1	40	5	42
Pumpkin P/N	31	1	31	5	33
TOT			71		75
TCS					
Sensors	20	1	20	5	21
TOT			20		21
ST&M					
ISIS space 2U structure	206	1	206	5	216
TOT			206		216

EPS					
Power conditioning and distribution	86	1	86	5	90
Solar array	50	7	350	5	368
Battery	268	1	268	5	281
TOT			704		739
Impact sensor					
Conductive Stripes	10	1	10	20	12
Structural frame	50	1	50	20	60
Dedicated board	25	1	25	20	30
Multiplexer	5	10	50	20	60
TOT			135		162
Micro-vibration sensor					
Micro- vibration sensor	67	1	67	5	70
Dedicated board	25	1	25	5	26
TOT			92		97
CCR					
Structure	80	1	96	20	96
CCR	2	5	10	5	11
TOT			90		107
Quantum Future payload					
Dedicated board	25	1	25	20	30
MRR	135	1	135	5	142
TOT			160		172

System total mass	2149
System total mass + 20% margin	2579

Power Budget

The reported power budget (Table 5) considers the subsystems active at the same time at their peak power consumptions. This assumption brings to a conservative power budget which considers the worst-case scenario. The maximum power generated on board is ~8 W. All the reported values are TBC.

Table 5: CubeSat preliminary power budget.

Subsystem	Idle mode [W]	Safe mode [W]	P/L Impact sensor [W]	P/L micro-vibrations [W]	P/L Quantum Future [W]	Communication mode [W]
ADCS	1.5	2.176	2.695	2.695	2.695	2.695
TTC	0.52	0.52	0.52	0.52	0.52	4.04
EPS	0.01	0.01	0.01	0.01	0.01	0.01
TCS	0.01	0.01	0.01	0.01	0.01	0.01
OBC	1.8	1.8	1.8	1.8	1.8	1.8
Impact sensor	/	/	0.2	0.2	0.2	/
Microvibration sensor	/	/	/	0.18	/	/
MRR	/	/	/	/	0.5	/
System total	3.84	4.516	5.235	5.415	5.735	8.555
System total + 20% margin	4.608	5.419	6.282	6.498	6.882	10.266

Pointing Budget

The pointing budget considers the pointing accuracy required from the subsystems for each axis (Table 6). In the satellite the subsystems that require pointing accuracy are: the impact sensor, the MMR, the CCR and the antenna. For each mode the pointing accuracy required by these subsystems is evaluated and the lower value is taken. All the reported values are TBC.

Table 6: CubeSat preliminary pointing accuracy budget.

	PL impact sensor mode	PL Quantum Future	Communication mode	CCR
Impact sensor [deg]	±5	±5	/	±5
MRR [deg]	/	±20	/	±20
CCR [deg]	±20	±20	±20	±20
Antenna [deg]	/	/	±20	/
Pointing accuracy	±5	±5	±20	±5

2.3 Payload & Subsystems Design Definition

2.3.1 Payload 1: Impact sensor

The aim of this payload is to collect in-situ measurements of the sub-mm space debris environment in LEO. At today, the sensor is in the preliminary design phase. The impact sensor has been developed starting from the literature and from the strong background vested in our university in this field. The impact sensor is inspired by DRAGONS experiment, which was mounted in 2018 on the ISS, in the SDS (Space Debris Sensor) payload. The DRAGONS experiment employs resistive stripes and acoustic sensors to detect an impact. The sensor is 1 m^2. The sensor under development is miniaturized to be hosted in a 2U CubeSat face. In addition, it does employs resistive stripes only. The flight model of the debris detector will be made by a multitude of thin, conductive stripes (material: copper) which are formed with fine pitch (pitch: 100 um TBC) on a thin film of nonconductive material (thickness: 12.5 um TBC, material: polyimide TBC). A dust particle impact is detected when one or more stripes are severed by the perforation hole. The sensor area will be approximately 9 cm x 18 cm. The flight model of the payload is expected to have a number of stripes up to 700 TBC.

Figure 7 shows the schematic view of the circuit design of the development model of the sensor. In the development model, the stripes are connected to ground on the bottom side, and to the control circuitry on the top side. The control circuitry consists of an Arduino Board

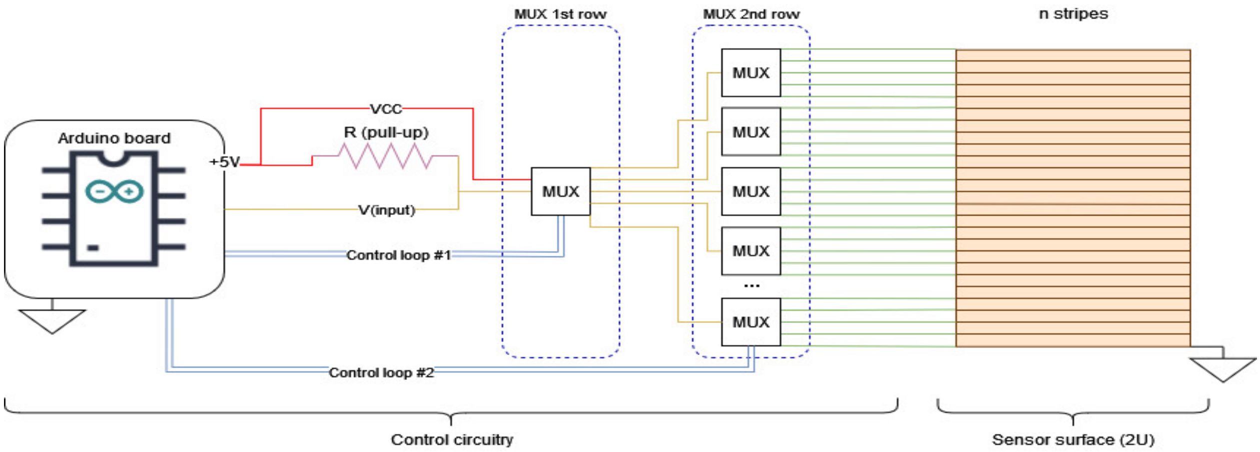


Figure 7: Schematic view of the development model of the impact sensor.

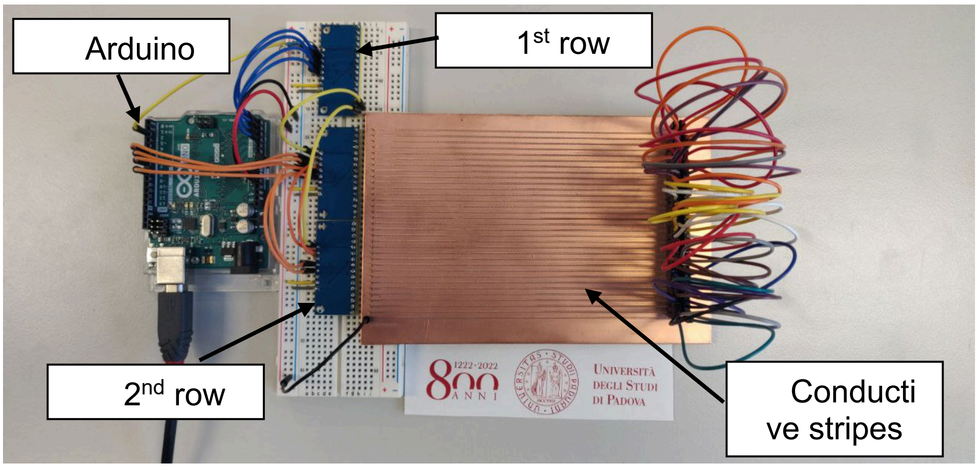


Figure 8: Prototype of the impact sensor. The jumper on the right are used to simulate the severing of a stripe.

and an array of multiplexers. In future model of the impact sensor, space qualified control board will be employed. The board loops through the stripes and registers their state (one bit per stripe: 0, intact; 1, broken). Then, it sends the data (binary word) to the OBC via an UART (TBC) connection. To select the stripe, the control board uses the select pins of the multiplexer. There are two rows of multiplexers connected in series. To read the stripes, the first row, which is made by a single mux in the current design, loops through the multiplexers of the second row selecting the mux to read. The selected multiplexer loops through the stripes in order to find if an impact has occurred. The loop of the first row is slower to let the time to read the stripes to the second multiplexer. When a stripe is selected, it is connected through the pull-up 10 kΩ resistor to the +5 V source, which allows reading. The impact sensor is turned on by the OBC. The sensor preliminary sample rate estimation is 1 Hz. If the sensor is composed of 700 stripes, the data passed to the OBC is 700 bits (1 bit per stripe). If additional information is added to the data passed to the OBC the total data volume expected is about 1 kB. When a new stripe is broken the data are provided to the OBC and memorized. The sensor keeps sampling after an impact. A prototype of the sensor is currently under development (Figure 8). It is completely representative of the functionalities of the sensor, with the only difference being that it is built with different materials (a standard PCB), and it has a lower number of stripes and multiplexers.

2.3.2 Payload 2: Micro-vibration sensor

The purpose of the payload is to collect information on micro-vibrations experienced by payloads on a CubeSat in different mission phases. A triaxial accelerometer will be used to measure micro-vibrations. The sensor shall be able to measure the acceleration experienced by a body along the three cartesian directions. The term micro-vibration usually refers to low-level mechanical vibration or disturbance in the microgravity environment, typically occurring at frequencies of 1 Hz up to 1 kHz. A crucial aspect is the sensitivity of the accelerometer which shall be in the range of milli-g. In order to achieve the objective, the sensor positioning on the satellite and its mounting system shall be considered. Indeed, the structure behave as a low pass filter. In future versions of the design, the sensor will be placed close to a possible interface with payload such as a camera; for the proposed mission the CCRs. The COTS accelerometer considered for the Development Model is the **Type 8688A Triaxial Miniature PiezoBeam** by Kistler. COTS MEMS accelerometers will be considered for the Engineering model and the ProtoFlight model. The Protoflight model of the payload will use a dedicated board for the signal analysis. The board is expected to interface with the OBC through I2C or UART (TBC) protocol. The estimated minimum acquisition time is 10 s (TBC) to ensure the frequency resolution and the sampling rate of the sensor is expected to be 5 kHz (TBC). A trade-off analysis is ongoing to find the best solution to deal with the data volume generated. On the one hand, raw data may be stored and transmitted to the GS saving RAM cost but increasing memory demand. On the other hand, data may be preliminary analysed onboard to reduce the data volume to be stored with the drawback of an increased RAM cost.

2.3.3 Payload 3: Corner Cube Reflectors (CCR)

The CubeSat will be equipped with an array of Corner Cube Retroreflector (CCR) to allow orbit and attitude determination through laser ranging from ground. An initial trade off analysis considering the dimensions, the number and the coating of the CCRs has been carried out.

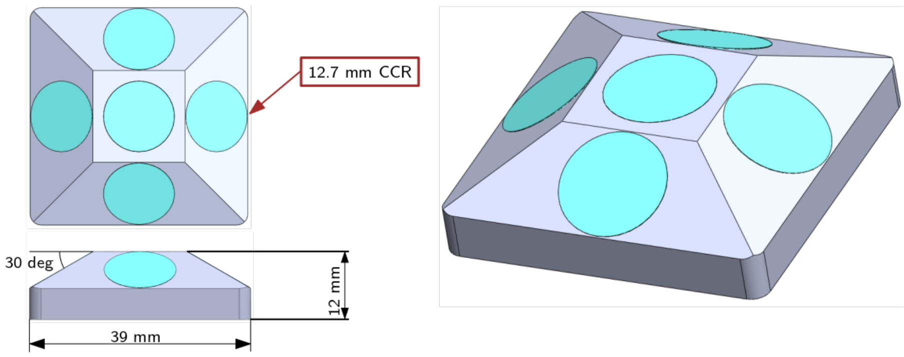


Figure 9: Preliminary configuration of the CCRs mounting system.

The CCRs considered, in terms of diameter, are: 7.16 mm, 10 mm, 12,7 mm and 25,4 mm. For each case both the coated and uncoated version were taken into account considering also multiple mounting configurations from 1 to 10 CCRs on the same face. For this preliminary trade off the laser ranging station situated in Matera was used as power source, all the technical data of the station can be retrieved from [22].

One of the key aspects during the decision was the volume constrain, including the limited external protrusion. The final choice fell on a configuration consisting of five aluminum coated 12.7 mm diameter CCRs arranged in a cross pattern. This type of structure is not a COTS and shall be implemented during the project. A first schematic view of the CCRs mounted on a dedicated structure is visible in Figure 9. The mounting configuration has an envelope of 39x39x12 mm, the central CCR is parallel with the CubeSat face, while the others are inclined by 30 deg. This configuration allows to achieve ground coverage without interruption in the field of view. The projection of the composed field of view on the ground can be schematized as a “flower” with four petals. The CCR will be mounted in the face of the CubeSat pointing nadir, +Z face. To comply with the constraints imposed by the CubeSat standard, the structure conceptual design allows a maximum protrusion from the face of 6.5 mm, the same protrusion as the rails (Figure 10). Considering a 7 days simulation, in a circular orbit at 500 km and inclination of 51.6 degree and the Matera laser ranging station as ground segment, an estimation of the duration and quality of the contact has been derived. The mean contact duration is 1.24 minutes with peaks over 4 minutes.

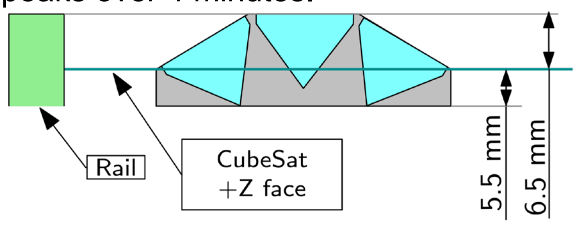


Figure 10: Schematic overview of the external protrusion of the system.

2.3.4 *Payload 4: Modulating-Retro Reflector (MRR)*

The main goal for the payload is to create a new method to test optical receiver for quantum communication such as ground stations (GS). Currently, GSs can be tested using the only available satellite for quantum communications i.e., Micius satellite developed by the Chinese Academy of Science [23]. Indeed, this creates a bottleneck in the GSs development. Alternative solutions can be CCRs of standard satellites, but this limits the test functionality losing the capability of checking synchronization and polarization alignment. Here, we want to develop a payload that is able to increase the test functionalities while keeping the SWaP cost low compared to Micius solution.

The possibility to have an active CCR, i.e., an MRR, can increase the capability of distinguish the returning single-photon from the background noise as proposed in [24]. In fact, from this point of view the MRR acts like a photon source on the CubeSat.

This solution (Scenario A) would represent a new technology as the MRR has never been integrated on a satellite and can lead to significant improvement in SQC (according to ESA roadmap ID 86). Furthermore, this will also imply a reduced setup on the GS as it will not be required a dedicated modulating system, thus simplifying its operability, and reducing its cost.

The identified MRR can be provided as a prototype (TRL 2) by Boston Micromachines (bostonmicromachines.com). Given the American provenance of this device, single point of failure might arise due, for instance, to shipping delays or import-export issues. Considering this possibility, a proper countermeasure should consider replacing the MRR with a standard corner cube-retroreflector (CCR). In this case (Scenario B), the GS should also be equipped with an additional modulating system. Indeed, this solution can still bring to excellent educational and scientific outcomes.

Method

GSs for quantum communication require as input an optical signal typically made by weak-coherent pulses as the quantum-key distribution case [25]. To emulate this signal, we propose the current method:

Scenario A

- The GS shines a high-power continuous laser to the satellite
- The incoming beam is collected and reflected by the MRR
- The MRR create an intensity modulation for the signal deflecting its trajectory
- The reflected beam goes back to the GS and is injected into the quantum receiver to test the synchronization and the single-photon detection apparatus

Scenario B

- The GS shine a high-power pulsed-modulated laser to the satellite
- The incoming beam is collected and reflected by the CCR
- The reflected beam goes back to the GS and is injected into the quantum receiver to test the synchronization and the single-photon detection apparatus

The feasibility of this experiment was evaluated by the constraint on the starting laser power of approximately 1 W. From a preliminary calculation, it was evaluated that the initial laser should be 5 orders of magnitude higher than the incoming quantum signal. This was obtained calculating the channel losses using the clipping of a gaussian beam

$$\eta_{clip} = 1 - e^{-\frac{d^2}{(2w)^2}}$$

and the approximation of the propagation of a gaussian beam

$$w(z) \approx \frac{z\,\lambda}{\pi\,wo}$$

with a LEO distance of 500 km, a lambda of 800 nm, an MMR of 0.0254 m diameter, and a GS with 0.4 m diameter.

Physical and functional architecture

The preliminary design of the payload is composed by two elements: an MRR and its electronic driver circuit (DRV-MRR). The MRR is a corner-cube reflector of 1 inch diameter where one of the mirrors is a deformable piezo-electric element. To control the piezo-electric extension, a high-voltage circuit (DRV-MRR) is used and is controlled by one digital signal (PWM) provided by a digital output of the OBC (QP.D.1). When the QP.D.1 is high, the DRV-MRR provides a high voltage signal (QP.D.3) to the input of the MRR activating the MEMS and, in turn, deflecting the incoming laser beam. When QP.D.1 is low, it releases the power from the piezo bringing it back to normal CCR behavior. The target frequency of the modulation is around 1 kHz but different frequencies (up to 10 kHz) will be tested thanks to the PWM. Also, the DRV-MRR feedbacks to the OBC an analog output ranging from 0 to 5 V (QP.D.2) that is an attenuated copy of the QP.D.3 signal for operativity check. The DRV-MRR will be powered by a regulated output from the batteries (QP.E.1) at 12 V. The payload power consumption includes both static and dynamic components, where the latter depends on the frequency modulation of the MRR and represents the major contribution. The maximum modulation frequency will be precisely evaluated after the final design taking into account the maximum available power.

The development model of the DRV-MRR will be composed by two main devices: a power supply converter and a high-voltage switching device. The identified models are:

- **Q Series by XP Power** which can convert a DC 5 V input to an DC output up to 10000 V with 5 mA. This perfectly fits the requirement on the MRR activation voltage (between 100 V 200 V) and the required current to charge its capacitor (between 300 pF and 1 nF) at the target modulation frequency. The dimensions of this device are 12.7x12.7x12.7 mm;
- **GS65011-EVBEZ** device by GaN Systems which can be used to switch on and off an output with very high voltage. By connecting the 200 V to the input of the device, it is possible to switch between 0 V and 200 V on its output which, in turn, is connected to the control pin of the MRR. The switching control is the one connected to the PWM pin from the OBC.

The electronic circuit is representative in terms of power required and functionalities of the one that will be used in the Protoflight model. Table 7 reports the preliminary details of the Quantum Future payload; all the value reported are TBC.

Table 7: Preliminary Quantum Future payload details.

Target modulation frequency	1000 Hz
Size	0.38 U
Elements in the satellite face	MRR visible from earth
Maximum power	0.5 W
Mass	160 g

2.3.5 *Attitude & Orbit Control (AOCS)*

The aim of the AOCS is to determine and control the attitude and orbital position of the satellite. However, the proposed mission does not require orbital positioning control. Hence, the AOCS provides only the pointing accuracy and stability required by the payloads, the antenna and the solar arrays during the different mission phases. Moreover, it is responsible for the satellite detumbling after the deployment and after unpredicted events such as impacts with space debris. The AOCS shall guarantee the 3-axis spacecraft stabilization to ensure the pointing accuracy required by the payloads. The catalogue selection of the AOCS is based on the estimation of the external disturbances acting on the satellite and the most demanding manoeuvres that the satellite shall perform. The expected environmental torques acting on the satellite, considering the gravity gradient, magnetic, solar radiation and aerodynamic effects, are estimated to be in the order of magnitude of 10^{-5} Nm. Supposing that the satellite had to achieve a slew angle of 140 deg in 150 s to track the position of a ground station during the passage above it, led to a required torque in the order of magnitude of 10^{-6} Nm. Moreover, by considering a uniform acceleration profile it was possible to estimate the maximum angular momentum to be in the order of magnitude of 10^{-4} Nms. A trade-off analysis has been performed considering different sensors and actuators. In the analysis, various solutions for AOCS were compared based on their performance, requirements compliance and mass, power and envelope budgets. As a result, the AOCS employs sun sensors, an earth sensor, magnetorquers and gyros to evaluate the satellite attitude and a momentum wheel and magnetorquers as actuators. The unit identified for the ADCS is the **CubeADCS Y-Momentum** produced by CubeSpace, which main characteristics is summarized in Table 8.

Table 8: CubeADCS Y-Momentum by CubeSpace main characteristics.

<table>
<tr><th rowspan="2"></th><th rowspan="2">Mass [g]</th><th rowspan="2">Volume [mm]</th><th rowspan="2">Peak Power [W]</th><th colspan="2">Temp. Range [°C]</th><th rowspan="2">Attitude Determination Accuracy [deg]</th><th rowspan="2">Attitude Control Accuracy [deg]</th></tr>
<tr><th>Min</th><th>Max</th></tr>
<tr><td>CubeADCS Y-Momentum</td><td>300</td><td rowspan="3">90x95x42.89</td><td>2.295</td><td>-10</td><td>+60</td><td rowspan="3">0.6 (3σ)</td><td rowspan="3">3 (3σ)</td></tr>
<tr><td>CubeSense - Sun sensor</td><td>30</td><td>0.2</td><td>-10</td><td>+60</td></tr>
<tr><td>CubeSense - Earth sensor</td><td>30</td><td>0.2</td><td>-10</td><td>+60</td></tr>
<tr><td>Total</td><td>360</td><td>90x95x42.89</td><td>2.695</td><td>-10</td><td>+60</td><td>0.6 (3σ)</td><td>3 (3σ)</td></tr>
</table>

This integrated ADCS unit consists of a dedicated computer, a momentum wheel, a magnetometer, 3 MEMS gyros, 10 coarse Sun sensors, 3 magnetorquers, an additional Sun sensor and an additional Earth sensor. The selected unit offers a 3-axis satellite stabilization and the possibility to perform pitch manoeuvres in the orbital plane. Moreover, being an integrated unit, this solution would increase the assembly easiness and overall system reliability. In the next iteration of the design, a GNSS receiver will be added. A trade-off analysis has been made and the **NovAtel OEM719 GPS kit** produced by GOMspace was selected. In future development steps, the mechanical and communication compatibility of the component with the system will be assessed.

Preliminary simulations in the MATLAB/Simulink environment have been performed to test the system performance. The B-dot algorithm was implemented to detumble the satellite from a certain attitude velocity to a threshold value in a given time. Moreover, the QUEST method was used to solve Wahba’s problem and test the achievable precision in attitude determination.

The AOCS has the detumbling mode and three operation mode: (1) coarse pointing, (2) precise pointing, (3) safe mode. The following table reports the components used for each mode.

Components	**Modes**			
	Detumbling	Coarse pointing	Precise pointing	Safe mode
Precise sun sensor			X	
Coarse sun sensors		X	X	X
Earth sensor			X	
Magnetometer	X	X	X	X
MEMS gyros		X	X	
Momentum Wheel			X	
Magnetorquers	X	X	X	X

2.3.6 Electrical Power (EPS)

The EPS generates and distributes electrical power to the whole system. During its working lifetime, the EPS shall:

- produce electrical power
- store exceeding electrical power
- regulate and control the electrical power
- distribute the electrical power to the loads in according with their needs.

The EPS will consist of solar arrays, a battery and a power control and distribution unit. Figure 11 shows the functional architecture of the EPS subsystem. It consists of a hybrid configuration. The Solar Arrays convert the solar flux into direct current. The current flux is directed to the Battery Charge Regulators (BCR). Each of them has an inbuilt Maximum Power Point Tracking (MPPT) in order to extract from the Solar Arrays the highest value of power. The outputs of the BCR are then connected together and supply charge to the battery and to the Power Conditioning Modules (PCM). The PCM network contains one unregulated battery voltage and three regulated voltage supplies (3.3, 5, 12 V) (TBC) each with a separate Latching Current Limiter (LCL). The LCL are connected to the loads.

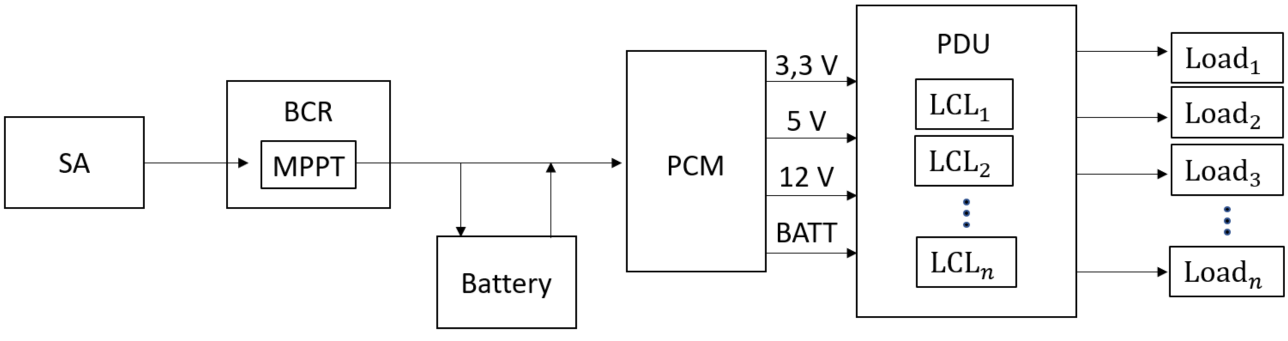


Figure 11: Electrical Power System functional architecture.

The components were chosen to fulfill the needs of the CubeSat design. An analysis has been carried out to estimate the power generation in orbit with solar panels.

Power Condition and Distribution Unit

The PCDU is the **Starbuck-Nano**, produced by AAC Clyde Space. The PDCU is based on a PC-104 motherboard. Its functional characteristics are reported in Table 9.

Table 9: Starbuck-Nano main characteristics.

Mass [g]	86
Dimensions [mm]	95.89 x 90.17 x 16.2
Regulated Power Buses [V]	3.3, 5, 12
Maximum Battery Voltage [V]	8.2
Thermal range [°C]	-40 to +85

Each power generation section is self-sufficient and does not require support from any other section within the EPS, the redundant solid-state isolation switches make it more reliable during launch operations. It includes the Maximum Power Point Tracking. Analog components and multiple inbuilt protection methods are employed to ensure safe operation during transport, launch and a full range of EPS telemetry via the I2C network. The PCDU will be switched on during the LEOP and will function continuously during the habilitation and operation phases until the disposal.

Battery

The battery is the model **Optimus-30** by AAC Clyde Space (Table 10), it features a daughterboard which matches the Starbuck-Nano PCDU. The battery is made of Lithium Polymer, built with a combination of strings connected in parallel with cell protection electronics, offering inherent redundancy. The battery is optimized for LEO missions with a maximum altitude of 850 km. It has an autonomous integrated heating system and inbuilt protections as undervoltage-, overvoltage- and string overcurrent protections. The battery is compatible with the ISS deployment safety requirements and it is qualified to the NASA standards EP-Wi-032.

Table 10: Optimus-30 main characteristics.

Mass [g]	268
Dimensions [mm]	95.89 x 90.17 x 21.55
EoC Voltage [V]	8.26
Full Discharge Voltage [V]	6.2
Charge Voltage Limit (max) [V]	8.4
Charge Voltage Limit (min) [V]	6.2
Charge/Discharge Current rate	1.53
Discharge Voltage Limit [V]	3
Quiescent Power Consumption [W]	< 0.1
Capacity [Wh]	30
Charge/Discharge Current [A]	1.95
Power Buses [V]	3.3, 5
Thermal Range [°C]	-10 to + 50

The battery is charged and turned on before the launch phase to ensure that there will be electrical power available at the deployment. The interface is the standard CubeSat PC-104.

Solar Arrays

The chosen model is the **CubeSat Solar panel DHV-CS-10**, the solar cells are the Azur Space 3G30C (Table 11). They use a triple-junction technology and are connected in series (2 for each solar panel). Both solar panels and solar cells are qualified for space applications. Solar cells are manufactured on a PCB and with integrated circuit board. The solar panels will be activated during the detumbling phase until the disposal, they will operate to feed the battery during the exposure time.

Table 11: CubeSat Solar panel DHV-CS-10 main characteristics.

Mass [g]	350
Dimensions [mm]	82.5 x 98 x 2.4
Open Circuit Voltage [V]	5.4
Short Circuit Current [A]	0.52
Voltage at max. power [V]	4.82
Current at max. power [A]	0.5
Efficiency [%]	30
Thermal range [°C]	-120 to +150

2.3.7 On-board Data Handling (OBDH)

The objective of the On-Board Data Handling subsystem is to manage and control the proper functioning of the other subsystems. In addition, OBDH shall manage the data collection and the distribution of commands received from ground to the various subsystems. The main unit responsible for processing data and controlling the satellite is the On Board Computer (OBC). The choice of the OBC is oriented toward the use of a two-module hardware. This consists of a single-board computer **BeagleBone Black (BBB)** hosted by the **Pumpkin Motherboard Module 2 (MBM 2)**. The choice resulted as a consequence of a trade-off analysis between the Pumpkin MBM 2+BBB and the ISIS On Board Computer (iOBC). The first one resulted to better fit during the preliminary analysis in terms of mass, thermal range, memory and mass storage data. The MBM 2+BBB has a wider thermal range, is more compact and has more storage data available than the other one, still showing similar features. Beside these advantages, challenges may come from (1) the delta TRL with the other COTS component and (2) the procurement of the component (extra EU vendor). If one of these issues arise during the development, the team will consider to replace the component with the ISIS iOBC. This has comparable performance but a more elevated TRL and is sold by an EU vendor.

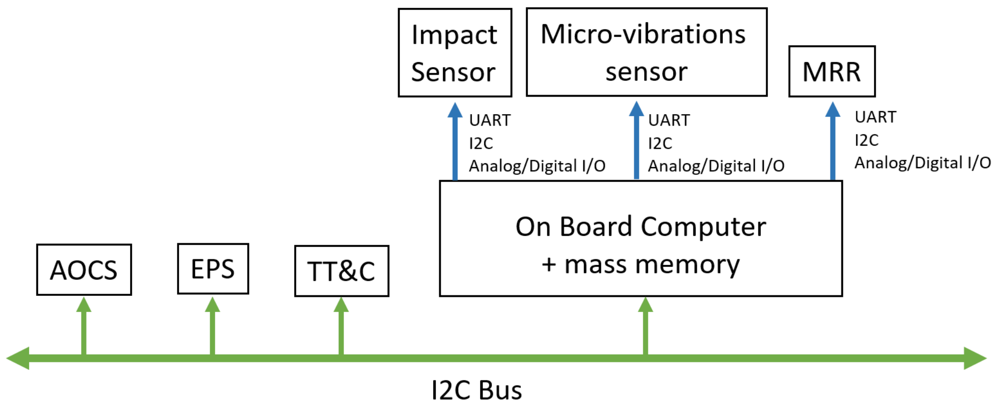


Figure 12: OBDH scheme

The features of the OBC are reported in Table 12. Table 13 shows the OBC subsystem characteristics. The OBC is compatible with the payloads and with the other subsystems. It will interface with them through UART or I2C protocol (TBC). The OBDH scheme is reported in Figure 12. The team has been made aware that the I2C protocol is not fault tolerant and has been reported as the source of mishaps to a number of satellite missions. Hence, in the next iterations of the design the team is considering to replace I2C with CAN bus.

Table 12: OBC components main features.

Processor	1 GHz ARM Cortex-A8
RAM	512 MB of DDR3
Flash storage	4GB eMMC

Table 13: OBC main characteristics.

Mass [g]	71
Dimensions [mm]	96 x 90 x 1.8
Peak power [W]	1.8
Thermal range [°C]	-40 to +85

2.3.8 On-board Software architecture (OBSW)

The OBC stores, loads, and executes the OBSW to perform actions like: controlling subsystems and payloads, manage communication channel by generating telemetry and commands and manage failure detection, isolation, and recovery (FDIR).
The main functional requirements of the OBSW are:

- control every other subsystems/payloads, monitoring and changing their status;
- perform every action prioritizing critical subsystems and following a defined time schedule;
- guarantee reliability of operations under specified conditions, especially if a hardware or software fault occur, providing a dedicated safe mode;
- protect data handled inside and outside the satellite, planning a backup code
- ensure maintainability over the course of the mission lifetime allowing software updates, corrections and parameter control from ground.

A backup version of the FS will be stored on board for redundancy and safety reason. The software architecture is the result of multiple patterns mixed together with the aim of fulfilling all software and hardware requirements. The OBSW is designed as a layered architecture (Figure 13) since upper layers must interact with other subsystems without concern about how lower layers, like BIOS, are working. The implementation of a service bus and the data handling procedures result in adopting a service-oriented architecture for the top layer, called Application, where every subsystem, as well as payloads, can provide data to the OBC and vice versa.

The BIOS is the lowest software layer, located in the nonvolatile memory, is used by the motherboard to initialize all hardware components and to load the OBSW to memory. The RTOS is the software system that manages the computer hardware and provides multitasking and scheduling. Both BIOS and RTOS are assumed to be already implemented as they are part of the purchased OBC. Communication is the software layer that allows interactions with external subsystems and the ground segment. The Application layer represents all the services that the OBSW should provide to the rest of the satellite. This software layer

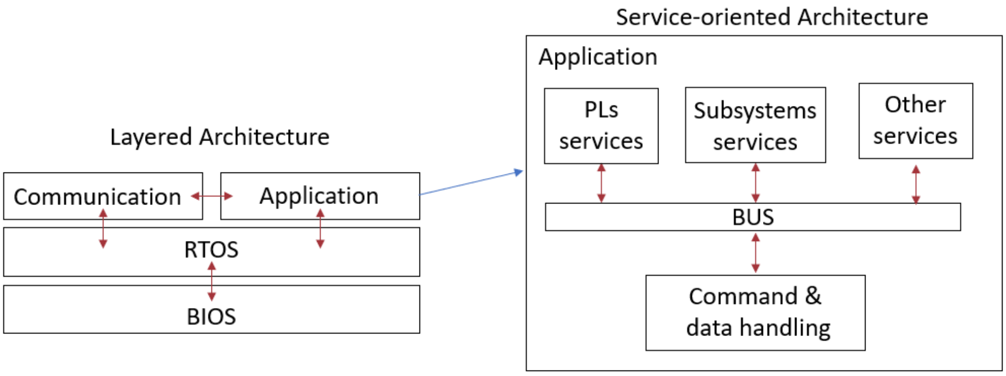


Figure 13: OBSW preliminary architecture

dialogues with all satellite components through a bus, transmitting payloads data, subsystems status, and other useful services like preloaded commands, operational modes and time-based schedules. Since the OBC is the core of the entire satellite, its software development process needs to rapidly adapt with the hardware and architectural changes. Hence, the team is choosing an Agile methodology approach iterating frequently instead of planning a precise roadmap. This fits the layered architecture as the Application layer and its services need to be a priority, while the hardware may change. The next milestone will be the development, and consequent verification and validation, of a software capable to reliably run a FlatSat version of our Cubesat.

2.3.9 Telemetry, Tracking and Communications (TT&C)

The system is constituted by two COTS components, namely the ISIS UHF/VHF Deployable antenna system and the ISIS VHF/UHF duplex transceiver.

ISIS VHF/UHF Deployable antenna system

Table 14 shows the main characteristics of the selected antenna (**ISIS VHF/UHF Deployable antenna system**). The antennas are deployed during the LEOP phase by melting a wire that keeps down the antenna lid. The procedure takes less than 3 s and requires 2 W of power. It is connected to the transceiver with a MMCX RF interface, while power and data (I2C protocol) are fed via an Omnetics Bilobe 9-pin connector.

Table 14: ISIS VHF/UHF Deployable antenna system main characteristics.

Mass [g]	89
Dimensions (stowed) [mm]	98 x 98 x 7
Dimensions (Extending fasteners) [mm]	102 x 102
Nominal power [W]	0.04 (2 W peak)
Thermal range [°C]	-20 to +60
Main beam gain [dB]	0
3 dB beamwidth [°]	80
Configuration	Turnstile or double dipole (TBC)

ISIS VHF/UHF duplex transceiver

Table 15 shows the main characteristics of the selected transceiver (**ISIS VHF/UHF duplex transceiver**). The transceiver is connected to the antenna with a MMCX RF interface and to the rest of the satellite with a CSKB connector, through which power and data are sent. When the transceiver is in transmitting mode, the power consumption is up to 4 W depending on the downlink signal strength desired. When it is in receiving mode, the power consumption is 0.48 W. In idle mode a beacon signal is transmitted every 10 s (TBC). From preliminary analysis (minimum elevation of 15 deg, 500 km of altitude) the mean downlink bandwidth is ~8000 Mbit/day with a single ground station. The selected COTS components have flown as parts of the TT&C subsystem in past missions.

Table 15: ISIS VHF/UHF duplex transceiver main characteristics.

Mass [g]	75
Dimensions (stowed) [mm]	90 x 96 x 15
Transmitter frequency [MHz]	435 - 438
Receiver frequency [MHz]	145.8 - 146
Transmit power [dBm]	27
Data rate [bps]	9600
Data layer protocol	AX.25
Temperature range [°C]	-20 to +60

2.3.10 Structures

The structure shall withstand launch loads and guarantee the right placement of components during all mission phases, so a COTS **ISIS 2-Unit CubeSat structure** has been chosen. The main advantages of this choice are the flight heritage and the acceptance testing executed by ISIS before the shipment of the unit. The mass of the structure is 170 g, that includes the structure itself and a panel that close the aft section of the CubeSat. The structure features two killer switch mechanisms.

2.3.11 Mechanisms

Three mechanism are present in the system: (1) the Momentum Wheel, (2) the deployable magnetometer and (3) the deployable antennas. All of them represent a single point of failure of the mission.

1. The loss of the Momentum Wheel results in the degradation of the pointing accuracy of the system. The team has estimated its loss as a major severity level with respect to the mission.
2. The failure of the deployment of the magnetometer implies a less accurate pointing determination. The team has estimated its loss as a minor severity level with respect to the mission
3. In case of a failure of the release mechanism of the antennas no communication with the ground station can be done. The team will procure the component in the early phases of the FYS! Programme in order to test the mechanism. This kind of failure is critical for the mission.

2.3.12 Thermal control

The aim of Thermal Control Subsystem (TCS) is to manage all the thermal loads to which the CubeSat is subjected during the mission. The Systema Software has been employed for the preliminary thermal simulations of the CubeSat during the mission.

One challenging aspect of the thermal control of the CubeSat is the impact sensor. The sensor is composed of a multitude of copper stripes, internally linked by several cables. In order to avoid the overheating of the system, the sensor will be thermally isolated from the space environment. For what concerns the simulation of the internal components, in the preliminary simulations it has been assumed that the heat power generated by the components is equal to the electrical power that they consume (Table 5).

Relying on the results of the simulations, the Worst Cold Case (WCC) and the Worst Hot Case (WHC) have been identified (Table 16).

Table 16: Results (WCC and WHC) of the preliminary thermal analysis. Margins of 30% are considered.

	Outer surfaces	Inner components
WCC [°C]	-75	-10
WHC [°C]	72	+40

The preliminary analysis showed that the satellite will not require active thermal control. Thermal sensors will be placed onboard to monitor the temperature inside the CubeSat. Thermal analysis will be conducted for each iteration of the design and, if required, electric heaters will be employed to guarantee the temperature range of the components.

More detailed analyses are currently ongoing. Thermal analysis will be conducted and updated throughout all the design phases.

2.3.13 Propulsion (when applicable)

No propulsion is requested.

2.3.14 Grounding Scheme (EMC/EMI)

The grounding scheme is TBD. The team will design the Grounding Scheme when the design of all the subsystems and payloads will be consolidated.

2.4 Ground Segment

A COTS Ground Segment will be purchased and assembled by Alba CubeSat UniPD, and its priority usage will be for Alba CubeSat UniPD project. The Ground Segment will be managed and maintained by the team members (from the SpaceLinker UniPD division). They will obtain the proper RF license prior its construction and the CubeSat launch. It will be placed in Padova (~ LAT 45°24′28″ N, LONG 11°53′09″ E) inside the University structures. The SpaceLinker UniPD division, mentioned above, is a designed group of students, members of the AlbaCubeSat UniPD project, whose objective is the design, assembly and future operation of the Ground Segment of the CubeSat. At the current time, the team is finalizing the design of the Ground Segment. As specified in Sec. 2.3.9, the Ground Segment will be designed to be interfaced with the CubeSat at a downlink frequency of 435 MHz (TBC) and at an uplink frequency of 145 MHz (TBC).

A bibliographic study has been performed, and the suitable components for the mission needs have been identified. Concurrently, a market study has been conducted for the components acquisition. The Ground Segment required components are resumed in the following list:

1. Antenna;
2. Software to control the antennas and the SDR radio
3. LNA;
4. Azimuth and Elevation Rotors;
5. SDR-based radio;
6. Power and Server rack.

An initial estimation of the Ground Station performance required is reported in the table below.

Component	Downlink performance	Uplink performance
Antenna	Figure 14	Figure 15
LNA	60 dB (TBC)	60 dB (TBC)
Rotator	Not necessary, omnidirectional antenna	Not necessary, omnidirectional antenna
SDR–based radio	VHF ranges	UHF ranges

A preliminary study has been conducted to assess the ground access availability (Table 17).

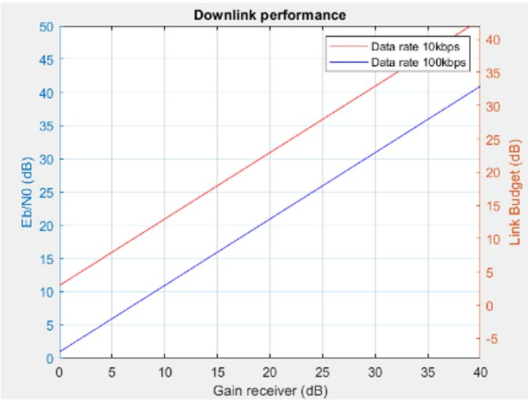


Figure 14: Trend of the gain and link budget depending on the data rate and Eb/N0.

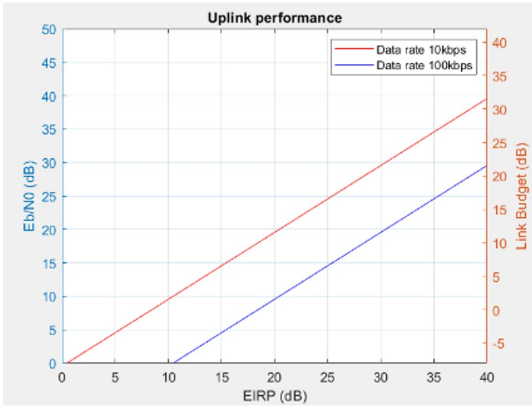


Figure 15: Trend of the EIRP depending on the data rate and Eb/N0.

For this analysis the ground contacts have been restricted to those with a minimum elevation of 10 degrees.

Table 17: Ground access preliminary analysis result.

Orbit	**Ground station**	**Mean number of contacts per day**	**Mean contact duration [min]**	**Max contact duration [min]**	**Max interval between contacts [hrs]**
500x500 SSO	Padova	3.5	5.9	7.7	13

The GS-Kit from Alén Space S.L. has been identified as a possible COTS solution. The GS is comprehensive of external structure and antennas with azimuth and elevation rotators, SDR rack, power rack and server. The GS will be acquired in the early stages of the next phase of the project.

For what concern the ground segment for Quantum Communication, the GS chosen is the University of Padova observatory located on top of the Department of Information Engineering (DEI). It is a German equatorial mount with 400 mm aperture equipped with an effective focal length of 3200 mm. The primary mirror has a diameter DRx of 410 mm and an obstruction, due to the secondary mirror, of diameter DObs=172 mm. The mirrors have an aluminum silicon dioxide (SiO_2) protective coating to guarantee high reflectivity in the band from 530 to 1600 nm. The GS was already tested for quantum-key distribution in a ground-to-ground link. Photos are shown in Appendix C. Taking the laser ranging measurements into consideration, the team will rely on the International Laser Ranging Service (ILRS). Our University has already got in touch with the Matera Laser Ranging Observatory (MLRO) in the framework of other research projects.

3 ASSEMBLY, INTEGRATION AND VERIFICATION

3.1 Model Philosophy

The model philosophy that will be followed during the CubeSat implementation includes three models, namely a Development Model (DM), an Engineering Model (EM) and a ProtoFlight Model (PFM). The team decided to follow this approach for two main reasons: (1) keep development costs low and (2) fast mission development. On one hand, the amount of the financial support that will be provided by the university and sponsors is still uncertain. So, the development costs must be kept low to ensure the full development of the mission. On the other hand, the mission development must be fast to make the mission ready to respond to upcoming launch opportunities. The DM+EM+PFM is used for all the system with the exception of the impact sensor. For this payload the approach DM+EQM+FM is employed. This is due to the fact that it is not a COTS component, hence a carefully development is required. For the impact sensor, the DM is used for functional test with a breadboard model of the sensor. Environmental tests (qualification level) will be performed on the EQM. The FM of the impact sensor will be submitted to environmental tests (acceptance level). Taking the whole system into consideration, a presentation of the models that will be developed is provided.

Development Model

A DM is developed for components and architecture test purposes. The DM is used for functional test of the components used in the CubeSat. Breadboards components are used in the DM, they have the same functionalities but not the same geometry of the ones used in the PFM. The DM consists of a FlatSat of the CubeSat. Since the early stages of the development, the FlatSat makes it possible to test the operation of the complete satellite while having the flexibility to swap out systems and try different configurations as well as develop and test user software. In this phase, the FlatSat will be made of breadboard models of the subsystems/payloads and a computer which will represent the OBC. In addition, the computer will be used to represent the communication with the GS. The tests of the breadboard models of the components on the FlatSat aim at verifying the preliminary budgets (data and power). Moreover, they will be used for preliminary assessment of the system functionalities. A preliminary version of the Flight Software will be tested using the DM. The results will be used to consolidate the design of the subsystems/payloads.

Engineering Model

The EM is used for system test purposes. The EM is a full-scale model made by COTS components representative in terms of functionalities, mass and envelope of the ones used in the ProtoFlight Model. These components will replace the DM of the subsystem/payloads on the FlatSat. The computer used in the development model will be replaced by a microprocessor representative of the OBC. The EM will be used to test the whole system and its functionalities. The FlatSat will be employed to test the Flight Software. The results will be useful to refine the system budgets and to finalize the design of the system.

Additionally, the EM will be used for testing the operations of the satellite. The operation tests shall be representative of all the mission phases. The FlatSat will communicate remotely with a Ground Station benchmark. These tests will be helpful to familiarize with the operations phase and for procedures and timelines assessment.

ProtoFlight Model

The PFM is subjected to a protoflight test campaign. After the functional tests of the components and subsystem/payloads, the system will be integrated and tested (PFM level).

The team will perform vibration tests (sinusoidal and random) and thermovacuum tests. After the test campaign the PFM will be launched. The team has decided to use the protoflight approach since the majority of the components of the CubeSat are COTS.

3.2 AIV Activities

The team is at the level of the Preliminary Design of the mission (Phase A). The AIV activities will be performed in the future phases of the project. In Phase B (Critical Design) the final design of the payloads and the components will be obtained. During this phase, the DM and the EM of the CubeSat will be developed.

The DM consists of the breadboard models of the components that constitute the satellite. These put together in a preliminary version of a FlatSat. Through the DM, the numerical simulations and the system budgets (e.g., data, link and power) will be verified and validated. The tests will start from component level to validate and verify the breadboard model. Then, the components will be joined together into a FlatSat to obtain the DM. This will permit to execute tests at system level. The tests that the team will execute will verify and validate:

- subsystems inter-communication
- preliminary flight software design
- subsystems and system functionalities
- system budgets (e.g., data, link and power)

The team plan to buy the components for the DM as soon as it is taken into the programme. This will make it possible to have the DM assembled in March and the tests results by the end of May. The output of these tests will be used to consolidate the design of the CubeSat. In addition, the tests will be employed for preliminary design validation and the verification the FlatSat.

After an intermediate design review, the DM will be upgraded to EM. The design of the FlatSat will be maintained but the breadboard models will be replaced with COTS components representative of the ones used in the PFM but not qualified for space flight. This development philosophy will permit a fast development of the EM since the FlatSat will be prepared for the upgrade. The major changes in this process are: (1) the computer will be replaced with a microprocessor representative of the one used in the CubeSat, (2) the FlatSat shall communicate remotely with a computer to simulate the communication with the GS. Through the tests executed on the EM the team will finalize the CubeSat design. The tests will aim at validating the design. System level tests will be executed for the functional qualification of the system (verification of electrical and software interfaces). The team aim at assembling the EM by the end of September 2023 in order to perform the test by the end of the year.

The TRL of the payload must be increased rapidly to close the delta-development with COTS subsystems. High priority will be given to the development of the payload to obtain the breadboarding models within February. The functional verification (TRL 4) will be obtained by the end of April in order to integrate the breadboard models in the DM. To reach a TRL of 5, the breadboards models will be tested in relevant environment: the impact sensor will be tested in vacuum using the hyper-velocity light-gas gun facility, the micro-vibrations sensor will be tested using a vibration testbed and the MRR and he CCR will be tested on an optical table. After the breadboards verification, the EQM/EM of the payloads will be produced/procured and tested in order to reach a TRL of 6. The team goal is to bring the TRL of the payloads to a level of 6 by the end of 2023. In the later stages of the project the payload will reach a TRL of 8 through the qualification tests.

3.3 Development Status Overview

Subsystem / element	Manufacturer (in-house/ COTS)	Model	At the Uni lab? (Y/N)	Status	Reference to test report(s)
Impact sensor	In-house	DM	Y	- Conceptual design - Prototyping	
		EQM	N		
		FM	N		
Micro-Vibration sensor	COTS	DM	N	- Conceptual design	
		EM	N		
		FM	N		
Corner Cube Reflector	COTS - Edmund Optics	DM	N	- Market analysis done - Conceptual design	
		EM	N		
		PFM	N		
Active Corner Cube Reflector	COTS - Boston Micromachines Corp.	DM	N	- Market analysis done - Conceptual design	
		EM	N		
		PFM	N		
EPS	COTS - AAC ClydeSpace/ DHV technoogy	EM	N	- Market analysis done - Preliminary study	
		PFM	N		
ADCS	COTS - CubeSpace	EM	N	- Market analysis done - Preliminary study	
		PFM	N		
TT&C	COTS- ISISpace	EM	N	- Market analysis done - Preliminary study	
		PFM	N		
OBC	COTS - Pumpkin	EM	N	- Market analysis done - Preliminary study	
		PFM	N		
Structure	COTS - ISISpace	EM	N	- Market analysis done - Preliminary study	
		PFM	N		
Ground Segment	COTS - Alén Space		N	- Market analysis done - Preliminary study	

3.4 Facilities and Ground Support Equipment

The University of Padova has several facilities that can be used for the development and testing of the CubeSat, see Table 18.

Table 18: Available facilities at the University of Padova. Figures representing the facility are reported in Appendix C.

Facility	Organization	Notes
Hyper-velocity light-gas gun	University of Padova	To test the impact sensor
Thermo-vacuum chamber	University of Padova	To test the components and the system
Vibration testbed	University of Padova	To test the components and the system To test the micro-vibration sensor
Optical testbed	University of Padova	To test MRR and CCR payloads
Clean room	University of Padova	For space payloads integration

The Quantum Future (QF) group has available seven state-of-the-art laboratories. Relevant to the project are the following tools: ground station and telescopes for free-space link tests and demonstrations including a 320 mm reflector and a 320 and 120 mm refractors; free-space and fiber optics components; several pulsed and CW diode lasers; a high power mode-locked Ti:Sa laser; a high power mode-locked Yb amplified laser system, several single photon detectors including SNSPD, SPAD with large active area (diameter of 200 micron); several time tagging units up to 1 ps temporal resolution; heralded photon sources by Spontaneous Parametric Down Conversion; FPGA controllers with programming and fast analog electronics; testing rig for fiber to photonics chip coupling and testing. Quantum Future has close collaboration with other Research Groups at the Department of Information Engineering, doing research on telecommunications, control system theory, photonics, microelectronics.

4 PROJECT ORGANISATION

4.1 Team Information

Team Leader

Name	Federico Basana
Job title or level of studies	PhD student
University	Università degli Studi di Padova
Department	Centre of Studies and Activities for Space “Giuseppe Colombo” (CISAS)

Endorsing Professor

Name	Alessandro Francesconi
Job title or level of studies	Professor
University	Università degli Studi di Padova
Department	Department of Industrial Engineering

System Engineer

Name	Federico Basana
Job title or level of studies	PhD student
University	Università degli Studi di Padova
Department	Centre of Studies and Activities for Space “Giuseppe Colombo” (CISAS)

List of Students

(1)	**Function** in the project (e.g. outreach, power subsystem)	(5)	**# of years** at University so far (e.g. BSc + MSc)
(2)	**Involved** in the CubeSat project **until** (MM YYYY) – expected date	(6)	**Field of study** (e.g. aerospace engineering) **& Specialisation** (e.g. Space Systems, Materials and Structures, etc.)
(3)	Preparing graduation **Thesis** within project (Yes/No)	(7)	**Nationality(s)** country code
(4)	Current **Level** of study (BSc/MSc/PhD)	(8)	**University** of enrolment

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)
Function	**Involved until**	**Thesis (if any)**	**Level**	**# of years**	**Field of study & specialisation**	**Nationality**	**University**
WP1100	Q3 2024	No	PhD	8	Aerospace engineering System engineering	IT	CISAS
WP1200	Q3 2025	Yes	BSc	4	Aerospace Engineering	IT	UNIPD
WP1200	Q3 2023	No	MSc	6	Aerospace Engineering	IT	UNIPD
WP1310	Q3 2024	No	MSc	4	Aerospace Engineering	IT	UNIPD
WP1310	Q4 2022	No	MSc	6	Mechanic Engineering	IT	UNIPD
WP1310	Q3 2023	No	MSc	5	Aerospace Engineering	IT	UNIPD
WP1310	Q4 2024	Yes	MSc	6	Aerospace Engineering	IT	UNIPD
WP1310	Q3 2023	No	MSc	6	Aerospace Engineering	IT	UNIPD
WP1310	Q3 2023	No	MSc	8	Aerospace Engineering	IT	UNIPD
WP1310	Q3 2024	Yes	BSc	3	Aerospace Engineering	IT	UNIPD
WP1320	Q4 2023	No	MSc	5	Aerospace Engineering	IT	UNIPD
WP1400 (CCR)	Q4 2024	No	PhD	8	Aerospace engineering System engineering	IT	CISAS
WP1400 (CCR)	Q3 2023	Yes	MSc	8	Aerospace engineering	IT	UNIPD
WP1400 (CCR)	Q4 2023	No	MSc		Aerospace engineering	IT	UNIPD
WP1400 (IS payload)	Q2 2023	No	MSc	5	Aerospace engineering	IT	UNIPD
WP1400 (IS payload)	Q4 2023	No	MSc	5	Aerospace engineering	IT	UNIPD
WP1400 (IS payload)	Q4 2023	No	MSc	5	Aerospace engineering	IT	UNIPD
WP1400 (IS payload)	Q3 2023	No	BCs	5	Electrical Engineering Integrated circuits	IT	UNIPD
WP1400 (micro-vibration payload)	Q3 2024	No	MSc	5	Aerospace Engineering	IT	UNIPD
WP1400 (micro-vibration payload)	Q3 2024	No	MSc	4	Aerospace Engineering	IT	UNIPD
WP1400 (MRR)	Q3 2023	No	PhD	10	Photonics Engineering Quantum Comunication	IT	UNIPD
WP1510 (AOCS)	Q2 2023	No	MSc	6	Aerospace engineering	IT/RO	UNIPD
WP1510 (AOCS)	Q2 2023	No	MSc	5	Aerospace engineering	IT	UNIPD

WP1510 (AOCS)	Q2 2023	No	MSc	5	Aerospace engineering	IT	UNIPD
WP1510 (AOCS)	Q2 2023	No	MSc	5	Aerospace engineering	IT	UNIPD
WP1510 (EPS)	Q1 2023	No	MSc	5	Energy and Nuclear engineering	IT	UNIPD
WP1510 (EPS)	Q4 2023	No	MSc	5	Aerospace Engineering	IT	UNIPD
WP1510 (EPS)	Q3 2023	No	MSc	5	Aerospace Engineering	IT	UNIPD
WP1510 (OBC)	Q3 2022	No	MCs	5	Aerospace engineering	IT	UNIPD
WP1510 (OBC)	Q3 2023	No	BCs	4	Computer engineering Aerospace engineering	IT	UNIPD
WP1510 (TT&C)	Q4 2024	No	MCs	5	Aerospace engineering	IT	UNIPD
WP1510 (TT&C)	Q4 2023	No	MCs	5	Aerospace engineering	IT	UNIPD
WP1510 (TT&C)	Q4 2023	No	MCs	4	Aerospace engineering	IT	UNIPD
WP1520	Q2 2024	No	BCs	4	Computer science	IT	UNIPD
WP1610	Q3 2023	No	MSc	5	Astrophysics and Cosmology	IT	UNIPD
WP1610	Q3 2023	No	MSc	5	Astrophysics and Cosmology	IT	UNIPD
WP1610	Q3 2024	No	MSc	4	Aerospace Engineering	IT	UNIPD

The Working Packages reported in the table are explained in detail in Sec. 4.3.2. The team will organize a recruiting to allocate new resources to the critical aspects of the project. For instance, the following students will be recruited: software engineer and computer scientists for the SW development, electrical engineers for assemble and integrate the development models into the FlatSat, information engineers for the TTC subsystem and for the GS development.

4.2 Organigram

The team is formed by a total of 37 students. They are Bachelor (5), Master (29) and PhD students (3). The team members have been divided in small workgroups; each one focuses on a subsystem of the satellite. The subsystem workgroups are led by experienced students (Master level). Together with the system engineer the workgroup leaders form the board of the project. Figure 16 shows the team organization.

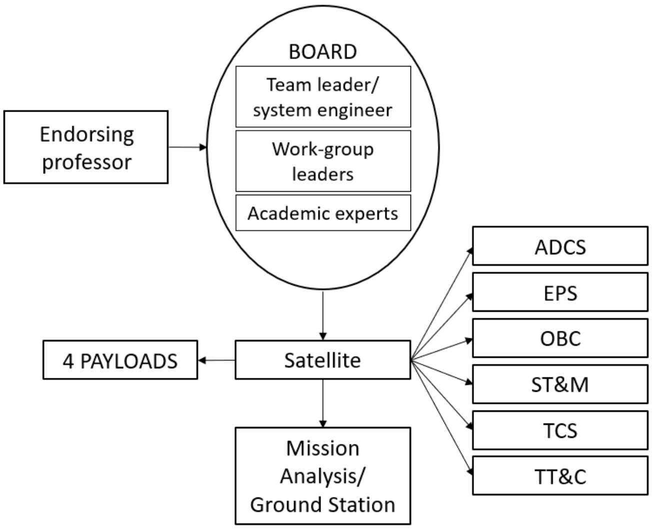


Figure 16: Scheme of the team organization.

The students dedicate approximately one afternoon per week to the project development. In addition, during weekly meetings, all the subsystem leaders expose the workgroup activities to the rest of the board. The team is supported by the professor Alessandro Francesconi who holds the course of Aerospace Systems and Satellite Attitude Control. A PhD student is the project manager and the system engineer. The project collaborates with a university project from the department of Information Engineering called Quantum Future which is responsible for the development of the quantum payload.

4.3 Project Schedule

4.3.1 Activities Performed

The project started its activities in December 2019. During its activities the team has followed the normal phase model used by ESA. A waterfall approach will be used where each phase end with a review milestone. The project phase and milestones are described in the following paragraphs. Figure 17 shows the GANTT chart of the activities performed.

Phase 0

The focus of this phase was the definition of the mission objectives. Starting from the analysis of the state of art, the team has found the technologies to use to achieve the mission goals and the gaps to be filled. The phase ended with the Mission Definition Review (MDR).

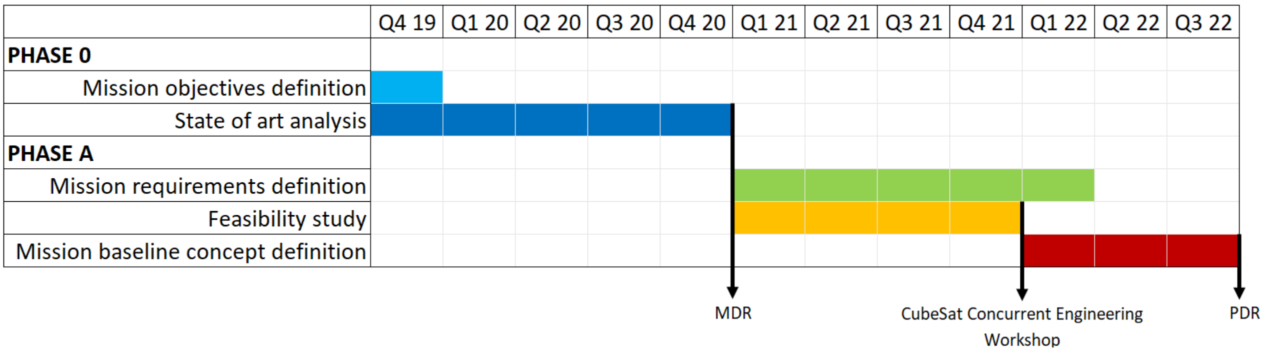


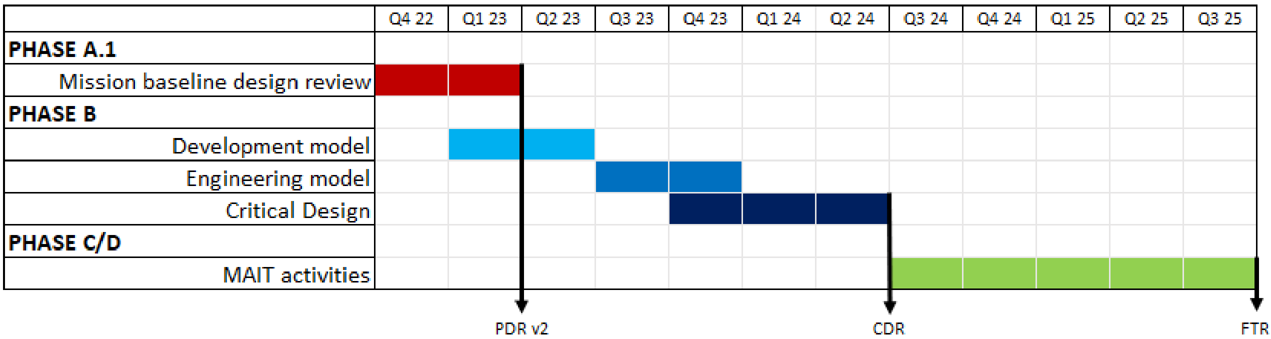
Figure 17: GANTT of the activities performed. The GANTT is organized in quarters.

Phase A

The Phase A objective is to define a baseline configuration of the satellite and the payloads, starting from the definition of requirements and the evaluation of the technical and programmatic feasibility of the project. During Phase A, the team participated in the CubeSat Concurrent Engineering Workshop. After the training week, the team reviewed the defined requirements and the analysis made and then started the mission baseline concept definition. Phase A ends with the Preliminary Design Review (PDR).

4.3.2 Planned Activities

The activities of the team will continue following the schedule proposed for the “Fly Your Satellite! - Design Booster” programme. Figure below shows the GANTT chart of the planned activities.



Phase A.1

The objective of this phase is to conduct a Baseline Design Review. The PDR of the mission will be reviewed in order to find issues in the design and to solve them. In this fashion, the team will enter the next phase of the project with a robust preliminary design. In this phase the components of to develop the DM of the CubeSat will be procured. This phase will end with an upgraded version of the Preliminary Design Review (PDR).

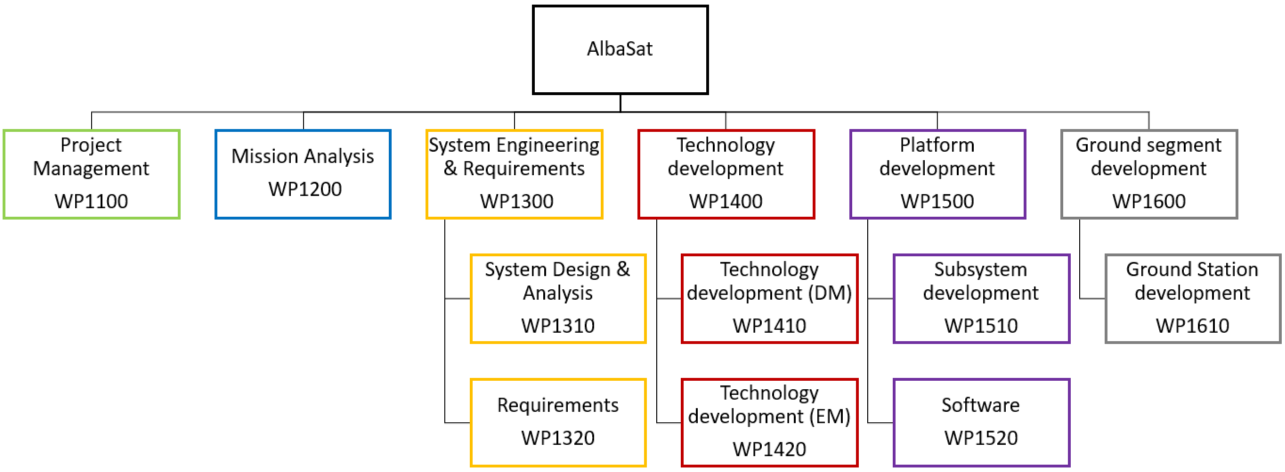
Phase B

Phase B will be based on the mission baseline configuration developed in the previous phase of the project. During this phase, the team will perform component level tests thanks to the FlatSat; the DM will be upgraded into the EM. The outcomes of the tests will be used for design validation and verification purposes. Phase B will end with the Critical Design Review (CDR).

Phase C/D

In Phase C/D the team will perform Manufacturing, Assembly, Integration and Testing (MAIT) activities. Phase C/D will end with the Functional Test Review (FTR).

A high-level Work Breakdown Structure is hereafter provided. The WBS of the project has been developed taking into account the need of clearly identifying the activity to be conducted and the entity assigned to each activity. The WBS will support project monitoring and control. The WPs explanation is provided in the tables below.



WP title	Project management		
ID	WP1100	**Duration**	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
○ WPs organization ○ Managing contacts with institutions (ESA, UNIPD) ○ Managing sponsors ○ Recruiting organization ○ Documentation			
Outputs			
Project management documentation, Critical Design Review (CDR)			

WP title	Mission analysis		
ID	WP1200	**Duration**	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
○ Ground Station coverage analysis ○ Communications link budget analysis ○ Risk Assessment ○ Debris mitigation guidelines			
Outputs			
Mission analysis report			

WP title	System Design and Analysis		
ID	WP1310	**Duration**	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
○ Budgets (mass, power, data, link, pointing, thermal) ○ CAD design of the satellite ○ Functional analysis ○ Thermal analysis ○ Structure analysis			
Outputs			
System budgets report, Thermal analysis report, structural analysis report			

WP title	Requirements		
ID	WP1320	**Duration**	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
○ Requirements tracking and verification			
Outputs			
System requirements report, subsystem requirements report			

WP title	Technology development (DM)		
ID	WP1410	**Duration**	5 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
This WP is divided in four WP, one for each payload. The goal of each WP is to reach a TRL of 4. ○ Preliminary design finalization ○ Drafting of test procedure and plan for payload ○ Payload breadboard manufacturing and assembly ○ Test campaign on payload breadboard			
Outputs			
Payloads Design Description Document, Payloads Breadboard			

WP title	Technology development (EM)		
ID	WP1420	**Duration**	8 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
This WP is divided in four WP, one for each payload. The goal of each WP is to reach a TRL of 6 by the end of 2023. ○ Critical design ○ Definition of test procedure and plan for payload ○ Payload manufacturing and assembly ○ Test campaign on payload			
Outputs			
Payloads Design Description Document, Payload critical design, Payloads EM (or EQM)			

WP title	Subsystem development		
ID	WP1510	**Duration**	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
This WP is divided for each subsystem of the satellite. ○ Trade-off analyses ○ Selection of subsystem modules ○ Functional analysis ○ Design of mechanical and electrical ground support equipment ○ FlatSat design, assembly and integration ○ Definition of test procedure and plan ○ Breadboarding models ○ DM test campaign ○ Engineering models ○ EM test campaign			
Outputs			
Subsystems design report, breadboarding models, Engineering Model			

WP title	Software		
ID	WP1510	**Duration**	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
○ Definition of on-board software requirements ○ Trade-off of on-board software architecture ○ Design of on-board software ○ Development of on-board software ○ On-board software validation ○ On-board software verification			
Outputs			
Software design report, OBSW			

WP title	Ground station development		
ID	WP1610	Duration	18 months
Inputs			
Mission objectives, Phase A documentation			
Tasks			
○ GS requirements definition ○ COTS GS selection ○ GS assembly and implementation ○ Definition of test procedure and plan ○ GS test			
Outputs			
Ground station report, Ground Station			

4.4 Cost Budget

A preliminary estimation of the cost of the mission is reported in the following table. The reported values are TBC and have been evaluated conservatively.

Subsystem	Cost [euros]
Payload 1: Impact Sensor	50.000
Payload 2: Micro-vibration sensor	5.000
Payload 3: (CCR)	10.000
Payload 4: Quantum future	30.000
AOCS	50.000
EPS	70.000
OBDH	10.000
OBSW	50.000
TT&C	20.000
Structures	5.000
Ground Segment	40.000
Environmental test facility rent	10.000
Clean room rent	5.000
Experts counsellors	36.000
Total	391.000
Total +20%	469.200

For the time being, a first search for sponsors has been initiated. Among others, we expect to receive contributions from the University of Padova, the Regione Veneto and to a variety of local industries that could be involved in the project.

4.5 External Parties

Quantum Future group is a research group based at the Department of Information Engineering, University of Padova. It was funded by Prof. Paolo Villoresi, who was the first to demonstrate the feasibility of Satellite Quantum Communication, in collaboration with the Matera Laser Ranging Observatory (MLRO) of the Italian Space Agency (ASI) and has twenty years experience in this field [26, 27]. Several of these experiments were conducted exploiting Corner Cube Retroreflectors to simulate an active source on a satellite [28, 29, 30, 31, 32].

The activities of the group are focused on Quantum Key Distribution (QKD) and Quantum Random Number Generation (QRNG). Latest results by Quantum Future includes several experimental demonstrations of practical QKD (both in free-space and fiber) [33, 34, 35, 36]. Quantum Future group also took part in several project related to Satellite Quantum Communication:

- CubeSat for QUANtum and 5G cOmmunication (QUANGO), an H2020 project that aims at developing a suitable breadboard platform for SQC (Project Manager: Unipd). [https://cordis.europa.eu/project/id/101004341/it]
- SeQBO, a project funded by the Italian Ministry of Defence, that aimed at the preparation of a setup to demonstrate the feasibility of satellite-to-ground QKD link (Project Manager: Argotec S.r.l.). [https://www.difesa.it/Amministrazionetrasparente/segredifesa/teledife/Documents/002_18Gen.pdf]
- PROtocols for Space sEcure Quantum cOmmunication (PROSEQO), an ESA's funded project that aimed a providing a model and a software for simulate satellite-to-ground links for QKD applications. (Project Manager: Unipd) [https://artes.esa.int/projects/proseqo]

QuantumFuture is also part of OpenQKD (https://openqkd.eu) and SAGA (https://www.esa.int/ESA_Multimedia/Images/2019/04/SAGA_for_quantum_key_distribution).

5 MAJOR CHALLENGES

The main technical challenge of the project is related to the development of the impact sensor, since all the other elements will be COTS. However, this development will not start from scratch, rather, it will be based on the strong background vested in our university in this field, with several impact sensor prototypes/breadboards already evaluated and calibrated in the CISAS hypervelocity impact laboratory (mostly in the frame of research contracts with ASI in support to the activities of the Inter-Agency Debris Coordination Committee).

Currently, the payloads have low TRL (Micro-vibration sensor, CCR and MRR have TRL 2 while impact sensor has TRL 3). The development of the payloads has to be speeded up in order to provide the information needed for the development of the satellite bus.

The management of the team is a key challenge. The team is mainly composed by Bachelor and Master students who have no practical expertise in developing a space mission. Even if the team members dedicated a lot to the project, the team has to be led in order to avoid delays due to the lack of experience and to the academic duties. For this reason, the activities of the team are scheduled in great advance taking into account the university exams sessions.

As Sec. 4.4 shows, the cost of the entire project represents a challenge for the student team. The team intends to cover the cost with both university funds and sponsors. We expect to receive contributions from a variety of local industries that could be involved in the project. The team will get in touch with these industries aiming at: (1) obtaining a sponsorship and (2) creating connections that will support the students in finding job opportunities.

Another critical challenge is represented by the OBSW implementation. The team has no expertise in this framework. Hence, a recruitment is scheduled to find experienced resources to be allocated on the WP. In addition, the team will ask for help to sector experts.

6 ACADEMIC RETURN

6.1 University experience and background

Since 2009, students from the University of Padova have been involved in several ESA hands-on activities (Rexus-Bexus, Drop Your Thesis, Fly Your Thesis). Four student projects participated in the Rexus/Bexus project, namely SCRAT, ARCADE, ARCADE R2, O-ZONE. In 2010, the project Spherical Compact Rechargeable Air Thruster (SCRAT) had the objectives to (1) develop a cold gas actuator and (2) test the actuator in a stratospheric flight. In 2011, in the framework of the project Autonomous Rendezvous Control And Docking Experiment (ARCADE), students developed and demonstrated the feasibility of a small scale docking system. In addition, they developed relative navigation sensors and attitude control systems for cooperative vehicles. In 2015, with the ARCADE -R2 project students developed a technology demonstrator that aimed to prove the feasibility of small-scale satellite and/or aircraft systems with automatic (a) attitude determination, (b) control and (c) docking capabilities. In 2021, students from our University studied atmospheric pollutants through the O-ZONE experiment.

The STAR team participated in the Drop Your Thesis! programme in 2016. The STAR team main objective was to design, build, and test an innovative tether deployment mechanism with retrieval capability. The system was conceived to be simple, compact, and light and thus be employable in small spacecraft platforms, such as CubeSats. The experiment was composed of a launch mechanism that deployed a tethered probe, a braking system which controlled the tether deployment, and a reeling mechanism that performed the tether retrieval once the deployment was complete.

In 2017, students from our University participated in the Fly Your Thesis! programme with the PACMAN experiment. It was a technology demonstrator whose main goal was to develop and validate in low-gravity conditions an integrated system for proximity navigation and soft docking based on magnetic interactions, suitable for small-scale spacecraft. This was accomplished by launching a miniature spacecraft mock-up towards a free-floating target that generates a static magnetic field; a set of actively-controlled magnetic coils on-board the spacecraft mock-up, assisted by dedicated localization sensors, were used to control its attitude and position relative to the target.

Currently, another team from University of Padova is participating in the Fly Your Thesis! programme. The ERMES team is designing and testing an autonomous docking maneuver between two free-floating CubeSat models. They completed the flight campaign in October 2022.

6.2 Educational and academic return of the project

The main purpose of the project is to develop a unique educational experience. The team was founded by students to support academic studies with hands-on activities. Since the early phases of the project, students have shown great interest in taking part in the team. During its activities, the Alba project engaged a great number of students (about 80) from several departments of the university, e.g., industrial engineering, information engineering, computer sciences, physics. In order to keep the number of members constant every year the team organizes a recruiting event to involve new students in the project.

The students involved in the project are strongly recommended to develop their thesis in the framework of the project. Table 19 reports the thesis developed in the framework of the Alba project. In addition, the participation in the workshop proposed by ESA is greatly recommended. Currently, team members have taken part in the following training week:

- Introduction to space law
- LadyBird Guide for Spacecraft Telecommunication Workshop
- Cubesat Concurrent Engineering Workshop
- CubeSat Hands-On Training Week
- Online Product Assurance Awareness Training Course
- Standardization Training Course
- CubeSat Summer School - livestream lessons

In addition, the team has presented an article in which the project is presented at the 4th Symposium on Space Educational Activities [37].

The lessons learned from the project have been used to support the Aerospace Systems university course. In addition, they will be used in the Space System Laboratory course that will be held starting from the next semester for Master students. During the laboratory course, hands-on activities similar to the one performed in the Alba project will be offered to the students. In this fashion, the students who attend the laboratory can cooperate with the Alba project. This will boost the activities of the team and will permit to increase the audience of the students that follow the Alba CubeSat team.

Table 19: Thesis developed in the framework of the Alba CubeSat project.

Name Surname	**Level**	**Thesis title**
Laura Callegari	BSc	Alba CubeSat: determinazione orbitale precisa attraverso Satellite Laser Ranging
Mattia Martello	BSc	Studio preliminare del sottosistema di controllo termico di un CubeSat 2U
Francesca Filippini	BSc	Progettazione strutturale preliminare di un CubeSat 2U
Mattia Sartor	BSc	Analisi dell’ambiente micro-vibrazionale su un CubeSat 2U
Enrico Bez	MSc	Sviluppo di un sensore per CubeSats per la rilevazione di impatti di piccoli detriti spaziali

6.3 Proposal Motivation

The “Fly Your Satellite! - design booster” programme represents a unique opportunity that will make our team grow. The team is majorly composed of BSc and MSc which have little experience on CubeSat mission development. During the activities, the team based the design choices on the literature and on what they learned in the academic courses. In addition, the team benefits tremendously from the workshops offered by ESA. The team believes that a dedicated programme such as “Fly Your Satellite!” will boost the design of the CubeSat.

The areas in which the team needs to improve its knowledge include the management of a CubeSat project, the familiarity with real CubeSat hardware and COTS components for nanosatellites, Flight Software implementation and the mission operations.

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APPENDICES

APPENDIX A: Technical Requirements Specification

Req ID	Requirement text
SYS-01	The system shall comply to the 2U standard CubeSat (CubeSat Design Specification Rev.13)
SYS-02	The project shall be developed according to tailored ECSS standards for CubeSat missions.
SYS-03	The shall accommodate the following subsystem: payloads, structure, EPS, OBDH, TT&C, AOCS, TCS
SYS-04	The system shall withstand the launcher induced mechanical environment (static, low and high frequency) during ground and launch.
SYS-05	The system shall be compatible with the orbital thermal environment during all phases of the mission
SYS-06	Single-point failures with catastrophic consequences in case of failure shall be prevented by design.
SYS-07	No single command function executed at the wrong time or in the wrong configuration shall lead to the loss of the mission
SYS-08	The system shall have a safe mode that is power positive, thermally safe and allows two-way communication with ground
SYS-09	The system shall be able to enter safe mode autonomously i.e., without ground intervention
AOCS-01	The attitude control shall be guaranteed throughout all the mission phases
AOCS-02	The attitude rate control shall be guaranteed throughout all the mission phases
AOCS-03	The attitude shall be evaluated throughout all the mission phases
AOCS-04	The attitude rate shall be evaluated throughout the mission phases
AOCS-05	The AOCS shall guarantee the transition from one mission mode to another
AOCS-06	The AOCS shall guarantee the satellite survival after reconfiguration (due to failure or anomaly) and after launcher separation
AOCS-07	The satellite shall be detumbled in maximum 48 hrs (TBC)
AOCS-08	The pointing accuracy of the satellite shall be smaller than 20 deg per axis during the MRR payload and communication mode (TBC)
AOCS-09	The pointing accuracy of the satellite shall be smaller than 5 deg w.r.t. the tangential velocity direction, during the Impact Sensor payload operation (TBC)
AOCS-10	The attitude of the satellite shall be estimated with an error lower than 1 deg. (TBC)
AOCS-11	The AOCS shall guarantee an attitude rate of the satellite lower than TBD rad/s in a TBD time
AOCS-12	The AOCS actuators shall be placed at a distance from the centre of mass of the Satellite lower than TBD.
AOCS-13	The AOCS shall be able to communicate with the OBC throughout all the mission phases
OBDH-01	The OBDH shall be able to store at least 2GB of data (TBC)
OBDH-02	The OBDH shall be able to code data ensuring data integrity
OBDH-03	The OBDH shall be able to elaborate data received from the ground

Req ID	Requirement text
OBDH-04	The OBDH shall be able to collect data from other subsystems every TBD s
OBDH-05	The OBDH shall register actions into log
OBDH-06	The OBDH shall communicate with the subsystems
MIS-01	The nominal operations minimum duration shall last at least 1 year (TBC)
MIS-02	Altitude shall comply with the ESA space safety regulations (United Nations Inter-Agency Space Debris Coordination Committee (IADC), European Code of Conduct on Space Debris Mitigation)
MIS-03	Probability of fatal impact shall comply with the ESA space safety regulations (United Nations Inter-Agency Space Debris Coordination Committee (IADC), European Code of Conduct on Space Debris Mitigation)
MIS-04	The minimum inclination shall be 45.4 deg (TBC)
MIS-05	The end-of-life strategy shall be passive de-orbiting
MIS-06	The time of eclipse shall be less than TBD s per orbit
TTC-01	The TTC system shall be able to have a closed link with the ground station throughout all the communication phase
TTC-02	The TTC system shall be able to transmit data stored on board to the GS
TTC-03	The TTC shall be able to relay periodically essential housekeeping information as a beacon message
TTC-04	The data shall be transferred to the ground station at a velocity of minimum 9600 bps (TBC)
TTC-05	The frequency of the downlink shall be between 435 (TBC) and 438 (TBC) MHz
TTC-06	The frequency of the uplink shall be between 145.8 (TBC) and 146 (TBC) MHz
TCS-01	The temperature of the Temperature Reference Point (TRP) shall be known with an accuracy of ± (TBD)°C
TCS-02	The temperature of the TRP shall be kept between -10°C (TBC) and +50°C (TBC)
TCS-03	The TCS shall ensure that all subsystems are maintained in their operating temperature ranges
TCS-04	The position of the TRP shall be TBD mm from the geometric centre of the CubeSat
TCS-05	The thermal control shall be guaranteed throughout all the mission phases
EPS-01	The EPS shall power the system throughout all the mission phases
EPS-02	The EPS shall store electrical power
EPS-03	The electrical power shall be provided at the voltage of 3.3 V, 5V, 12V (TBC)
EPS-04	The battery shall be charged before the launch operations
EPS-05	The battery capacity shall be of 30 Wh minimum (TBC)
EPS-06	The electrical power generated shall be distributed through the PCDU
IS-01	The Impact Sensor (IS) shall execute in-situ measurements of impacts from particles of sub-millimetric dimensions
IS-02	The Impact Sensor shall be placed along the direction of the satellite motion
IS-03	The Impact Sensor shall have a power consumption of maximum 0.2 W (TBC)
IS-04	The Impact Sensor shall survive in the space environment
IS-05	The Impact Sensor shall survive the launch loads
IS-06	The sampling rate shall be equal to 1 Hz (TBC)
IS-07	Data generated by the Impact Sensor shall be memorized on board

Req ID	Requirement text
IS-08	The Impact Sensor shall be able to identify false positives
CCR-01	The CCRs shall return to the laser ranging ground station at least the TBD% of the shined laser beam
CCR-02	The CCRs shall have a FOV equal to TBD deg
CCR-03	The structure of the CCRs shall guarantee the structural integrity throughout all the mission phases
MRR-01	The laser GS shall shine a laser beam of power of at least 1 W (TBC)
MRR-02	The laser beam shall have a wavelength of 1550 nm (TBC)
MRR-03	The MRR shall have 0.0254 m (TBC) of diameter
MRR-04	The laser GS shall have 0.4 m (TBC) diameter
MRR-05	The MRR shall be modulated at 100 Hz (TBC)
MRR-06	The MRR shall have a FOV equal to TBD deg
MVS-01	The micro-vibration sensor shall measure vibrations between 1 Hz (TBC) and 1500 Hz (TBC)
MVS-02	The micro-vibration sensor shall generate TBD kB of data
MVS-03	The micro-vibration sensor shall have a sample rate of 5 kHz (TBC)
MVS-04	The micro-vibration sensor acquisition time shall be of 10 s (TBC)

APPENDIX B: CubeSat Design Specification – Compliance Matrix

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
General requirements	3.1.1	CubeSats which incorporate any deviation from the CDS will submit a DAR and adhere to the waiver process (see Section 1.3 and Appendix A).	NA	Design Booster teams will follow ESA Education waiver process
	3.1.2	All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created.	C	
	3.1.3	No pyrotechnics shall be permitted.	C	
	3.1.4	Any propulsion systems shall be designed, integrated, and tested in accordance with AFSPCMAN 91-710 Volume 3.	NA	No propulsion is requested
	3.1.5	Propulsion systems shall have at least 3 inhibits to activation.	NA	No propulsion is requested
	3.1.6	Total stored chemical energy will not exceed 100 Watt-Hours.	NA	No propulsion is requested
	3.1.7	CubeSat hazardous materials shall conform to AFSPCMAN 91-710, Volume 3.	C	
	3.1.8	CubeSat materials shall satisfy the following low out-gassing criterion to prevent contamination of other spacecraft during integration, testing, and launch. A list of NASA approved low out-gassing materials can be found at: http://outgassing.nasa.gov 3.1.8.1 CubeSat materials shall have a Total Mass Loss (TML) < 1.0 % 3.1.8.2 CubeSat materials shall have a Collected Volatile Condensable Material (CVCM) < 0.1%	C	

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.1.9	The latest revision of the CubeSat Design Specification will be the official version which all CubeSat developers will adhere to. The latest revision is available at http://www.cubesat.org. 3.1.9.1 Cal Poly will send updates to the CubeSat mailing list upon any changes to the specification. You can sign-up for the CubeSat mailing list here: www.cubesat.org/index.php/about-us/how-to-join	NA	Student teams must report compliance to the CubeSat Design Specification rev 13, which will be used as the basis for selection
	3.1.11	The CubeSat shall be designed to accommodate ascent venting per ventable volume/area < 2000 inches.	C	
CubeSat Mechanical Requirements	3.2.1	The CubeSat shall use the coordinate system as defined in Appendix B for the appropriate size. The CubeSat coordinate system will match the P-POD coordinate system while integrated into the P-POD. The origin of the CubeSat coordinate system is located at the geometric center of the CubeSat.	C	
	3.2.1.1	The CubeSat configuration and physical dimensions shall be per the appropriate section of Appendix B.	C	
	3.2.1.2	The extra volume available for 3U+ CubeSats is shown in Figure 6.	NA	The CubeSat form factor is 2U
	3.2.2	The –Z face of the CubeSat will be inserted first into the P-POD.	C	
	3.2.3	No components on the green and yellow shaded sides shall exceed 6.5 mm normal to the surface.	C	
	3.2.3.1	When completing a CubeSat Acceptance Checklist (CAC), protrusions will be measured from the plane of the rails.	NA	Design Booster teams will follow FYS Interfaces Verification Procedure

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.2.4	Deployables shall be constrained by the CubeSat, not the P-POD.	C	
	3.2.5	Rails shall have a minimum width of 8.5mm.	C	
	3.2.6	Rails will have a surface roughness less than 1.6 μm.	C	
	3.2.7	The edges of the rails will be rounded to a radius of at least 1 mm	C	
	3.2.8	The ends of the rails on the +/- Z face shall have a minimum surface area of 6.5 mm x 6.5 mm contact area for neighboring CubeSat rails (as per Figure 6).	C	
	3.2.9	At least 75% of the rail will be in contact with the P-POD rails. 25% of the rails may be recessed and no part of the rails will exceed the specification.	C	
	3.2.10	The maximum mass of a 1U CubeSat shall be 1.33 kg	NA	The CubeSat form factor is 2U
	3.2.11	The maximum mass of a 1.5U CubeSat shall be 2.00 kg.	NA	1.5 U form factor not included in the scope of Design Booster
	3.2.12	The maximum mass of a 2U CubeSat shall be 2.66 kg.	C	
	3.2.13	The maximum mass of a 3U CubeSat shall be 4.00 kg.	NA	The CubeSat form factor is 2U
	3.2.14	The CubeSat center of gravity shall be located within 2 cm from its geometric center in the X and Y direction.	C	
	3.2.14.1	The 1U CubeSat center of gravity shall be located within 2 cm from its geometric center in the Z direction.	NA	The CubeSat form factor is 2U
	3.2.14.2	The 1.5U CubeSat center of gravity shall be located within 3 cm from its geometric center in the Z direction.	NA	1.5 U form factor not included in the scope of Design Booster

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.2.14.3	The 2U CubeSat center of gravity shall be located within 4.5 cm from its geometric center in the Z direction.	C	
	3.2.14.4	3U and 3U+ CubeSats' center of gravity shall be located within 7 cm from its geometric center in the Z direction.	NA	The CubeSat form factor is 2U
	3.2.15	Aluminum 7075, 6061, 5005, and/or 5052 will be used for both the main CubeSat structure and the rails.	C	
	3.2.16	The CubeSat rails and standoff, which contact the P-POD rails and adjacent CubeSat standoffs, shall be hard anodized aluminum to prevent any cold welding within the PPOD.	C	
	3.2.17	The 1U, 1.5U, and 2U CubeSats shall use separation springs to ensure adequate separation.	C	
	3.2.17.2	The compressed separation springs shall be at or below the level of the standoff	C	
	3.2.17.3	The 1U, 1.5U, and 2U CubeSat separation spring will be centered on the end of the standoff on the CubeSat's –Z face as per Figure 7.	C	
	3.2.17.4	Separation springs are not required for 3U CubeSats.	NA	The CubeSat form factor is 2U
Electrical Requirements	3.3.1	The CubeSat power system shall be at a power off state to prevent CubeSat from activating any powered functions while integrated in the P-POD from the time of delivery to the LV through on-orbit deployment. CubeSat powered function include the variety of subsystems such as Command and Data Handling (C&DH), RF Communication, Attitude Determine and Control (ADC), deployable mechanism actuation. CubeSat power	C	

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
		systems include all battery assemblies, solar cells, and coin cell batteries.		
	3.3.2	The CubeSat shall have, at a minimum, one deployment switch on a rail standoff, per Figure 7.	C	
	3.3.3	In the actuated state, the CubeSat deployment switch shall electrically disconnect the power system from the powered functions; this includes real time clocks (RTC).	C	
	3.3.4	The deployment switch shall be in the actuated state at all times while integrated in the PPOD.	C	
	3.3.4.1	In the actuated state, the CubeSat deployment switch will be at or below the level of the standoff.	C	
	3.3.5	If the CubeSat deployment switch toggles from the actuated state and back, the transmission and deployable timers shall reset to t=0.	C	
	3.3.6	The RBF pin and all CubeSat umbilical connectors shall be within the designated Access Port locations, green shaded areas shown in Appendix B.	C	
	3.3.7	The CubeSat shall include an RBF pin.	C	
	3.3.7.1	The RBF pin shall cut all power to the satellite once it is inserted into the satellite.	C	
	3.3.7.2	The RBF pin shall be removed from the CubeSat after integration into the P-POD	C	
	3.3.7.3	The RBF pin shall protrude no more than 6.5 mm from the rails when it is fully inserted into the satellite.	C	
	3.3.8	CubeSats shall incorporate battery circuit protection for charging/discharging to avoid unbalanced cell conditions	C	

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.3.9	The CubeSat shall be designed to meet at least one of the following requirements to prohibit inadvertent radio frequency (RF) transmission. The use of three independent inhibits is highly recommended and can reduce required documentation and analysis. An inhibit is a physical device between a power source and a hazard. A timer is not considered an independent inhibit. 3.3.9.1 The CubeSat will have one RF inhibit and RF power output of no greater than 1.5W at the transmitting antenna's RF input. 3.3.9.2 The CubeSat will have two independent RF inhibits	C	
Operational Requirements	3.4.1	Operators will obtain and provide documentation of proper licenses for use of radio frequencies.	C	
	3.4.1.1	For amateur frequency use, this requires proof of frequency coordination by the International Amateur Radio Union (IARU). Applications can be found at www.iaru.org.	C	
	3.4.2	CubeSats will comply with their country's radio license agreements and restrictions.	C	
	3.4.3	CubeSats mission design and hardware shall be in accordance with NPR 8715.6 to limit orbital debris. 3.4.3.1 Any CubeSat component shall re-enter with energy less than 15 Joules. 3.4.3.2 Developers will obtain and provide documentation of approval of an orbital debris mitigation plan from the FCC (or NOAA if imager is present).	NA	Teams in Design Booster shall comply to ESA's and their national space rules on space debris mitigation. This will be superseded by an FDS requirement.

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.4.4	All deployables such as booms, antennas, and solar panels shall wait to deploy a minimum of 30 minutes after the CubeSat's deployment switch(es) are activated from PPOD ejection.	C	
	3.4.5	No CubeSats shall generate or transmit any signal from the time of integration into the PPOD through 45 minutes after on-orbit deployment from the P-POD. However, the CubeSat can be powered on following deployment form the P-POD.	C	
	3.4.6	Private entities (non-U.S. Government) under the jurisdiction or control of the United States who propose to operate a remote sensing space system (satellite) may need to have a license as required by U.S. law. For more information visit http://www.nesdis.noaa.gov/CRSRA/licenseHome.html. Click on the Application Process link under the Applying for a License drop down section to begin the process.	NA	Teams are responsible to achieve compliance with applicable international and domestic law & regulations. This will be superseded by an FDS requirement.
	3.4.7	Cal Poly will conduct a minimum of one fit check in which developer hardware will be inspected and integrated into the P-POD or TestPOD. A final fit check will be conducted prior to launch. The CubeSat Acceptance Checklist (CAC) will be used to verify compliance of the specification (Found in the appendix of this document or online at http://cubesat.org/index.php/documents/developers).	NA	If applicable, fit-checks will be conducted according to the FYS Interfaces Verification Procedure
Testing Requirements	4.1	Random Vibration Random vibration testing shall be performed as defined by the launch provider	NA	Random Vibration requirements will be

CubeSat Design Specification Rev.13 (for 1U, 2U and 3U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
				given by the FDS specification
	4.2	Thermal Vacuum Bakeout Thermal vacuum bakeout shall be performed to ensure proper outgassing of components. The test specification will be outlined by the launch provider.	NA	Thermal vacuum bakeout requirements will be given by the FDS specification
	4.3	Shock Testing Shock testing shall be performed as defined by the launch provider.	NA	Shock testing requirements will be given by the FDS specification
	4.4	Visual Inspection Visual inspection of the CubeSat and measurement of critical areas will be performed per the appropriate CAC (Appendix C).	NA	Design Booster teams will follow FYS Inspection Procedure
	4.5	CubeSat Testing Philosophy	NA	Testing philosophy requirements will be given by the FDS specification

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/NA	**(5) Comments**
General Requirements	3.1.1	All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris will be created.	NA	The CubeSat form factor is 2U
	3.1.2	No pyrotechnics shall be permitted.	NA	The CubeSat form factor is 2U
	3.1.3	Any propulsion systems shall be designed, integrated, and tested in accordance with AFSPCMAN 91-710 Volume 3.	NA	The CubeSat form factor is 2U
	3.1.4	Propulsion systems shall have at least 3 inhibits to activation.	NA	The CubeSat form factor is 2U
	3.1.5	Total stored chemical energy will not exceed 100 Watt-Hours.	NA	The CubeSat form factor is 2U
	3.1.6	CubeSat hazardous materials shall conform to AFSPCMAN 91-710, Volume 3.	NA	The CubeSat form factor is 2U
	3.1.7	CubeSat materials shall satisfy the following low out-gassing criterion to prevent contamination of other spacecraft during integration, testing, and launch. A list of NASA approved low out-gassing materials can be found at: http://outgassing.nasa.gov 3.1.7.1 CubeSat materials shall have a Total Mass Loss (TML) < 1.0 % 3.1.7.2 CubeSat materials shall have a Collected Volatile Condensable Material (CVCM) < 0.1%	NA	The CubeSat form factor is 2U
	3.1.9	The CubeSat shall be designed to accommodate ascent venting per ventable volume/area < 2000 inches.	NA	The CubeSat form factor is 2U

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
CubeSat Mechanical Requirements	3.2.1	The CubeSat shall use the coordinate system as defined in Appendix B. The origin of the CubeSat coordinate system is located at the geometric center of the CubeSat.		
	3.2.1.1	The 6U CubeSat configuration and physical dimensions shall be per the appropriate section of Appendix B.	NA	The CubeSat form factor is 2U
	3.2.2	The –Z face of the CubeSat will be inserted first into the 6U Dispenser.	NA	The CubeSat form factor is 2U
	3.2.3	No components on the yellow shaded faces shall exceed 10 mm normal to the surface.	NA	The CubeSat form factor is 2U
	3.2.3.1	When completing a CubeSat Acceptance Checklist (CAC), protrusions will be measured from the plane of the rails.	NA	Design Booster teams will follow FYS Interfaces Verification Procedure
	3.2.4	Deployables shall be constrained by the CubeSat, not the 6U Dispenser.	NA	The CubeSat form factor is 2U
	3.2.5	Rails shall have a minimum width of 8.5mm.	NA	The CubeSat form factor is 2U
	3.2.6	Rails will have a surface roughness less than 1.6 μm.	NA	The CubeSat form factor is 2U
	3.2.7	The edges of the rails will be rounded to a radius of at least 1 mm	NA	The CubeSat form factor is 2U
	3.2.8	At least 75% of the rail will be in contact with the 6U Dispenser rails. 25% of the rails may be recessed, and no part of the rails will exceed the specification.	NA	The CubeSat form factor is 2U

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.2.9	The maximum mass of a 6U CubeSat shall be 12.00 kg.	NA	The CubeSat form factor is 2U
	3.2.10	The CubeSat center of gravity shall be located within 4.5 cm from its geometric center in the X direction, within 2 cm from its geometric center in the Y direction, and within 7 cm from its geometric center in the Z direction	NA	The CubeSat form factor is 2U
	3.2.11	Typically, Aluminum 7075, 6061, 6082, 5005, and/or 5052 are used for both the main CubeSat structure and the rails. If materials other than aluminum are used, the CubeSat Developer should contact the Mission Integrator or dispenser manufacturer.	NA	The CubeSat form factor is 2U
	3.2.12	The CubeSat rails and standoff, which contact the 6U Dispenser rails and ejector plate, shall be hard anodized aluminum to prevent any cold welding within the 6U Dispenser.	NA	The CubeSat form factor is 2U
Electrical Requirements	3.3.1	The CubeSat power system shall be at a power off state to prevent CubeSat from activating any powered functions while integrated in the P-POD from the time of delivery to the LV through on-orbit deployment. CubeSat powered function include the variety of subsystems such as Command and Data Handling (C&DH), RF Communication, Attitude Determine and Control (ADC), deployable mechanism actuation. CubeSat power systems include all battery assemblies, solar cells, and coin cell batteries.	NA	The CubeSat form factor is 2U

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.3.1.1	Powered-On Real Time Clocks (RTC) may be permitted, if they satisfy the following requirements. 3.3.3.1.1 Real time clock circuits shall be isolated from the CubeSat's main power system. 3.3.3.1.2 Real time clock frequencies shall be less than 320 kHz. 3.3.3.1.3 Real time clock circuit(s) shall be current limited to less than 10 mA.	NA	The CubeSat form factor is 2U
	3.3.2	The CubeSat shall have, at a minimum, one deployment switch, which is actuated while integrated in the 6U Dispenser.	NA	The CubeSat form factor is 2U
	3.3.2.1	In the actuated state, the CubeSat deployment switch shall electrically disconnect the power system from powered functions.	NA	The CubeSat form factor is 2U
	3.3.2.2	The deployment switch shall be in the actuated state at all times while integrated in the 6U Dispenser.	NA	The CubeSat form factor is 2U
	3.3.2.3	If the CubeSat deployment switch toggles from the actuated state and back, the satellite shall reset to a pre-launch state, including reset of transmission and deployable timers.	NA	The CubeSat form factor is 2U
	3.3.3	The CubeSat shall include a RBF pin. The RBF pin shall cut all power to the satellite once it is inserted into the satellite.	NA	The CubeSat form factor is 2U
	3.3.3.1	The RBF pin shall protrude no more than 10 mm from the rails (or within the allowable CubeSat volume on tabbed dispensers, see planetarysystemscorp.com for details) when fully inserted into the satellite.	NA	The CubeSat form factor is 2U

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/ NA	**(5) Comments**
	3.3.3.2	Access to the CubeSat is not guaranteed during or after integration. The RBF pin shall be removed from the CubeSat before integration into the 6U Dispenser, if the dispenser does not have access ports.	NA	The CubeSat form factor is 2U
	3.3.4	CubeSats shall incorporate battery circuit protection for charging/discharging to avoid unbalanced cell conditions. Additional manufacturer documentation and/or testing will be required for modified, customized, or non-UL-listed cells.	NA	The CubeSat form factor is 2U
	3.3.4.1	In the actuated state, the CubeSat deployment switch will be at or below the level of the standoff.	NA	The CubeSat form factor is 2U
	3.3.5	The CubeSat shall have at least three independent RF inhibits to prohibit inadvertent radio frequency (RF) transmission. An inhibit is a physical device between a power source and a hazard. A timer is not considered an independent inhibit.	NA	The CubeSat form factor is 2U
Operational Requirements	3.4.1	Operators will obtain and provide documentation of proper licenses for use of radio frequencies.	NA	The CubeSat form factor is 2U
	3.4.1.1	For amateur frequency use, this requires proof of frequency coordination by the International Amateur Radio Union (IARU). Applications can be found at www.iaru.org.	NA	The CubeSat form factor is 2U
	3.4.2	The CubeSat will comply with their country's radio license agreements and restrictions.	NA	The CubeSat form factor is 2U
	3.4.3	CubeSats mission design and hardware shall be in accordance with NPR 8715.6 to limit orbital debris.	NA	Teams in Design Booster shall comply to ESA's and their

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	(2) Requirement ID	(3) Requirement Text	(4) Expected Compliance C/NC/U/ NA	(5) Comments
		3.4.3.1 Any CubeSat component shall re-enter with energy less than 15 Joules. 3.4.3.2 Developers will be ready to provide orbital debris mitigation data if requested by the licensing agency or Mission Integrator.		national space rules on space debris mitigation. This will be superseded by an FDS requirement.
	3.4.4	All deployables such as booms, antennas, and solar panels shall wait to deploy a minimum of 30 minutes after the CubeSat's deployment switch(es) are activated during6U Dispenser ejection.	NA	The CubeSat form factor is 2U
	3.4.5	No CubeSats shall generate or transmit any RF signal from the time of integration into the 6U Dispenser through 45 minutes after on-orbit deployment from the 6U Dispenser. However, the CubeSat can be powered on following deployment from the 6U Dispenser.	NA	The CubeSat form factor is 2U
	3.4.6	Private entities (non-U.S. Government) under the jurisdiction or control of the United States who propose to operate a remote sensing space system (satellite) may need to have a license as required by U.S. law. For more information visit http://www.nesdis.noaa.gov/CRSRA/licenseHome.html. Click on the Application Process link under the Applying for a License drop down section to begin the process.	NA	Teams are responsible to achieve compliance with applicable international and domestic law & regulations. This will be superseded by an FDS requirement.
	3.4.7	The Mission Integrator may conduct a fit check in which the CubeSat Developer hardware will be inspected and integrated into the 6U Dispenser or Test 6U Dispenser. A final fit check will be conducted prior to launch. The	NA	If applicable, fit-checks will be conducted according

6U CubeSat Design Specification Rev.1 (for 6U CubeSats)				
(1) CDS Section	**(2) Requirement ID**	**(3) Requirement Text**	**(4) Expected Compliance** C/NC/U/NA	**(5) Comments**
		CubeSat Acceptance Checklist (CAC) can be used to verify compliance of the specification (CAC can be found in the appendix of this document or online at http://cubesat.org).		to the FYS Interfaces Verification Procedure
Testing Requirements	4.1	Random Vibration Random vibration testing shall be performed as defined by the launch provider	NA	Random Vibration requirements will be given by the FDS specification
	4.2	Thermal Vacuum Bakeout Thermal vacuum bakeout shall be performed to ensure proper outgassing of components. The test specification will be outlined by the launch provider.	NA	Thermal vacuum bakeout requirements will be given by the FDS specification
	4.3	Shock Testing Shock testing shall be performed as defined by the launch provider.	NA	Shock testing requirements will be given by the FDS specification
	4.4	Visual Inspection Visual inspection of the CubeSat and measurement of critical areas will be performed per the appropriate CAC (Appendix C).	NA	Design Booster teams will follow FYS Inspection Procedure
	4.5	CubeSat Testing Philosophy	NA	Testing philosophy requirements will be given by the FDS specification

APPENDIX C: Pictures

Figure 18**:** In these photos are shown on the Galyqeye: on the left we have the german mount, on the right the external dome on top of the building DEI/G, part of the Department of Information Engineering at University of Padova.

Figure 19: In these photos, the MRR realized by Boston Micromachines (prototype) is shown inside a cylindrical structure used for the exposition.

Figure 20: Hyper-velocity light-gas gun.

Figure 21: Vibration testbed at the University of Padova

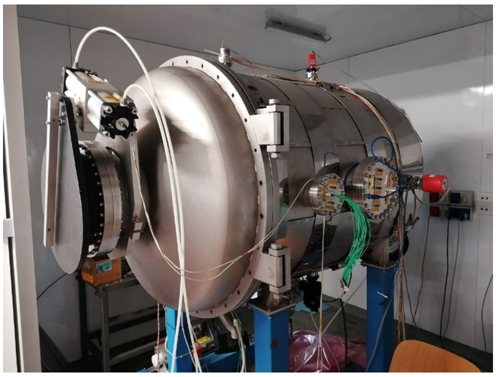
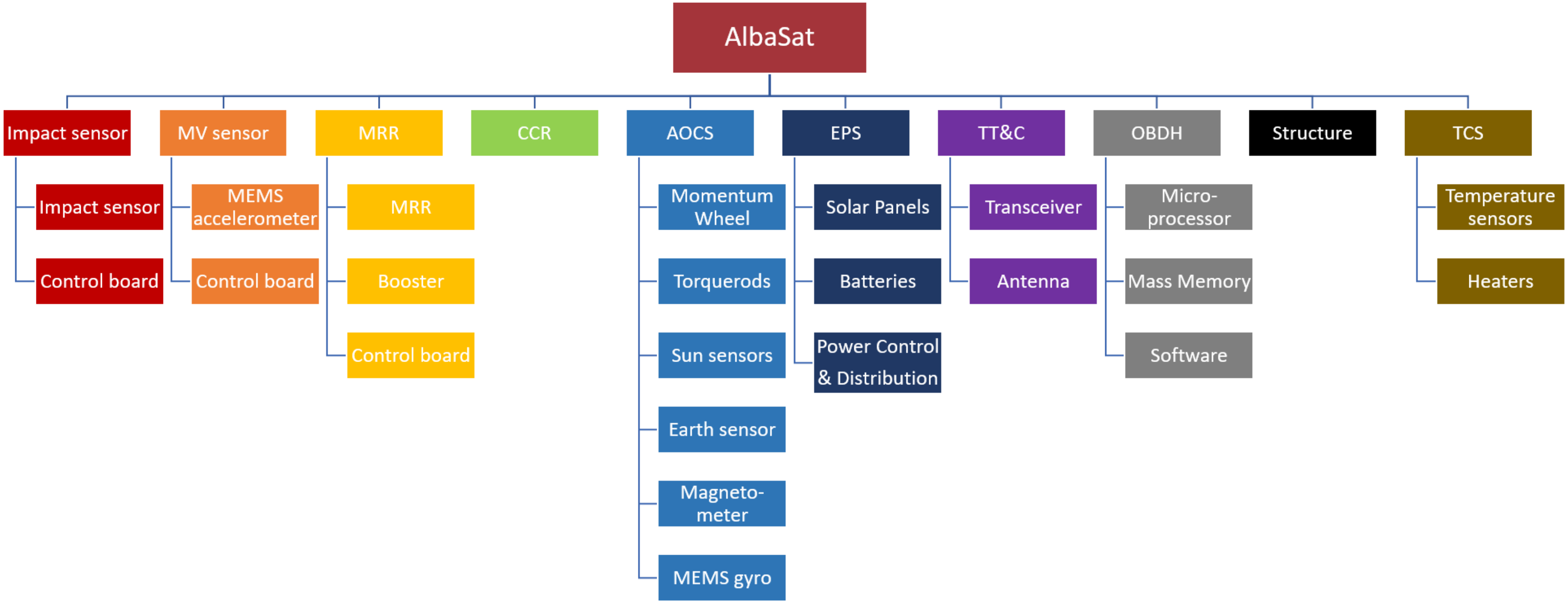


Figure 22: Thermo-vacuum chamber

APPENDIX D: Product breakdown tree



Subsystem	Component	Commercial Name	Seller	Nationality	TRL level
ADCS	Integrated ADCS	CubeADCS Y-momentum	CubeSpace	South Africa	9
EPS	PCDU	Starbuck-nano	AAC Clyde Space	Sweden	9
	Battery	Optimus-30	AAC Clyde Space	Sweden	9
	Solar Arrays	DHV-CS-10	DHV Technology	Spain	9
TT&C	Antenna	ISIS VHF/UHF deployable antenna system	ISISPACE	The Netherlands	9
	Transceiver	ISIS VHF/UHF duplex transceiver	ISISPACE	The Netherlands	9
OBDH	On-Board Computer	MotherBoard Module 2	Pumpkin	USA (California)	
		Beagle Bone Black	Texas Instruments	USA (Texas)	7
Structure	Structure	ISISPACE 2-Unit CubeSat structure	ISISPACE	The Netherlands	9
CCR	Corner Cube Retroreflector	Corner Cube Retroreflector	EdmundOptics	USA (New Jersey)	9
MRR	Modulating Retro Reflector	Modulating Retro Reflector	Boston Micromachines	USA (Massachusetts)	4
Impact sensor	Impact sensor	/	/	Italy	3
MV sensor	3 axes MEMS Accelerometer	TBD	TBD	TBD	TBD
GS	Ground Station	Alén Space Ground Station kit	Alén Space	Spain	-

APPENDIX E: Analysis Results

Radiation Environment Analysis

A preliminary radiation environment analysis has been performed. The ionizing dose of the radiation environment has been assessed using Spenvis *SHIELDOSE-2* model. The absorbed dose for a preliminary geometry (20x10x10 cm parallelepiped) of the CubeSat has been analysed using *Systema* and *Oltaris*. The total dose absorbed by the satellite components in one year has been estimated to be:

Orbit	Outer walls	Inner components
400x400 51 deg	1600 Gy	40 Gy
550x550 97 deg	1300 Gy	30 Gy

Space Debris collision analysis

Two different software have been employed for space debris collision analysis, namely MASTER and DRAMA. MASTER is employed for probability of impact assessment while DRAMA is used to evaluate the probability of penetration and ballistic curve.

MASTER

We exploit MASTER (Meteoroid and Space Debris Terrestrial Environment Reference) to compute the flux distribution assuming different mission configurations and starting epochs (selecting 4 7-month spaced dates starting from 01.01.2023). The two circular orbits considered are

- h = 450 km, i = 51.6 deg
- h = 500 km, i = 97.7 deg, RAAN = [0, 45, 90] deg

The propagation is performed over 1 year with a 3-month resolution. A lower cut-off of 10^{-5} m has therefore been imposed on the diameter to assure precise enough predictions.

We consider an impact plate orthogonal to the velocity vector and the following propagation coefficients:

Drag	2.2
Solar Radiation Pressure (SRP)	1.8
Cross section (Drag)	0.02 m^2
Cross section (SRP)	0.025 m^2
Mass	2.2 kg

MASTER integrates the orbit accounting for the proper perturbations and evaluates the distribution of the debris flow according to the selected impacting detrital population. In this appendix the results of the 500 km SSO (RAAN = 45 deg) are reported.

Figure 23 shows the debris distribution. One can notice that error bars get bigger for some classes of debris diameter. Since micro debris cannot be tracked, the statistical uncertainty on lower diameters tends to get higher. There is a wide 1-σ uncertainty on the number count, stressing again the need for more precise measurements of the debris population. Here the

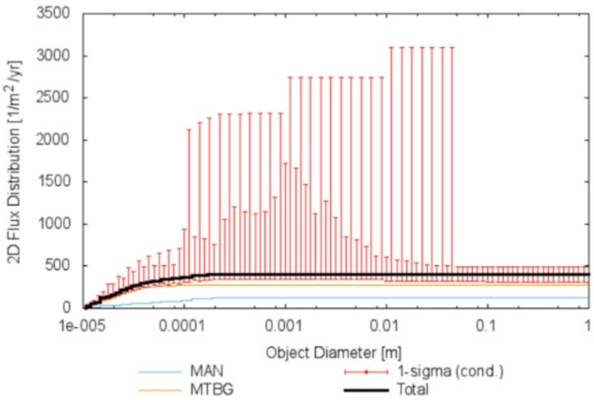


Figure 23: Debris population for a 500 km (RAAN 45 deg) SSO.

results for the overall population distribution are presented, while next section will focus on how this population will affect the spacecraft during nominal mission duration. MASTER also provides a first instance of safety test. The expected flux distribution is shown in Figure 24 for various values of the ballistic limit, expressed as a critical radius, plotting it both classically binned and as a reverse-cumulative distribution. Assuming a ballistic limit lower than 1 mm.

Furthermore, one can look at the distribution in velocity space, as shown in Figure 25. The distribution is peaked at velocities of the order of 15 km/s.

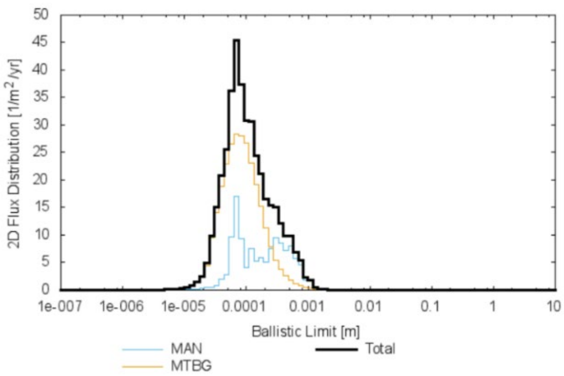


Figure 24: Expected debris flux distribution for various values of the ballistic limit (500 km, 45 deg RAAN, SSO)

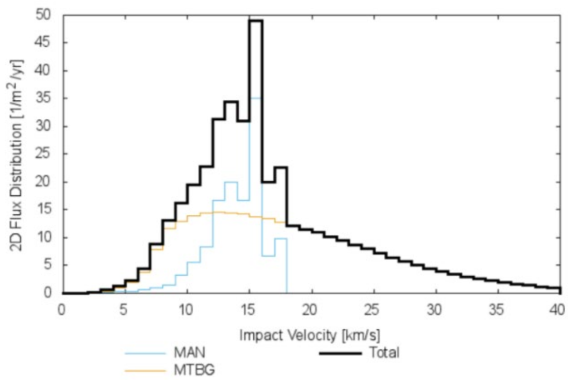


Figure 25: Velocity distribution of the impacts (500 km, 45 deg RAAN, SSO)

DRAMA (MIDAS)

MIDAS has been employed over the same satellite/orbit configuration for a 1-year propagation. Two thresholds have been imposed on the debris sizes, a lower one at 10^{-5} m and an upper one at 1 m. The goal is to find probability of collision, failure flux and probability of penetration as functions of the debris diameter. The main plots refer to the velocity-side results since they provide the worst-case scenario for potentially fatal impacts.

Figure 26 shows the probability of impacts as a function of impactor diameter. One can notice that for smaller sizes, the meteoroid population provides a higher contribution to the total count. Higher diameters have negligible probability to result in impacts, while an encounter with a smaller object is expected.

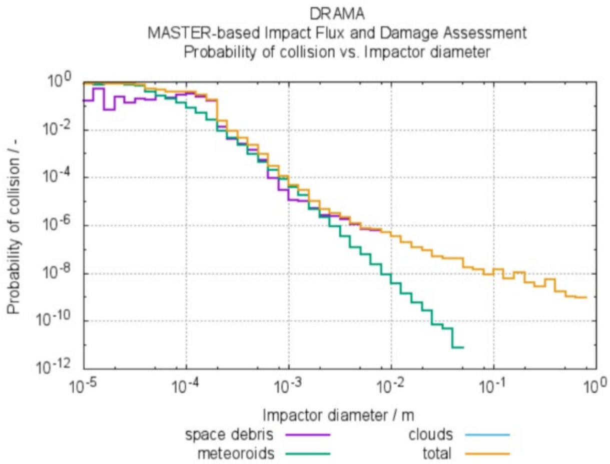


Figure 26: probability of impacts as a function of impactor diameter.

It is fundamental to assess whether the predicted impacts may be fatal to the mission. The CubeSat surfaces have been modelled according to the expected structural design. Specifically, 1-2 mm thin plates (solar panels and spacecraft walls) have been placed on all sides. We assume as “mission failure” each total-penetration event that may occur. Figure 27 shows the number of penetrations and the connected probability of penetration. A peak is present for sizes of about 1 mm, which is the region of interest of the mission in diameter-space. Nevertheless, it should be noted that the flux is consistently lower than 10^{-2} for each diameter, dropping exponentially in other size ranges. This assures that even if a collision event will occur, the spacecraft can survive. Sub-millimeter and millimeter debris are more dangerous in terms of penetration probability. However, the probability of penetration is in the order of magnitude of 10^{-4} %.

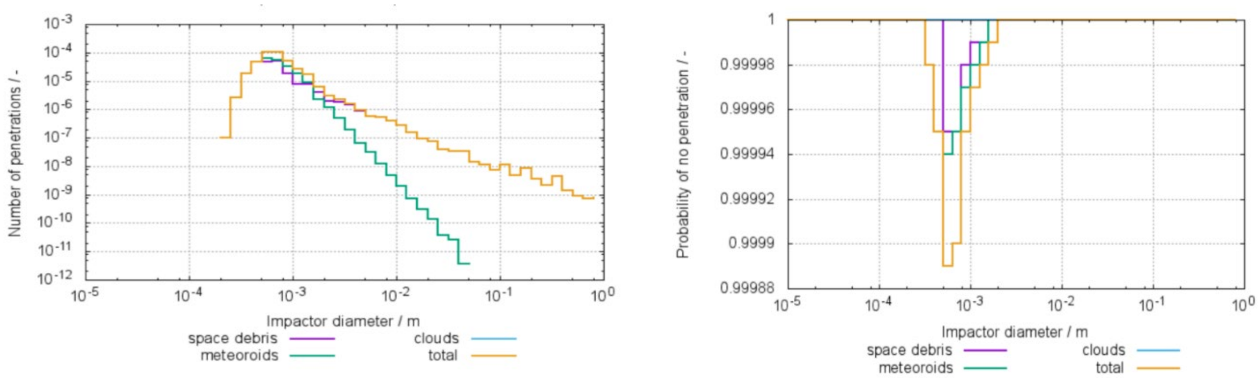


Figure 27: Number of penetrations (left) and the connected probability of penetration (right).