POLITECNICO DI TORINO

Master Degree in Aerospace Engineering

Master Thesis

Procurement activities for the design of a 6U CubeSat for Earth Observation in LEO

Academic Supervisors:
Prof.ssa Sabrina CORPINO
Prof. Fabrizio STESINA
Company Supervisor:
Eng. Simone SIMONETTI

Candidate:
Andrea SICA

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Contents

List of Figures

List of Tables

Abstract

A preliminary design of a 6U CubeSat platform is presented for an Earth Observation mission in Low Earth Orbit. Assuming a mission statement, this work aims to find the preliminary requirements of the LEO environments that apply to the sub-systems of a spacecraft, as compatible as possible with the optical payload of the CubeSat market.

These will be used to select a set of possible choices to define a platform baseline configuration useful to speed up the study and development process of a satellite in a future proposal phase of a similar mission. The combination of the mentioned choices will then provide to Argotec an approximate price and lead time of the total effort.
Carrying out this analysis will be a way to identify the reference suppliers and their products of interest through several procurement actions, thanks to which it will be possible to outline a purchasing strategy.

Chapter 1

Introduction

In the context of spaceflight, a satellite is an object placed into orbit. In order to distinguish them from natural satellites, like Earth's Moon, they're called *artificial satellites*, but the adjective is usually omitted.

The first artificial satellite, *Sputnik* 1, was launched on 4 October 1957 by the Sovietic Union and the main purpose of the mission was to place a radio transmitter in orbit around the Earth. It orbited successfully during three weeks until its battery died and then it orbited silently for 2 months before falling back into the atmosphere. This event established the beginning of the so called *Space Age*.

Since then, thousands of satellites have been launched into various orbit, for different application and with distinctive configuration.

1.1 State of the Art

Within these three mentioned domains, satellites can be divided per type. Among the applications, we can find:

- *Weather* satellites, used to gather information on meteorological conditions of Earth wide areas, permitting to obtain weather forecasts. They also help to detect phenomena like fires, effect of pollution, sand or wind storms, etc;

- *Telecommunication* satellites, used for television, telephone, radio, internet and military applications. They relay radio signals via a system that create a communication channel between a transmitting source and a receiving terminal;

- *Navigation* satellites, which determine the geographic location, speed and direction of target objects. A navigation system with global coverage is called Global Navigation Satellite System ($GNSS$). The United State NAVSTAR Global Positioning System (GPS), the Russian Globalnaya Navigazionnaya Sputnikovaya Sistema ($GLONASS$), the European $GALILEO$ and the Chinese $BEIDOU$ are examples of this satellite system;

- *Earth Observation* satellites, designed to observe the Earth from orbit and that include environmental monitoring, cartography, natural disaster detection, military strategies and others;

- *Astronomical* satellites, used for observe distant galaxies, stars, planets and other celestial object. Being in orbit above the Earth, the satellite's vision is not clouded by the gases that make up the Earth's atmosphere, and its infrared imaging equipment is not confused by the heat of the Earth. The *Hubble Space Telescope* is probably the most famous system of this category.

Other applications may be performed by *Biosatellites, Exploration Probes, Space Station or Crewed Spacecraft.*

Subsequently, we can define satellite orbits based on the values of their characteristic parameters.
According to *eccentricity* (e), there are:

- Elliptical orbit, where $e < 1$. That's the orbit of the planets around the Sun in the Solar System;
- Circular orbit, where $e = 0$;
- Hyperbolic orbit, where $e > 1$. That's an open trajectory used in interplanetary missions to leave the planets influence sphere;
- Parabolic orbit, where $e = 1$. This theoretical element represent the boundary between the aforementioned orbit, but it isn't really achievable.

According to *inclination* (i), there are:

- Equatorial orbit, where $i \approx 0°$;
- Polar orbit, where $i \approx 90°$;
- Retrograde orbit, where $i > 90°$;

According to *altitude* (h), there are:

- Low Earth orbit (LEO), where $h < 2000km$;
- Medium Earth orbit (MEO), where $2000km < h < 35786km$;
- High Earth orbit (HEO), where $h > 35786km$;
- Geostationary orbit (GEO), where $h = 35786km$. Choosing this value, for an equatorial and circular orbit, permits to match the orbit period with the Earth rotation period, and so to stand always above the same point.

One particular LEO is called *Sun-Synchronous orbit* (SSO), in which inclination and altitude are selected in order to match its orbital precession rate with the mean motion of the Earth about the Sun. This combination allow the satellite to pass over a spot on the Earth's surface at the same local *mean solar time* each time, and thus having the same illumination angle.

Lastly, different classes of satellite can be identified by their total orbiting mass. Table 1.1 follows the FAA definition in *The Annual Compendium of Commercial Space Transportation* of 2018.

Satellite class	MASS Range (kg)
Extra Heavy	>7000
Heavy	5401-7000
Large	4201-5400
Intermediate	2501-4200
Medium	1201-2500
Small	601-1200
Mini	201-600
Micro	11-200
Nano	1,1-10
Pico	0,1-1
Femto	0,01-0,09

Table 1.1: Satellite classification by mass

The green rows in table 1.1 indicate the classes being part of the bigger family of *SmallSats*. They are characterized by a lighter mass, and thus, smaller dimensions.Interest upon them has grown since early 2000s, and they are now widely used for every application thanks to their lower cost and shorter production time.

1.2 Launch Campaign

In order to bring a satellite into orbit, a *launch vehicle* is necessary. It is a rocket divided in different stage, each of which has the task of propelling the payload further along the trajectory designed to reach to required orbit. When its own propellant runs out, the single stage separates from the rest of the launch vehicle to lighten the burden.The satellite/spacecraft is placed inside the fairing and it is planned to be released in its orbit by eliminating the constraints that bond it to the structure.

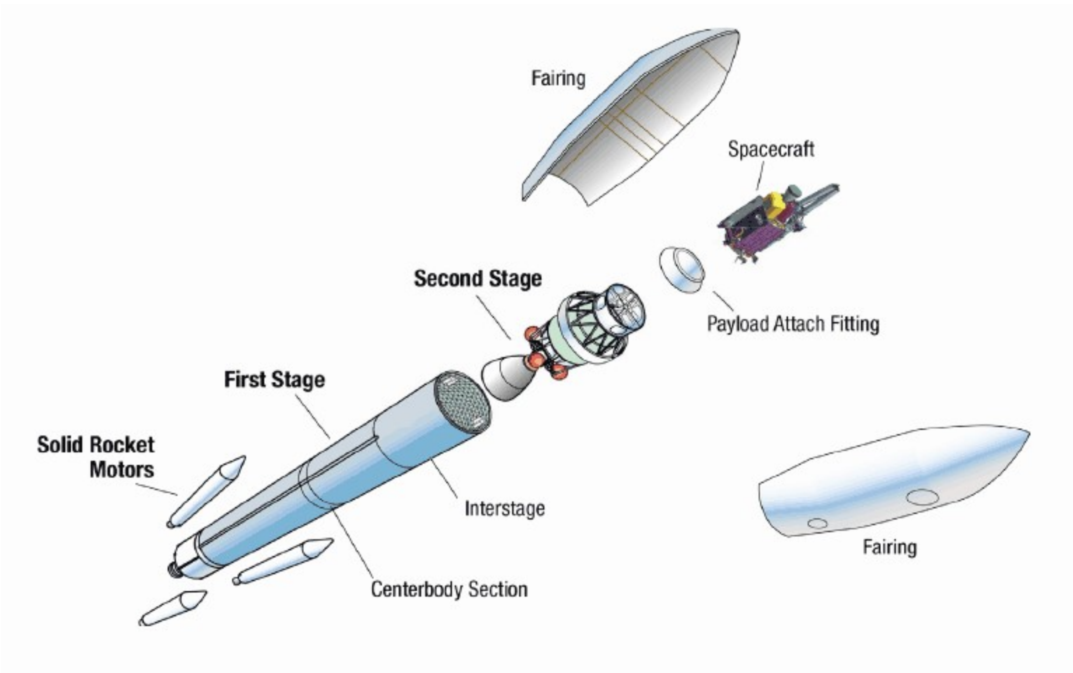


Figure 1.1: Delta II configuration for SMAP observatory [2]

The entity which develops the spacecraft and deals with launcher service provider is called primary customer. Since the fairing volume capability usually exceeds the primary customer needs, secondary customer inclusion has been considered and exploited. Secondary customers typically occupy the surplus volume with SmallSats For this purpose, adapters and dispenser (see section 2.2) has been created to accommodate secondary spacecrafts on launchers. This procedure, known as *rideshare*, permits to have incomparable prices even for the launch phase of SmallSats.
To give an example of the launcher capabilities, on the 3rd December of 2018 Space-X Falcon 9 rocket launched 64 small satellites into orbit.

Dedicated launches are performed too, i.e. launches where only SmallSats are present and thus where they are the primary spacecrafts.

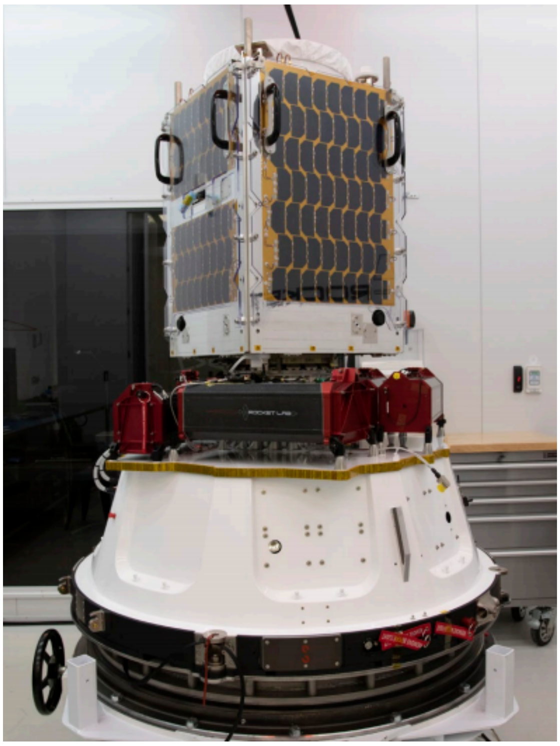


Figure 1.2: Primary (top) and secondary (bottom) spacecrafts accommodated on the Electron payload plate. Secondary spacecrafts are contained in Rocket Lab dispenser [3]

Figure 1.3: Vega-C example render for Small Spacecraft Mission Service [5]

1.3 SmallSats

As anticipated in the former section, developing and launching SmallSats has increasingly attracted organizations working within the space industry. Figures 1.4, 1.5, 1.6 and 1.7 illustrate the trends from 2012 to 2019 regarding operators, applications and launches of the SmallSat industry.
Graphs show that the main operators are the commercial ones, increasing over the years from almost not participating to leading. Universities continue their constant activities as the years go by, as well as governmental entities.
Remote sensing and technology developments have always been the dominant applications, but, in the last years, communication satellite are significantly growing in number.

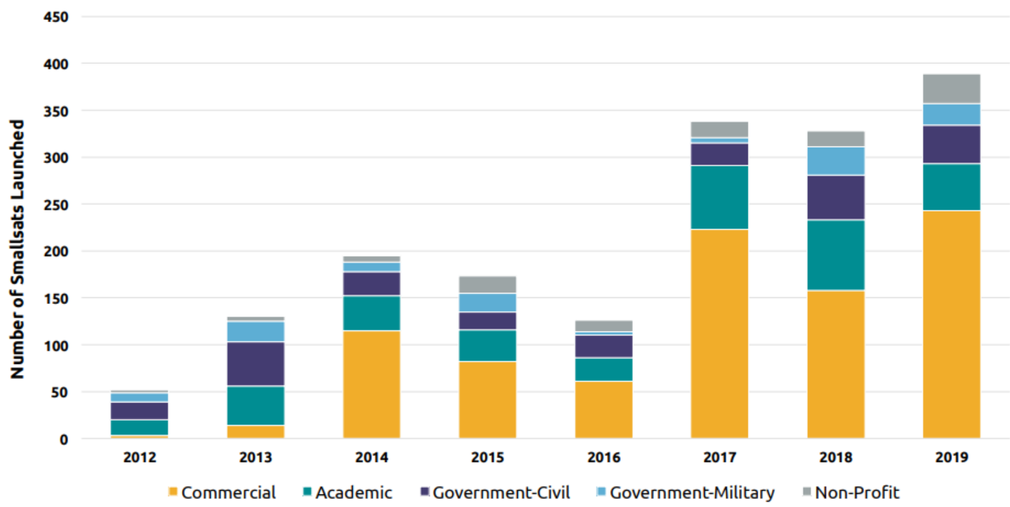


Figure 1.4: Number of SmallSats launched by operator type [6]

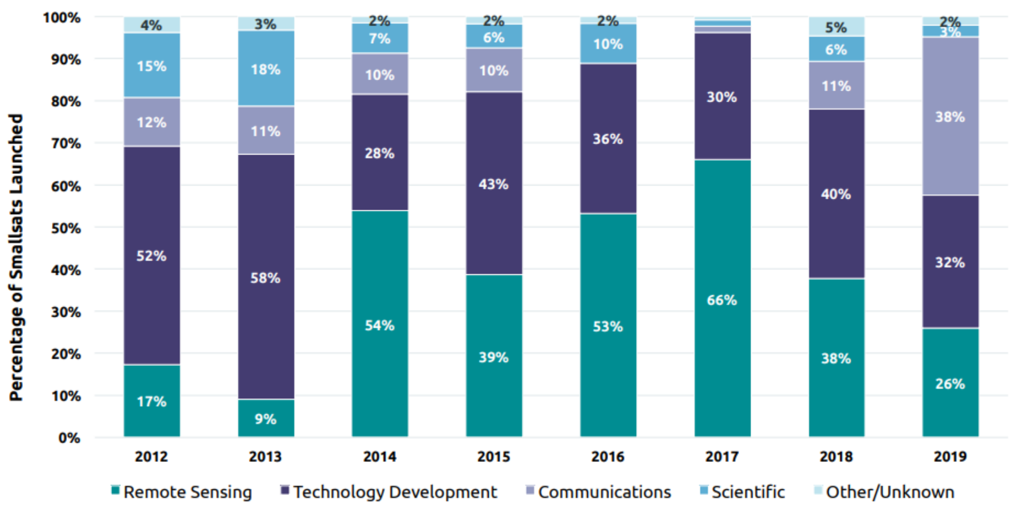


Figure 1.5: Percentage of SmallSats launched by application type [6]

The number of launches without SmallSats remains almost the same during the analyzed years, while those where they are present doubled from 2017 to 2018 and the growth was confirmed in 2019. This finds an answer in the last graph which says that dedicated launches have increased in the mentioned years, more than rideshare did.

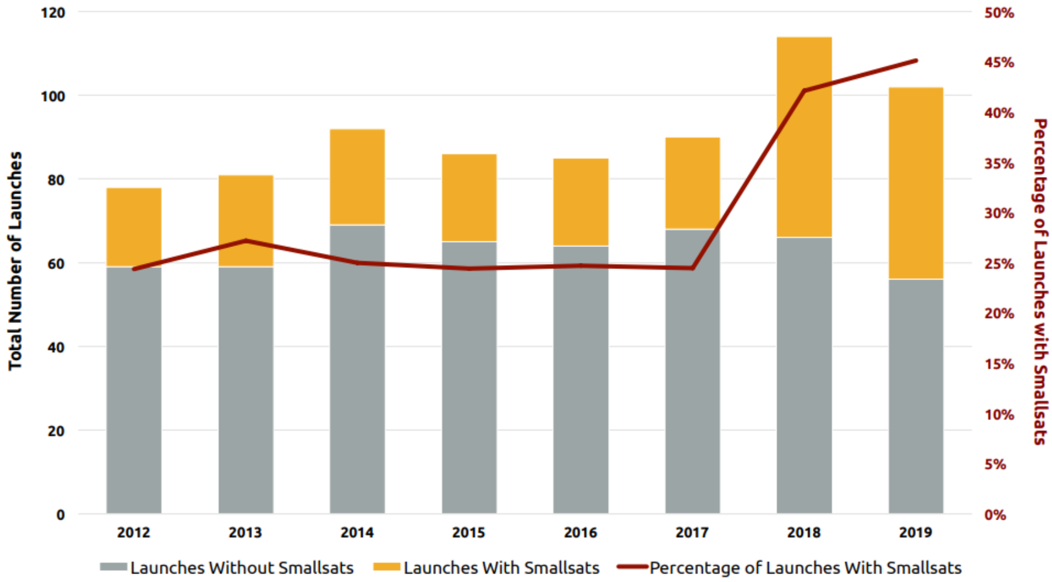


Figure 1.6: Launches with and without SmallSats [6]

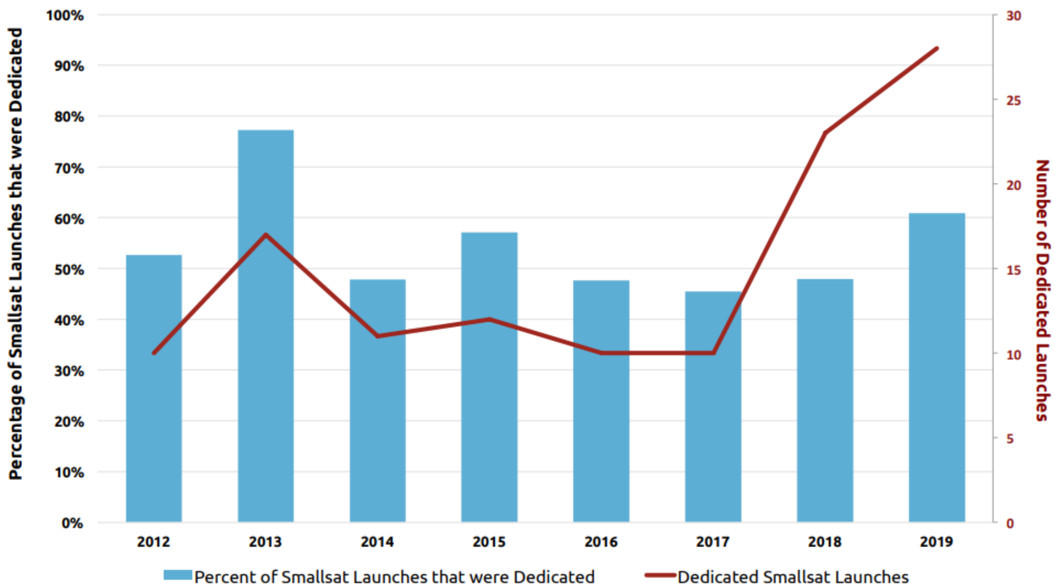


Figure 1.7: Percentage and absolute number of SmallSat launches that were dedicated [6]

1.3.1 CubeSats

A particular type of SmallSat it's called *CubeSat* and its configuration is based on standardized units of mass and volume. The concept was originally developed in 1999 by California State University and Stanford University for space exploration and research as an academic program. One unit is a 10x10x10 cm cube, also known as 1U. During the following years different volume units has been exploited, from 1U to 27U.

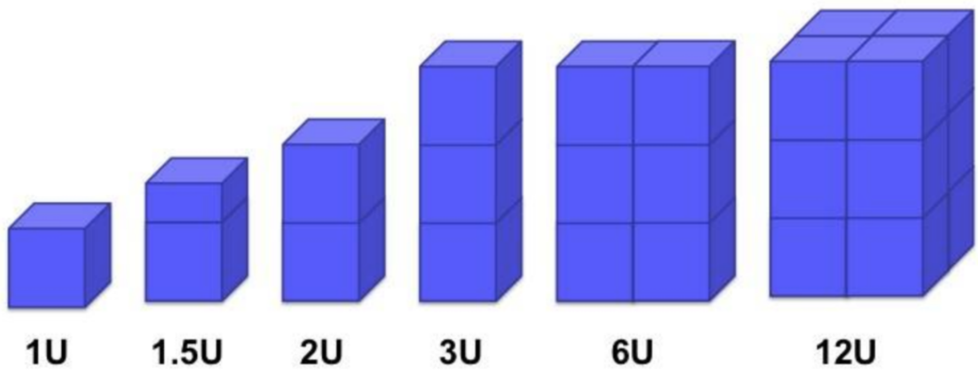


Figure 1.8: Examples of CubeSat configuration [1]

Figure 1.9 illustrate how preponderant has been the development and launch of CubeSats in the SmallSat industry from 2012 to 2019, showing a peak of almost 90% in 2017.

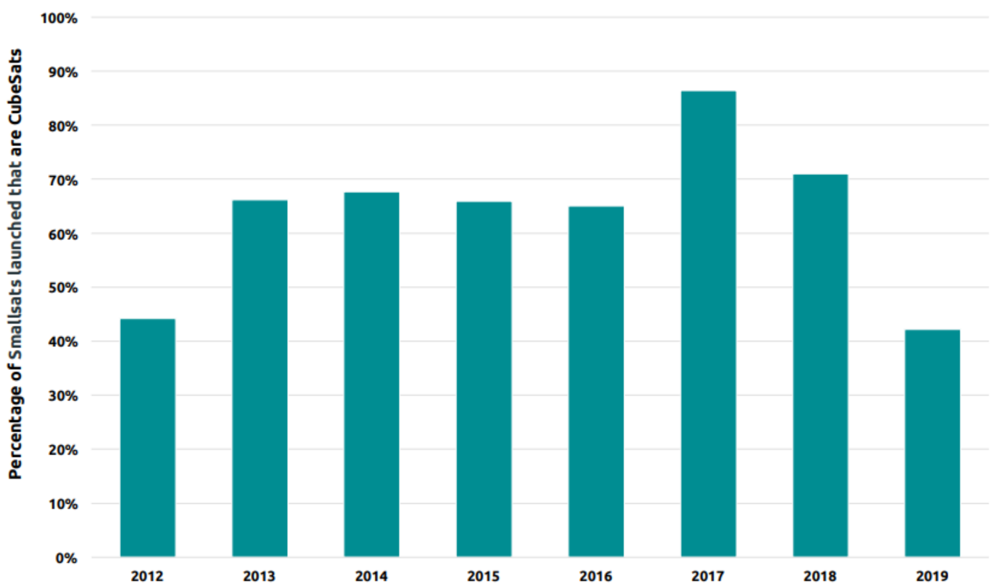


Figure 1.9: Percentage of SmallSats launched that were CubeSats [6]

How much different configurations are used is reported in figure 1.10. 3U CubeSats are the most employed as they represent more than half of the total, followed by 1U CubeSats. Bigger volume are expected to be exploited in the next years with the deployment of SmallSat constellation, primarly for *Telecomunication* and *Internet of Things* applications.

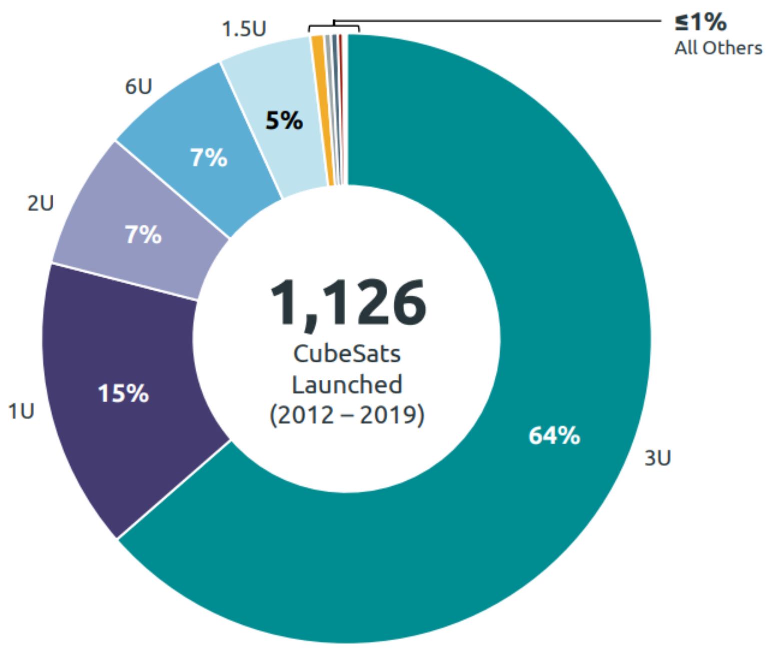


Figure 1.10: Percentage of CubeSats launched by dimension [6]

1.4 Drivers

In this context, during the last decade, *Argotec* has specialized in the development of high technology CubeSats that operates in LEO and Deep Space environment, collaborating with established entities like the European Space Agency (ESA), the Italian Space Agency (ASI) and the National Aeronautics and Space Administration (NASA).

The mentioned entities work in the scientific field, while other clients may have purely commercial objectives. Considering the latter, it is very useful for the company to possess a product whose preliminary characteristics are already established, as well as their estimated cost and production time. This would allow, at the time of a potential client's request, to gain much of the time that the study of the solution would require, and therefore to have it available for tailored modifications, if necessary for the project.

Thus, the purpose of this work is to study a reference mission in order to prepare the design and the configuration of a complete CubeSat platform that is compatible with the constraints of the space environment and of the industry.
To do so, it is necessary to define some initial assumptions. The mission that will be studied concerns a 6U CubeSat platform that operate in a LEO orbit for Earth Observation purposes. The selected mission duration is 3 years. Choices on the elements and subsystems will be influenced not only by technical and quality requirements, but also by cost and time optimization, as it obviously is important both for the company and the client.

The study will be developed by interpolating the engineering inputs with an analysis of the products in the market and an evaluation of their suppliers.

Chapter 2

System Engineering

System Engineering is an interdisciplinary approach employed to design, realize, manage, operate and retire a system. *System* is defined as the combination of all the elements that cooperate together to meet the requirements of a need.
The components include hardware, software, equipment, facilities, personnel, processes and procedures needed for this purpose. The value of the overall system is mainly due to how the components interact with each other. Therefore, the purpose of systems engineering is to create an operating system capable of meeting the requirements imposed by a need, while attempting to balance the contributions of all disciplines involved in the development of the system.

The *Systems Engineer* must develop skills that are useful in identifying the efforts needed to optimize the overall vision of the system, without favoring one or more subsystems at the expense of others, while simultaneously meeting project objectives. The expert must know where his knowledge is limited, and thus, where to let the specialist work. Then, he is not a specialist of the specific phase, but he isn't even a generalist; the system engineer is a complexity specialist, to whom the global vision is entrusted.

2.1 Space Environment

The first necessary step to underline the high-level mission requirements is to define a mission scenario. Therefore, in the following sections the point that has been touched are:

- Orbital Parameters;
- Eclipse Events Evaluation;
- Radiation Analysis;
- Drag Evaluation;
- Temperature Evaluation;

2.1.1 Orbital Parameters

To calculate the orbital parameters concerning the mission ESA's SPace ENVironment Information System (SPENVIS) has been used. The online interface needs these input:

- Mission duration;
- Solar pressure parameter;
- Orbit type;
- Mission start date;
- Altitude.

The mission duration is specified in section 1.4, while the solar pressure parameter is obtained through the tool's guidelines. Sun-Synchronous orbit has been selected for his advantages and for his high availability, as highlighted in figure 2.9. Same considerations for the altitude, selected in the range from 500 km to 600 km. The 1st of January 2022 has been inserted as a reference
Inclination is what makes an orbit sun-synchronous, permitting its angular precession ($\frac{\Delta\Omega}{T_{sat}}$) to match the Earth mean motion around the Sun ($\frac{2\pi}{T_E}$).

$$\Delta\Omega = -3\pi \frac{J_2 R_E^2}{p^2} \cos i$$

$$T_{sat} = 2\pi \sqrt{\frac{a^3}{\mu}}$$

Where p is the semi-latus rectum of the orbit, J_2 is the coefficient for the second zonal term (1.08263×10^3) related to the oblateness of the Earth, R_E is the Earth's mean radius and μ is the standard gravitational parameter of the Earth.

Since the SSO can be approximated as a circular orbit, the *semi-latus rectum (p)* equals the *semi-major axis (a)*.

$$\frac{\Delta\Omega}{T_{sat}} = -\frac{3}{2}\pi \frac{J_2 R_E^2 \sqrt{\mu}}{a^{\frac{7}{2}}} \cos i$$

The right term is then equalled to $\frac{2\pi}{T_E}$ so that the inclination (i) can be obtained knowing the semi-major axis, or radius, of the SSO orbit.

As said in the beginning, SPENVIS did this calculation, along with plenty of others, returning the outputs, resumed in table 2.1. The relation between sun-synchronous altitude and inclination is shown in figure 2.1.

Report file	
Orbit type	Heliosynchronous
Local Time of Ascending Node	6:00 am
Apogee	500-600 km
Perigee	500-600 km
Inclination	97,40-97,79°
R.A.A.N.	190,24°
Period	1,58-1,61 hrs
Semi-latus rectum	6879,16-6978,16 km
Semi-major axis	6879,16-6978,16 km
Eccentricity	0,00

Table 2.1: Orbit parameters

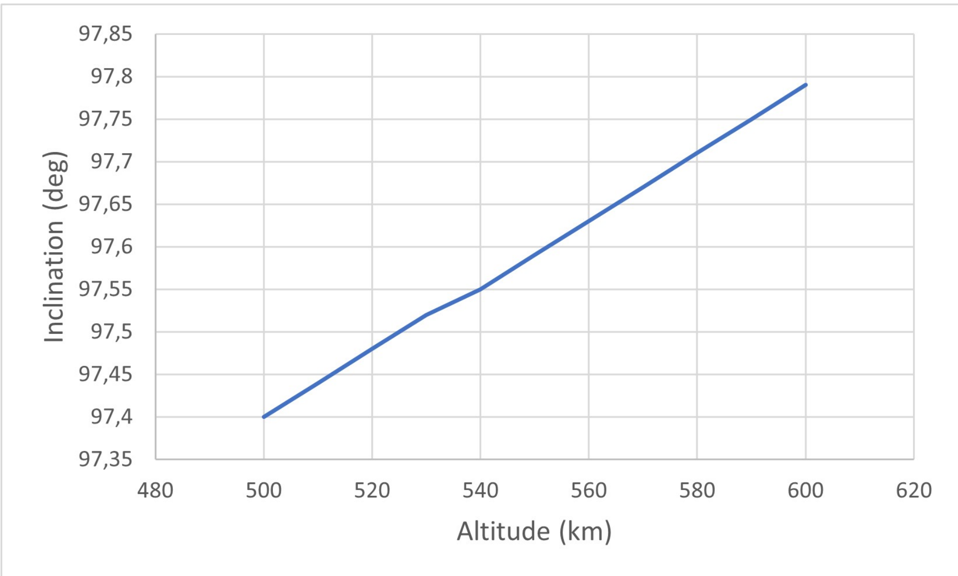


Figure 2.1: Inclination changes between 500 km and 600 km

Following the definition of the trajectory, it is fundamental to study the different problematic areas: eclipse events, radiation dose, atmospheric drag, temperature range and battery cycles.

2.1.2 Eclipse Events Evaluation

The eclipse analysis aims to analyse and define the number and the duration of the eclipse events occurring during the nominal operation phase of the mission. The analysis has been simulated through GMAT (General Mission Analysis Tool) and it has been performed considering the lifetime goal of 3 years to design the system to optimally operate for that time. The program has been run different time to analyze the range spacing from 500 km to 600 km and both the Earth and the Moon has been considered as occulting bodies.
There are two type of eclipse event: *Umbra*, when the satellite is completely behind the occulting body and therefore none of its part is illuminated by the Sun, and *Penumbra*, when the satellite is still partially illuminated by the Sun.

The number of these events and their duration drive the spacecraft design since they define the charge/discharge profile of the battery as well as the definition of the mission profile. The maximum and minimum duration, instead, are necessary to individuate the temperature range which the spacecraft will be subjected. These data behaviour between 500 km to 600 km are reported in figures 2.2, 2.3 and 2.4.

The worst scenario, i.e. the worst case for every data, has been considered, even if belonging to different altitude, so that the platform could be design to fully operate in the whole range considered.

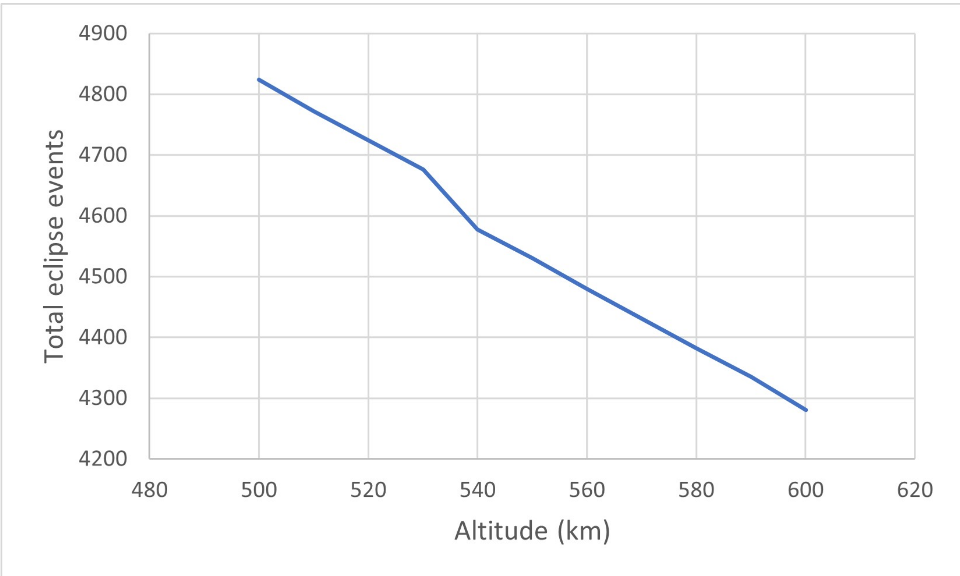


Figure 2.2: Total eclipse events when varying altitude

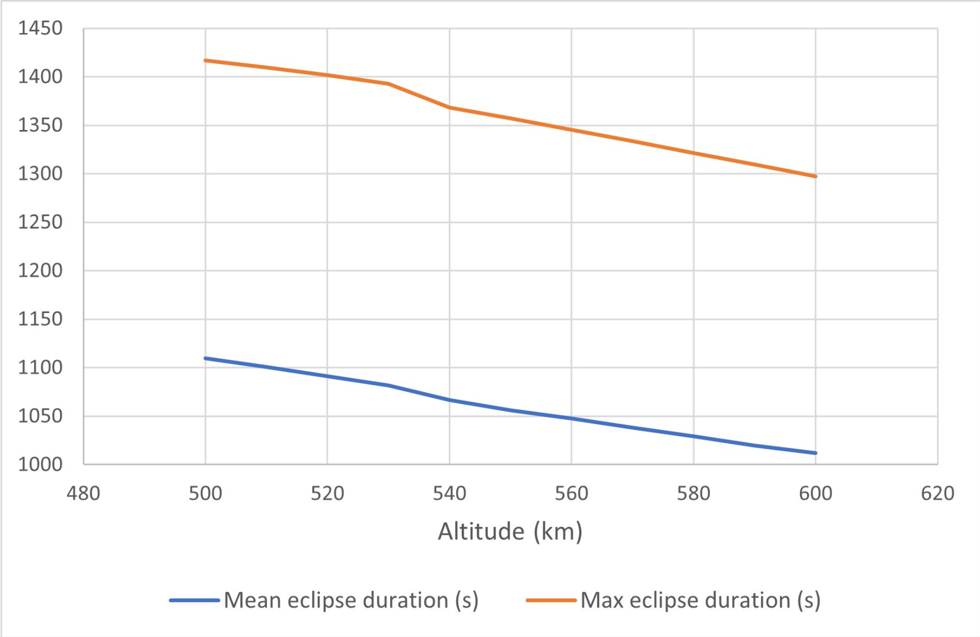


Figure 2.3: Mean and maximum eclipse duration when varying altitude

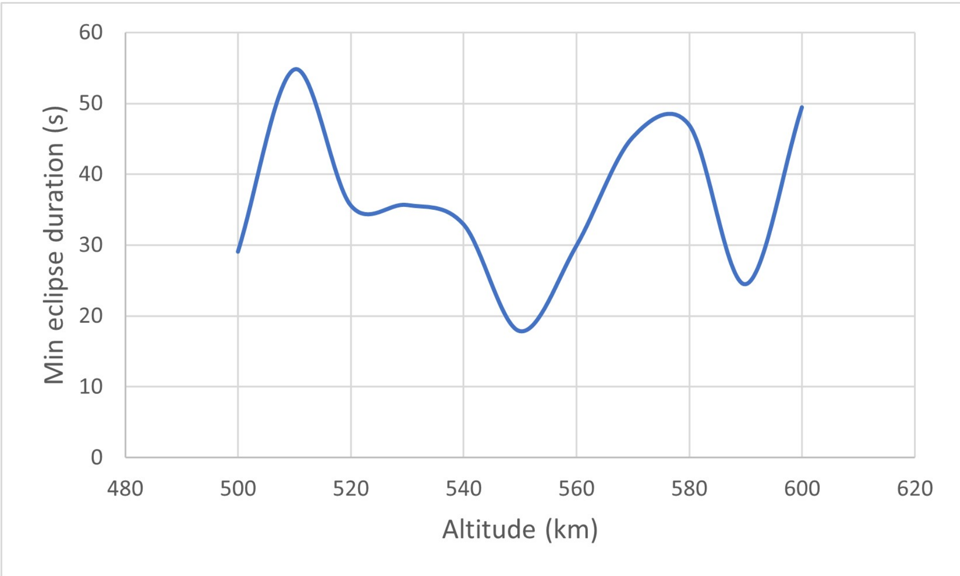


Figure 2.4: Minimum eclipse duration when varying altitude

Total eclipse events	4824
Mean eclipse duration	1109,76 s
Maximum eclipse duration	1416,75 s
Minimum eclipse duration	17,86 s

Table 2.2: Eclipses - worst-case scenario

2.1.3 Radiation Analysis

Since the satellite shall withstand the LEO radiation environment for the ionizing dose during the mission, the following paragraphs show the outcomes of the radiation analysis performed, using the SPENVIS tool, considering the selected mission time of 3 years. The Total Ionizing Dose (TID) allows defining the capacity of the satellites' subsystem to withstand the space environment considering different contributing factors which are:

- Trapped protons and electrons;
- Secondary Bremsstrahlung protons;
- Solar flare protons;
- Galactic cosmic ray ions.

The TID, that can result in device failure, is measured by a unit called rad (Radiation Absorbed Dose) and, from the spacecraft point of view, varies according to the material and thickness of the structure. Aluminum has been selected as material. TID, in terms of dose depth curve, has been calculated selecting the following models:

- AP-8 Solar Maximum for trapped protons;
- EP-8 Solar Maximum for trapped electrons;
- CRÈME 96 for the short-term Solar Particle fluxes considering ions from Hydrogen to Uranium and worst-week conditions;
- ESP-PSYCHIC for the Solar Particle fluence (confidence level 95%) considering ions from Hydrogen to Uranium;
- ISO 15390 for Galactic Cosmic Ray fluxes considering ions from Hydrogen to Uranium.

The worst-case scenario has been researched in the altitudes-inclination range defined in 2.1. Results are shown in figure 2.5.

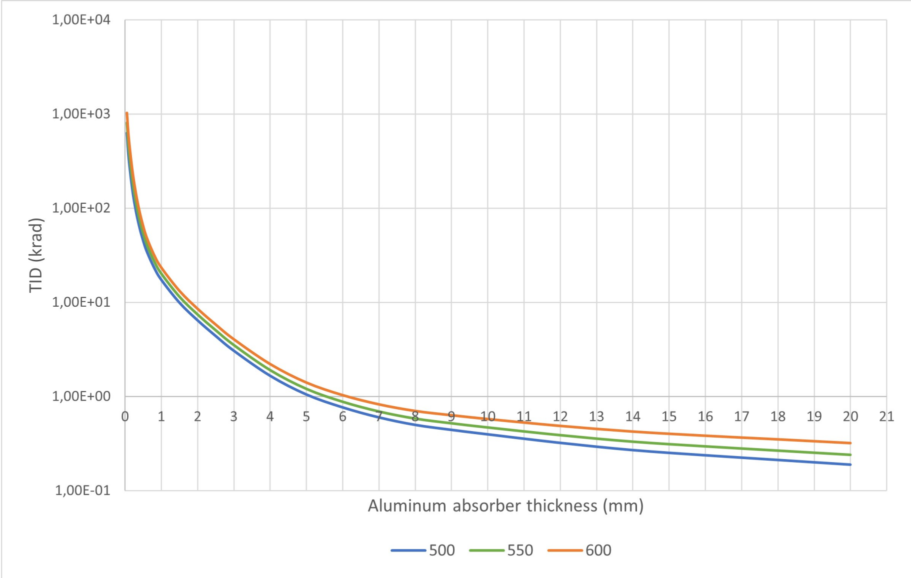


Figure 2.5: Total Ionizing Dose when varying thickness

Inclination of 97.79 degrees at 600 km of altitude feature the higher TID level. Choosing a structure thickness of 3 mm permits to have an overall TID in 3 years below 5 krad at any altitude, as figure 2.6 affirms.

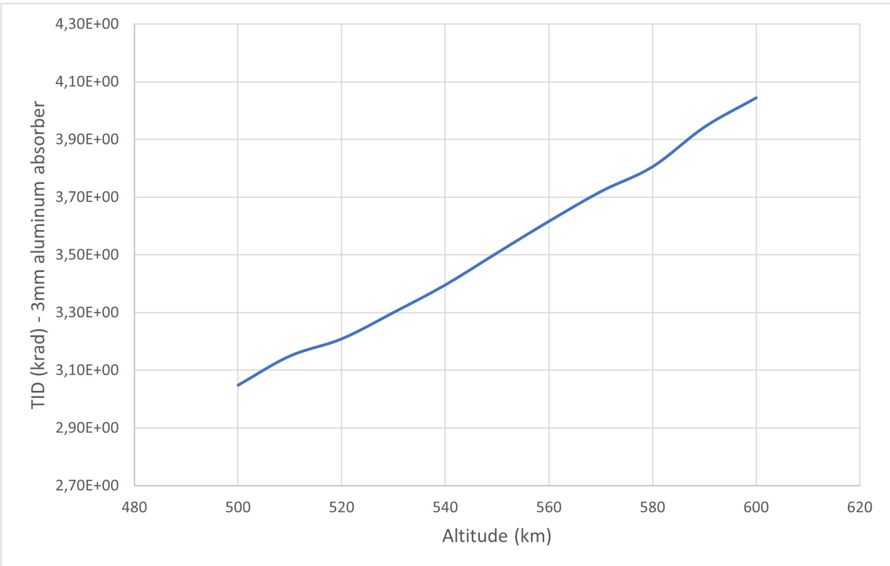


Figure 2.6: Total Ionizing Dose for a 3mm structure thickness

2.1.4 Drag Evaluation

A satellite that orbits in low altitude around the Earth interacts with the upper atmosphere. In particular, the gas molecules collide with the surface of the satellite, causing an overall drag force. For a short duration interaction, for example for a few days, the effects would not be interesting, but for missions whose duration is measured in years, this phenomenon causes the satellite to decay, i.e. to periodically decrease its orbit altitude.
The disturbing acceleration caused by the Atmospheric Drag resulting on an orbiting satellite can be modeled as:

$$\mathbf{a}_{atm} = -\frac{1}{2}\frac{\rho C_D A}{m}\dot{r}^2\hat{\dot{\mathbf{r}}}$$

where ρ is the atmosphere density, C_D the drag coefficient, A the cross-sectional area of the satellite perpendicular to its direction of motion, m the satellite mass and $\dot{\mathbf{r}}$ the satellite velocity. $\hat{\dot{\mathbf{r}}}$ is the unit vector along the direction of the satellite velocity.

Moreover, every object exposed to solar radiation, perceives a mechanical pressure resulting from the interaction between the photons and the affected surface.
The disturbing acceleration caused by the Solar Radiation Pressure resulting on an orbiting satellite can be modeled as:

$$\mathbf{a}_{SRP} = -p\frac{A}{m}k\hat{\mathbf{U}}$$

where p is the intensity of the radiation pressure, k is the reflectivity and $\hat{\mathbf{U}}$ the unit vector in the direction to the Sun. The intensity of the radiation pressure varies inversely with the square of the distance to the Sun s and following this equation:

$$p = p_o(\frac{s_o}{s})^2$$

where p_o is the intensity at the mean earth-sun distance s_o. At 1 A.U. (the Earth-Sun mean distance) $p = 4,5\text{x}10^{-6}\frac{N}{m^2}$.

Depending on where the satellite is along the orbit, and so its relative position with respect to the Sun, the force acting on the satellite can raise or decay the orbit, while the drag caused by the atmosphere always tends to decay the orbit. During the nominal life of the satellite, it's necessary to counteract the orbit decay to let it perform its task correctly. Differently, when the mission ends these drags are useful to accomplish the satellite de-orbiting.
IADC (Inter-Agency Space Debris Coordination Committee) recommends "*... to ensure that the lifetime after disposal will not exceed 25 years*" [20]. To demon-

strate that the platform complies with this requirement, a simulation has been performed in GMAT.

While the orbit has already been defined in section 2.1, for drag calculation, it's necessary to give in input the mass of the satellite,chosen in section 2.2 and its coefficient of drag. The geometry of the satellite has been assumed to be a filled sphere, and thus the surface affected by the Atmospheric Drag it's always half of the sphere's surface. For the Solar Radiation Pressure instead, considering that the solar panels will be mounted on the biggest face, and the SPA dimensions defined in 2.3.1, a trade off has been made: the surface affected during the eclipse would be zero, while during the rest of the orbit, being the satellite in Sun-pointing mode, it would be the maximum one, i.e. the biggest face plus the SPA surface.

Earth has been chosen as Primary Body, *JGM-2* till the fourth zonal harmonica as Gravity Model, *JacchiaRoberts* as Atmospheric Model and *Spherical* as Solar Radiation Pressure Model for generating the Force Model. Input argument are resumed in table 2.3.
Results are shown in figure 2.7, where it can be seen that the satellite, after ending its mission, re-enter the atmosphere in approximately 4644 days, i.e. 12 years and 261 days.

Mass (kg)	11
Coefficient of Drag	2,2
Drag Area (m^2)	0,0798
SRP Area (m^2)	0,27

Table 2.3: Satellite Ballistic/Mass Parameters

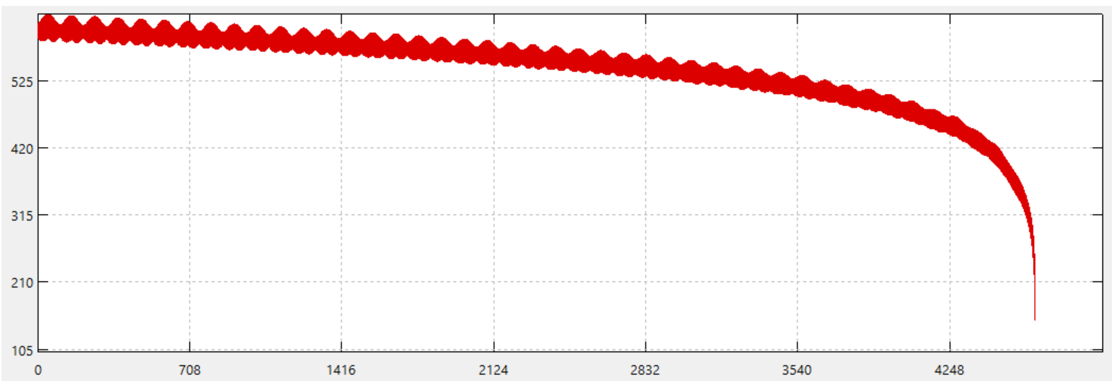


Figure 2.7: Orbit Altitude vs Mission Elapsed Days

2.1.5 Temperature Evaluation

Thermal environment is an important issue in designing a space platform orbiting in LEO. Its temperature profile needs to be studied and kept under control to assure that it will works correctly during every phase of the mission. All spacecraft components have a range of allowable temperature that must be maintained in order to meet survival and operational requirements and thus different active or passive techniques are used to regulate the temperatures throughout the platform's sub-systems.
In CubeSat design, the most common mitigation techniques involves the use of paint with favorable thermal properties; optimized disposition of the sub-system also helps managing the spacecraft thermal control. For example, a components that needs to have a relative hot temperature could be place adjacent to the battery.

As a preliminary evaluation, the platform will be considered as a single node with a uniform temperature, as modeled in figure 2.8. The previously mentioned factors are indicated as Q_{ext}, while Q_{int} is the internally dissipated energy, considered as 20% of the spacecraft required power (plus or minus a margin of 15% whether we consider the hot or the cold case), and Q_{out} is the energy exchanged from spacecraft to space. T_{sink} is the space temperature while $T_{S/C}$ is the unknown parameter. Cold case refers to the eclipse time.

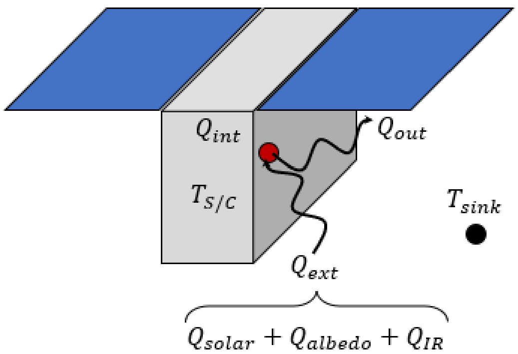


Figure 2.8: Spacecraft single node model

The actors to be considered are the *Solar Flux*, the solar energy radiated into space by the Sun, the *Albedo*, the portion of solar energy reflected from the Earth surface back into space, and the *Earth Infrared Radiation*, the energy emitted from the Earth into the space as its temperature is above 0 K. Following NASA guidelines defined in [22], worst case parameters are reported in table 2.4 and 2.5. For the last two actors the worst case has been considered as the affected area, i.e. the 0.2x0.3 face.

-	Heat Flux $[W/m^2]$	$A_{affected}$ $[m^2]$	Energy [W]
Solar Flux	1422	0,03	$Q_{solar} = 42{,}66$
Albedo	497,7	0,06	$Q_{albedo} = 29{,}86$
IR Radiation	266,6	0,06	$Q_{IR} = 15{,}99$

Table 2.4: Thermal Load - hot scenario worst case

-	Heat Flux $[W/m^2]$	$A_{affected}$ $[m^2]$	Energy [W]
Solar Flux	0	0,03	$Q_{solar} = 0$
Albedo	0	0,06	$Q_{albedo} = 0$
IR Radiation	214,2	0,06	$Q_{IR} = 12{,}08$

Table 2.5: Thermal Load - cold scenario worst case

The equation governing the heat exchange is the following:

$$\Delta Q = \Sigma Q_{ext} + \Sigma Q_{int} - \Sigma Q_{out}$$

The hot case has been treated as steady, considering the sun exposition enough to reach the thermal equilibrium. Thus, with ΔQ being equal to zero, an heat balance has been calculated to obtain the hottest temperature the platform will register.

$$(Q_{solar} + Q_{albedo})\alpha + \epsilon Q_{IR} + Q_{int} - \epsilon\sigma A_{S/C}(T_{S/C}^4 - T_{sink}^4) = 0$$

$$T_{S/C_{hot}} = (\frac{Q_{solar} + Q_{albedo}}{\epsilon\sigma A_{S/C}}\alpha + \frac{Q_{IR}}{\sigma A_{S/C}} + \frac{Q_{int}}{\epsilon\sigma A_{S/C}} + T_{sink}^4)^{\frac{1}{4}}$$

The cold case instead, has been treated as transient, because the eclipse time is limited and not sufficient to reach the equilibrium between spacecraft and environment.

$$Q = \epsilon Q_{IR} + Q_{int} - \epsilon\sigma A_{S/C}(T_{S/C}^4 - T_{sink}^4)$$

$$\Delta T = \frac{T_{eclipse}\Delta Q}{c_{p_{Al}} m_{S/C}}$$

$$T_{S/C_{cold}} = T_{S/C_{hot}} + \Delta T$$

In the previous equations, α and ϵ are respectively the solar absorptance and the emittance of the affected surfaces.
The data used to calculate the temperatures are reported in table 2.6. To be

conservative, the worst operative mode in terms of power consumption, highlighted in 2.3.6, has been considered for Q_{int}, as if it would last during the whole orbit.

T_{sink}	4 K
$A_{S/C}$	0,22 m^2
P_{hot}	60 W
P_{cold}	32 W
$Q_{int_{hot}}$	13,8 W
$Q_{int_{cold}}$	5,44 W
$T_{eclipse}$	1416 s
$m_{S/C}$	11 kg
$c_{p_{Al}}$	880 $\frac{J}{kgK}$

Table 2.6: Thermal Analysis input arguments

The calculation has been performed first considering polished aluminum as the platform material and then, the same material with the white painting *YB71*. Their thermal properties are listed in table 2.7.

-	Polished Aluminum	YB71 white paint
α	0,14	0,18
ϵ	0,03	0,90

Table 2.7: Surfaces Thermal properties [23]

Considering the platform surfaces to be made of polished aluminum the results of the analysis are definitely harsh, reaching temperatures that wouldn't let the systems survive. As anticipated at the beginning of this section, coatings helps the satellite to mitigate the heat absorption and emission phenomenons, as shown in table 2.8.

T_{max}	8,39 °C
T_{min}	4,55 °C

Table 2.8: Temperature Range - YB71 white paint

Results confirms how the higher emittance of the white paint helps the platform exchange heat with the outer space and achieve a lower temperature. To obtain this acceptable range, it has been decided to use the paint in the 0.2x0.3 faces only.

2.2 Launch Environment

In this section, the environment in which the platform will be integrated at the end of its design and development process has been studied.
To bring the satellite to the designated orbit, it's necessary to analyze the vector, i.e. the launch vehicle, and the adapting feature which will accommodate the product in it. The separation systems in form of containers minimize the risks for the primary payload and for the launch vehicle.

As explained in section 1.2, launch providers offer the possibility to exploit the exceeding fairing capability to secondary payloads like the ones discussed in section 1.3.
SpaceX Falcon 9, Arianespace VEGA C, Soyuz and Ariane 6, and ISRO (Indian Space Research Organisation) PSLV are example of launch vehicle that allow this optimization, while Rocket Lab ELECTRON is an example of vehicle that also offers SmallSat dedicated launches, where heavier class of satellite aren't present.
During launch phases, primary and secondary payloads are subject to mechanical and acoustic loads due to propulsion and vibrations. Together with electrical interfaces, they represent the *Launch Vehicle Requirements*. Loads on the satellite could be amplified or mitigated by the adapter interface between them and the launcher. Thus, during qualification, the satellite and adapter assembly should be tested.

Dispensers are thought specifically for CubeSats, which present a form factor and thus, standardized dimensions. In the same way, dispenser are identified by the volume they can host, as figure 2.9 shows.

Figure 2.9: ISIS CubeSat deployers [9]

Right now, dispensers in the market are distinguished by the feature which permits the satellite the be fixed during storage, and to slide out during deployment. Two are the main configuration:

- Tabs along two edges;
- Rails along the four edges;

The deploy mechanism is the same: the CubeSat is pushed down through the dispenser and rests on top of a spring platform which holds the elastic energy for deployment; the CubeSat is enclosed by a spring-loaded door that can be released with a bolt separation or split spool-based system; once the door is fully open, the spring platform push the CubeSat out of the dispenser.

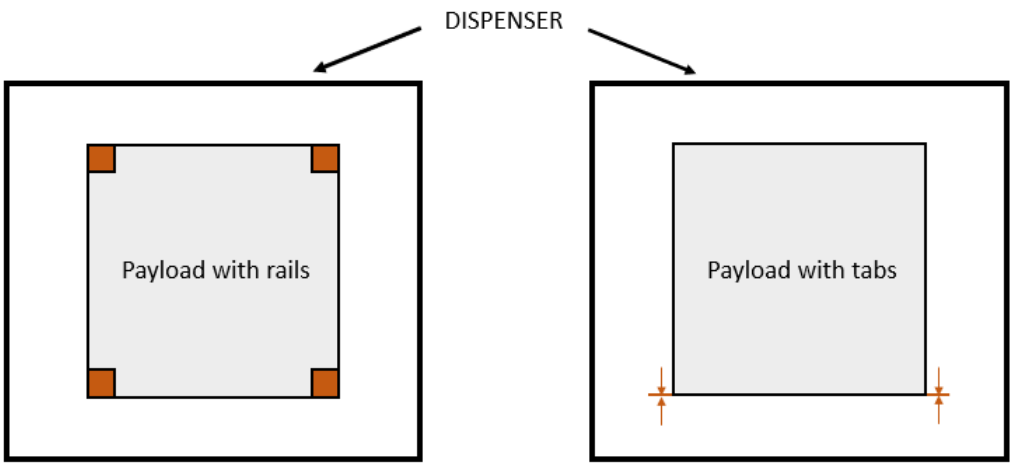


Figure 2.10: Rails an tabs configuration illustration

The first configuration was patented by *Planetary Systems Corporation*, which therefore is the only manufacturer. Closing the dispenser door automatically preloads the tabs, creating a modeled load path so that strength at critical locations can be accurately calculated. Preloaded tabs also avoid the payload to jiggle and damage itself. Figure 2.11 shows an example of the tabs as they should feature on the CubeSat.

Figure 2.11: Argotec 6U platform with detailed zoom on tabs

The railed configuration instead, is offered by many companies and with different features. Being constrained to all the four edges the payload minimizes its vibrations. The GTM 12U bus in figure 2.12 is an example of the this configuration.

Figure 2.12: GTM 12U platform [10]

A third configuration exists and it's manufactured by NASA Wallops Flight Facility. It utilizes slotting pins at the key positions at the top and the sides of the payload structure creating a more predictable loading environment.

Loaded dispenser are then attached to the launch vehicle in different ways: bolted to a plate (see figure 1.2), using an adapter, such as the ESPA ring in figure 2.13. The latter can hold CubeSats up to 24U volume, but special interfaces with the launch vehicle are required.

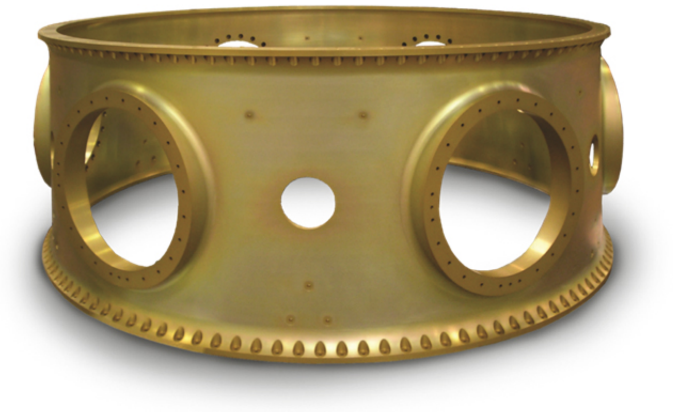


Figure 2.13: Evolved Expendable Launch Vehicle Secondary Payload Adaptor [11]

2.2.1 Launcher

In this section, Falcon 9, VEGA C and ELECTRON requirements will be analyzed. The approach consist to evaluate each launcher's requirements individually, and then to individuate the meshed requirements that satisfies all others, i.e. the strictest. Acceleration loads, the accumulation of absolute acceleration values over a specified period, shock loads, transient loads of very high amplitude and short duration (stage separations, fairing deployment, etc.), and random vibration, whose absolute value is not predictable at any point in time are the main behaviour that needs to be verified.

Falcon 9 has two different adapters, the *Dispenser Ring* and the *Starlink Adapter* (see figure 2.14), which present different acceleration loads profile, as illustrated in figure 2.15. As stated in [16], payload axial and bending mode fundamental frequencies should be greater than 40 Hz.
Shock loads and random vibrations for Rideshare service are found in figures 2.16 and 2.17. There is no differences depending on the adapter employed. The electric interface require that the spacecraft batteries must be switched off during launch.

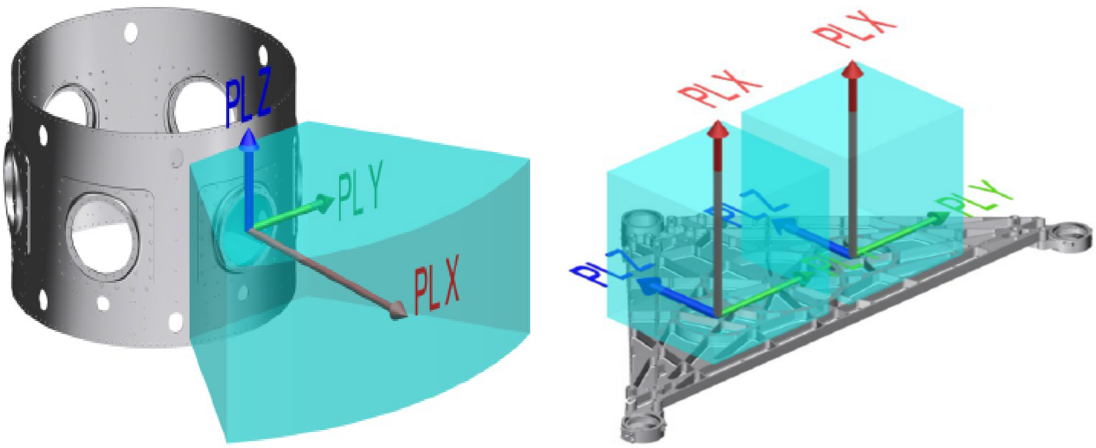
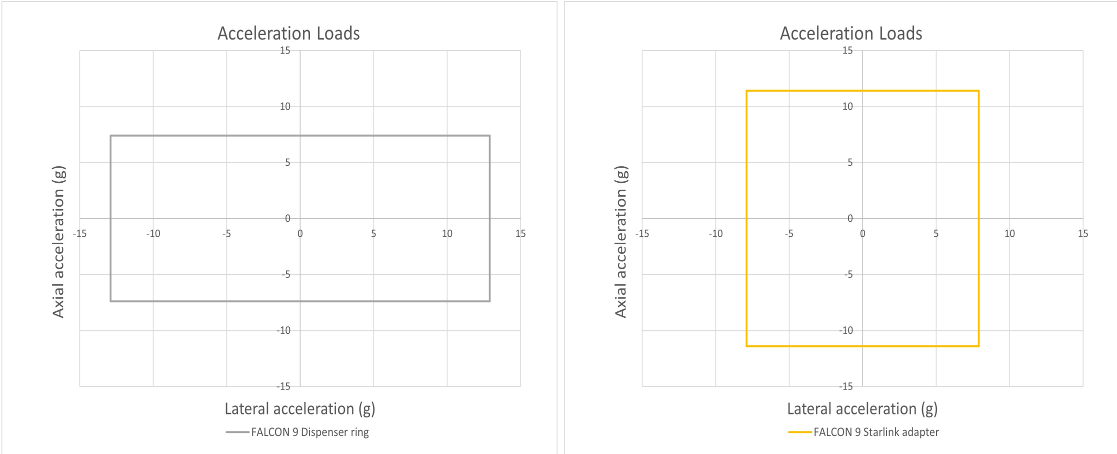


Figure 2.14: Dispenser Ring (left) and Starlink Adapter (right) models



(a) Dispenser Ring Load Factor (b) Starlink Adapter Load Factor

Figure 2.15: Falcon 9 Rideshare Load Factors

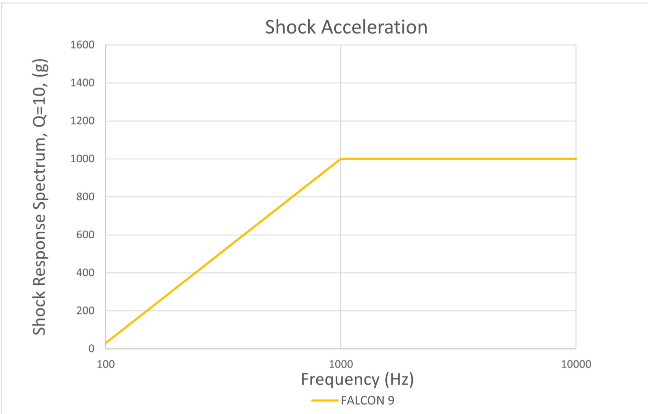


Figure 2.16: Falcon 9 induced Shock Loads

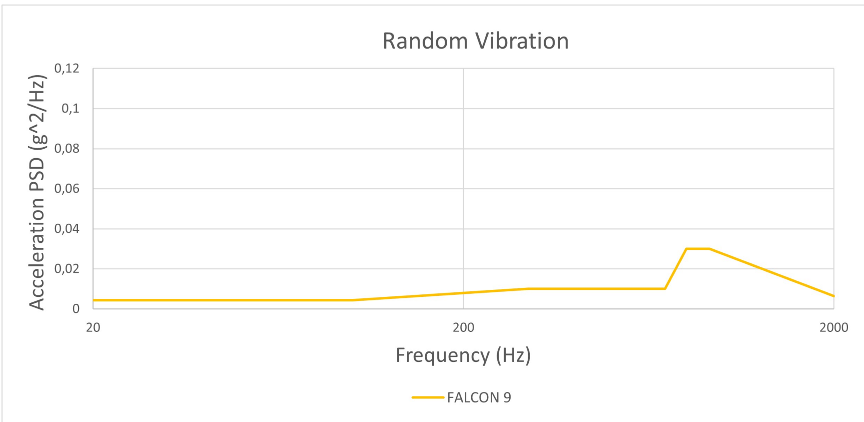


Figure 2.17: Falcon 9 Random Vibration Maximum Predicted Environment

VEGA C provide two possible accommodations for CubeSat deployers: *SSMS Hexagon* and *Tower* positions (see figure 2.18). Acceleration loads are position-independent, while shock loads and random vibration present two different profile depending whether the payload is place in one position or another. To prevent any dinamic coupling with fundamental modes of the launch vehicle, in case of a CubeSat deployer, the assembly shall be designed with a structural stiffness which ensures the longitudinal and lateral main frequencies are greater than 115 Hz. The allowable mass is comprised between 10 and 35 kg. Moreover, the spacecraft shall be inert during the final countdown and ascent phase until after the it separates from the launcher.

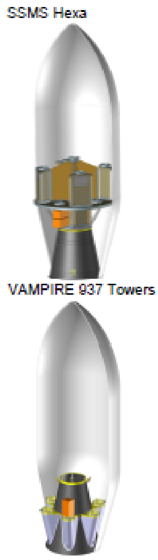


Figure 2.18: SSMS Hexagon (top) and Tower (bottom) positions

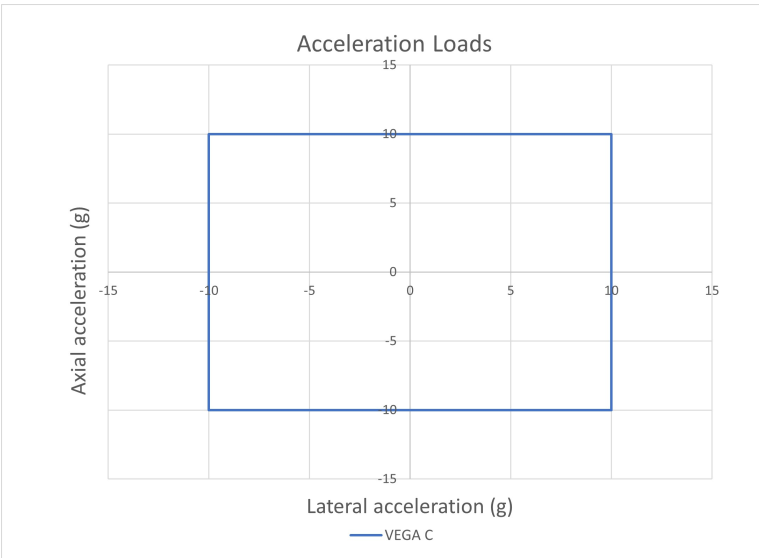
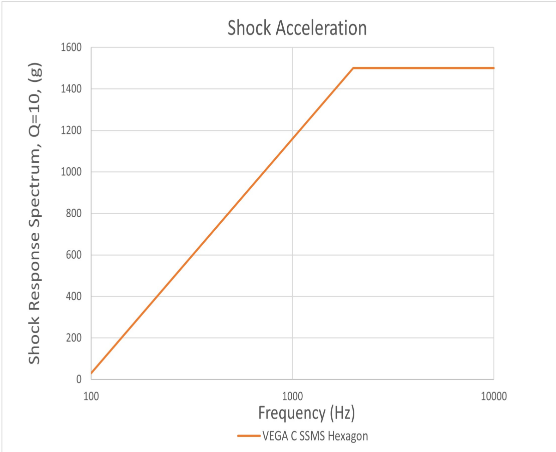
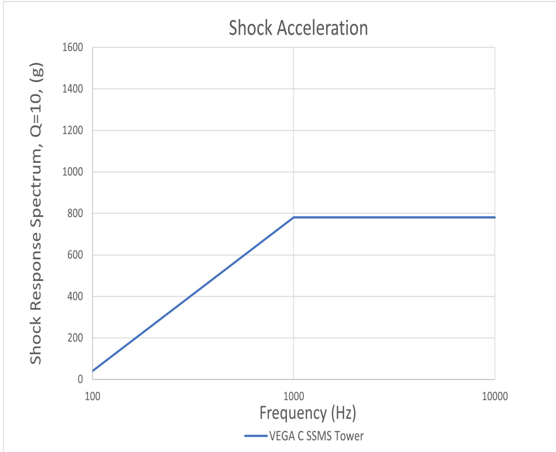


Figure 2.19: VEGA C Loads Factor for SSMS

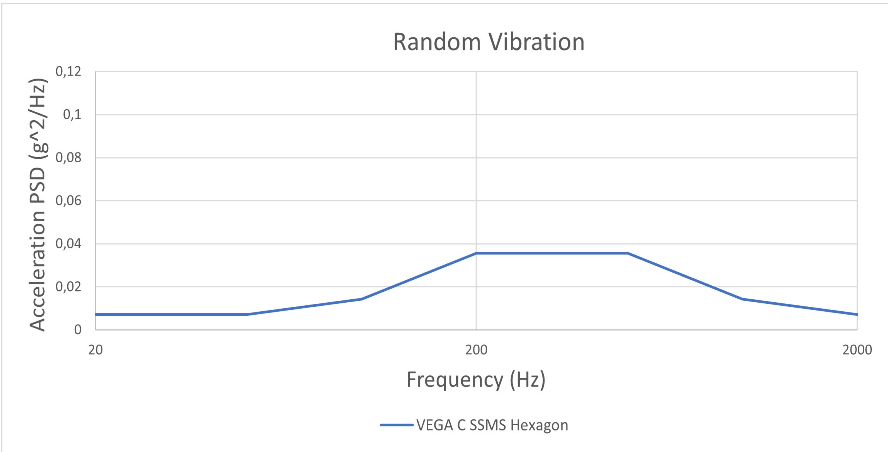


(a) SSMS Hexagon induced Shock Loads

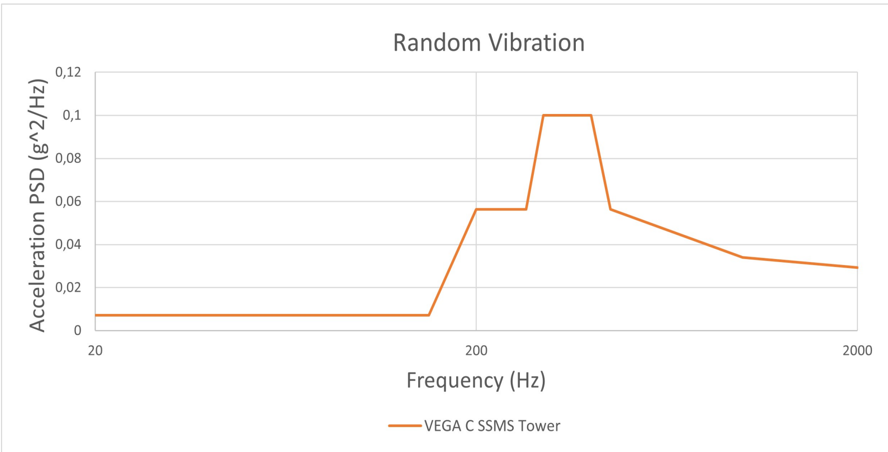


(b) Tower induced Shock Loads

Figure 2.20: VEGA C Shock Loads for SSMS



(a) SSMS Hexagon Random Vibration Maximum Predicted Environment



(b) Tower Random Vibration

Figure 2.21: VEGA C Random Vibration for SSMS

ELECTRON's unique Kick Stage is designed to deliver especially small satellites to orbits, and so loads perceived by the payloads are softer than the ones already saw, as illustrated in figures 2.22, 2.23 and 2.24. In [3] there aren't specification concerning the main frequencies.

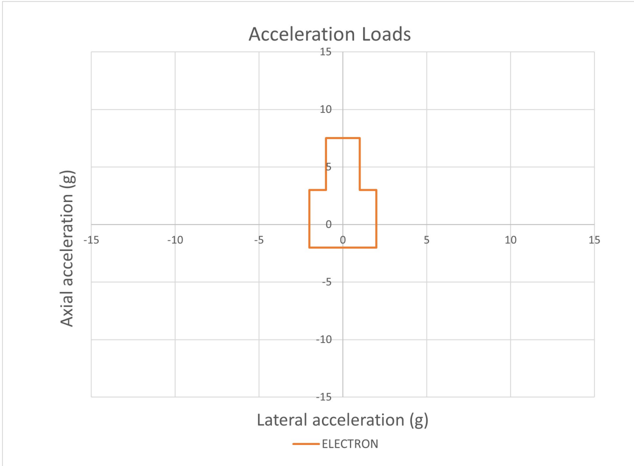


Figure 2.22: ELECTRON Loads Factor

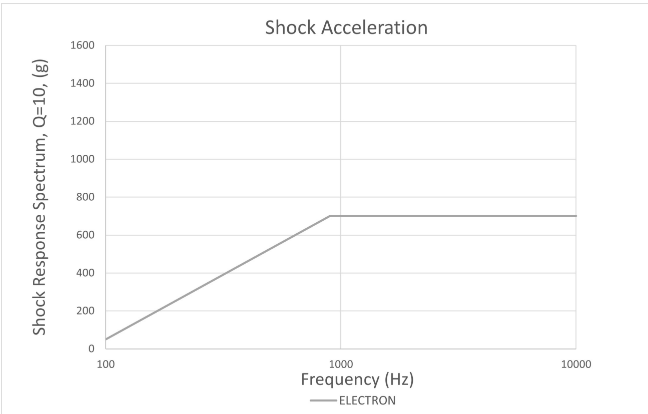


Figure 2.23: ELECTRON induced Shock Loads

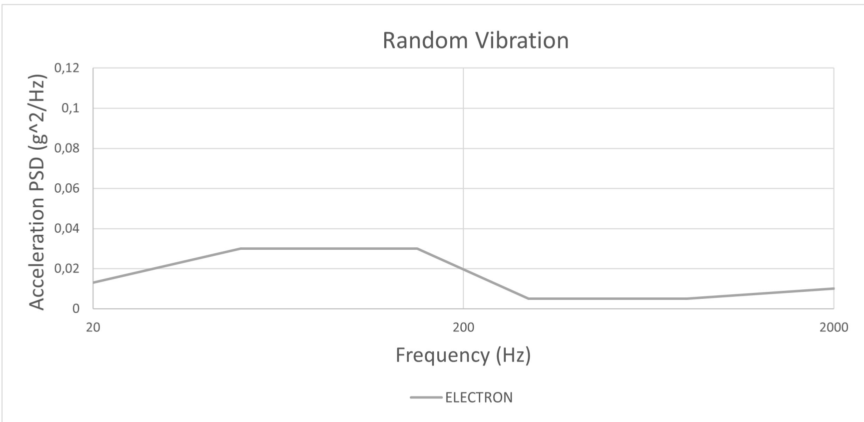


Figure 2.24: ELECTRON Random Vibration Maximum Predicted Environment

After the requirements of all selected launch vehicles have been analyzed, they are meshed to obtain the overall requirements. Satellite axial and bending mode fundamental frequencies should be greater than 115 Hz and mechanical switches to inhibit all satellite functionalities shall be thought. Mechanical requirements are reported in figures 2.25, 2.26 and 2.27.

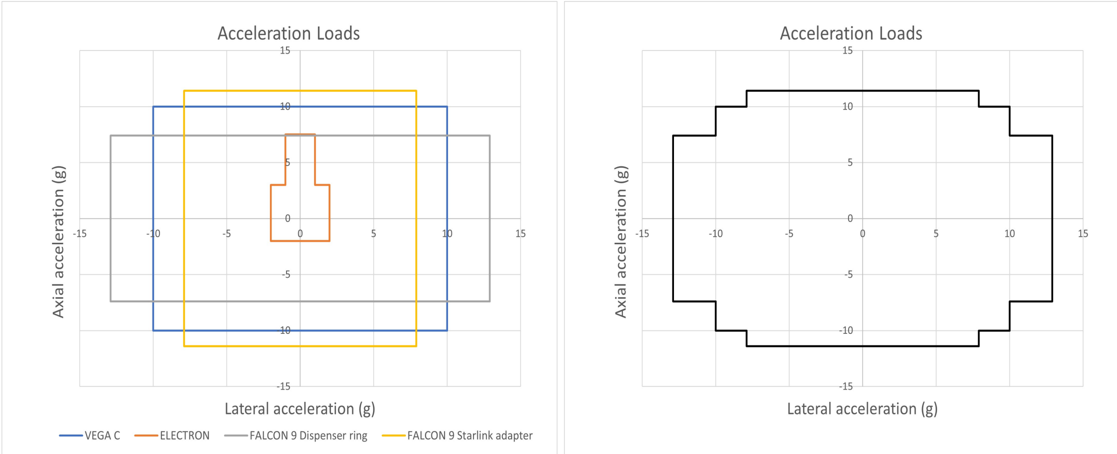


Figure 2.25: Launch Vehicles Loads Factor, all (left) and overall (right)

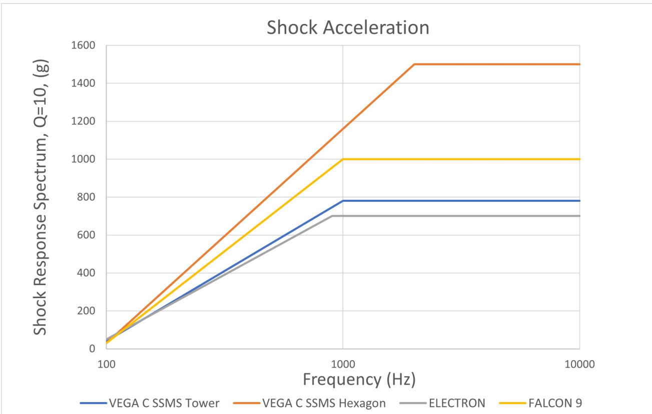
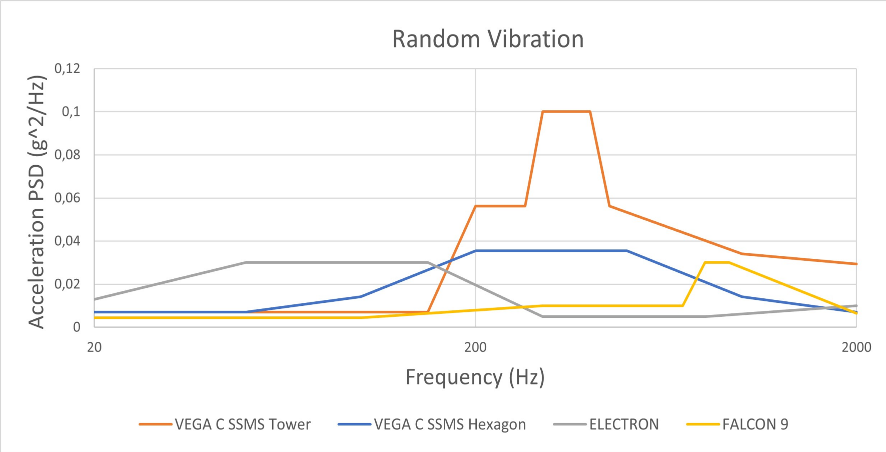


Figure 2.26: Launch Vehicles induced Shock Loads



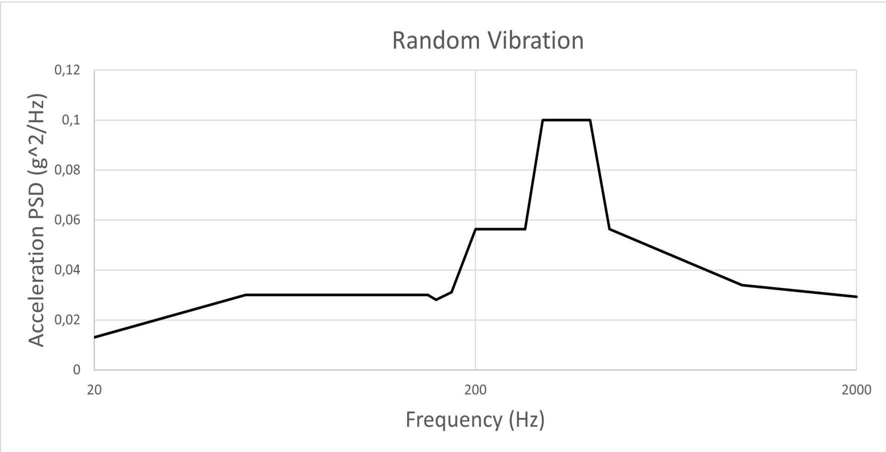


Figure 2.27: Launch Vehicles Random Vibration Maximum Predicted Environment, all (top) and overall (bottom)

Arianespace Small Spacecraft Mission Service and SpaceX Rideshare scheduled flights can be consulted in their web pages. These can give an idea of the most common orbit used in commercial activity and specifically in mission shared with SmallSat secondary payloads. Table 2.9 reports the flights opportunities.

Launch Vehicle	Date	Orbit	Altitude (km)
Falcon 9	December 2021	SSO	500-600
Falcon 9	March 2022	SSO	500-600
Falcon 9	June 2022	SSO	500-600
Vega C	Q3 2022	SSO	600
Falcon 9	October 2022	SSO	500-600
Vega C	Q4 2022	SSO	550
Vega C	Q4 2022	SSO	520
Falcon 9	April 2023	SSO	500-600
Falcon 9	June 2023	SSO	500-600
Soyuz	Q2 2023	SSO	400
Vega C	Q3 2023	SSO	600
Falcon 9	Q4 2023	SSO	500-600
Vega C	Q2 2024	SSO	700
Vega C	Q3 2024	SSO	750
Vega C	Q4 2024	SSO	750
Ariane 6	Q3 2025	SSO	500

Table 2.9: Arianespace and SpaceX scheduled flights

As already anticipated in section 1.1, Sun-Synchronous orbit are arranged so that every time that the satellite is overhead, the surface illumination angle on the planet underneath it will be nearly the same. This consistent lighting is a useful characteristic for satellites that image the Earth's surface in visible or infrared wavelengths. Special cases of the orbit are the *noon/midnight* orbit, where the local mean solar time of passage for equatorial latitudes is around noon or midnight, and the *dawn/dusk* orbit, where the local mean solar time of passage for equatorial latitudes is around sunrise or sunset, so that the satellite rides the terminator between day and night. Other advantages are that satellites in this orbit can help in monitoring activities near the poles, for low altitude they provide good ground resolution, they cover the entire globe on regular basis and provides repetitive coverage on periodic basis.

These reasons, together with the high availability of SSO commercial missions, make this orbit, in the range of 500 km to 600 km, an optimal solution for Earth Observation.

2.2.2 Dispenser

To develop a platform whose design can be suitable for most of the dispenser in the market, they will be analyzed in terms of allowable platform dimension and total envelope, which differ because the latter considers also the lateral thickness available for deployable features, such as solar panels.

Exolaunch EXOpod, ISIS ISIPOD, Rocket Lab Maxwell Dispenser, Planetary Systems Corporation CSD and Nanoracks NRDD are the products taken into account in this section, in 6U configuration, as said in section 1.4. All information have being gathered from [15], [9], [12], [13] and [14]. Platform dimension are reported in table 2.10, while total allowable envelope are represented in the following figures.

Product	Allowable dimension (mm^3)	Allowable mass (kg)
EXOpod	100x226,3x340,5	11
ISIPOD	100x226,3x340,5	12
Maxwell Dispenser	100x226,3x366	11
CSD	116,2x239,4x366	-
NRDD	100x226,3x366	12

Table 2.10: Platform dimension and mass for commercial dispenser

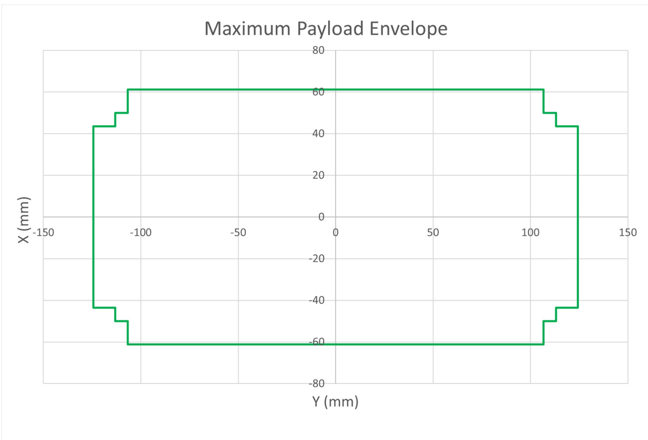


Figure 2.28: EXOpod's payload envelope

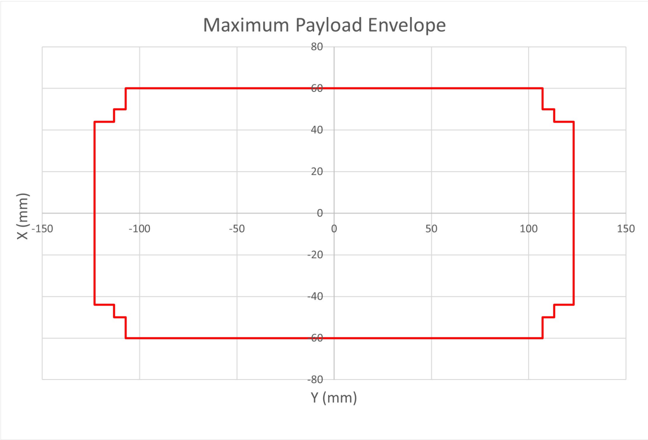


Figure 2.29: ISIPOD's payload envelope

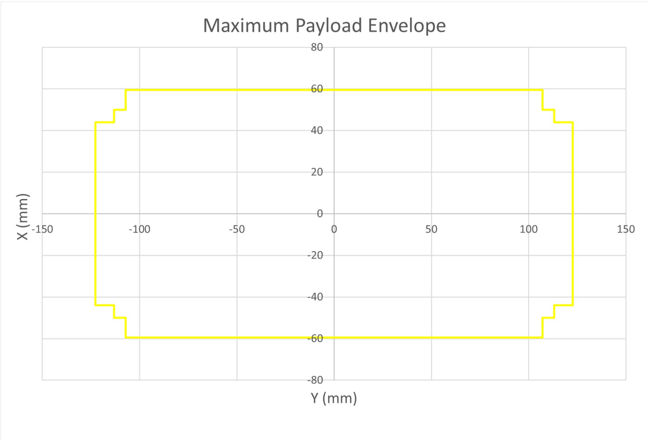


Figure 2.30: Maxwell Dispenser's payload envelope

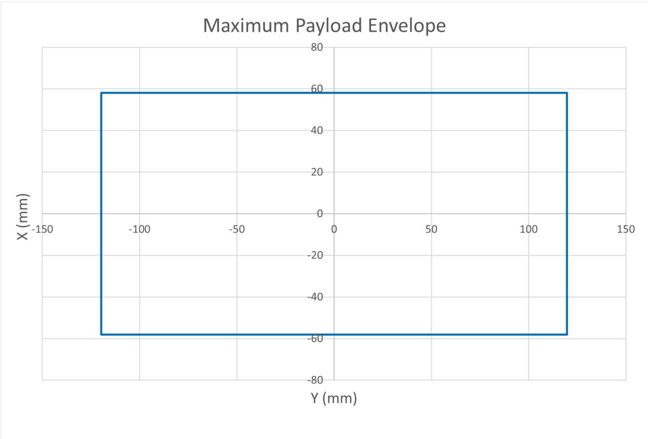


Figure 2.31: CSD's payload envelope

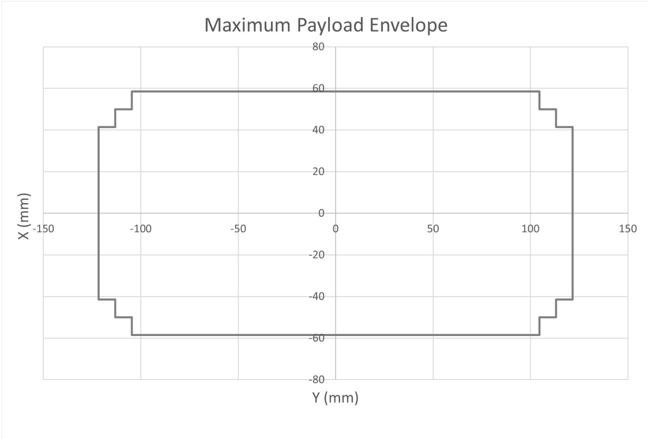


Figure 2.32: NRDD's payload envelope

For the CSD, platform dimension and total envelope are the same, because, unlike the other dispenser, no room should be left for rails. Planetary Systems Corporation does not indicate an allowable mass because "claiming a dispenser can accommodate a certain payload mass is not productive ... every payload has a unique dynamic response", and therefore only states that the payload response due to all loading shall not exceed $3,560\ N$.

Finally, the meshed payload envelope is illustrated in figure 2.33.As third dimension (along Z), the smaller should be taken, i.e. $340,5\ mm$. It guarantees the 6U platform can be hosted by all the mentioned dispensers.

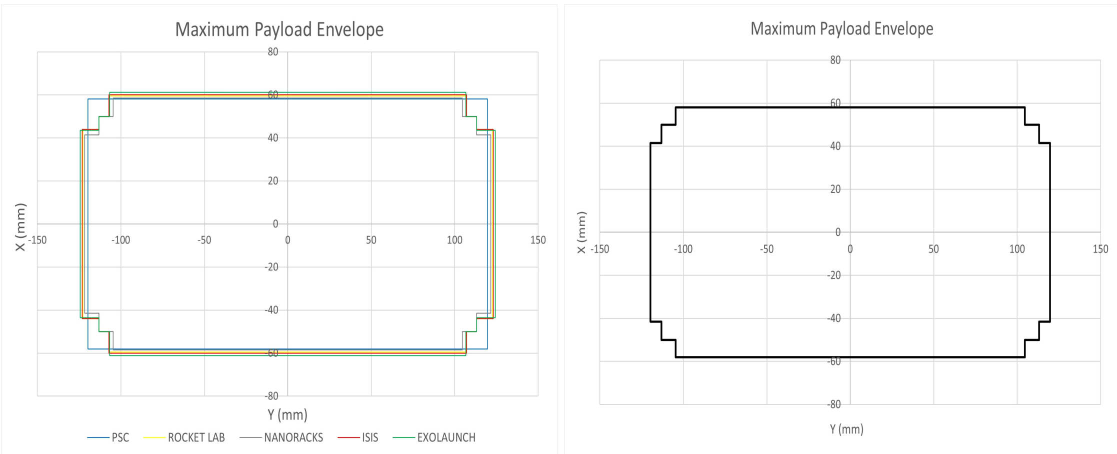


Figure 2.33: Dispenser payload envelope, all (left) and overall (right)

The inner rectangle represent the platform maximum dimension, reported together with the additional lateral thicknesses in table 2.11, and rails shall be designed to fit within it. The platform response due to all loading shall not exceed 3,560 N and its mass shall not exceed 11 kg.

Platform dimension	Additional volume (top-bottom)	Additional volume (left-right)
100x226,3x340,5 mm^3	8,1x209,3x340,5 mm^3	83x6,55x340,5 mm^3

Table 2.11: Overall platform dimension

2.3 Platform Architecture

The platform architecture studied to perform an Earth Observation mission is composed of the following subsystems:

- Electrical Power System;
- On-Board Computer & Data Handling;
- Telemetry Tracking & Command;
- Attitude Determination and Control System;
- Propulsion System;

The following sections describe the items that has been selected to constitute the platform, looking to comply with the high-level requirements underlined in the previous sections. These have been chosen by analyzing the market solutions and opportunities listed in chapter 3.3.

2.3.1 Electrical Power System

The Electrical Power System (EPS) is in charge of generating, storing and converting power to supply the satellite. It is composed of three elements:

- Power Conditioning and Distribution Unit;
- Battery Unit;
- Solar Panel Array.

Power Conditioning and Distribution Unit

As its name says, the PCDU converts the power arriving from the Solar Panels and the Battery in order to distribute it to all the subsystems and payload meeting their demands.The battery pack is needed as a secondary power source to provide power during the eclipses and to manage the power peak requests.
The power sources are connected to the main bus at 21-29V with a switching regulator controller that implements a charging profile for the battery.
Using secondary DC-DC converters make 4 isolated buses available at three common output voltages: one at 3V3, one at 5V and two at 12V. The converters are connected to the main bus through an EMI filter, that allows to reach an aggregated output power of 100W. Table 2.12 resume the four output rails information. A block diagram of the power rails configuration is shown in figure 2.34 and the main characteristic of the PCDU can be found in table 2.13.

Rail Output Voltage	Power Rating	Anti-Latch-Up Rails
3,3 V	20 W	1
5 V	30 W	2
12 V	40 W	1
12 V	40 W	3

Table 2.12: Output Buses characteristics

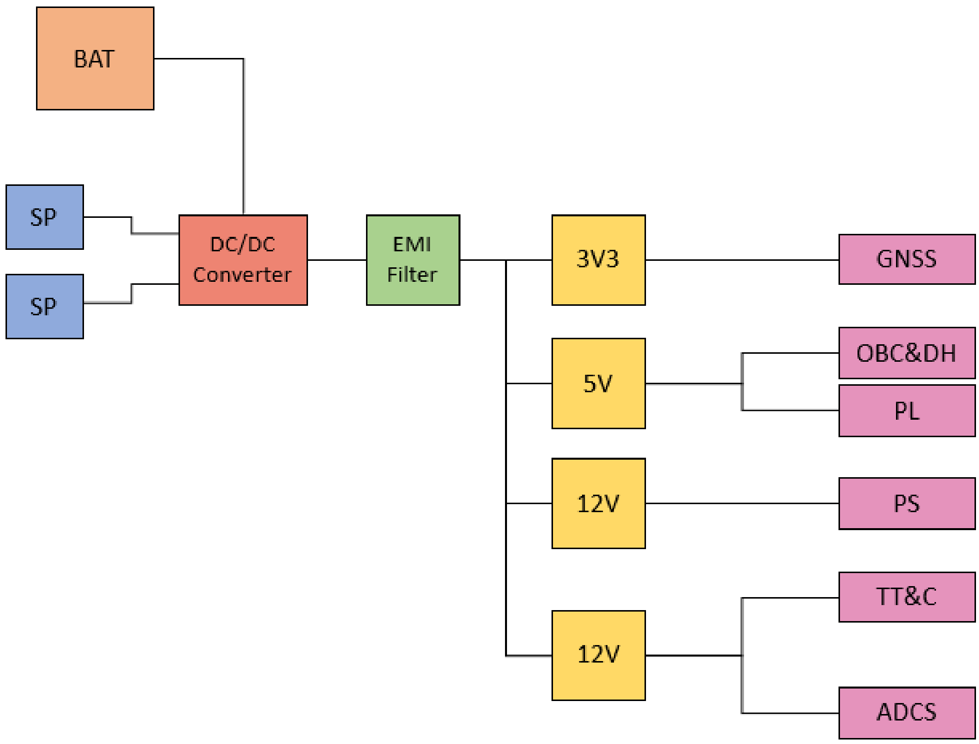


Figure 2.34: PDCU Preliminary Block Diagram

Solar Panels Rail	15-25 V
Battery Power Rail	21-29 V
Output Voltage	3V3, 5V, 12V
Operating Temperature	-20°C to +60°C
Radiation Tolerance	$>$ 30 krad
Mass	880 g
Volume	0.6 U
Power Consumption	6 W
Lifetime	$>$ 5 years

Table 2.13: PCDU parameters

Battery Unit

The battery unit is composed of two modules of Li-ion cells in a 8s1p configuration, i.e. one strings of 8 cells of 11.5 Wh each. The choice of using two modules depends on wanting a redundancy and to increase the confidence to meet the lifetime of the service even if 1 battery fails. Its design specification are mainly driven by the power budget in the eclipse mode analyzed in section 2.4.4. Then, the power requirement consider the mean eclipse duration and a DoD value of 40,00%, resulting in a capacity requirements @BOL of 28,12 Wh. Therefore the item selected greatly exceeds the requirements. This will allows the battery unit to manage the power peaks during the other modes.
The main characteristic of the battery unit are resumed in table 2.14.

Module Capacity	92 Wh
Discharge Operating Temp.	-20°C to +60°C
Charge Operating Temp.	0°C to +45°C
Mass	1310 g
Voltage	24,4-33,3 V
Lifetime	5 years
TRL	9

Table 2.14: Battery Unit parameters

Solar Panel Array

The SPA constitutes the energy source of the system, and it is part of the EPS of the platform, together with the Battery Unit and the PCDU.
The design of the SPA architecture is driven by the EPS needs and specific mission targets, in order to generate the required power and save mass and volume. With reference to the power budget reported in section 2.4.4, the proposed solution is characterized by:

- 2 wings configuration with 2 panels (6U sized) each;
- Available generated power @BOL of 85 W;
- Stowed envelope of 5 mm;
- Cells with a cover glass to reduce a degradation over time in the radiation environment;

Avg. Efficency	29,6% V
Number of cells	80
Power @EOL	80 W
Mass	1000 g
Bus Voltage	15,6-23,7 V
Lifetime	$>$ 3 years
TRL	9

Table 2.15: SPA parameters

2.3.2 On-Board Computer & Data Handling

The On-Board Software (OSW) runs on the CPU of the OBC&DH subsystem and is based on a Real-Time Operating System with hard real-time task scheduling. The application layer of the OSW is responsible for controlling the operating modes of the system as well as its subsystems. It controls the TMTC subsystem to guarantee communication with ground according to the requested planning. The lower layers of the OSW contain the interface drivers and communication stacks for the serval satellite subsystems. The OSW can receive commands from ground and store system telemetries that can be later retrieved for analysis. It implements the Packet Utilization Standard (PUS) according to ECSS-EST-70-41C for telecommands (TC) and telemetries (TM). It uses the PUS "on-board operation scheduling" service to manage a list of time-tagged commands to be executed even if communication with ground is not in place. Dedicated PUS services are implemented for every subsystem, providing a way to interact with the single subsystem from ground. Furthermore, the OSW implements a Fault Detection, Isolation and Recovery (FDIR) strategy: a set of monitoring services to ensure the environmental parameters of the system (power and thermal parameters) and the basic functionalities of the subsystems are correct. It is characterized by the storing capability on a non-volatile mass memory (at least 16GB), through dedicated NAND Flash memories connected to the FPGA and low and high data rate interfaces to the payload. In case the OSW detects a critical anomaly, it can either try to recover the proper functionality, or at least bring the system in a configuration (safe-mode) that minimizes the risk for the satellite. Characteristics of the OBC&DH can be found in table 2.16.

Input Voltage	5 V
Mass	500 g
Volume	0.5 U
Power Consumption	11 W
Operating Temperature	-20°C to +60°C
Radiation Tolerance	$>$ 30 krad
Lifetime	$>$ 5 years
TRL	8

Table 2.16: OBC&DH parameters

2.3.3 Communication System

The TT&C subsystem is in charge of handling command reception and telemetry and science data transmission in the Earth-to-spacecraft link. The hardware of TT&C includes:

- S-Band transceiver managing both downlink and uplink communications;
- S-Band diplexer;
- Coaxial cable for RF connections;
- S-Band Antenna;
- Power and control cables.

Figure 2.35 shows a high-level block diagram of the TT&C subsystem and its connections to the OBC&DH and EPS subsystems. Table 2.17 and 2.18 resumed the main characteristics of the TT&C solutions chosen.

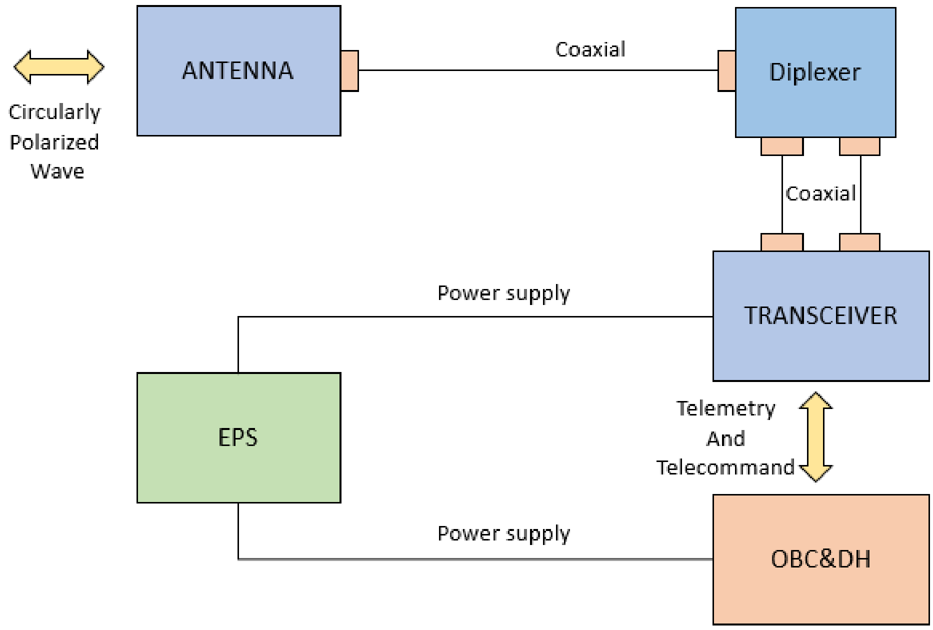


Figure 2.35: TT&C Preliminary Block Diagram

Band	S
Rx Frequency Range	2025-2100 MHz
Tx Frequency Range	2200-2290 MHz
Input Voltage	12 V
Transmitting Power	+30 dBm
Mass	275 g
Volume	0.25 U
Power Consumption	13 W (Tx+Rx) - 3 W (only Rx)
Operating Temperature	-20°C to +50°C
Lifetime	> 3 years
TRL	9

Table 2.17: Radio parameters

Band	S
Frequency Range	2025-2500 MHz
Gain	6 dBi
Polarization	RHCP
Mass	65 g
Volume	0.1 U
Operating Temperature	-30°C to +60°C
Lifetime	> 3 years
TRL	9

Table 2.18: Antenna parameters

2.3.4 Attitude Determination and Control System

The ADCS aims at stabilizing and orienting the satellite toward the desired directions during the overall mission despite of the external torques acting on it. The ADCS is composed of:

- 1 ADCS controller which provides the attitude control and determination using the inputs coming from the other ADCS components;
- 1 Star Tracker to determine the CubeSat attitude;
- 3 orthogonal Reaction Wheels for three-axis attitude control providing a high pointing accuracy;
- 3 orthogonal Magnetorquers to perform the desaturation makeunders of the reaction wheels;
- 1 IMU to detect movements and measure the intensity of movemnts in terms of acceleration and rotational speeds;
- 1 Magnetometer designed to measure low-strength magnetic fields;
- 2 Sun Sensors used to provide the sun positioning to orient the SPS
- GNSS receiver and antenna to acquire the attitude position from GPS. Performance of the ADCS can be found in table 2.19.

Pointing Accuracy	0,008°
Maximum Torque	0,002 Nm
Magnetic Moment	0,4-0,5 Am^2
Input Voltage	5-15 V
Mass	1300 g
Volume	07 U
Power Consumption	2 W
Operating Temperature	-40°C to +45°C
Radiation Tolerance	45 krad
Lifetime	$>$ 5 years
TRL	8

Table 2.19: ADCS parameters

2.3.5 Propulsion System

The propulsion system is utilized to provide the necessary Delta-V for station-keepin and de-orbiting if necessary. Additionally, the propulsion system can be used, with thrust vectoring, as a redundant measure for reaction wheel desaturation, which is primarily performed by magnetorquers. Performance of the selected system are resumed in 2.20

Propellant Type	LMP-103S
Propellant Mass	420 g
Thrust	100 mN
Specific Impulse	190 s
Input Voltage	12 V
Dry Mass	1100 g
Volume	1 U
Power Consumption	12 W
Radiation Tolerance	13 krad
Lifetime	5 years
TRL	9

Table 2.20: Propulsion System parameters

ΔV provided by 420 g of propellant for a 11 kg spacecraft is 73 m/s. Thus, confronting it with the required in section 2.4.1 there is a margin of 22,63 m/s to perform any contingency manoeuvre if needed.

2.3.6 Operative Modes

During its mission and along its orbit the satellite shall perform different tasks and withstand different events. For these reason it's useful to define the operative modes, that resume which sub-systems are active while the spacecraft carries out its duty.
The following modes have been identified:

- Science Mode (SCM): the satellite is fully operative. It's stabilized and points towards the Earth in order to get mission data and to provide mission service autonomously;

- Communication Mode (COMM): the satellite communicates with the ground station. It receives command in uplink and sends telemetry e mission data in downlink;

- Sun-Pointing Mode (SPM): the satellite solar panel assembly points directly towards the Sun for the purpose of recharging the battery. This is obtained by means of the ADCS that autonomously changes the satellite attitude;

- Safe Mode (SM): the satellite is placed in its safest configuration, in which the payload is off and the SPA points toward the Sun. Communication system is available in uplink mode.

- Eclipse Mode (EM): during the eclipse, the SPA is pointing where the sun is going to be when the spacecraft will exit from the eclipse event. Communication is on in both mode, ready to receive instruction and to send telemetry regarding its status;

- Station-keeping Mode (SKM): the propulsion system works actively to correct the spacecraft altitude.

2.4 System Budgets

This section provides an overview of the system budgets and the associated margins used for estimating the mass, delta-v, power, and link for the platform. Every margin considered follows the guidelines defined in "Margin philosophy for science assessment studies"[26].

2.4.1 ΔV Budget

The Delta-v budget defines the capabilities of the platform to perform the main mission manoeuvres. Considering the nominal operational scenario, station-keeping has been considered as the only manoeuvre the spacecraft will do and it has been calculated in the worst-case scenario of 500 km of orbit altitude and 3 years. Specifically, a manoeuvre to correct the apogee to the value of 500 km when reaching the threshold of 485 km, and another to then correct the eccentricity to 0.
De-orbiting manoeuvres has not been considered since the spacecraft will be albe to de-orbit thanks to the drag sources, as discussed in section 2.1.4.
GMAT software has been used to simulate this impulsive manoeuvres and its results can be seen in figure 2.36. Tolerances and step size are shown in 2.37.
ΔV budget results are reported in table 2.21.

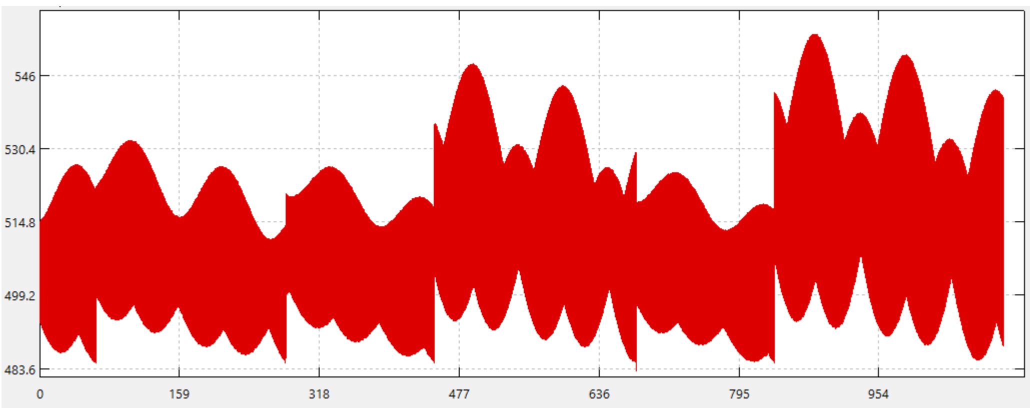


Figure 2.36: Altitude vs Mission Duration

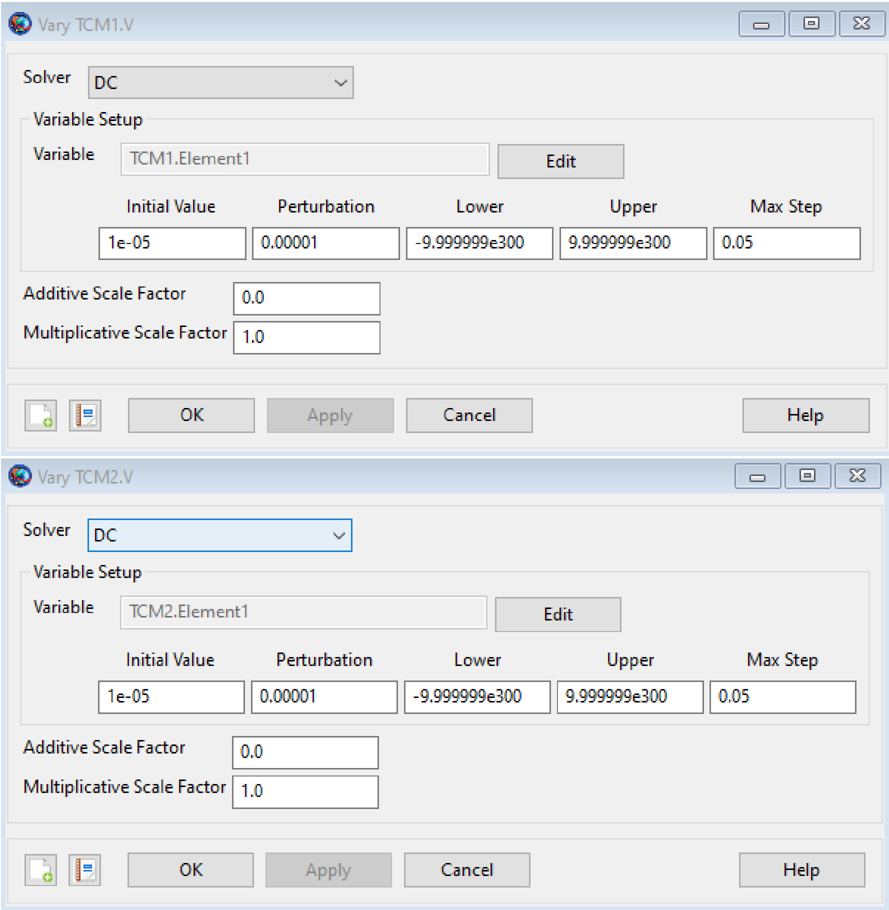


Figure 2.37: Simulation parameters for apogee (top) and eccentricity (bottom) correction

Manoeuvre	ΔV	Margin	Margined ΔV
Station-keeping	47,97 m/s	5%	50,37 m/s

Table 2.21: ΔV budget

2.4.2 Momentum Budget

The momentum budget aims to evaluate the perturbations which the spacecraft will be subject during its operative lifetime. The main sources of momentum in LEO environments are the following:

- Atmospheric Drag;
- Solar Radiation Pressure;
- Magnetic Torque;
- Gravity Gradient.

The perturbation deriving from the first mentioned source is calculated following the procedure described in 2.1.4 and adding a multiplication factor that is r_{cp}, the distance between the barycenter and the center of pressure of the spacecraft. Same thing for the one related to the Solar Radiation Pressure.

$$M_{atm} = \frac{1}{2}\frac{\rho C_D A}{m}\dot{r}^2 r_{cp}$$

$$M_{SRP} = p\frac{A}{m}kr_{cp}$$

The Magnetic Torque is due to the Earth magnetic field, which it's generated by electric currents that arise due to the movement of convection currents in the mixture of the Earth's outer core. The intensity (B) of this field varies from 30 μT to 65 μT. Its value for our orbit it's obtained starting from the typical value of 30 μT at 200 km and considering an inverse cubic trend.

$$B = 30(\frac{6578,16}{6878,16})^3 \mu T$$

$$M_{mag} = Bd_m$$

Where d_m is the residual magnetic dipole of the spacecraft. The Gravity Gradient is the effect caused by the gravity acting between the Earth and the spacecraft, due to the mass distribution of the latter. Its disturbing torque id calculated as follow.

$$M_{grad} = 3\frac{\mu}{a^3}I = 3n^2 I$$

All the contributions have been evaluated considering an orbit of altitude of 500km, as it is the worst cause due to the intensity of the magnetic field. Results are reported in table 2.22.

Atmospheric Drag Torque	1,12 x 10^{-8}
Solar Radiation Pressure Torque	3,41 x 10^{-8}
Magnetic Torque	2,62 x 10^{-6}
Gravity Gradient Torque	7,37 x 10^{-7}
Total Torque	3,41 x 10^{-6}

Table 2.22: Momentum budget

2.4.3 Link Budget

The calculation of a preliminary link budget is needed to assure that the selected TMTC solution can establish a communication link between the spacecraft and the ground station, both to transmit science and telemetry data and to receive command.

The link budget considers a reference altitude of 600 km as nominal altitude and maximum distance of 2045km at the edge of coverage (see figure 2.38). The former is a "Nominal" communication scenario, the latter is the "Worst Case" condition. Moreover, the link budget in the wort case scenario considers:

- Antenna gain reduced by 3 dB, assuming that the direction corresponds to the edge of the Half-Power Beam-Width of the antenna;
- Diplexer loss of 1.5 dB;
- Cable loss of 1 dB;
- Polarization loss of 0.5 dB (max loss) account for 3 dB axial radio on both antennas involved in the link;
- The link margin shall be more than 6 dB.

For the link performance, these are the assumption made for the ground station:

- Antenna Gain of 30.5 dB;
- Equivalent Noise Temperature of 19.6 dB;
- Transmitter Power of 12.5 dBW.

Calculations are reported in figures 2.39 and 2.40.
The selected items would permits to exchange up to 24 MB in the average communication window of about 390 seconds.

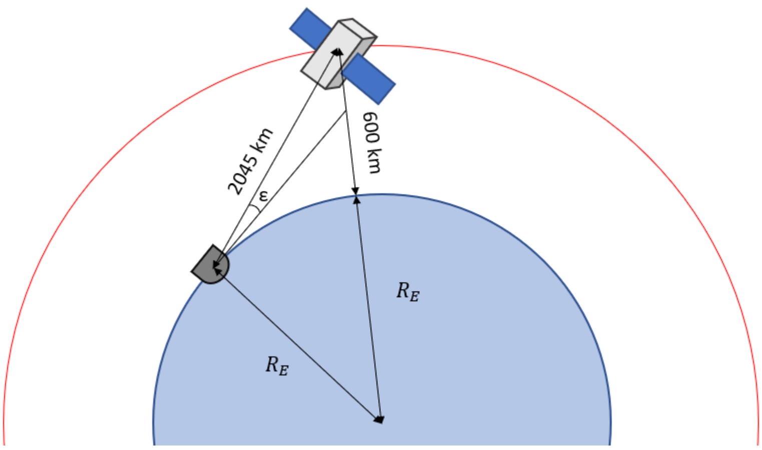


Figure 2.38: Link Budget scenario

#	Parameter	Symbol	Worst Case	Nominal	U
	TMTC Downlink (2200 - 2290 MHz)				
	INPUT PARAMETERS				
1	Distance from Earth	d	2045	600	km
2	Frequency	f	2,200		GHz
3	Boltzmann Constant	k	-228,6		dB/(K*Hz)
4	Datarate	r	512	512	kbps
5	Encoding		None		
6	Symbol Rate	r_s	512	512	ksps
7	Carrier tracking		Suppressed		
8	Speed of Light	c	299792		km/s
	TRANSMITTER PARAMETERS				
9	Transmitter Power	P_{tf}	0		dBW
10	Antenna Gain	G_t	6		dBi
11	Diplexer Loss	L_d	1,5		dB
12	Cable Loss	L_c	1		dB
13	Antenna pointing loss	$L_{point/GS}$	3	0	dB
	PATH PARAMETERS				
14	Atmospheric Losses at Zenith	L_{az}	0,046	0,042	dB
15	Elevation angle	α_{elev}	10	90	deg
16	Atmospheric Losses	La	0,26	0,04	dB
17	Free Space Path Losses	FSL	165,51	154,86	dB
18	Polarization Losses	LM	0,5		dB
	RECEIVER PARAMETERS				
19	Antenna Gain	G	30,5		dBi
20	Equivalent Noise Temperature	T_{eq}	19,6	19,6	dBK
21	Antenna pointing loss	$L_{point/GS}$	0,5	0,5	dB
	CHANNEL PERFORMANCE				
22	Telemetry Power at LNA Receiver	Pd	-135,3	-121,4	dBW
23	Received P_d/N_0	P_d/N_0	73,7	87,6	dBW-Hz
24	Energy per Bit-to-Noise Power-Density Ratio	E_b/N_0	16,63	30,51	dB
26	Uncoded Channel Bit Error Rate	BER	4,1E-22	0,0E+00	dB
25	Link Margin for 1e-6 BER (uncoded)	M	6,13	20,01	dB

Figure 2.39: Downlink Budget

#	Parameter	Symbol	Worst Case	Nominal	U
	TMTC Uplink (2025 - 2110 MHz)				
	INPUT PARAMETERS				
1	Distance From Earth	d	2045	650	km
2	Frequency	f	2,100		GHz
3	Boltzmann Constant	k	-228,6		dB/(K*Hz)
4	Bitrate	r	64	64	kbps
5	Encoding		None		
6	Symbol Rate	r_s	64	64	ksps
7	Carrier Tracking		Suppressed		
8	Speed of Light	c	299792		km/s
	TRANSMITTER PARAMETERS				
9	Transmitter Power	P_{tf}	3,00		dBW
10	Antenna gain	G_t	30,5	30,5	dBi
11	Antenna pointing loss	$L_{point/GS}$	0,5		dB
	PATH PARAMETERS				
12	Atmospheric Losses at Zenith	L_{az}	0,046	0,042	dB
13	Elevation angle	α_{elev}	10	90	deg
14	Atmospheric Losses	L_a	0,26	0,04	dB
15	Free Space Losses	FSL	165,11	155,15	dB
16	Polarization Loss	L_p	0,5		dB
	RECEIVER PARAMETERS				
17	Antenna Gain	G	6		dBi
18	Antenna pointing loss	$L_{poin/SC}$	3	0	dB
19	Antenna Temperature	T_{ant}	290		K
20	Receiver Noise Figure	F	5	4	dB
21	Equivalent Noise Temperature	T_{eq}	29,6	28,6	dBK
22	Diplexer Loss	L_d	1,5		dB
23	Cable Loss	L_c	1		dB
	CHANNEL PERFORMANCE				
24	Signal Power at LNA Receiver	P_t	-132,37	-119,19	dBW
25	Signal Power-to-Noise Ratio	P_t/N_0	66,61	80,78	dB-Hz
26	Energy per Bit-to-Noise Power-Density Ratio	E_b/N_0	18,5	32,7	dB
28	Uncoded Channel Bit Error Rate Incl. Link Margin	BER	2,93E-33	0,0E+00	dB
27	Link Margin for 1e-6 BER (uncoded)	M	8,04	22,2	dB

Figure 2.40: Uplink Budget

2.4.4 Power Budget

The platform generates up to 85 W at Beginning-of-Life (BOL) conditions and 80 W at End-of-Life (EOL). The power budget has been evaluated considering the power consumption of each subsystem according to the different operative modes. A relevant margin has been left for all the operative modes except for the eclipse mode, that will be managed by the battery unit. A 15 W power consumption has been considered for the Earth Observation payload, but it can be raised if necessary. Figure 2.41 shows the power budget for every mode, considering the Science Mode (SCM) the one concerning the use of the payload.

Description	EM	SM	SCM	SPM	SKM	COMM
Payload [12V]	0,00	0,00	15,75	0,00	0,00	0,00
ADCS [5V]	3,15	3,15	3,15	3,15	3,15	3,15
GNSS [3.3V]	0,95	0,95	0,95	0,95	0,95	0,95
OBC&DH [5V]	11,55	11,55	11,55	11,55	11,55	11,55
Propulsion System [12V]	0,00	0,00	0,00	0,00	12,60	0,00
TT&C TX+RX [12V]	0,00	0,00	0,00	0,00	0,00	13,65
TT&C RX [12V]	3,15	3,15	3,15	3,15	3,15	0,00
PCDU Internal Loss	6,60	6,60	6,60	6,60	6,60	6,60
PCDU Primary Side Loss	0,00	8,64	8,64	8,64	8,64	8,64
PCDU Secondary Side Loss	4,52	4,52	7,82	4,52	7,16	6,72
Total Required Power [W]	29,92	38,56	57,61	38,56	53,80	51,26
Max Available Power @EOL [W]	0,00	80,00	80,00	80,00	80,00	80,00
Power Margin [W]	-29,92	41,45	22,40	41,45	26,21	28,75

Figure 2.41: Operative Modes - Power Budget

2.4.5 Mass Budget

The mass budget considers as maximum platform mass the value of 11 kg as defined in section 2.2.2.
The mass distribution of the platform is reported in figure 2.42, which shows the percentage of each subsystem's mass with respect to the maximum allowable mass of 11kg. The platform can accommodate a payload mass up to 1.66 kg.

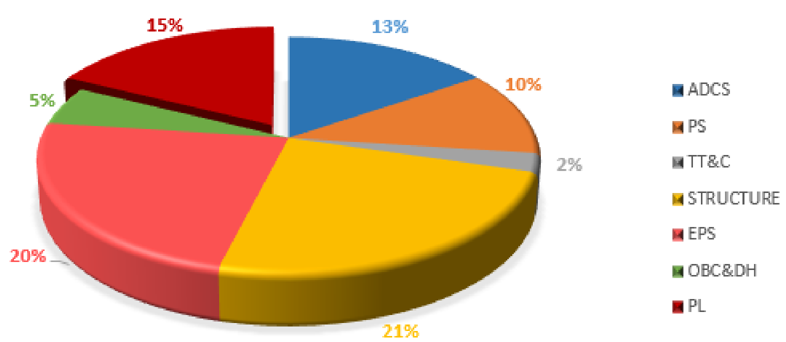


Figure 2.42: Platform Mass distribution

2.5 Platform Compatibility

The platform defined in section 2.3 has the possibility to accommodate different payload solutions.
Looking at the current optical payload market there is two possible way to chose an items. There are companies, such as *Dragonfly Aerospace*, *Simera Sense* and *SCS Space*, that produce complete solutions including sensor and lens, and other like *Raptor Photonics* and *Schneider Optics* that only one of the two components.

Figure 2.43 resumed the mentioned market opportunities for optical payload and their relevant characteristics.

Manufacturer	Product Name	Input Voltage	Power Cons.	Mass
Simera Sense	TriScape100	5 V	< 6 W	1,2 kg
GOMSpace	NanoCam C1U	3,3 V	< 1 W	0,3 kg
SCS Space	Chameleon	5 V	< 10 W	1,6 kg
DRAGONFLY	Mantis	5 V	< 4,6 W	0,5 kg
DRAGONFLY	Caiman	5 V	< 10 W	1,5 kg
Raptor Photonics (sensor) Schneider Optics (lens)	Owl 1280 TELE-XANAR	12 V	< 8 W	0,5 kg

Figure 2.43: Optical Payload for 6U CubeSat

The above listed solutions are found to be preliminary compatible with the platform architecture defined. Different voltage requirements can be resolved by changing the allocations in the power rails for subsystems that allow it.

Chapter 3

Procurement Engineering

The purchase of technical products is surely an important issue inside a company, because it directly affect the quality, the pricing and the delivery timing of the final product. Fields involved by this activity are many and varied, such as consumables, machinery, software, tools and systems, as well as shipping and general services. The main aspects the Procurement Engineer has to deal with are:

- Economic: everything has a budget in a project . The more it can be save respecting the requirements, the better;
- Timing: some items may be critical for the project, thus the shorter is their leading time, the sooner the project can be completed successfully;
- Strategic: having an understanding of the proposal process and a knowledge of the supply chain of the industry can speed up and optimize the project phases;
- Quality: the final product has to meet particular quality standards to perform the tasks it has been crafted for. To do that, it's necessary to chose the right tools, systems and processes;
- Logistic: purchasing from one country or another could present several issues linked to their specific handling and control laws.

He has the skill to think beyond the actual order, having a total view of the company needs, with specific regard to each of its departments. Thus, having the Procurement in a company can be an helpful strategy to support all the different departments in defining their needs and identifying the possible partners to negotiate with. This can be obtained only establishing a constant dialogue and a continuous exchange of information between them.

3.1 Proposal

Outside of ordinary expenses, the situation in which the Procurement Unit is most needed surely is he one when the company decided to respond and to participate to a "Call for Proposal", whether it is solicited by a commercial or governmental entity. When this occurs, three main phases can be identified in the Procurement's work:

- Bid Phase: in order to submit the best proposal, it is critical to identify the right sub-contractor and supplier, to define their involvements and their costs;
- Customer Contract Award Phase: once acquired the project, the Procurement has to select the products that perfectly match with the technical, contractual, programmatic and financial aspects;
- Sub-Contractor execution Phase: at this point the Procurement has to keep trace of the supplies, follow-up of the supplier and the sub-contractor while giving contractual support to the other involved departments.

During the first phase there are several actions the Procurement has to carry out. These includes the management of the NDA (*Non Disclosure Agreement*) regarding the companies it gets in touch with during the scouting activity, the management of RFI , RFQ and RFP (*Request For Information/Quotation/Proposal*) and the application of the product strategy in order to create the HM (*Hardware Matrix*), including the Make/Buy policy.

NDA Management

Usually, when discussing with another company, the parts are brought to talks about their projects and plan so that they can better understand the needs and the capabilities of them. These arguments involved the strategies, the investments and sometimes even the discoveries in terms of researches. That's why it's common to sign an NDA from both parts, a legal contract that outlines the information that they will shared between them,and at the same time it establishes the boundaries of their disclosure. The contents of the NDA are often the same, what can varies is the duration of the contract and the court whose law will be valid in the event of a dispute.

RFI/RFQ/RFP

These type of requests are sent to the companies that can develop goods and technologies for which you don't have the capabilities. They are specialized in their field and therefore during the bidding phase they can help you understand deeper

your project, giving you tips on the configurations. Doing this also clearly helps the potential sub-contractor to be put under contract, establishing a relationship between the companies and the a bond between the final product and the sub-system.
Typically, as its name says, the RFI is sent to obtain information you may need to proceed in your research and to program your project. It consists in a exchange of knowledge through emails, calls or meetings and if the results is positive, it is followed by a RFQ. It is an official documentation that present the offer for the good. It will specify how the item will be crafted, which will be the tests to which it will be subjected, the technical requirements and quality standards it will have to meet in order to be accepted and, moreover, worth as a contract there will be reported the delivery date, the type of payment and how the latter will be partitioned, and last but not least, for how long will the offer be valid. Finally, the RFP is the last variant and consist in an official document the company sends to potential partners, proposing to collaborate in the bigger context of a project where there may be the necessity to provide services so different that the single entities won't be able to satisfy.

Hardware Matrix

Once the previous processes are finished, it is possible for the Procurement to provide an estimation of the cost for the development of the space system of interest. It is resumed in the so called Hardware Matrix, where for every component will be indicated the product name, the manufacturer, the Quote and the Lead Time. This tool will permit to have an estimation, more or less accurate depending on the information, of the total effort in terms of time and price, highlighting all the suppliers to rely on.

The present work will concentrate on the Bid Phase. In relation to the mission assumed in chapter 2, suppliers of the main platform sub-systems has been identified and analyzed. As second step, a skimming of their products has been done, looking for the ones meeting the systems preliminary requirements.

3.2 Supplier Analysis

This section aims to outline an overview of the actual CubeSat industry supply chain. It has been mainly focused on the EU and US market, whose actors have been identified and evaluated through different criteria, as mentioned in the previous section.
The web sources used to investigate suppliers has been the *SmallSat Conference*,whose section named *Exhibit Hall* was essentially a showcase for any company that wanted to put itself under the spotlight and the *State of the Art of Small Spacecraft Technology* [1] on the NASA website, where a general explanation of CubeSats is given, while presenting the industry pioneers and the current technologies developed.
Other opportunities to get to know about new realities arise frequently throughout the year, during the space conferences or exposition, like this year's *International Astronautical Congress (IAC 2021)* or *Space Tech Expo Europe*. This events are though so that the representatives of every companies can share information about future projects and possible collaborations that may arise from the needs of these.

Through these different tools and opportunities, a first major scenario of the space industry suppliers focused in the SmallSat sector can then be defined. Figure 3.1 present all the different actors that has been discovered and contacted during the study.

Figure 3.1: SmallSat indutry Suppliers

Among the whole supply chain, it is possible to define some players differently with respect to others. There are companies which are specialized in the development and production of a particular good, of a single subsystem, and companies whose experience and path has led them to produce multiple subsystems and eventually integrated space platforms. Among the latter type, there is not such a company that produces every sub-system of a platform by itself, but usually relies on specialized companies to develop the goods that their production capacity cannot produce.

When evaluating suppliers, valuable information relates to corporate strength, and thus years in business, growth, and number of employees; logistics, and thus their geographic location; industry experience, and thus the number of projects they have participated in and the government entities they have partnered with.
Since Argotec is used to work with NASA and ESA, for a potential partner/supplier, having the same knowledge of the standards of these two agency is an important attribute during its evaluation. This can only be acquired when directly collaborating with them.
Corporate strength is potentially a guarantee of successful collaboration; where an emerging team might fail, an established company already knows how to address and resolve emergencies.

3.2.1 Competitors

Among the research studies carried out by the R&D department and the consulting activity performed by the other Argotec's units, the company's current main business concerns the development of integrated CubeSat platforms. Thus it is possible to classified Argotec as being part of the second category mentioned before, and define the other companies being part of the same group as *competitors*. That is because during the bid phase these companies will likely compete to try and win the project object of the bidding.
In this category there can be included *Blue Canyon Technologies*, *Space Inventor*, *Spacemanic*, *Surrey Satellite Technology*, *AAC Clyde Space*, *GomSpace*, *ISISpace Hyperion Technologies*, *Berlin Space Technologies* and *EnduroSat*. Table 3.1 resumed the main information of Argotec's competitors.

Figure 3.2: Argotec's HAWK platform

Company Name	Country	Employees	Heritage	Products
Blue Canyon Technologies	US (CO) 2008	48	>15 missions NASA, APL ...	ADCS, EPS, TT&C
Space Inventor	DK 2015	19	ESA	ADCS, EPS, OBC&DH, TT&C
Spacemanic	SK 2014	7	ESA	ADCS, TT&C, OBC&DH
Surrey Satell. Technologies	UK 1985	>300	>50 missions NASA, ESA ...	ADCS

Company Name	Country	Employees	Heritage	Products
ACC Clyde Space	SE 2005	>100	9 missions NASA, ESA ...	EPS, TT&C, OBC&DH
GomSpace	DK 2007	>200	7 missions ESA	ADCS, EPS, OBC&DH TT&C
ISISpace	NL 2006	>100	10 missions ESA ...	ADCS, TT&C, OBC&DH EPS
Hyperion Technologies	NL	-	-	ADCS, PS
Berlin Space Technologies	DE 2010	38	>100 systems provided	ADCS
EnduroSat	BG 2015	>70	2 missions ESA	EPS, TT&C, OBC&DH
NanoAvionics	US (IL) 2014	>100	>90 missions	EPS, TT&C, OBC&DH, PS, ADCS

Table 3.1: Competitors valuable information

3.2.2 Specialists

The companies that were not mentioned in the previous table are therefore part of category named *specialists*, because they focus their studies and production on single or multiple sub-systems without engaging in the development of integrated platforms. Table 3.2 resumed the main information of the market's specialists. Since Argotec's expertise permits the company to develop solution like power distribution unit and on-board computer itself, the specialists considered mainly focus on the other sub-systems.

Company Name	Country	Employees	Heritage	Products
Cube Space	ZA 2017	24	>80 systems NASA...	ADCS
New Space System	ZA 2014	>30	NASA, ESA, ...	ADCS
CubeSat Pointing	BE 2020	8	ESA	ADCS
Benchmark	US (VT) 2017	>30	-	PS
Busek	US (MA) 1985	>50	NASA, ESA, ...	PS
Enpulsion	AT 2016	49	-	PS
T4i	IT 2014	30	ESA	PS
VACCO	US (CA) 1954	>200	20 system NASA, ...	PS

Company Name	Country	Employees	Heritage	Products
Tethers	US (WA) 1994	>50	NASA	PS, TT&C
Exotrail	FR 2017	30	-	PS
Lift Me Off	UK 2018	9	ESA	PS
CU Aerospace	US (IL) 1998	20	-	PS
DHV	ES 2013	>100	-	EPS
MMA Design	US (CO) 2007	40	17 system NASA	EPS
Pumpkin	US (CA) 1995	>20	100 systems	EPS
Ibeos	US (VA) 2013	13	NASA	EPS
ABSL	UK	9000	>130 systems	EPS
SAFT	AU 1918	>1000	-	EPS
Anywaves	FR 2017	22	ESA	TT&C

Company Name	Country	Employees	Heritage	Products
Sirlinks	FR 2011	>100	ESA	TT&C
Innoflight	US (CA) 2004	>50	-	TT&C
Arralis	IE 2013	>20	ESA	TT&C
IQ Wireless	DE 1999	>40	-	TT&C
Helical Comm. Tech	US (FL) 2012	5	-	TT&C
L3 Harris	US (FL) 1890	>10000	NASA,...	TT&C

Table 3.2: Specialists valuable information

3.3 Strategy

This section wants to present and lists the great majority of the products the companies previously analyzed have the capabilities to develop. Every sub-system of a CubeSat platform have been researched in the market, looking for the items that comply the high-level requirements underlined in section 2.1. By doing these, the main supplier has been mapped, so that the company could understand to which one it can rely on.

The products have been discovered through a scouting activity with a particular focus on the items that can be defined as a COTS. This term stays for *Commercial Off-The-Shelf* and identifies those products, hardware or software, that are already characterized by a basic configuration, already studied and successfully proven. The choice of this type of products can imply a saving of time and money, both due to the absence of a complete study to begin with, typical of custom solutions. However, these items, even if already defined, will always need at some changes, that can be minor or major and thus changing the total price and lead time in a relative way.
Besides this definition, the goods are characterized by the TRL (*Technology Readiness Level*) scale. It has been developed and introduced by NASA during the 1970s, and measure the technology from 1 to 9, where TRL1 is the stage in which the scientific principle has been observed and formulated, while TRL9 is the stage that denote a system which has been flight proven trough successful mission operations with documented results, as explained in figure 3.3.

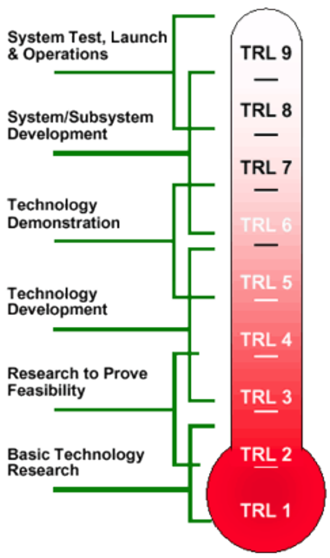


Figure 3.3: NASA TRL scale [25]

3.3.1 Electrical Power System

As already said in section 3.2.2, since Argotec have the capability to develop and produce in-house the Power Conditioning & Distribution Unit, thus being part of the EPS, its market opportunities has not been investigated.
Looking to the Argotec's PCDU characteristics, commercial solutions for the Solar Panel Assembly and the Battery Unit has been searched. The main strategic supplier has been mapped as in figure 3.4 and 3.6, and their products, which could be interesting for our mission, are resumed in figure 3.5 and 3.7.

Battery Unit

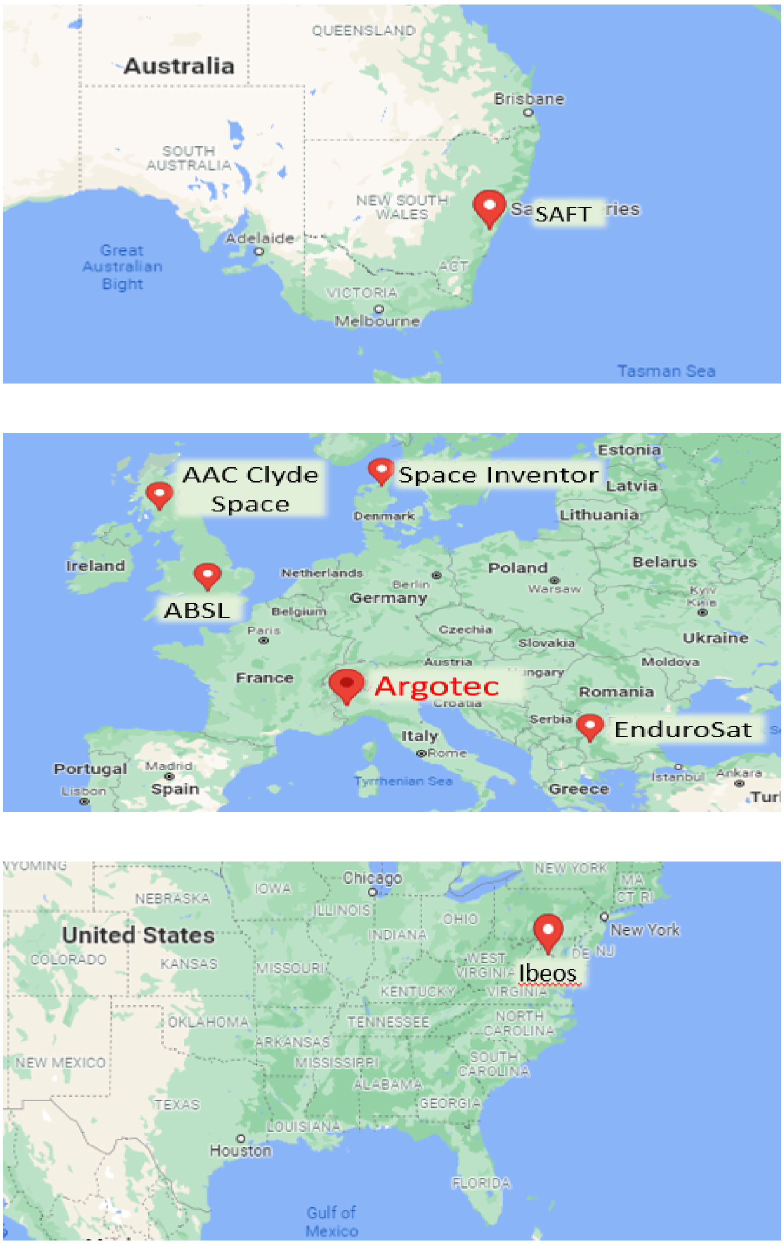


Figure 3.4: Battery Unit - Strategic Supplier

Manufacturer	Product Name	TRL	Capacity @BOL	Volume	Mass
AAC Clyde Space	OPTIMUS series	9	30-80 Wh	< 0,5 U	250-700 g
Berlin Space Technologies	BAT-110	-	~ 180 Wh	2 U	3 kg
EnduroSat	EPS I Plus	9	~ 20 Wh	-	< 300 g
EnduroSat	Battery Pack	9	~ 80 Wh	-	<1,5 kg
Pumpkin	Lithium Battery Module	-	up to 100 Wh	-	-
Space Inventor	BAT100-P3	9	~ 80 Wh	< 0,5 U	< 700 g
Ibeos	Modular SmallSat Battery		~ 50 Wh	< 0,3 U	< 400 g
Ibeos	B28-135		135 Wh	< 1 U	< 1 kg
SAFT	VES16	9	~ 20 Wh	-	< 120 g
ABSL	18650NL	9	~ 120 Wh	< 0,5 U	< 1 kg

Figure 3.5: Battery Unit solutions

Prices for the battery unit approximately range from 5.000 €, for COTS items, to even 60.000 € for custom solutions. AAC Clyde Space, EnduroSat, Space Inventor and Ibeos propose COTS solution which can be compatible with Argotec's PCDU voltage specifications. ABSL and SAFT have a great know-how a years of experience specifically in this field and can provide custom batteries for every type, as well as Ibeos.

Solar Panel Assembly

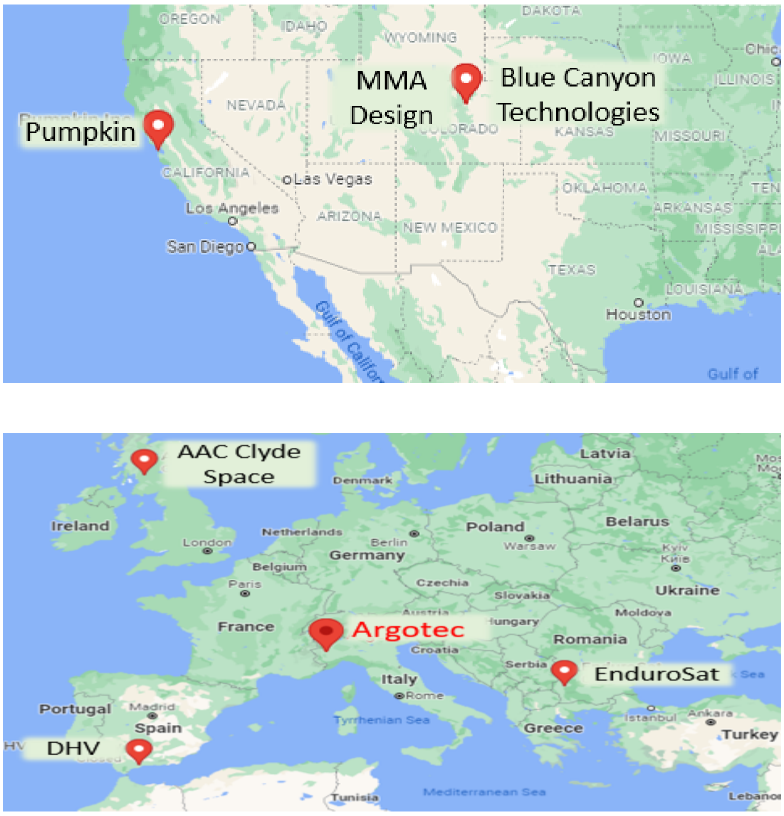


Figure 3.6: Solar Panel Assembly - Strategic Supplier

Manufacturer	Product Name	TRL	Panel Size	Power Gen. @BOL	Thickness	Mass
AAC Clyde Space	PHOTON	9	3 U	9 W	< 4 mm	135 g
Blue Canyon Technologies	-	9	6 U	24 W	-	-
DHV	6 U double deployable	9	6 U	20 W	< 2mm	< 300 g
EnduroSat	6U Deployable Solar Panel	9	6 U	19,2 W	< 2 mm	285 g
GOMSpace	NanoPower TSP		6U	15 W	-	250 g
MMA Design	HAWK	8	6U	23 W	< 4 mm	< 300 g
Pumpkin	Fixed Solar Panels	9	3U	7 W	<1 mm	160 g
SPACEMANIC	RA	9	1 U	2,3 W	< 2 mm	50 g

Figure 3.7: Solar Panel Assembly solutions

Prices for the Solar Panel Assembly are driven by the number of cells which constitute it, thus its dimension and power requirements. For this reason it is easier to find already designed solution that may surpass the mission needs and being cheaper at the same time. For the needs specified in section 2.4.4 the price range is between 100.000 € and 180.000 €. For the assumed mission it was not taken into account the use of SADA (Solar Array Drive Assembly), which would definitely make the price inflate.

3.3.2 Communication System

There are a lot of different solutions when talking about the communication subsystem. It is essential to understand the frequency ranges that would be useful based on the orbit type. Increasing the frequency range of the items typically their price.
For the tasks of the communication system of the assumed mission it was considered useful to focus on the VHF, UHF and S bands. The main strategic supplier has been mapped as in figure 3.8 and 3.10, and their products, which could be interesting for our mission, are resumed in figure 3.9 and 3.11.

Radio

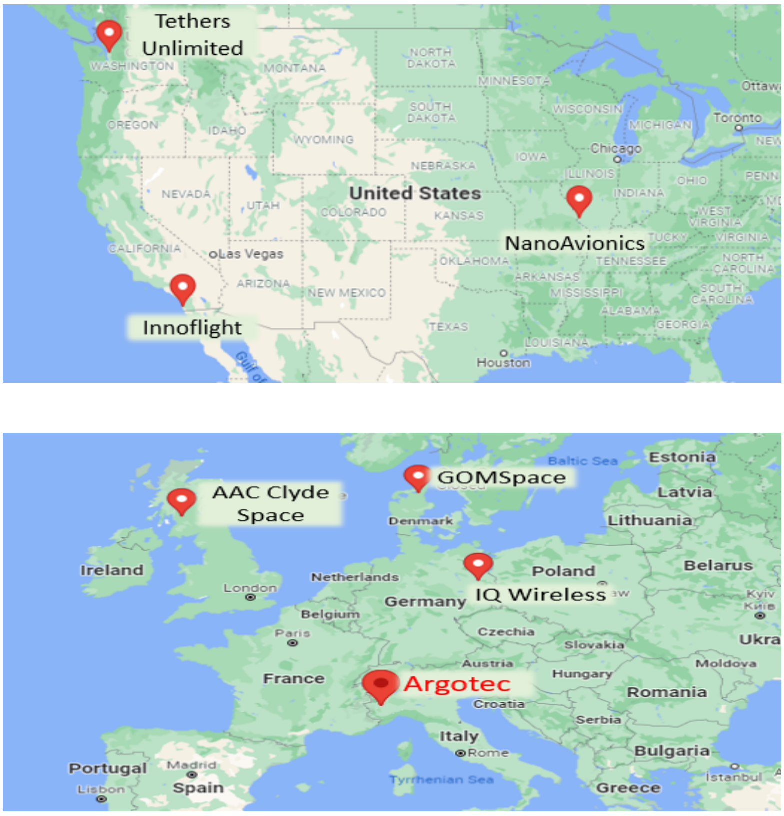


Figure 3.8: Radio - Strategic Supplier

Manufacturer	Product name	TRL	Type	Band	Input Voltage	Data Rate	Mass	Volume
Tethers Unlimited	SWIFT-UTX		Transmitter	UHF	9-34.6 VDC	-	< 300 g	< 0,4 U
ACC Clyde Space	PULSAR-STX	9	Transmitter	S	6-12 VDC	up to 10 Mbps	< 100 g	< 0,2 U
ACC Clyde Space	PULSAR-VUTRX	9	Transceiver	VHF/UHF	3.3-5 VDC	9.6 kbps	< 100 g	< 0,2 U
EnduroSat	S-BAND TRANSMITTER	9	Transmitter	S	9-15 VDC	Up to 17 Mbps	< 200 g	< 0,2 U
GOMSpace	NANOCOM AX2150		Transceiver	S	5.5 VDC	Up to 90 kbps	< 100 g	< 0,1 U
GOMSpace	NANOCOM AX100	9	Transceiver	VHF/UHF	3.4 VDC	Up 38.4 kbps	< 100 g	< 0,1 U
ISISpace	TXS	9	Transmitter	S	7-20 VDC	Up to 10 Mbps	< 150 g	< 0,2 U
NanoAvionics	SATCOM TP0	9	Transceiver	VHF/UHF	5 VDC	-	< 100 g	-
IQ Wireless GmbH	SLink-PHY	9	Transceiver	S	7-18 VDC	Tx up to 4 Mbps Rx up to 64 kbps	< 300 g	0,5 U
Innoflight	SCR-100		Transceiver	S	9-13 VDC	Tx up to 2 Mbps Rx up to 100 kbps	< 400 g	< 0,5 U
L3Harris	CXS-1000	9	Trasponder	S	28 VDC	Tx up to 20 Mbps Rx up to 1 Mbps	< 1,5 kg	1 U

Figure 3.9: Radio solutions

Prices for Radio solutions approximately range from 4.000 € to 20.000 € for COTS items, of the type of receiver, transmitter and transceiver.
SDR (Software Defined Radio) instead are characterized by higher costs, reaching the range between 100.000 € and 200.000 €. This is due to the higher flexibility of the solution, where common hardware components like mixer, filter, amplifiers etc., are implemented at the software level.

Antenna

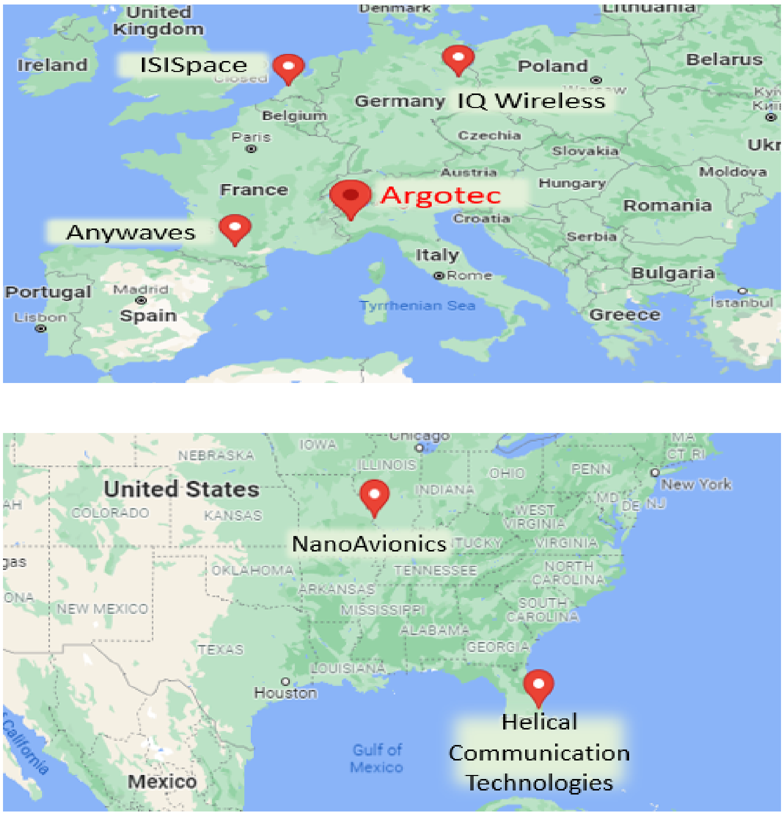


Figure 3.10: Antenna - Strategic Supplier

Manufacturer	Product name	TRL	Type	Band	Gain	Mass	Dimension
AAC Clyde Space	PULSAR-SANT		Patch	S	~ 7 dBi	< 50 g	80x80
EnduroSat	UHF Antenna III	9	Deployable Omnidirectional	UHF	< 1 dB	< 100 g	-
EnduroSat	S-Band Antenna Commercial	9	Patch	S	~ 7 dBi	< 100 g	-
GOMSpace	AM2150P/PS		Patch	S	~ 7 dBi	< 100g	-
GOMSpace	ANT-6F VHF		Deployable Omnidirectional	VHF	< 1 dB	< 100 g	200x100
IQ Wireless GmbH	S Band Antenna	9	Patch	S	~ 6 dBi	< 100 g	100x100
Helical Comm Tech	QHA	9	Deployable	UHF	~ 3 dBi	< 200 g	0,4 U
ISISpace	Deployable Antenna System	9	Deployable	VHF/UHF	-	< 50 g	0,3 U
ISISpace	S Band Patch Antenna		Patch	S	~ 7 dBi	< 50 g	80(Diameter)
Anywaves	S-Band TT&C Antenna	9	Patch	S		< 200 g	80x80

Figure 3.11: Antenna solutions

Prices for the antennas in the market are similar to the radio's ones, ranging from 3.000 € to 12.000 €. Helical Comm. Tech. is specialized in the design of deployable antennas, while the other mainly propose "patch" antennas, which are very useful when trying to save space. It should be noted that Anywaves and IQ Wireless has the capacity to mass produce this type of equipment.

3.3.3 Attitude Determination & Control System

Technologies for the attitude control are numerous, the chosen configuration strongly depends on the application and thus the accuracy the spacecraft needs during its mission in order to perform its tasks.

The ADCS market gives two possible solutions: buying all the different components to assemble or buying an integrated system.
Integrated solutions are more cost and time effective, for this reason the scouting focused on these.

Figure 3.12: ADCS - Strategic Supplier

Manufacturer	Product Name	TRL	Pointing Acc.	Mag. Moment	Input Voltage	Power Cons.	Volume
Blue Canyon Tech	XACT-50	8	<< 1°	0,2 Am^2	12V	3W	0,75 U
CUBE Space	3-Axis	9	< 1°	0,48 Am^2	3.3V - 5V	3W	0,75 U
CubeSat Pointing	ARCUS ADCS	9	< 1°	-	3.3V - 5V	1,4W	0,75 U
GOMSpace	ADCS assembly	-	< 1°	-	3.3V - 5V	6,9W	0,75 U
Space Inventor	ADCS-R3	9	< 1°	0,2 Am^2	7-28V	3W	0,4 U
Hyperion Tech	iADCS400	9	<< 1°	0,4 Am^2	5V	2W	0,7 U

Figure 3.13: Integrated ADCS solutions

There are not many supplier which propose integrated solutions for this subsystem as figure 3.13 shows, but anyway, the ones that can be found have demonstrated to be reliable as their TRL assure.
Prices for these type of technologies ranges from 80.000 € to even 200.000 € considering the best sensors to achieve the maximum accuracy.

3.3.4 Propulsion System

As per the communication system, the propulsion system present many possibilities in the CubeSat market. The type of technologies implemented is crucial and depends on the specific needs of the mission. In this case, chemical thrusters have been scouted, looking for high thrust and medium specific impulse, to perform station-keeping manoeuvres. Supplier of the discussed sub-system and their products are shown respectively in 3.14 and 3.15.

Figure 3.14: Propulsion System - Strategic Supplier

Manufacturer	Product Name	TRL	Sp. Impulse	Thrust	Input Voltage	Power Cons	Wet Mass	Volume
Benchmark Space Systems	STARLING	-	100-150 s	10 mN-1 N	-	15W	1,5 kg	1 U
Benchmark Space Systems	HALCYON	9	150-300 s	100 mN-22 N	-	15W	1,5 kg	1 U
CU Aerospace	PUC	6	~ 100 s	4,5 mN	12V	15 W	1,5 kg	1 U
CU Aerospace	MPUC	4	~ 200 s	160 mN	12V	3 W	2,462 kg	1,5 U
GOMSpace	NanoProp20000	9	~ 50 s	10 mN	5V -12V	20 W	0,380 kg	0,5 U
Tethers	HYDROS-C	9	~ 300 s	> 1,2 N	-	25 W	2,7 kg	2,5 U
VACCO	PUC	-	~ 100 s	5,4 mN	5 V	15 W	1,5 kg	1 U
VACCO	MiPS	9	~ 200 s	100 mN	12 V	12 W	2,5 kg	1,5 U
LMO	LEROS-MHTP	6	~ 150 s	350 mN	28 V	<50 W	3,86 kg	2 U
BUSEK	BGT-X5	-	200-250 s	500 mN	12 V	-	1,5 kg	1 U

Figure 3.15: Propulsion System solutions

All the suppliers identified as potential resources are located in the US. They all have years of experience in the industry and propose different technologies besides the chemical ones.
The scouted solutions price ranges from 100.000 € to 250.000 €, for already designed configuration, while the custom ones could even reach 500.000 €.

3.4 Hardware Matrix

Finally, considering the sub-system described in section 2.3, chosen from the items scouted in section 3.3, an Hardware Matrix can be developed. This helps define the total cost of the platform for the assumed mission, as well as the time needed to delivery of every systems. The latter it's fundamental to prepare the overall project phase, but this is the project manager's job.
Figure 3.16 shows the total price of the platform, as well as the one of every single sub-system considered to assemble it.

Sub-System	Item	Quotation	Lead Time
ADCS	ACS	191.000,00 €	9 months
	GNSS Receiver	5.450,00 €	2 months
	GNSS Antenna	9.000,00 €	5 months
TT&C	Radio	15.000,00 €	6 months
	Antenna	4.000,00 €	6 months
EPS	SPA	140.000,00 €	11 months
	PCDU	150.000,00 €	11 months
	Battery	10.000,00 €	2 months
OBC&DH	OBC&DH	90.000,00 €	11 months
PS	PS	$ 160.000,00	10 months
STRUCTURE	Structure	15.000,00 €	3 months
Total Price	Platform	767.050,00 €	11 months

Figure 3.16: Hardware Matrix - EO Platform

Prices of the PCDU and the OBC&DH certainly have an important weight in the total. This is mainly due to the fact that Argotec's products has been designed for Deep Space missions, where the environment and thus the requirements and qualification of every components are much aggressive. Typical prices for these system could be widely lowered for specific LEO design. Therefore components like the ADCS, the PS and the SPA mainly driven the order of magnitude of the total platform price.
As the Lead Time it has been taken the higher between the sub-system, but it is important to specify that this isn't the TTM (*Time To Market*) of the platform. It will needs more months for the assembling to be completed and fully tested before being ready to be delivered to a costumer. The analysis of the time and money that this phases would need is not the subject of the following work and therefore it has not been evaluated.

Conclusion

This work has had several results, both in the field of Procurement and System Engineering. For the second one, high-level requirements for what concerns the space environment of LEO orbits have been identified, in particular for altitudes ranging from 500 km to 600 km. These can be used as a starting point for further study of the mission under study, or for undertaking the study of future missions involving similar considerations.
Moreover, the architecture of a CubeSat platform has been defined, capable of operating in a typical environment for space missions and to host payload with a mass up to 1.66 kg and with power requirements of 15 W and more. This configuration serves as a starting point, and to be consolidated it would be necessary to consider in more detail the interfaces of the individual subsystems in order to ensure that they can operate together.

On the Procurement side, instead, reference suppliers have been defined in the American and European panorama according to the sub-system whose purchase is required. The activity allowed to have a first contact with the representatives of these companies and to create the basis for a relationship that can provide future collaborations, useful to increase the business of Argotec as well as the partner's one.
Products from these suppliers were analyzed and key information was gathered to accelerate business processes regarding participation in commercial and scientific projects. These have been used to build an hardware matrix which results in a total price of 767.050,00 € with 11 months required to receive all the items.

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