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BRIDGE: A Telemedicine Satellite System, Feasibility Study

Relatore: Prof. MICHELLE LAVAGNA

Tesi di laurea di:

ANDREA DI CARLO

Matr. 765585

PATRIK DELLA VEDOVA

Matr. 766616

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Sommario

Lo sviluppo di sistemi satellitari a basso costo rappresenta al meglio l'interazione fra il mondo universitario, tipicamente mancante di risorse, ed il settore industriale. In particolare la progettazione e l'integrazione di tali sistemi garantisce il continuo sviluppo tecnologico e congiuntamente provvede ad una formazione più ampia del personale coinvolto.

Il presente lavoro si inserisce in questo ambito, sviluppando uno studio di fattibilità di un sistema satellitare rispondente agli standard CubeSat, applicato ad un progetto di telemedicina. Essa consiste nella raccolta e digitalizzazione di dati medici, in modo da provvedere ad una loro analisi a distanza. Specificamente, l'azione di Patologi oltre Frontiera verte sulla produzione di referti patologici di popolazioni che risultano sprovviste di adeguati programmi di prevenzione oncologica.

Un iniziale analisi di missione ha consentito il successivo approfondimento dei singoli sottosistemi e delle loro reciproche interconnessioni, successivamente al quale si è provveduto ad un dimensionamento iniziale del sistema, volto ad analizzarne la possibilità di realizzazione.

I risultati ottenuti dimostrano una generale efficacia dello strumento concepito, con particolare attenzione ad una sua implementazione per missioni di tipo umanitario.

Parole Chiave: Telemedicina, CubeSat, Design Sistema Satellitare, Ingegneria dei Sistemi, Analisi di Missione, Telecomunicazione in banda S, Data-Rate Elevati, Patologi Oltre Frontiera, Regione Sub Sahariana

Abstract

The development of a low cost satellite system represents at its best the interaction between academia, typically lacking in resources, and the industrial sector, namely private companies. In particular, the design and integration of these systems ensures the continuous technological development and jointly provides a more extensive training of the personnel involved.

This work of theses fits in this area, developing a feasibility study of a satellite system compliant with CubeSat standards, and applied to a telemedicine project. Telemedicine consists in the collection and digitization of medical data in order to provide them for a remote analysis. Specifically, the mission of Patologi oltre Frontiera focuses on the production of pathology reports from people who are deprived of adequate programs for cancer prevention.

An initial mission analysis has allowed the gradual deepening of the individual subsystems and their mutual interconnections, after which the design of each subsystem took place, which aims to analyze the possibility of realization of the system itself.

The results obtained show an overall efficacy of the instrument conceived, with particular attention to its implementation for humanitarian missions.

Key Words: Telemedicine, CubeSat, Satellite Design, System Engineering, Mission Analysis, S-Band Telecommunication, High Data-Rate, Patologi Oltre Frontiera, Sub Saharan Region

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Chapter 1

Introduction

African population not only suffer famine and diseases, a big portion is subjected to tumors. Compared with African population the percentage of pathologists, that can analyze the patient and make a diagnosis, is rather low. In order to provide an efficient diagnosis service *Patologi Oltre Frontiera* needs a link between African laboratories that permits the transfer of patient informations efficiently and promptly.

> *Telemedicine is the interactive transmission of medical images and data to provide better health care for people in remote or medically underserved locations. The concept has been around since the 1920s and its general viability has been demonstrated since the 1950s, but wide adoption was slowed by high costs and technological shortcomings.*
>
> *NASA*

1.1 Patologi Oltre Frontiera

Patologi Oltre Frontiera (Pathologists Beyond Borders) is a non-governmental association founded in Venice in 1999 with the aim of realizing projects for developing pathological anatomy and oncologic diagnostics in the South of the world. Born for an initiative of a group of Anatomic pathologists, began in the same year its activity, taking part in a project aimed at creating a Pathologic Anatomy Service in Tanzania. In 2001 *Patologi Oltre Frontiera* was officially acknowledged as ONLUS, while in 2006, following the acknowledgement of the Ministry of foreign affairs, was included in the list of qualified non governmental organizations [1].

Patologi Oltre Frontiera are operative in different regions around the world, For a decade they have been realizing projects in Africa, Latin America, Europe and Middle East.

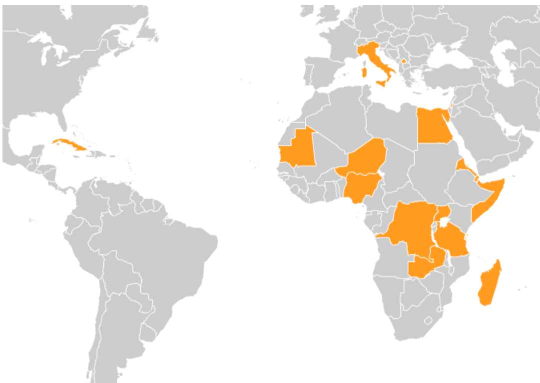


Figure 1.1: Patologi Oltre Frontiera Missions Map

1.1.1 Telepathology

Telepathology is a branch of the telemedicine which envisages the possibility of transferring, from a point to another, images (macroscopic or microscopic images) allowing the sharing of information or even the distance formulation of a diagnosis.
Today the technology allows more ways of action:

- **Static Telepathology:** the pathologist A (less expert) choices some microscopic fields in a slide and takes some images at the microscope with a camera. Through internet, he sends images to the pathologist B (more expert) for an advice. The pathologist B answers, via e-mail, giving his opinion. It is the cheapest system: it needs a cheap equipment and it works with a satellite dish; but, it requests, from both parties, the presence of pathologists with a good experience. This mode was applied for the vaginal cytology in projects in Zambia and Madagascar and in the project Tanzania for the quality control of the histological diagnostics.

- **Dynamic Telepathology:** the pathologist A (less expert) puts the slide on the robotized microscope which, via internet, can be manipulated at distance by the pathologist B. This last can, as he likes, increase or decrease magnifications and moves on the substance. Once a diagnosis is made, he invites, via phone or chat, the pathologist B to change the slide to make another diagnosis. This mode was used at the beginning of the project Zambia and Palestine but then it was abandoned both for the costs and for organizational problems. The costs are at an intermediate level between the first and the third solution.

- **Static and Dynamic Telepathology:** in a lab in seat A, where a pathologist is not present, a technician includes in a special scanner some slides which are quickly scanned with a high magnification and transformed in a virtual slide. The images obtained (too large, 4-5 gigabytes) of the virtual slide are downloaded in a server. The pathologist of seat B, regardless the presence of pathologists or technicians of seat A, can access with a password the server and sees the slide, moving himself, modifying the magnifications, and taking photos. This solution is the most expensive and was adopted in the projects Zambia and Madagascar, since locally a pathologist was not available.

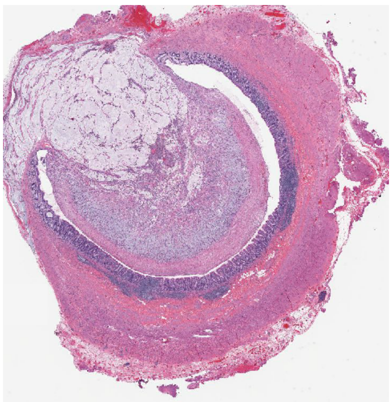


FIGURE 1.2: Slide Example

Patologi Oltre Frontiera have to propose a project to the *World Health Organization*, where the objective is to cover the sub-Saharan region, about 54 states, with a diagnosis service. The project plans to install, in each state, this kind of configuration:

- A headquarter in the capital with some pathologist (Pathologist B).
- A set of 3 to 4 technical center spread around the state with no doctors, only technicians (Pathologist A) who will perform the tissue sampling.
- Connection between technical centers and the capital by means of telemedicine, where Europe/Italy will act only as a support and quality-check center.

Patologi Oltre Frontiera started by simply using flash drives to transport images from villages to the capital, as of now they started to rely on already existing satellite service, provided by an italian private company, which offer an internet connection with acceptable data rates, about 512/256 kbps download/upload constrained by a CR of 1:10.

1.2 A Glimpse of a Dedicated Satellite System

Satellite system are especially well suited for the provision of telecommunications service over large or disperse service areas, where countries cover vast territories and have formidable natural obstacle, a scattered population and a rudimentary infrastructure, satellites can provide a means to rapidly establish a telecommunication network.

System of this type are often set by leasing or purchasing one or more available transponders in an existing satellites. However when the traffic increases, the implementation of a dedicated satellite system may prove profitable.

Setting up a new satellite system, which may be either a regional system with the participation of a group of countries, or a purely national (domestic) system, is obviously much more difficult than using an existing system: in fact, the decision to implement a new satellite system usually results from a long-term process, which may be preceded by the phases outlined below:

- Utilization of the space segment of an existing satellite, usually by leasing space segment capacity.
- Preliminary economic and technical studies of the validity and profitability of a new system considering the traffic growth and the possible need for new telecommunication services.

- Technical and operational preliminary experiment e.g. by using an existing satellite, if available, or even by launching an experimental or pre-operational satellite.

In comparison with a optical fiber cable service the implementation of the ground segment of a satellite network is relatively simple because the number of physical installation is minimal. To install a satellite network, a planner need only consider the sites where service is required. In comparison, the installation of an optical fiber cable system or of microwave links requires first that the right-of-way be secured from organizations such as governments, utility companies, and rail roads. Hundred or even thousands of sites must be provided with shelter and power (and even access roads in the case of terrestrial microwave). After the entire system is installed and tested, all the equipment must maintained to assure continuous service. Even then, one outage along the route could put the chain out of service until a crew and equipment can arrive on the scene to effect repairs [2].

1.3 BRIDGE: A Telemedicine Satellite System

Given the contractor's needs and knowing the advantages of a satellite system, this thesis deals with the study and design of a dedicated system capable of providing a telemedicine service.

Figure 1.3: BRIDGE Logo

Besides making *Patologi oltre Frontiera* independent from private services, the design of such a system permit also to extend the range of possible users to other organizations operating in the equatorial region, whether in the african continent or elsewhere in the world.

Given the humanitarian nature of *Patologi Oltre Frontiera* mission, the principal constraint is the lack of capital, which is mainly obtained through international funds or fundraising campaigns.

Even though at first glance this can be seen as a limitation, it can also be an opportunity for students of Politecnico di Milano to develop a satellite system, with the possibility of putting in practice the knowledge acquired during their studies, and gain experience in actual, hardware oriented spacecraft design and operations

It is in this perspective, of all the possible satellite architecture, that BRIDGE system design will be based on a CubeSat architecture, whose simplicity and standardization satisfy the most of the previous statements, both technical and cost related.

1.3.1 Cubesat Overview

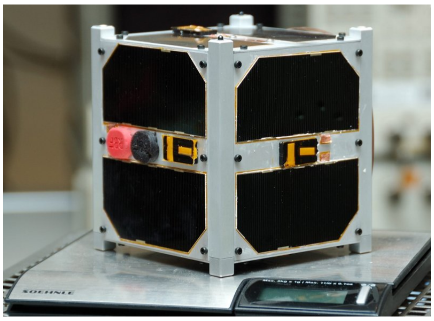


Figure 1.4: 1U CubeSat

1U CubeSat*: A 10cm cube with a maximum mass of 1.33kg.*

CubeSat Program started in 1999 from the collaboration of California Polytechnic State University and Standford University, aiming to give the opportunity to other universities and companies a more affordable access to space.

The purpose of the CubeSat program is to provide a standard for the design of picosatellites with the purpose to reduce costs and develop time, increase accessibility to space, and sustain frequent launches. At present the program count more than 100 international collaborations between universities, private firms, with payload spacing from scientific, private and governmental.

The CubeSat program has proven to be a reliable, practical and cost-effective program that allows launch opportunities to everyone, launched as piggyback using excess capability on larger launch vehicle. It has regulations on size and configuration, standardizing the deployment interface and providing also participants with the necessary documents and export licenses for launching their system in space [3].

Thanks to the lower budget needed, government and industry develop an increasing interest in CubeSats, requiring CubeSats capabilities to become more demanding.

1.4 Thesis Outline

The subject of this thesis deals with the feasibility study of a system capable of fulfilling the contractor request. The process starts with the tradeoff analysis of the orbit and architecture, driven by the telecommunication payload. It will then follow a subsystems design phase in order to understand which are the key factors that affect BRIDGE development. The process is briefly illustrated in figure 1.5.

Table 1.1: ESA Development Phases

Phase	**Description**
0	*Mission Analysis, Need Identification*
A	*Feasibility*
B	*Preliminary Definition*
C	*Detailed Definition*
D	*Production, Ground Qualification Testing*
E	*Utilization*
F	*Disposal*

Given the broad objectives that BRIDGE has to achieve, a list of requirements has been developed. Those requirements will guide the whole design procedure, they are showed in appendix A.

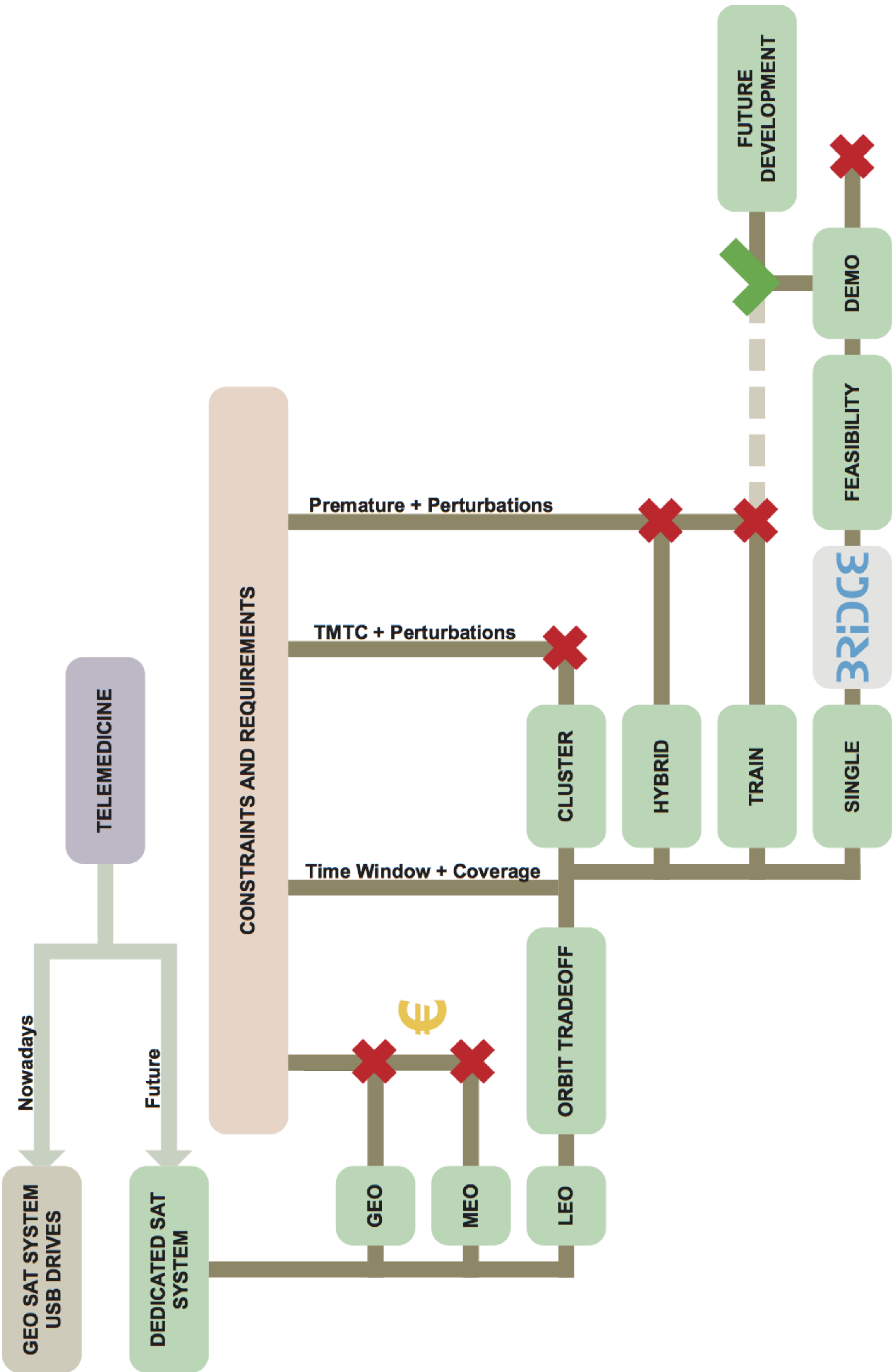


FIGURE 1.5: Feasibility Study Outline

Chapter 2

Mission Analysis Payload Driven

Before proceed with the sizing of the satellite system, it's essential to first consider which are the broad targets and constraints of the mission and which are the possible solutions, that can lead to achieve those objectives in the most efficient way available, and of course, also in the cheapest way.

The design challenge is an iterative, evolving process, based on costumer needs, applicable technology and cost constraints. In this chapter is described this process, applied to the architecture and orbit selection. Process that, given contractor's main objective, will have as main driver the telecommunication payload. After a requirements consideration, different mission concepts are evaluated, finally closing the selection to one architecture and a small range of orbits.

2.1 Mission Requirements

Requirements are used to understand how well the system must perform to meet objectives, how the system operates and how users interact with it to achieve its broad objectives, and which are the constraints, limit cost, schedule, and implementation techniques available to the system designer.

The system shall meet certain requirements based on both customer request and international regulation. The backslides on mission analysis and payload are listed hereafter.

Table 2.1: Mission Analysis Requirements

Mission Analysis Requirements		
ID	Requirement	Notes
	Mandatory	
MA-1	The system shall operate for at least 2 years	-
MA-2	The system shall cover the sub-Saharan region	Based on customer needs
MA-3	The system shall operate without orbit correction	Retain KISS philosophy
MA-4	The system shall not use a dedicated launch system.	Cost saving
MA-5	The system shall foresee de-orbiting	Fulfil International regulations
MA-6	The system shall limit orbital lifetime in LEO after mission completion or maneuvering to a disposal orbit	Fulfil International regulations
MA-7	The system shall limit the human casualty risk from space system components surviving reentry as a result of postmission disposal	Fulfil International regulations
MA-8	The system shall limit the debris hazard posed by tether systems	Fulfil International regulations
MA-9	The system shall limit the probability of impact with other objects in orbit	Fulfil International regulations
	Optional	
MA-10	It would be nice to have an european access	Quality control on diagnosis
MA-11	It would be nice to launch the system before the WHO-project starts.	-
MA-11	It would be nice to have an available window long enough to perform a diagnosis	Based on customer needs

Table 2.2: Payload Requirements

Payload Requirements		
ID	Requirement	Notes
	Mandatory	
PL-1	The system shall provide the quality needed to perform a diagnosis	-
PL-2	The system shall be capable to transfer contents	Based on customer needs
PL-3	The system shall be built with off-the-shelf equipment, even if not space qualified	Cost saving, retain KISS philosophy
PL-4	The system shall manage contents coming from ground equipment	APERIO scanner
PL-5	The connection shall be available at least during working hours (daylight)	Workability
PL-6	The system shall sustain a store-and-forward architecture	Ref. PL-2
	Optional	
PL-7	It would be nice to have the possibility to share the service with different users	Cost sharing
PL-8	it will be nice to stream contents from the servers to the end-users	Customer primary need
PL-9	It will be nice to use all available time window	-

2.2 Payload Overview

The satellite system under design process, is practically a telecommunication satellite which transmit medical data, in short, a Telemedicine System. The biggest part of the data to be transmitted consist of images coming from *Aperio* scanner presented below, whereas a small portion of commands, to and from ground, are present.

2.2.1 Aperio Scanner

The Scanner used by the *Patologi Oltre Frontiera* is a system, made by *Aperio*. ScanScope digital scanners are precision instruments that scan glass slides and create seamless, true color digital slide images of entire glass slides in a matter of minutes. A brief table of its characteristic is shown:

TABLE 2.3: Aperio Specifications

Specification	**Figure**
Scanning magnification	20x, 40x
Resolution	20x @ 0.50μ m, 40x @ 0.25μ m
Image compression	JPEG2000, JPG
Image File Format	TIFF (SVS), CWS, JP2
Scanning region	26.3mm x 60mm, 50mm x 60mm

This system allows pathologist to present in more “fluid” manner while provides low power whole images for more flexibility that can be easily zoomed to high power for relevant details; enables navigation through slides more efficiently, which makes it the perfect tool for telemedicine works.
Aperio scanner produce digital images variable in size, depending on the type of magnification and compression selected. The images are then stored in a server accessible both from the resident facility of *Aperio* scanner and from the outside environment, i.e. any system linked to this server [4].

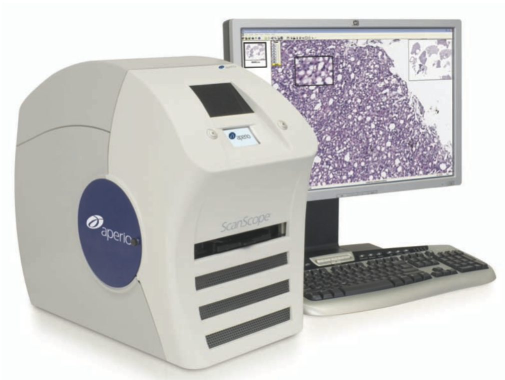


FIGURE 2.1: Aperio Scanner

2.2.2 Digital Images and Telecommands

Aperio scanner digital images transmission is essentially the core of the mission. The output images of the *Aperio* Scanner are gigantic, semi-lossless images, which dimensions can be as high as 15Gb (80000 x 70000, 40x magnification) to 800Mb (16000 x 19000, 20x Magnification).

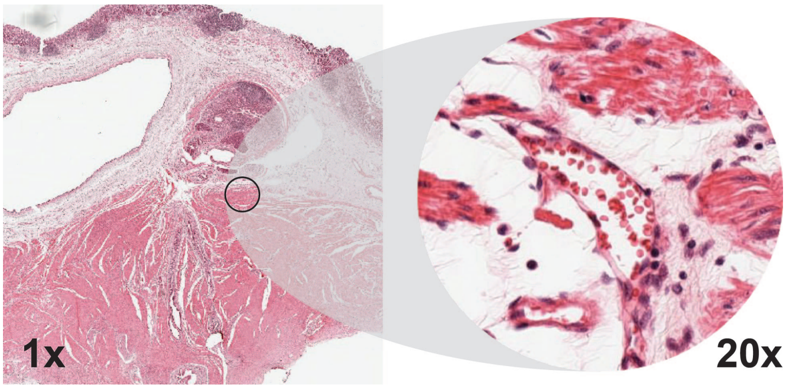


FIGURE 2.2: Aperio Scanner Image Example

Although the system shall be designed to work as autonomous as possible, a set of commands are needed to be transmitted to the satellite to perform BOL and EOL tasks, and attitude checks, as well as telemetry information from the satellite to ground. For the following study telecommands aren't considered as a driving parameters, so they aren't discussed in details.

2.2.3 Transfer Modes

Patologi Oltre Frontiera request the possibility to access to african servers where the *Aperio* scanner images are stored and analyze them in real time. To satisfy this request two possible way are considered.

- **Streaming**: Based on the idea of a constant or semi-constant connection availability, the images are streamed to the pathologist directly from the server located in the technical center. From an interview with a pathologist a mean analysis time of 30 to 20minutes has been estimated. It follows that in order to provide a useful service a minimum connection time must be granted.

  Large time-windows can be achieved by rising orbit altitude or by using a multi-satellite system where each single window can be combined in a bigger one.

- **Complete Transfer:** This second option consider the complete transfer of the images from the server located in the peripheral villages to the server in the capital, without loosing image quality. Even if this mode doesn't give the possibility of a real time connection, it has the advantage of not being constrained by a minimum time-window, since the pathologist can access to the capital server simply by using the internet, which is still not available in the villages.

  To completely transfer images in a lossless way, a possibility is to split the huge images into tiles that can be handled by the satellite communication system. In this way images are transferred tile by tile to the capital server where the image is finally recomposed and available for analysis via internet connection.

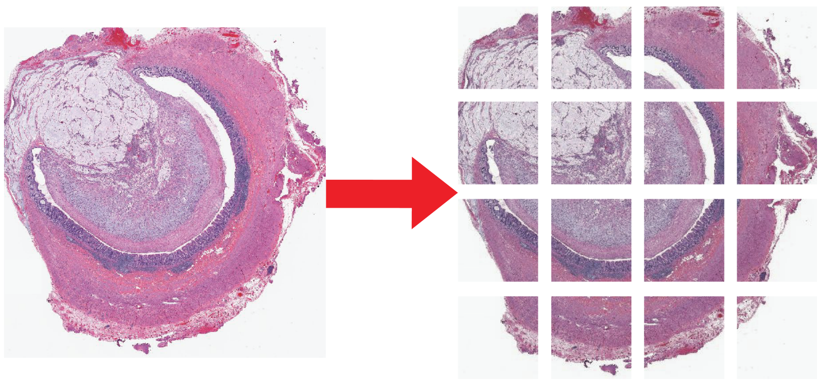


FIGURE 2.3: Image Splitting Example

2.3 Mission Architectures and Concepts

Before proceed with concepts illustration, the two following tables summarize the mission elements subject to trade and main option for each negotiable elements considered.

Table 2.4: Elements of Decision

Mission Element	**Can Be Traded?**	**Reason**
Mission Concept	Yes	Mission goal is constrained by the customer but not the mode
User	TBD	Main customer is fixed, possibility to share the service
Payload	Yes	Complexity and frequencies choices are flexible
Spacecraft Bus	No	CubeSat design configuration
Launch System	Yes	Based on availability and cost
Orbit	Yes	LEO, MEO, GEO with varying satellites number
Ground System	TBD	Upon customer locations
Mission Operations	Yes	Degree of automation
Communications	Yes	Given by mission operations

Table 2.5: Negotiable Elements Options

Mission Element	**Option Area**	**Options**
Mission Concept	Data Delivery	Streaming, Complete image transfer
Payload	Complexity Size	Upon orbit selection Upon orbit selection and geometry constraints
Spacecraft Bus	Propulsion Orbit Control Navigation ADCS Power	No TBD TBD Upon antenna pointing Solar and batteries
Launch System	Launch Vehicle Launch Site	Upon availability and cost Equatorial
Orbit	Type Altitude Inclination Constellation Config.	Circular, Geosynchronous, Sun-Synchronous LEO, MEO, GEO Upon customer location Number of satellites depending on visibility time requirements
Ground System	Type	Dedicated
Communication	Timeliness Control and Data Relay mechanism	Real-time link, Store and forward Multiple ground station, direct to user, user commanding Satellite to satellite interlink
Mission Operations	Automation level Autonomy level	Part-time operations, full-time operations Partial autonomy, full ground command and control

To implement the *Static and Dynamic Telepathology Method* illustrated before, different architectures types have been considered, based on mission analysis and payload requirements.

2.3.0.1 Single Satellite

This mode consist in the implementation of a system composed by a single satellite. It will provide a complete transfer of the images from villages to the capital, but it won't provide a streaming capability. The transfer of the images will be performed by a fragmentation of the big image into more little packages, which will be transferred piece by piece at each available connection.

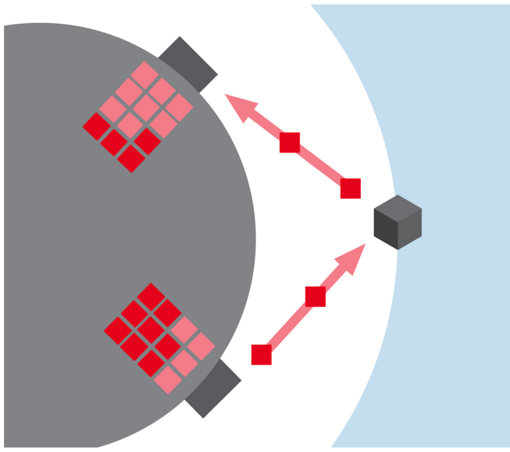


FIGURE 2.4: Single Satellite Concept

2.3.0.2 Train

In order to obtain a sufficient large time-window for a streaming service, a train of satellites configuration has been considered. This concept is based on the idea of take each satellite time-window and assemble them into a bigger one.

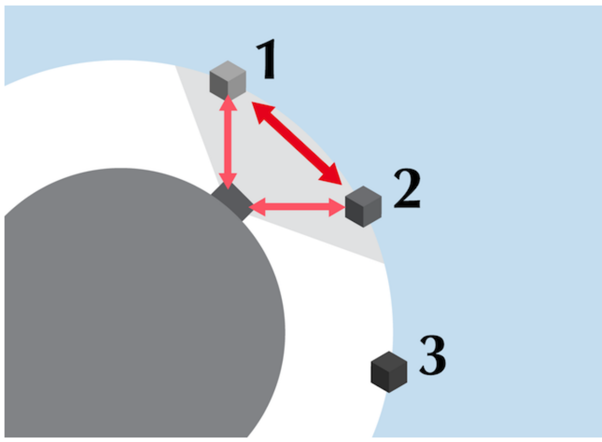


FIGURE 2.5: Train Concept

2.3.0.3 Hybrid

Combining the first two concepts, single and train, is possible to develop another concept: using a single satellite to transfer a low resolution image to the capital allowing the pathologist to perform a preliminary analysis, then use the enlarged time window for a focused analysis using the streaming mode.

2.3.0.4 Cluster

A cluster configuration has been taken into consideration, as for the train concept, it requires a group of satellites. Three satellite are necessary, the external ones are those prescribed to communicate with Earth, whereas the internal one is demanded to provide a inter-satellite link with the external ones and make them work synchronously. For sake of simplicity the three satellite are named HeadSat, CoreSat and TailSat. This configuration leads to cheaper ISL, as well as a dedicated beam for the head and tail to communicate with ground.

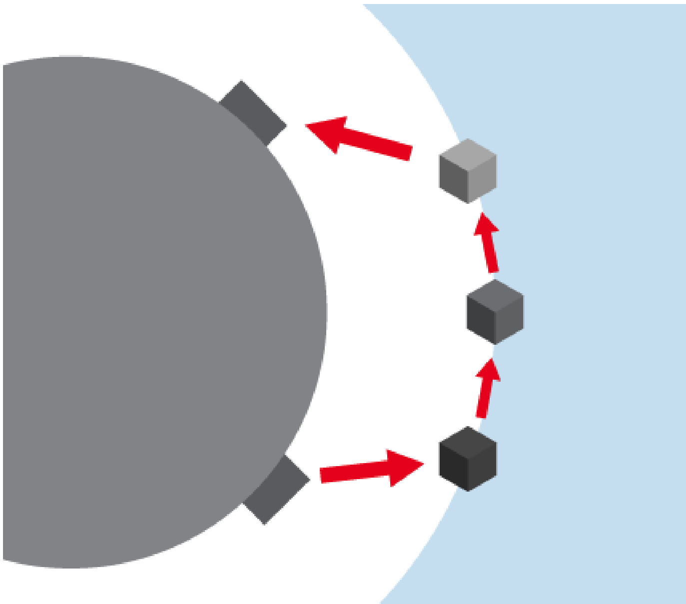


Figure 2.6: Cluster Concept

2.4 Architecture and Orbit Tradeoff

BRIDGE mission is mainly based on the time-window available between 2 nodes: the capital where the main lab is located and the villages where the images are captured. In order to determine the mission configuration an orbits analysis is carried out spacing thorough different inclinations and altitudes, considering the resulting time-windows of each combination.

The orbit is assumed to be a circular, since even if an elliptical orbit could give attractive time-windows during the apogee, a certain regularity in the connection timing is required which an elliptical orbit cannot provide, moreover, from a link budget point of view the distances involved in an elliptical orbit are prohibitive for the telecommunication system considered.

An ideal solution would be a system orbiting in a geosynchronous orbit, but due to altitude and architecture constraints given by the choice of use a CubeSat design, this kind of solution, as for other high altitude orbits, is discarded.

Analysis Scenario Overview:

Node-1 Capital:	Kinshasa (-4.322°, 15.321°)
Node-2 Technical Center:	Butembo (0.128°, 29.288°)
Inclination:	0° to 90°
Altitude range:	700km to 11200km
Propagation time:	1 year

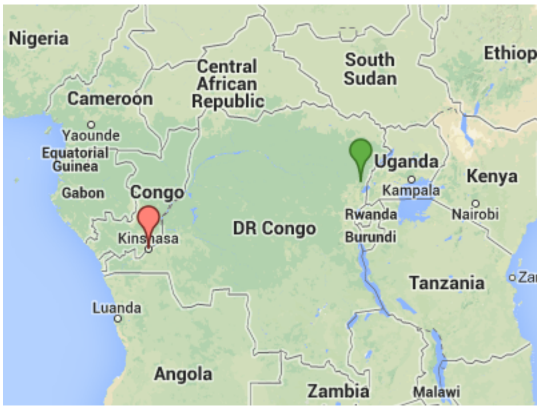


FIGURE 2.7: Node Map

Based on this scenario, the following time-window data have been obtained. Constraints of 10 minutes as minimum acceptable time-window and 2300km as the maximum altitude matching a CubeSat possible capability, have been applied. The latter limit is also dictated by environmental constraints, since at higher altitude the system will be exposed to a high radiation doses due to the presence of the Van Allen belts.

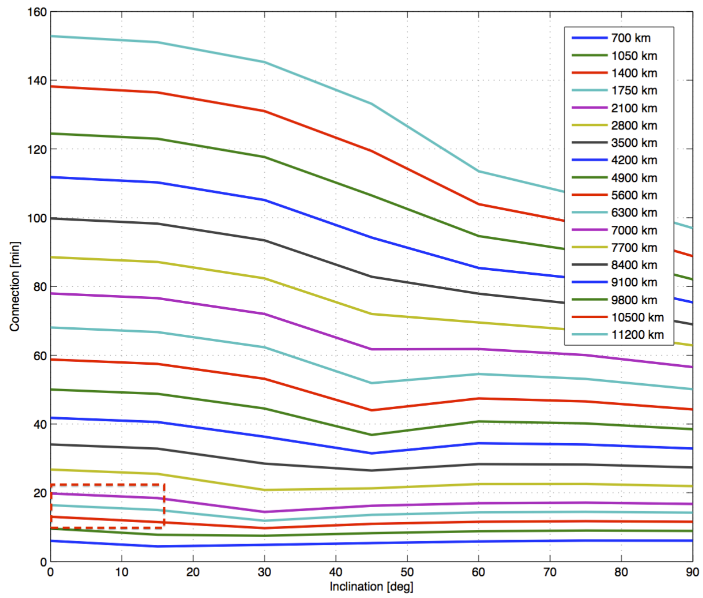


FIGURE 2.8: Mean Connection Duration VS. Inclination

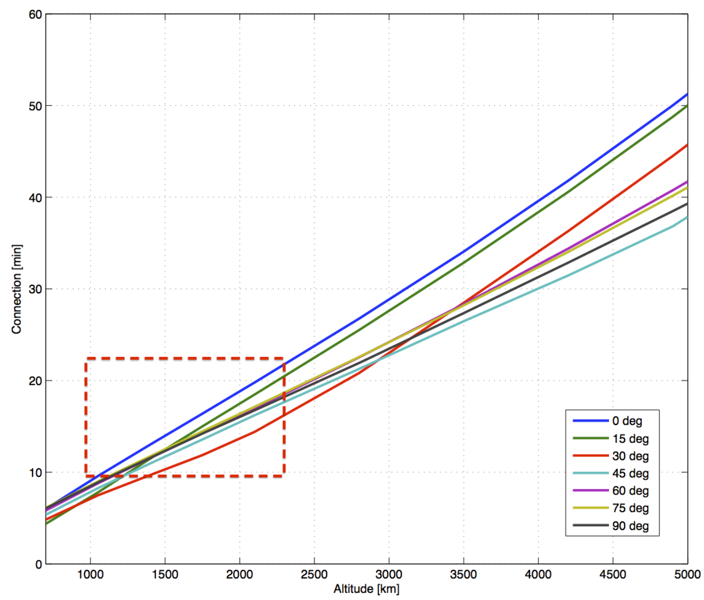


FIGURE 2.9: Mean Connection Duration VS. Altitude

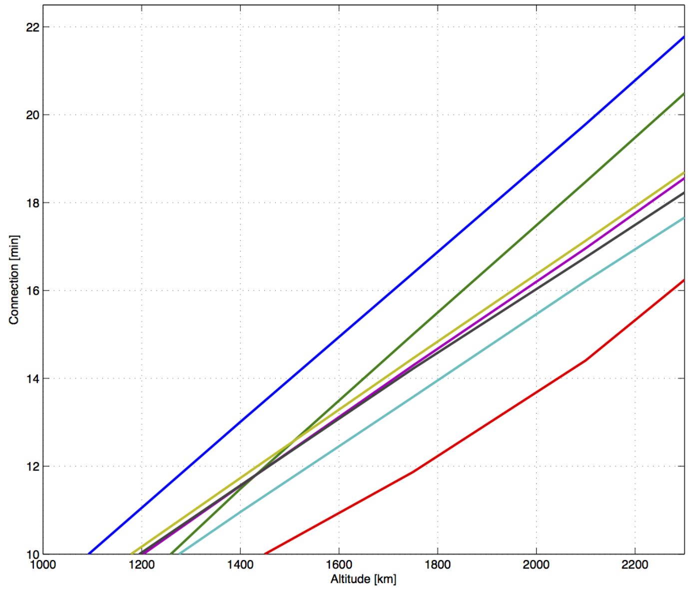


FIGURE 2.10: Mean Connection Duration VS. Altitude (Detail)

Due to the geographical location of the nodes that need the connection, the orbit inclination should stay between 0° and 15°; as it can be noticed from the following image, where as example an orbit 1400km has been considered, clearly the number of connection per day increase with the decreasing of the orbit inclination.

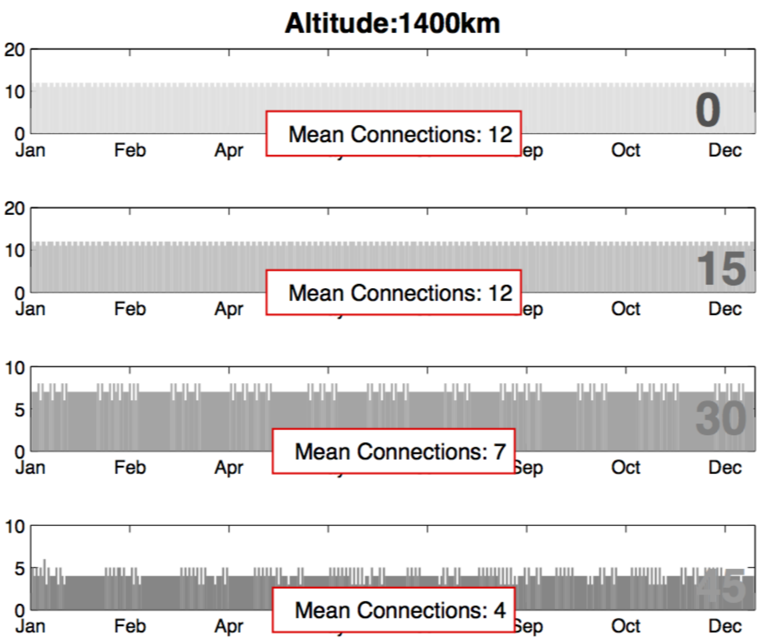


FIGURE 2.11: Number of Connection per Day VS. Inclination

From this considerations is possible to confine the orbit selection in the following ranges:

Altitude	1000km to 2300km
Inclination	$0°$ to $15°$
Eccentricity	0

2.4.1 Perturbation Analysis

Since satellites relative position is essential for the success of the Train and Cluster concepts, orbital perturbations represent for them the main issue. It's so essential to perform a perturbation analysis in order to evaluate mission performances and degradation under this condition.

In the real world a body in the space, i.e. a spacecraft, undergoes a series of actions that are very general and vary. A list of the main perturbations that are likely to affect an orbiting body is given hereinafter:

- *Atmospheric Drag:* dissipative interaction with Earth's atmosphere, important below 1000km.
- *Solar Pressure:* the effect of solar radiation on particles moving through interplanetary space.
- *Gravity perturbation:* due to Earth's oblateness and non-uniform mass distribution.
- *Lunisolar Gravitational Attractions (Third body action):* interaction with the sun and the moon.

The perturbations analysis has been carried out using the orbit propagation tool STK-HPOP (High Precision Orbit Propagator) which numerically integrates the orbital equations of motion taking into account the above effects.

In the analysis the following fundamental parameter are considered: *Semimajor axis, Eccentricity, Inclination, RAAN* and *Argument of Perigee.*

2.4.1.1 Short Period Results

The first analysis is made over a period corresponding to a rotation of the spacecraft around Earth.

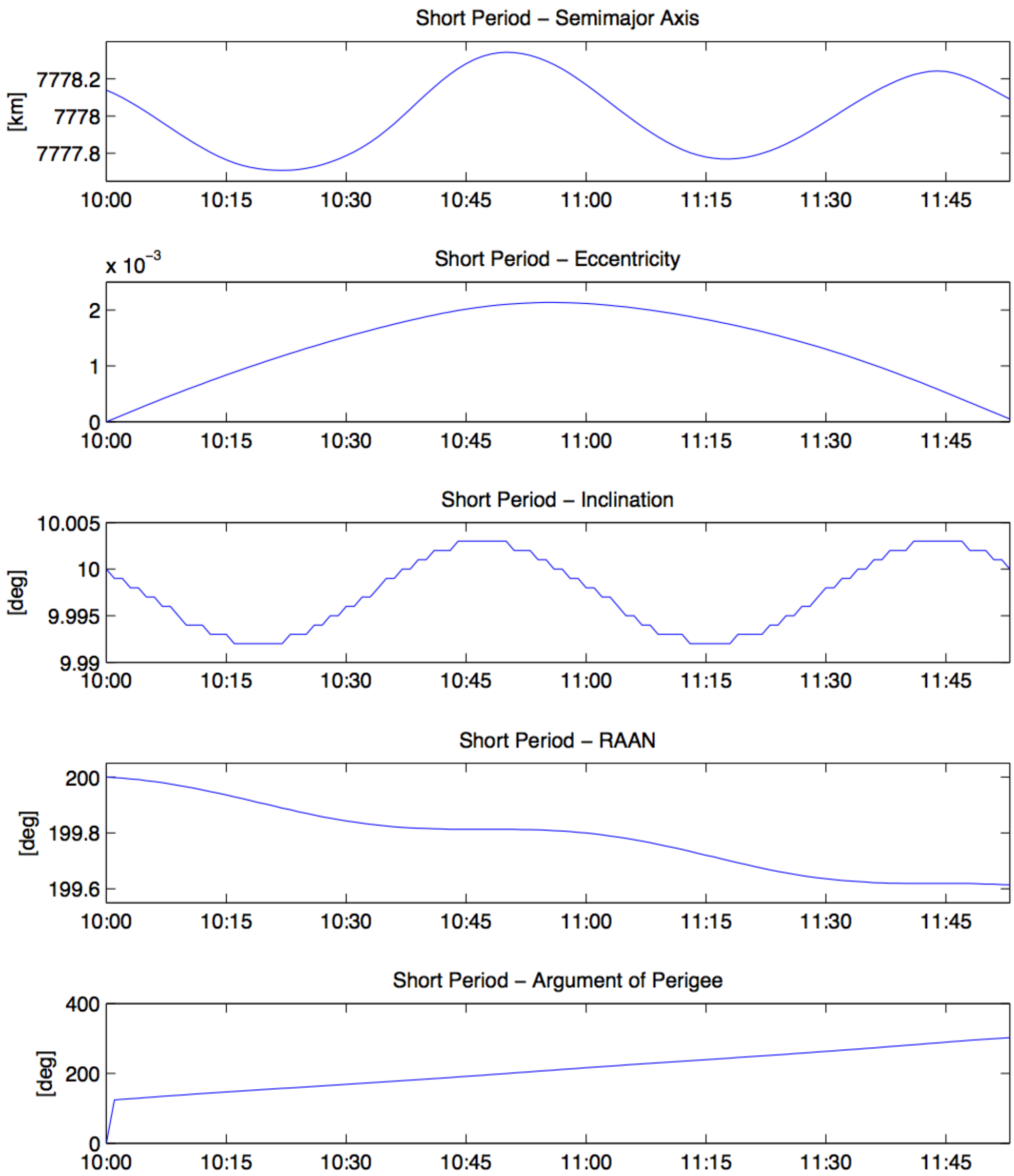


FIGURE 2.12: Short Period Perturbation

Over a satellite period the variations of the keplerian parameters are way under 1%, therefore they can be considered negligible.

2.4.1.2 Long Period Results

A second analysis has been performed over a period of 2 years, corresponding to the expected lifetime of the system.

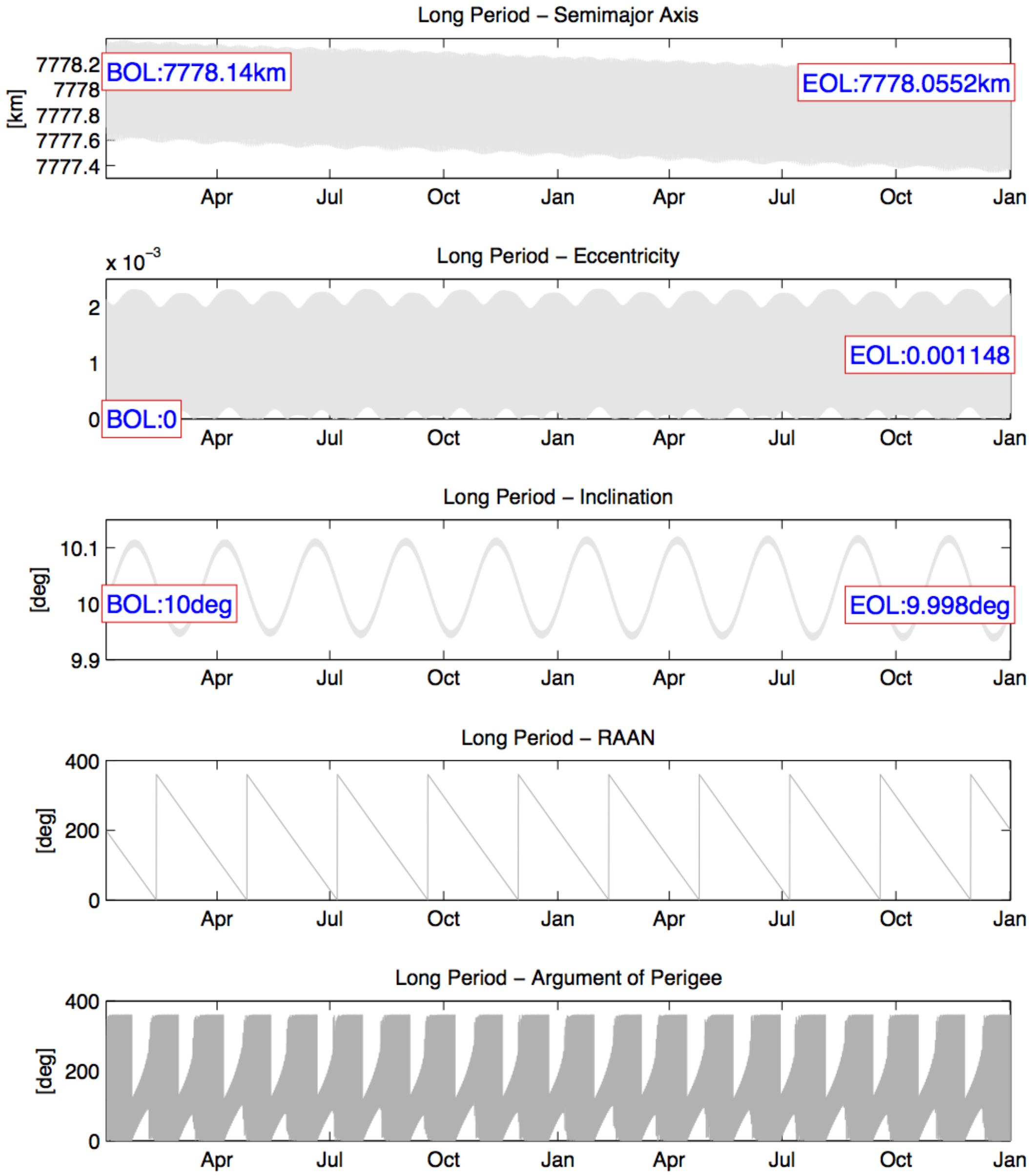


FIGURE 2.13: Long Period Perturbation

For a single satellite architecture the keplerian parameters variations over the system lifetime doesn't affect the design process. Unfortunately as it will showed in following section, this is not the case for the other architectures taken into consideration.

2.4.2 Train Architecture Evaluation

In order to better asses the train architecture performances, several analysis has been conducted, in order to make a sensitivity analysis the following scenario is considered.

Altitude range: 1050km, 1400km, 1750km, 2100km;
Inclination range: 0°, 5°, 10°, 15°;
Satellites number: 4 units

To have an overlap in the time-windows, in order to obtain a continuous bigger window, each satellite must be shifted from each other by a proper amount of true anomaly, considering an overlap long enough to coordinate the connection switch between two consecutive satellites in duty. The ranges of true anomaly deltas are listed below:

TABLE 2.6: Train Orbit Separation

Altitude [Km]	$\Delta\vartheta$
700	15°-8°
1050	28°-21°
1400	35°-31°
1750	41°-40°
2100	47°-46°

As example here is reported the results obtained for a train of satellites orbiting at 1400km with 10° of inclination, it's easy to appreciate how this architecture can provide large time-windows.

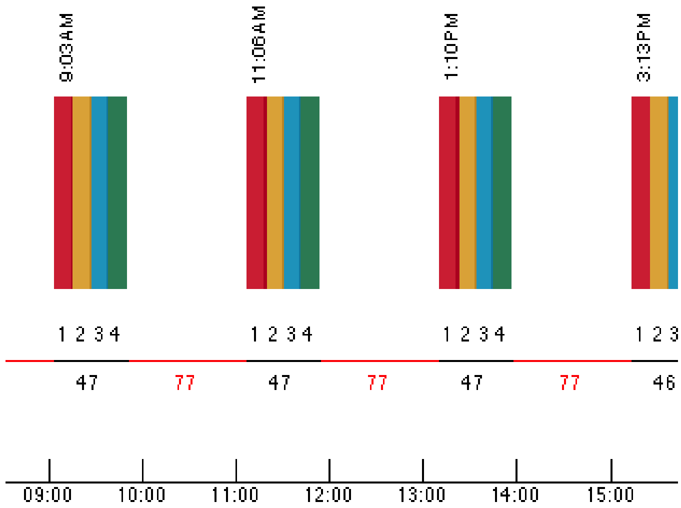


FIGURE 2.14: Train Scheduling Example

However, due to orbit perturbations, the train configuration will little by little drift, loosing in this way the overlaps between the different time-windows and consequently loosing the continuous time-window. It's clear that this kind of mode require the presence of orbit maintenance in the system design.

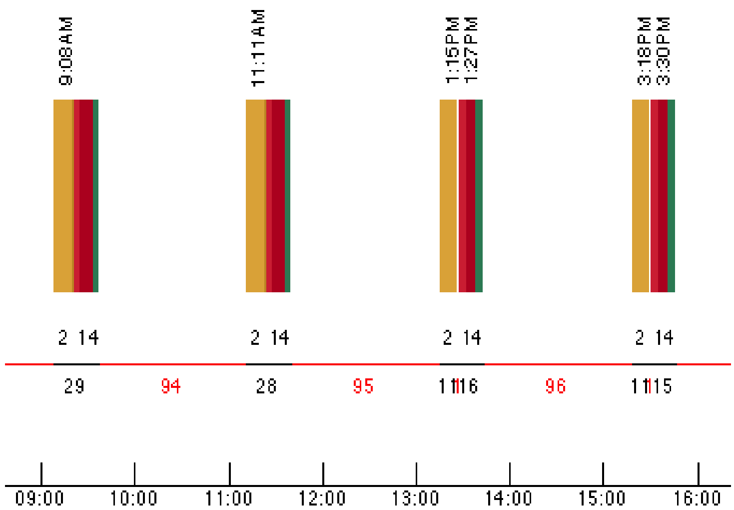


FIGURE 2.15: Train Scheduling Example - Perturbed

2.4.3 Cluster Architecture Evaluation

Cluster mode has been modelled considering the three satellites shifted by a true anomaly of 7°, corresponding to the latitude difference between the capital and the far East village

that has been considered, maximising in this way the visibility of the cluster by the two nodes on ground.
Here are reported the results in time-window duration for different altitudes considering an equatorial orbit.

TABLE 2.7: Hybrid Time Window

Altitude [Km]	**TimeWindow[min]**
1400	17.7
2100	25.2
3000	35
4000	47

Compared with Train mode, to obtain an appealing window the system should orbit above the imposed altitude limit of 2300km; considering the level of complexity of this kind of system and the fact it doesn't satisfy the altitude constraint, this mode has been discarded.

2.4.4 Train Rephasing Analysis and Budget

As stated before, perturbation will little by little change the relative position of the satellites. An Analysis of the Δv needed for maintaining the relative distances has been carried out, evaluating the re-phasing maneuvers costs.

The Δv for a phasing maneuver is determined by:

$$\Delta v = \sqrt{\frac{\mu}{r}}\left[1 - \sqrt{2 - \left(\frac{2\pi(n+1)}{2\pi(k+1)-\phi}\right)^{\frac{2}{3}}}\right] \tag{2.1}$$

Whereas the time elapsed for the maneuver is:

$$T = 2\pi\sqrt{\frac{r^3}{\mu}}\left[\frac{2\pi(k+1)-\phi}{2\pi(n+1)}\right] \tag{2.2}$$

Where:

- ϕ is the rephase angle between the two satellite;

- **k** is the number of complete revolutions of the forward satellite do before the final burn that complete the manoeuvre;
- **n** is the number of times that the walking orbit is completed by the backward satellite, literature suggest to use $n = k$ [5].

Analysis using STK proves that satellites shift from each other by an average of $1°$/month. A first guess approach to the control strategy implies to rephase the satellites every month.

Plotting $\Delta v = f(n = k)$ and $\Delta T = f(n = k)$ it can be seen that for $n > 20$ the total manoeuvre time reach its asymptote 2.16. Moreover selecting $n = 14$, i.e. one day per month will be dedicated to rephasing, the total time is 0.2% of its asymptotic value 2.17. Increasing the value of n leads to waste time in terms of telemedicine purposes.

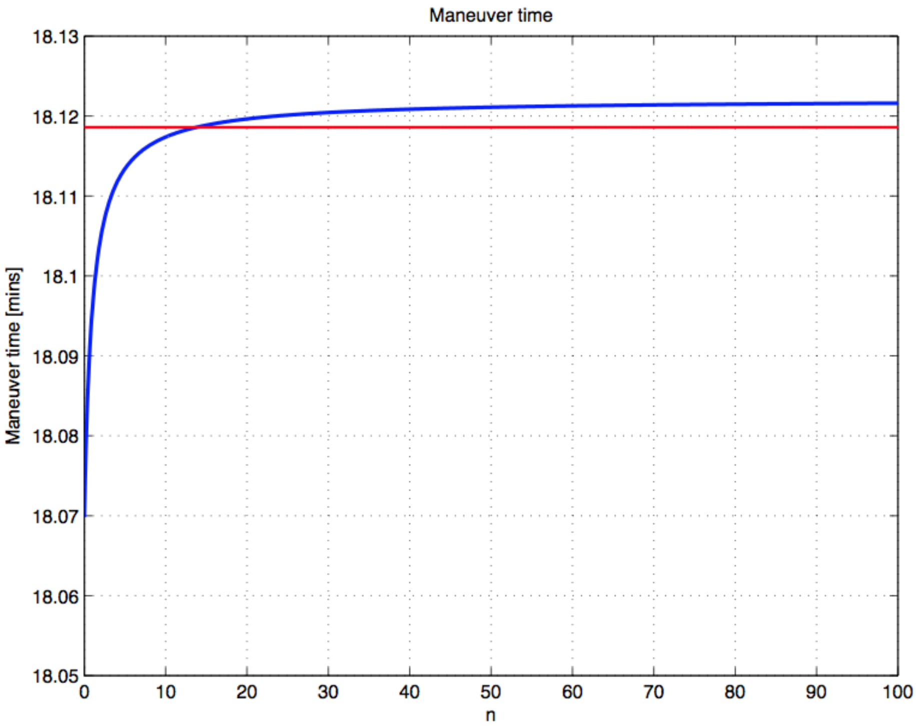


FIGURE 2.16: Maneuver Time Vs. n

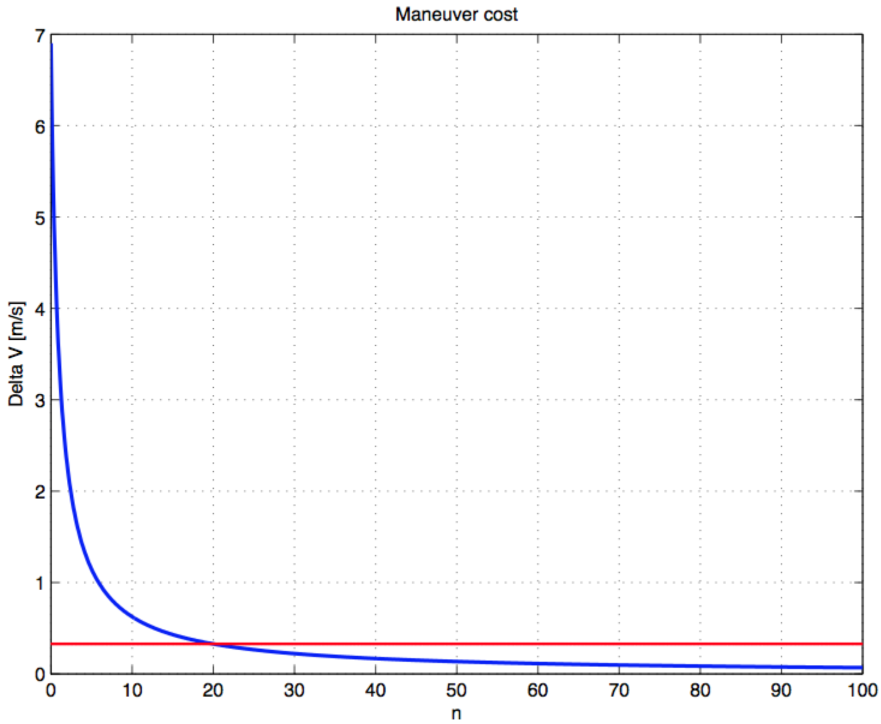


FIGURE 2.17: Maneuver Cost Vs. n

TABLE 2.8: Rephasing Parameters

	Asymptote	n=k=14
ΔT [min]	18.1219	18.1186
Δv [m/s]	0.0172	0.4586

Total maneuver cost can be further lowered by consuming more time in this phase, and consequently decrease the telecommunication performances of the satellite itself. Further analysis must be conducted taking into consideration or estimating the real working time of the system in its main mission objective. As stated above, an average of 1°/month lead to a maneuver cost of 0.4586 m/s.

CubeSats usually don't have a propulsion subsystem, but some study has been conducted by several universities and private company to develop an affordable yet robust solution. Typically they deal with cold gas systems, which are both cheap and simple, even though their performances are not as good as other common chemical propulsion systems. Its specific impulses are in the range of 70s.

Performing a burn per month, the total fuel mass needed for orbit maintenance is:

Table 2.9: Rephasing Budget

Mission duration [years]	1	2	3
Δv per burn [m/s]	0.4586		
Fuel mass [kg]	0.2084	0.8416	1.6835

This results add the constraint to the mission duration to not exceed two years, having fixed the CubeSat configuration.

2.5 Launchers

Here are listed a set of compatible launchers, capable to put the system into the required orbit ranges. Moreover the injection accuracy of each launcher has been considered.

Table 2.10: Launcher Sensitivity Summary

Launcher	**Parameters**	**Accuracy** [$\pm 1\sigma$]
Vega	Altitude	± 5 Km
	Inclination	± 0.05 deg
	Argument of perigee	N.A.
	RAAN	± 0. deg
Antares	Altitude	± 10 Km
	Inclination	± 0.05 deg
	Argument of perigee	N.A.
	RAAN	N.A.
Falcon 9	Altitude	± 20 Km
	Inclination	± 0.1 deg
	Argument of perigee	N.A.
	RAAN	± 0.15 deg
Falcon 1	Altitude	± 10 Km
	Inclination	± 0.1 deg
	Argument of perigee	N.A.
	RAAN	N.A.

2.6 Conclusions

To summarize:

- The cluster concept results into a non-efficient and too complex architecture, so it's discarded.
- Train and single satellite concepts are the most appealing solutions, moreover they give an interesting flexibility to the mission implementation. First start with a single satellite, that could behave as demonstrator, and then after its validation build up a constellation to implement the train concept.

Given the statements above, the subject of this thesis from now on regards the design of a single satellite. This satellite will be the benchmark of a future implementation of constellation capable of both streaming, e.g. train, or an improved traditional transfer.

Chapter 3

Telecommunication System

The telecommunication subsystem is the only mean that connect the satellite with Earth, and as already mentioned, it represent the main payload of BRIDGE system. In this chapter a link budget analysis is carried out, based on the assumptions and considerations made in the mission analysis chapter. Two cases are analyzed, one with no active attitude control and one with an active pointing, results are then illustrated for both case in terms of link margin and time windows restriction.

3.1 Equipment

Before starting to evaluate the link budget and the feasibility for a telecommunication network between the space segment and the ground one, i.e. the Earth stations, it's mandatory to present the antenna system chosen for both segments. Frequency and transmitter power are assigned by transmitter manufacturer, and particularly the power is essential to be as low as possible, to be compliant with the power generation on board.

3.1.1 Space Segment

The CubeSat configuration does not allow the implementation of a parabolic, or dish, antenna, mainly due to geometric and structural constraints, for this reason the choice fell back on what is available today on the CubeSat market. There are studies of alternative antenna architectures, like inflatable, that can provide more powerful connection, bust since they are still in a conceptual development phase they are not considered in this studies [6, 7].

The actual configuration is made of a S-Band patch antenna and transmitter from *Clyde Space*® [8] taken as reference for a set of patch antennas to be further analysed once the feasibility study is finished and approved.

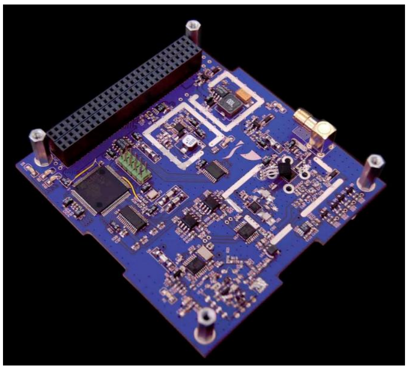


FIGURE 3.1: Clyde Space S-Band Transmitter

The transmitter is designed for CubeSat missions and compatible with the CubeSat standard. It implements QPSK modulation with transmission data rates of up to 2 Mbps, matching BRIDGE mission requirements.

TABLE 3.1: S-Band Transmitter Specifications

Total power consumption:	6 W (at maximum RF power output)
Transmission data rates:	2 Mbps (1/2, 1/4 and 1/8 rate modes)
Supported frequencies:	2.4 - 2.5 GHz Amateur
	2.2 - 2.3 GHz Commercial
Encoding and modulation:	QPSK / OQPSK
Transmit output power:	21 dB - 30 dB (3 dB steps)

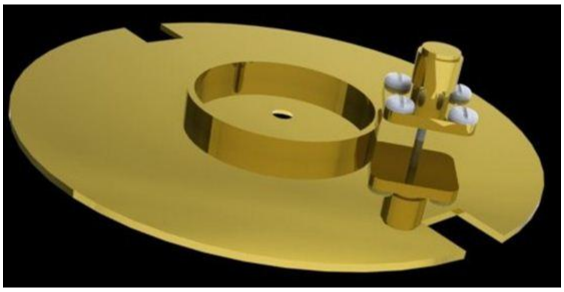


FIGURE 3.2: Clyde Space S-Band Patch Antenna

An S-Band patch antenna can also be incorporated into the CubeSat design. It will be mounted on the nadir face of BRIDGE satellite, here's the antenna specifications [8]:

Table 3.2: S-Band Patch Antenna Specifications

Temperature Range:	-25°C to 85°C
Mass:	50 g
Diameter:	76 mm
Standoff Height:	3.8 mm
Frequency Range:	2.4-2.483 GHz or 2.2-2.3 GHz
RF Power:	2 W
Gain:	8 dB
Beamwidth:	60°
S11:	-10 dB
Polarisation:	Left/Right Hand Circular

3.1.2 Ground Segment

The ground segment need a specific equipment capable of providing an high gain, that balances the low gain at the satellite end. Therefore for the purpose of this analysis an ideal parabolic antenna is selected, ideal as it represent a set of antennas that can be bought of the shelf nowadays.

Table 3.3: Ground Segment Specifications

Antenna:	Parabolic
Diameter:	1m
Gain:	25dB
Efficiency:	55%
Beam-width:	7°
Minimum Elevation:	10°

The following table outline the equipment selected for the link budget analysis:

TABLE 3.4: Equipment Summary

	Space Segment	**Ground Segment**
Antenna Type	Patch	Parabolic
Frequency	2.2 - 2.5 GHz	
Gain	8 dB	25 dB
Beamwidth	60 °	7 °
Efficiency	70%	55%
RF Power	1 - 2 W	10 W
Data Rate	up to 2 Mbps	10 Mbps
Polarization	Circular Left/Right-Hand	
Modulation	QPSK	
Diameter	76 mm	1 m

3.2 Link Budget

The Link Budget analysis as been carried out with the support of the STK software, which incorporate a link budget design tool. To perform the analysis both ground segment and space segment has been modelled according to equipment specification. The STK tool take into account the following link losses:

- *Propagation path loss*: The power density reduction of an electromagnetic wave as it propagates through space.
- *Rain attenuation*: The absorption of a microwave RF signal by atmospheric rain, snow or ice.
- *Troposheric scintillation*: Caused by fluctuations of air refractive index that can produces random fades and fluctuations of the received signal amplitude.
- *Noises*: Related to the presence of the atmosphere, Sun and cosmic background.

The following scenario has been considered:

Altitude:	1400km, 2000km
Inclination:	10°
Attitude:	Nadir pointing, Target pointing
Transfer mode:	Download, Upload.

The download case use a radio frequency power of 1W for the transmission, while for the upload the transmission power is set to 10W, since the ground station is clearly not subjected to power limits as the satellite. The data rate is kept constant at the same value offered by the private service used by *Patologi Oltre Frontiera* at the moment, 512kbps, which is a good starting point since its positive review by the contractor.

The analysis consider a time span of 1 month, results are showed in the following graphs, where in each graph this quantities are illustrated:

- A link margin threshold of 3dB of Eb/No highlighted in red.
- Percentage of connection time above Eb/No threshold, considering a minimum time window of 10 minutes.
- Mean Eb/No, EIRP and BER.

3.2.1 Nadir Pointing Analysis

The satellite is assumed to simply point toward Earth along its nadir direction without any active pointing action.

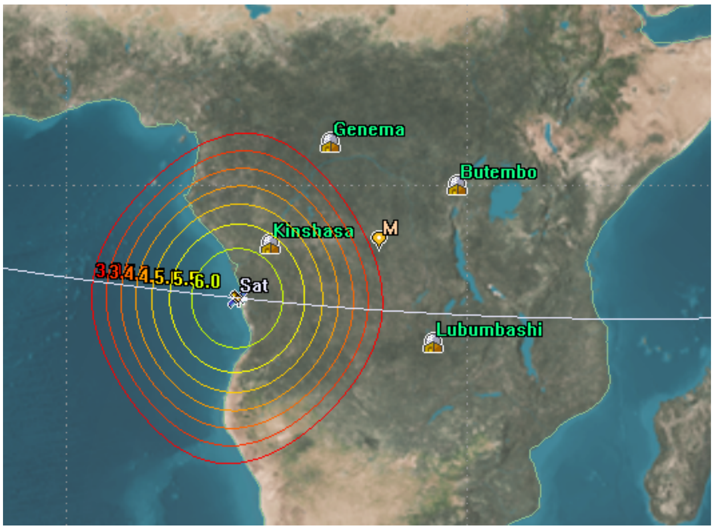


FIGURE 3.3: Nadir Pointing Map

3.2.1.1 Nadir Pointing Downlink

- *Nadir Pointing, 1400 Km*

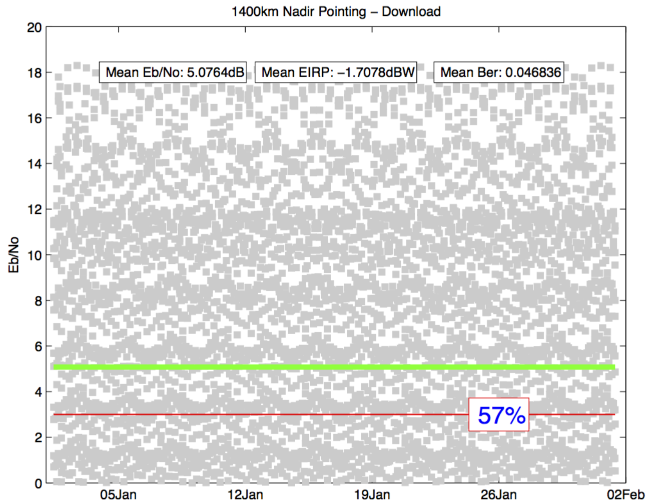


FIGURE 3.4: Downlink Nadir Pointing 1400 km

- *Nadir Pointing, 2000 Km*

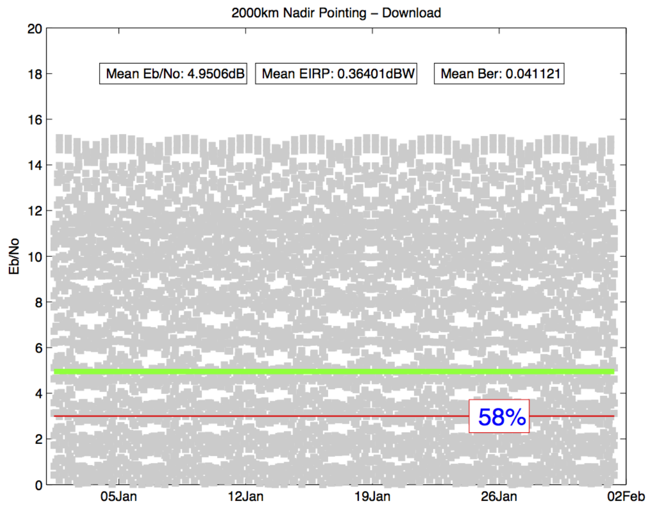


FIGURE 3.5: Download Nadir Pointing 2000 km

Results show positive values for Eb/N0 though not so high to induce an high fidelity of the overall system. The problem lies in the BER figure which its value is too high to set up a robust telecommunication link, in particular there is a chance close to 5% that any received bit is corrupted. Proper value for telecommunication network are given in literature [9, 10] and staying in the range of 10^{-5} for data, and 10^{-7} for telemetry.

Moreover results shows that only a portion (49% to 58%) of the access time actually stays over the margin of 3dB taken as threshold.

3.2.1.2 Nadir Pointing Uplink

- *Nadir Pointing, 1400 Km*

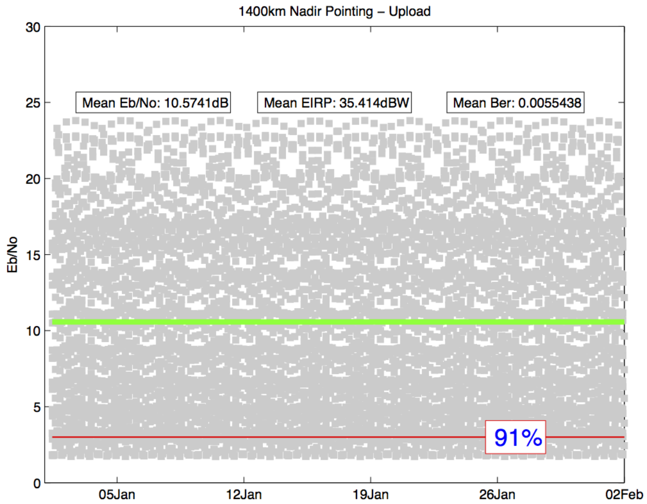


FIGURE 3.6: Upload Nadir Pointing 1400 km

- *Nadir Pointing, 2000 Km*

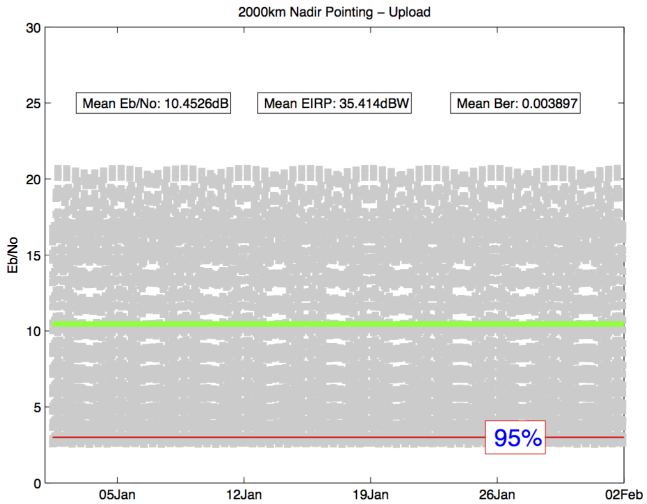


FIGURE 3.7: Upload Nadir Pointing 2000 km

In this case satisfying Eb/N0 values are obtained, as well as access time usability with respect to threshold. Even in the uplink phase BER is a key factor, without pointing it falls under the prescribed limits.

3.2.2 Area Pointing Analysis

The satellite is assumed to be equipped with and attitude control system which allow the spacecraft to point toward the specific area of interest in order to embrace at best al the nodes considered (see point M in figure 3.8).

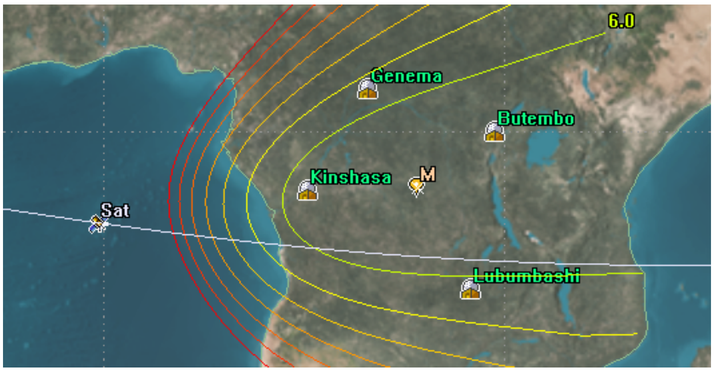


FIGURE 3.8: Area Pointing Map

3.2.2.1 Area Pointing Downlink

Applying an attitude control the situation becomes much more efficient in terms of both Eb/N0 and BER. The high margins at higher elevations can also overcome peaks in scintillation in presence of particularly hostile weather conditions.

- *Area Pointing 1400 Km*

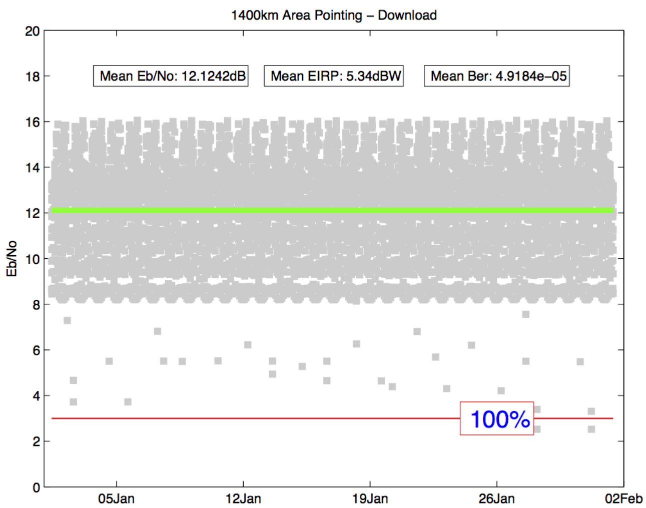


FIGURE 3.9: Download Area Pointing 1400 km

- *Area Pointing 2000 Km*

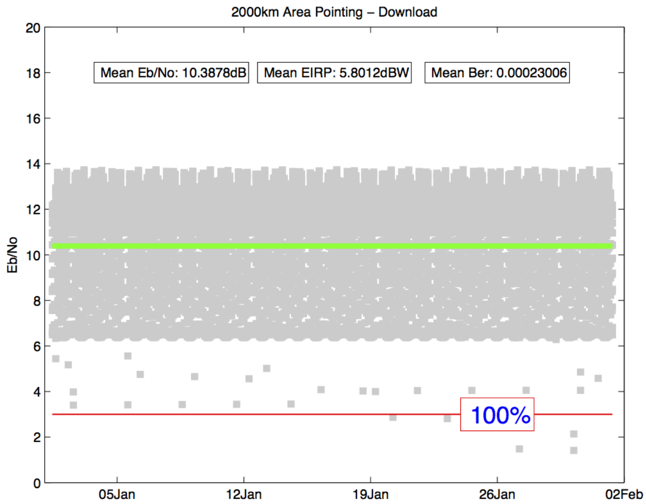


FIGURE 3.10: Download Area Pointing 2000 km

Eb/N0 is much higher than a Nadir pointing architecture, as well as BER reach a proper value for data transmission. Furthermore results show that all the access time is over the 3dB threshold, giving the system both robustness and fidelity quality. Problems lies in higher altitudes mainly due to BER considerations, limiting the whole system to lower altitudes.

3.2.2.2 Area Pointing Uplink

- *Area Pointing, 1400 Km*

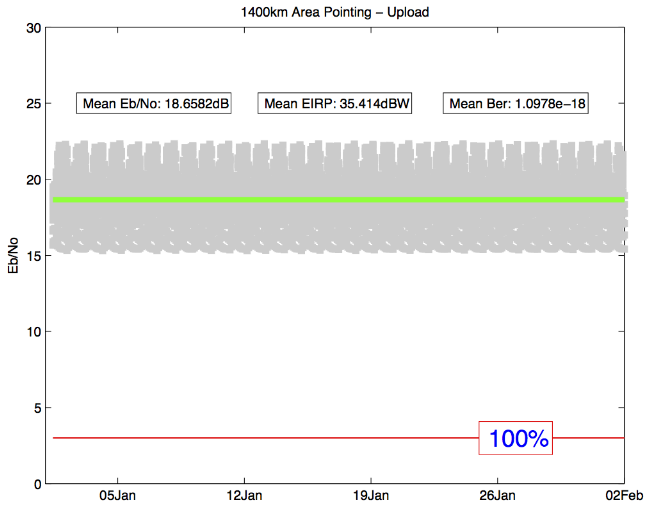


FIGURE 3.11: Upload Area Pointing 1400 km

- *Area Pointing, 2000 Km*

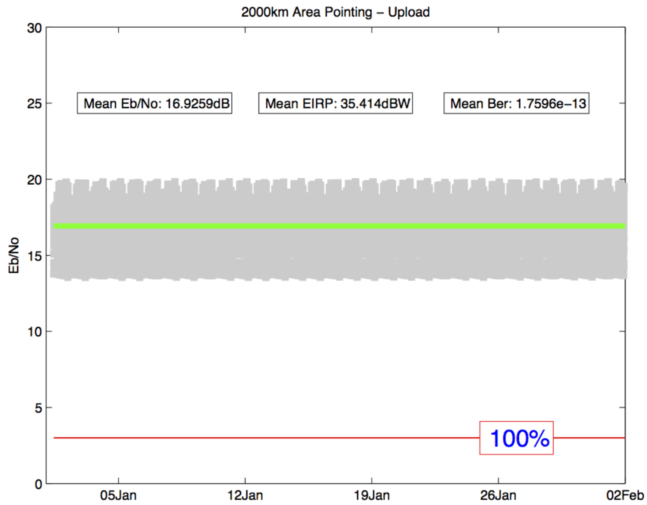


FIGURE 3.12: Upload Area Pointing 2000 km

Results show excellent performances regarding Eb/N0, BER and usability time for both altitudes considered. Uplink does not limit the overall system characteristic (in terms of keplerian parameters), but it remarks the necessity of a pointing system.

3.3 Conclusions

The results yields few considerations:

- BRIDGE will require an attitude control system to reduce the losses coming from the pointing offsets, which can be rather high at lower elevations;
- ground stations need to be equipped with an auto tracking system, to maintain a good pointing through each passage, and maximize the time-window available at each passage.

Chapter 4

Eletric Power System

The Electric Power System is demanded to provide the power needed by every subsystem during each phase of BRiDGE lifetime. In this chapter the analysis on the power production generated by the system is carried out, with particular attention to the most demanding phase, i.e. the telecommunication one.

4.1 Phases

For design and managing purposes it's useful to define and decompose the satellite life into different phases. Phases can be operational or simply a sum of tasks done in a common time-window as it will be seen later. The definition process has been evolved into eight phases presented hereafter:

1. **Launch**: every task the spacecraft need to perform during the launch phase.
2. **Release**: short time frame in which the satellite disengage from the launcher's adapter.
3. **Commissioning**: comprises the check and test of every operational component of the satellite, from individual functions, such as instruments and equipment, up to complex amalgamations such as modules, subsystems and the system itself.
4. **Communication**: include the tasks needed to perform the correct link between the satellite and the ground stations, both for scientific data transmission and telemetry.
5. **Light**: any moment under Sun illumination in which the satellite is not communicating with ground. It's a sort of stand-by mode.

6. **Eclipse**: when the spacecraft is covered by the Earth and not receive direct Sun illumination.

7. **Safe**: safety mode in which the satellite enters when one or more subsystems undergo a major failure.

8. **Deorbit**: time frame starting from the end of operative life of the satellite until its atmospheric re-entry and subsequent destruction.

4.1.1 Power Demands

Each phase requires a specific amount of power based on the tasks accomplished by each subsystem. A brief analysis based on preliminary assumptions on subsystems power needs has been performed; a short specification of the subsystem involved is shown:

- *EPS*: comprehend the power unit board consumption.
- *TX/RX and Antenna*: peak power demanded by Transmitter/Receiver unit and the antenna.
- *OBDH*: peak power demands, and average consumption are considered.
- *ADS*: comprehend possible integrated sensors consumption.
- *ACS*: active control though reaction wheels or magnetorquers.
- *TCS*: passive control system does not require any power.
- *PS (chem)*: chemical propulsion system.
- *PS (electric)*: electric propulsion system.
- *AEOLDOS*: Aerodynamic End-of-life Deorbit system, mechanism only.

Power demands for each phase are shown in table 4.1. Applied margins comes from values given by AIAA recommended power contingencies, selecting from the Bid Class the subcategory of existing spacecrafts , i.e. 13% increment of the total power [11]. As seen in the table, and quite expected, the communication phase is the most power consuming, hence this phase happens to be the design case for the subsystem.

TABLE 4.1: Power Demands per Phase

Subsystem	**BOL**			**Operations**				**EOL**
	Launch	*Release*	*Commissioning*	*Communication*	*Light*	*Eclipse*	*Safe*	*Deorbit*
EPS	0.25	0.25	0.25	0.25	0.25	0.25	0	0.25
Transmitter and Antenna	6	6	6	6	0	0	0	6
Receiver and Antenna	0	8	8	8	0	0	0	8
OBDH	5	5	9	9	5	5	5	5
ADS	0	0.82	0.72	0.46	0.46	0	0.46	0.46
ACS (ReactionWheels)	0	0	5.4	5.4	5.4	0	0	0
TCS	0	0	0	0	0	0	0	0
PS (Chemical)	0	0	0	0	0	0	0	1
PS (Electric)	0	0	0	0	0	0	0	3
AEOLDOS	0	0	0	0	0	0	0	0
Total Power	11.25	20.07	21.97	23.71	5.71	5.25	5.46	20.71
Total Power (Margins)	13.5	22.68	24.83	26.79	6.45	5.93	6.17	23.40

4.2 CubeSat Solar Panels

The design phase for the solar arrays starts with the selection of common CubeSat solution provided by *AzurSpace* [12], which meets both system level requirements of space qualified and off-the-shelf products.

Phase	**Current [mA]**
Efficiency	28.3% at BOL
Type	3J-GaInP/GaAs/Ge
Cell Area	30.18 cm^2

TABLE 4.2: AzurSpace SolarCell Main Specifications

FIGURE 4.1: AzurSpace Solar Cell

4.2.1 Solar Panels Power Analysis

The modelling of the solar panels and the satellite substructure has been performed using *Google Sketch-Up*, one of the software capable of exporting the model into a format manageable by STK. The three-dimensional model need also to be adapted to be read by the *Solar Panel Tool*. In order to accomplish this a *.anc* script file, containing the solar arrays characteristics has been written down.

The *Solar Panel Tool* enables to model the exposure of solar panels mounted on spacecraft, aircraft, and ground vehicles over a given time interval. The result of the analysis can be used to determine varying availability of electrical power for operations to be performed by the vehicle and onboard apparatus. Moreover it computes solar illumination over time by animating the scenario and periodically counting the pixels corresponding to illuminated portions of the solar panels under consideration [13].

Two types of solar panels configurations has been analyzed: a 0° and a 45° orientation of the top solar panels.

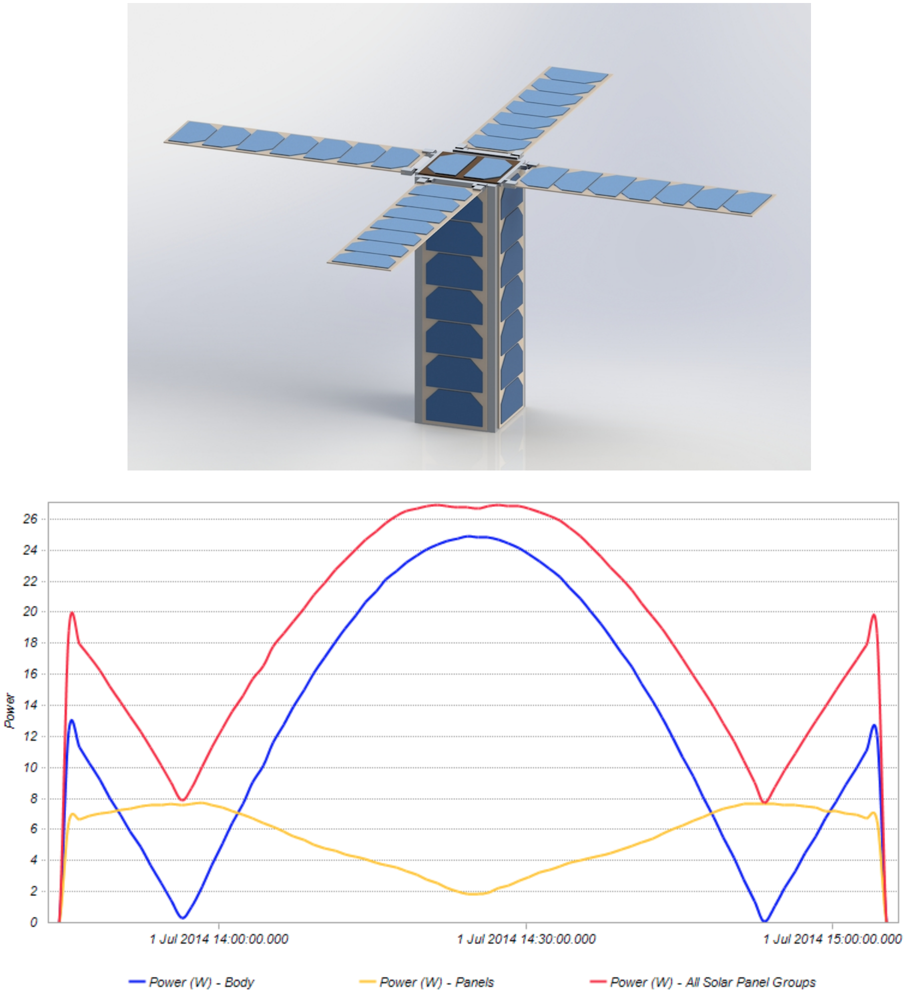


Figure 4.2: Solar Panels at 0 degree

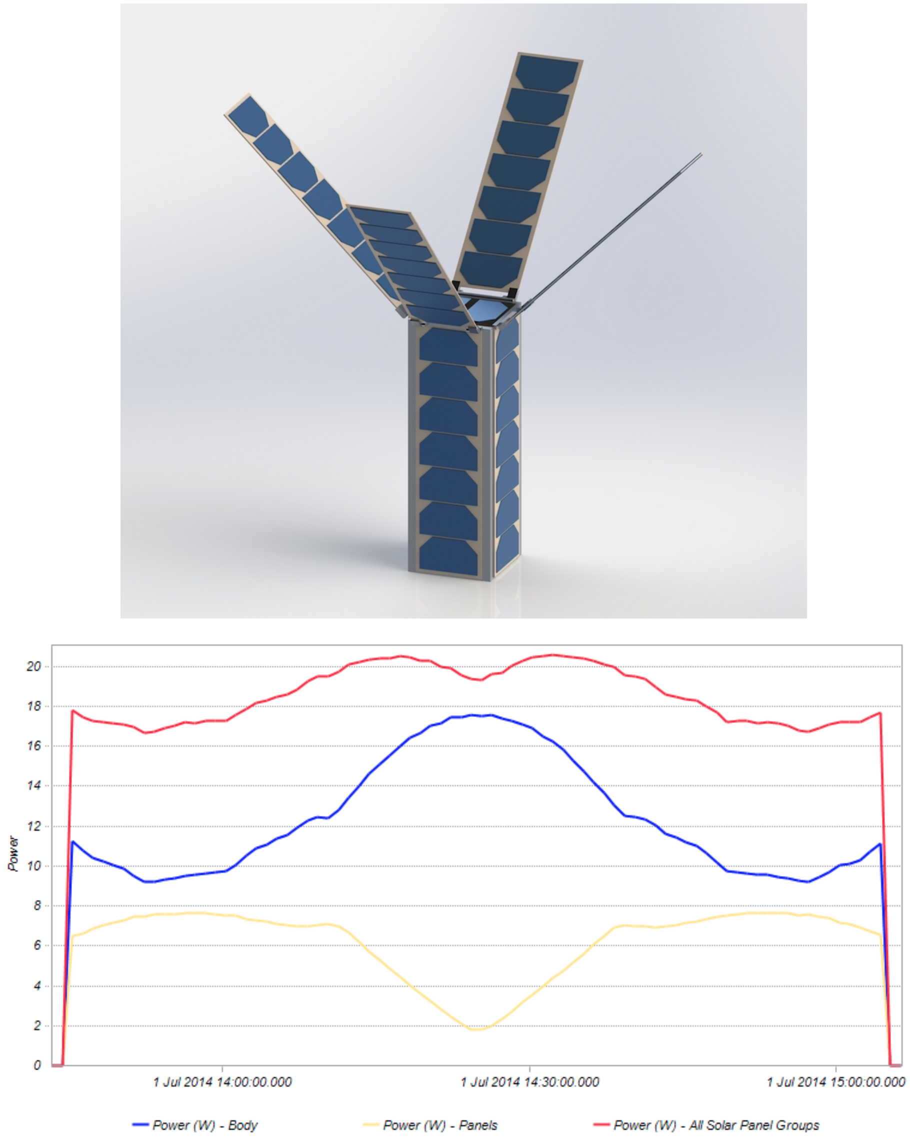


FIGURE 4.3: Solar Panels at 45 degree

It is clear that using a pointing spacecraft, the 0° configurations provide an higher peak power production, whereas the 45° configuration exploit the angle between the Sun and the panels themselves to generate a steadier power outcome, even though lower in absolute value.

4.2.2 Solar Panels EOL Considerations

Solar panels performances degrade over time do to the irradiation from electrons and protons. To quantify this degradation a equivalent fluence of monoenergetic 1MeV electrons and protons has been estimated [14].

Table 4.3: Solar Cells Damage Equivalent 1MeV

Particel	**MeV/year**
Electrons	$8\ 10^{12}$
Protons	$3\ 10^{14}$
2 Years Total Dose: $6.14\ 10^{14}$	

Then, based on solar cell manufacturer data [12] the efficiency degradation of the cell has been calculated, from 28% at BOL to a 25.5% at EOL. It is shown after 2 years a loss of about 2.5 W.

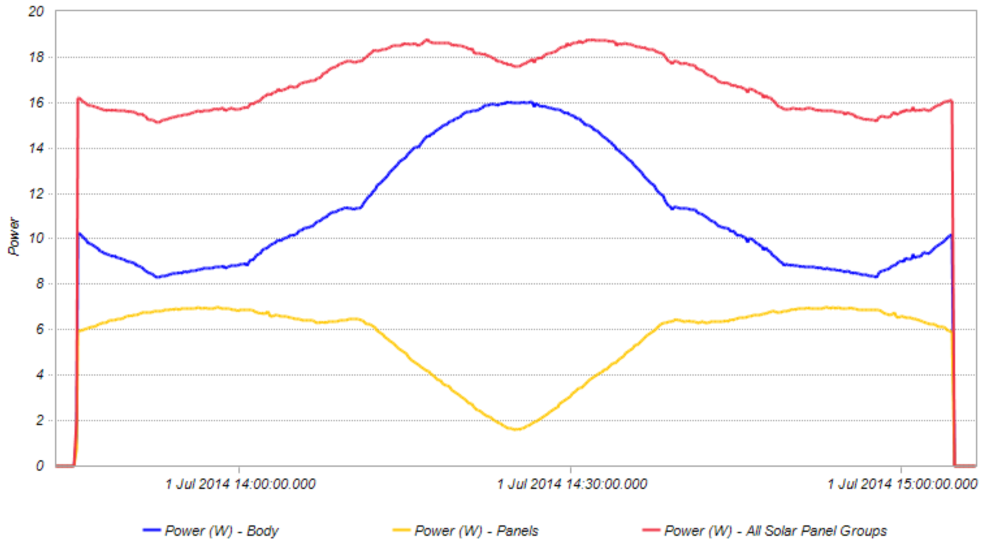


Figure 4.4: Solar Panels EOL Degradation

4.3 Batteries

From the power demands per phase it's crystal clear that the communication phase absorb the maximum power, table 4.1. The solar panels themselves cannot provide entirely this phase demands, therefore it grows the necessity of secondary rechargeable batteries, to overcome the difference.

Moreover the need of batteries raises from eclipse phase power request from the system. Secondary batteries recharge in sunlight, provided that they are not used, and discharge during eclipse and peak power loads.

4.3.1 Time Period Definition

We can consider three types of battery usage:

1. **Streaming Mode**: Daylight only battery usage. Based on the assumption that the contractor will use the service only during working hours (e.g. from 8 a.m. to 18 p.m., about 8 connection per day).
2. **Eclipse Mode**: Eclipse only battery usage.
3. **Transfer Mode**: The system works at each available connection.

4.3.2 Batteries Sizing

The sizing process starts form the typical energy storage formula as given by literature [9]:

$$N = \frac{P_e T}{DODnCr} \tag{4.1}$$

The values used in the sizing are taken from *Clyde Space Ltd.* data-sheets of secondary batteries [8], whereas the the DOD and the charging efficiency is deducted from a typical Li-ion battery currently used on ground [15–17].

Table 4.4: Batteries Sizing

Parameter	**Mode**		
	1	**2**	**3**
T [hr]	0.386	0.562	0.386 @ 10.70W - 0.562 @ 5.93W
Cr [Whr]	30		
$P_e [W]$	10.70	5.93	10.70W @ 0.386hr - 5.93W @ 0.562hr
Cycles	5840	9490	
DOD [%]	0.12	0.07	
n [%]	0.93		
Nr. Batteries	1.23	1.71	3.82
Nr. Real Batt.	2	2	4

4.3.3 Battery Charge and Discharge Model

An analysis on the batteries charging and discharging has been performed using a simulink model. The Battery block implements a generic dynamic model parameterized to represent most popular types of rechargeable batteries. The equivalent circuit of the battery is shown below:

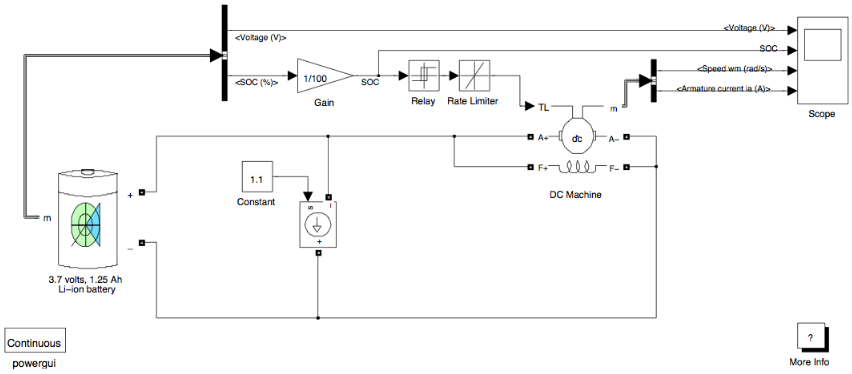


FIGURE 4.5: Lion Battery Model

A typical discharge curve is composed of three sections, as shown in the next figure.

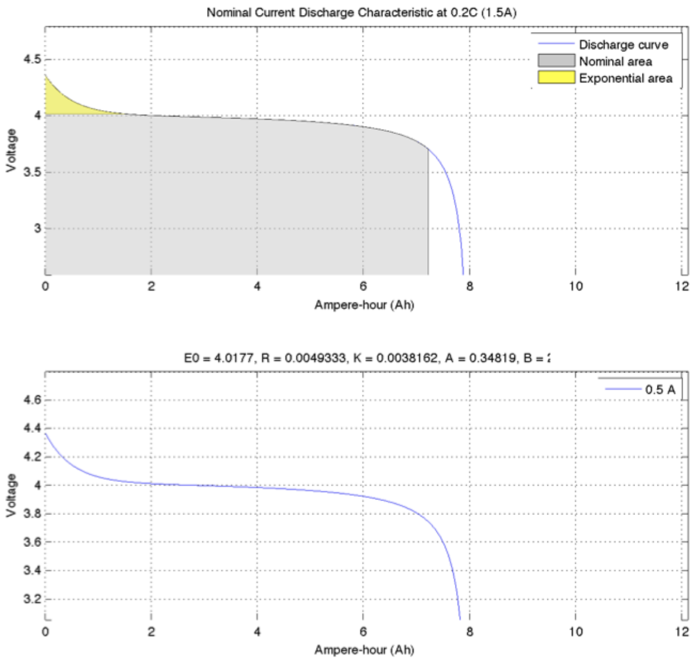


FIGURE 4.6: Battery Discharge Curve

The first section represents the exponential voltage drop when the battery is charged. Depending on the battery type, this area is more or less wide. The second section represents the charge that can be extracted from the battery until the voltage drops below the battery nominal voltage. Finally, the third section represents the total discharge of the battery, when the voltage drops rapidly. When the battery current is negative, the battery will recharge. In this model a 95% threshold of the battery charge has been set to start using them, and when the charge falls below 10% the recharge phase starts [18].

An assessment of the total current consumption has been made, based on data available. Current consumption, with margin [11], settle themselves at values reported hereafter, considering average values from the subsystem installed.

TABLE 4.5: Current Consumption per Phase

Phase	**Current [mA]**
Peak power	1450
Eclipse	1100

4.3.3.1 Begin of Life Performances

At system's BOL batteries perfectly fulfil their role, as can be seen from these figures.

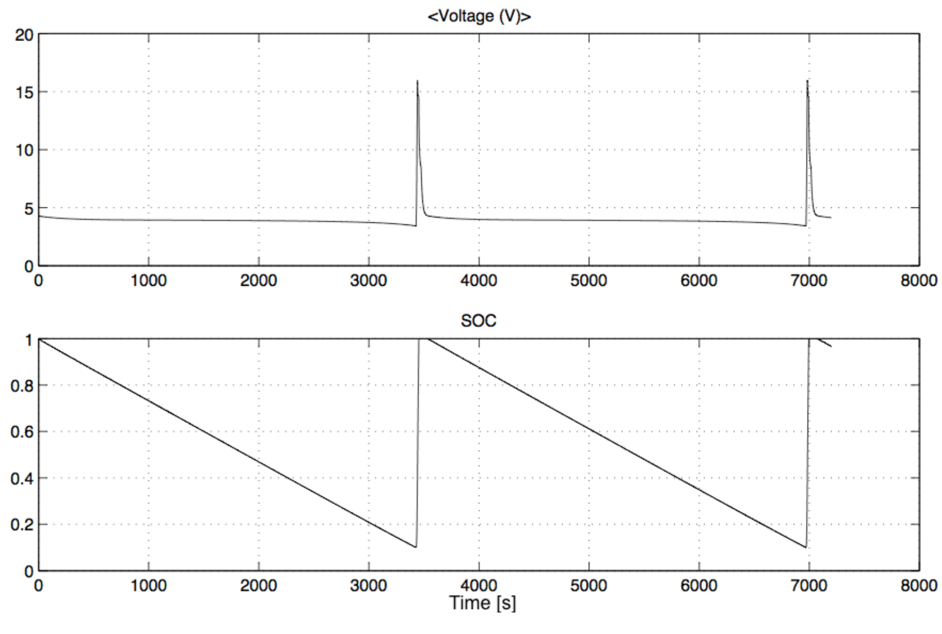


FIGURE 4.7: Battery State of Charge: BOL PeakPower

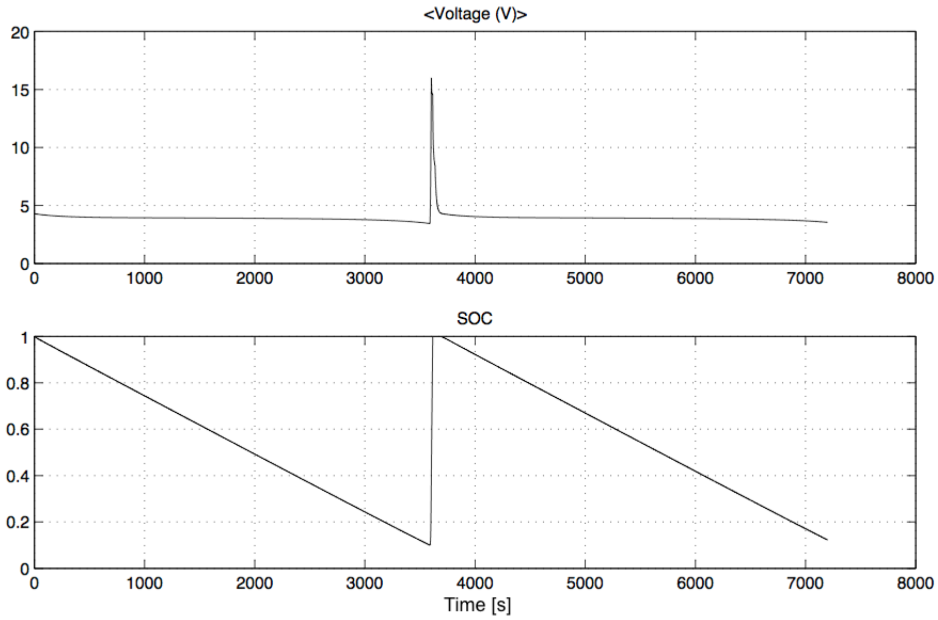


FIGURE 4.8: Battery State of Charge: BOL Eclipse

In particular, discharge times per phase are:

TABLE 4.6: BOL Discharge Time per Phase

Phase	**Time** [s]
Peak power	3430
Eclipse	3600

It can be seen that the batteries are capable of providing the power needed for both phases, in particular the discharge time during the peak power request is greater than the minimum communication time window, set at ∼10 minutes; moreover the system can satisfy the power required for the whole eclipse phase, since for the orbit considered it lasts ∼35 minutes.

4.3.3.2 End of Life Performances

At system's EOL batteries do not perform as well as BOL, in fact a DOD of 12% has been set and results shows criticalities.

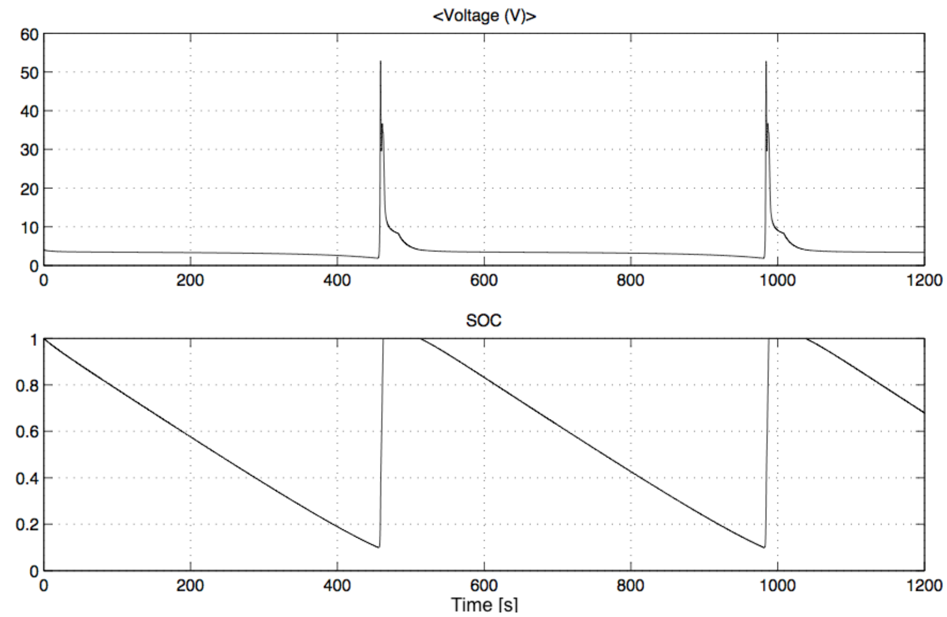


FIGURE 4.9: Battery State of Charge: EOL PeakPower

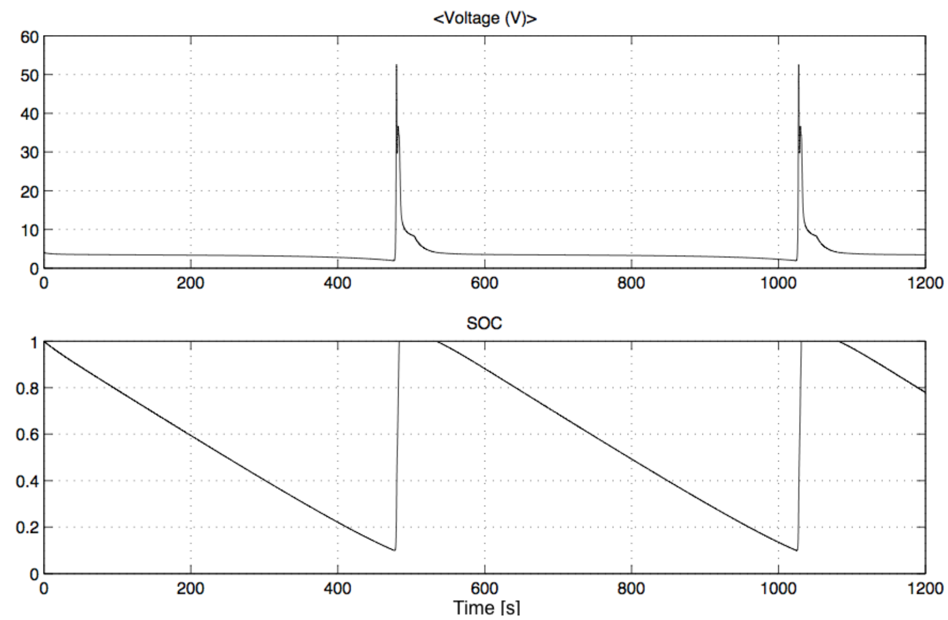


FIGURE 4.10: Battery State of Charge: EOL Eclipse

In particular, discharge times per phase are:

TABLE 4.7: EOL Discharge Time per Phase

Phase	**Time [s]**
Peak power	460
Eclipse	480

At EOL the peak power request cannot be satisfied for its entire duration, but only for a fraction of it. This means that during mission lifetime, the whole system usage shall diminish in time. Moreover, eclipse phase cannot be sustained unless using more batteries. This contrast CubeSat regulations and force designers to face mass limits.
To overcome a full eclipse phase, DOD shall not exceed 58%, or 10 batteries shall be added to the system. The DOD constraint limit each battery usage to a maximum of 950 cycles, whereas the number of batteries lead to mass limitation and exceed regulations limits in terms of chemical power installed.

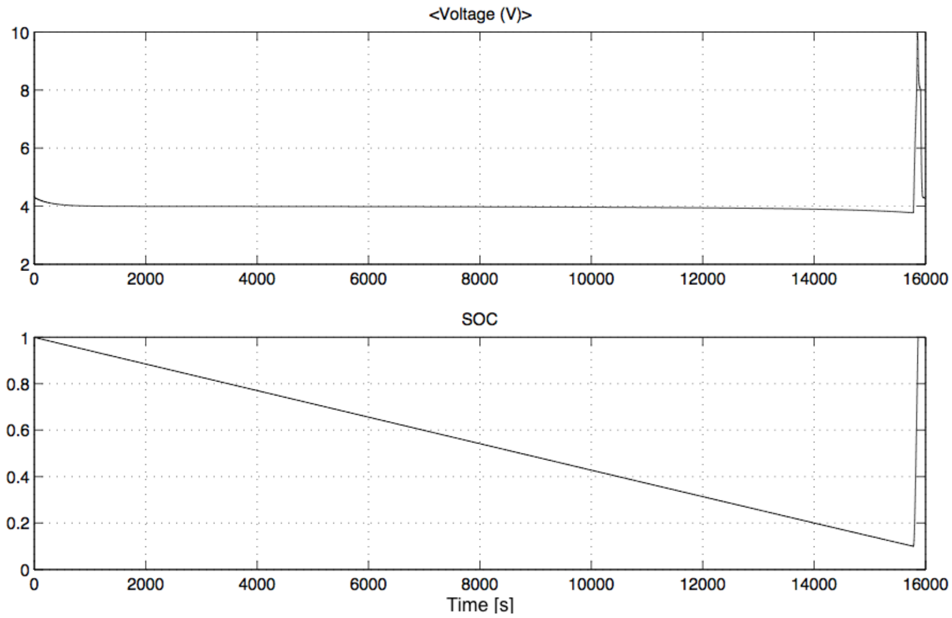


FIGURE 4.11: Battery State of Charge: BOL 10 Batteries

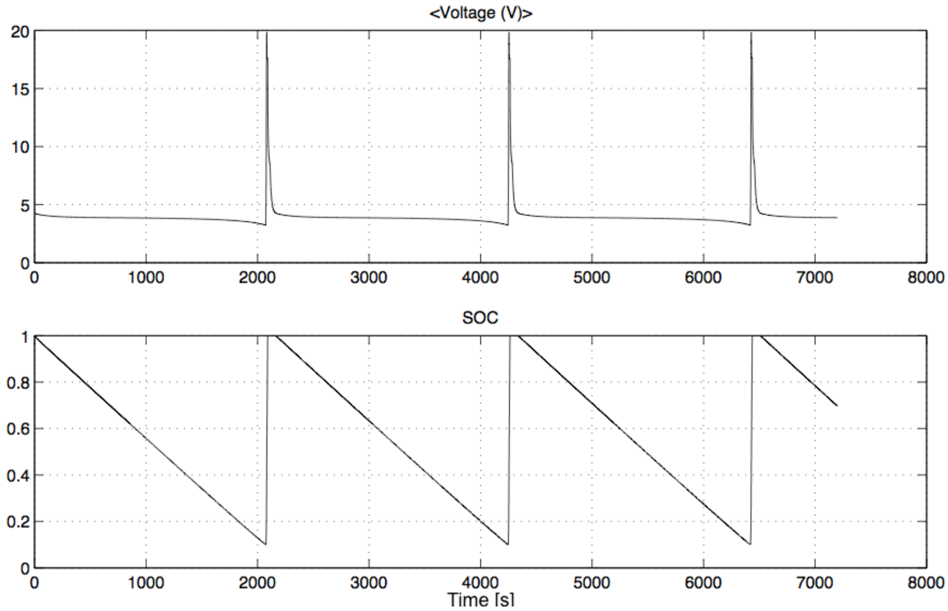


Figure 4.12: Battery State of Charge: EOL 10 Batteries

All things considered, the design approach led to adding 6 more batteries to allow the system to survive during eclipse phase at its end of life. Moreover this selection gives the system even more robustness at its begin of life.

4.4 Mass Budget

A mass budget for the subsystem has been performed combining data coming from datasheets [8], derived from specific (with respect to mass) figures of merit, and from literature [9] [19] for what concern harnesses. A summary table is shown hereafter:

Table 4.8: EPS Mass Budget

Mass [kg]	**Batteries**	**Solar Panels**
Item	1.16	0.19
Harness	0.1	
Total	1.450	
Total with margins	1.51	

Margins are taken from literature [11].

4.5 Conclusions

From the analysis above, the selection of two batteries used for the peak power request and twos for the eclipse phases, gives the system the robustness and redundancies qualities which are the cornerstones of any satellite subsystem design.

The requirements of CubeSat design prescribe that the total amount of chemical power on board shall not exceed 100 Whr. The design process lead to 120 Whr of chemical power installed, hence passing the limits.

The actual choice has been selected after a trade-off analysis based on total time available for connection and CubeSat requirements. Since the access windows are narrow, to avoid the use of batteries during the peak power requests (which is the phase for which design exceed the prescribed limits), the connection time with ground shall be lowered drastically, making the overall system inappropriate for telemedicine purposes.

Chapter 5

Thermal Control Subsystem

Space is a challenging environment, especially from a thermal point of view. In order to maintain the whole satellite and its subsystems within their operative temperature limits, a thermal analysis must be carried out, aiming to provide a thermal control system that will guarantee safe temperature ranges to the spacecraft, during all its mission phases. Nonetheless the design of a thermal control is also fundamental to ensure that the temperature gradient requirements are met, avoiding large gradient that could cause structural deformation an eventually the mission failure.

Due to the small size and small amount of power available to the satellite, and due to design requirements, the thermal control options are limited to passive devices. Keeping the thermal design as simple as possible and by avoiding the use of active components, relying only on surface finishes and insulation blankets will lead to a lighter system, far less expensive to build, more reliable, and easier to test.

5.1 Requirements

The following table resume the thermal requirements of all vehicle elements, by means of operative temperatures ranges based on the technology available nowadays on the market. This requirements will drive the thermal control design.

TABLE 5.1: Equipment Thermal Requirements

Subsystem	**Operative Temperature Limit**	
EPS	-40°C	85°C
Battery	-10°C	40°C
Solar Panels	-85°C	100°C
Transmitter	-25°C	85°C
Receiver	-25°C	85°C
Antenna	-25°C	85°C
OBDH	-20°C	60°C
Reaction Wheel	-40°C	80°C
General Electronics	-40°C	85°C

5.2 Thermal Analysis Background

Typically heat transfer occur in three ways: *convection*, *conduction* and *radiation*. However, since at very high altitudes the available air for natural cooling is negligible, in space is all about conduction and radiation between components of the satellites and the external environment. It follows that a good knowledge and understanding of space thermal environment is fundamental in order to consider every heat transfer phenomena that will occur in space during the whole mission. For usual Earth orbits, the most significant phenomena concerning heat transfer are the following:

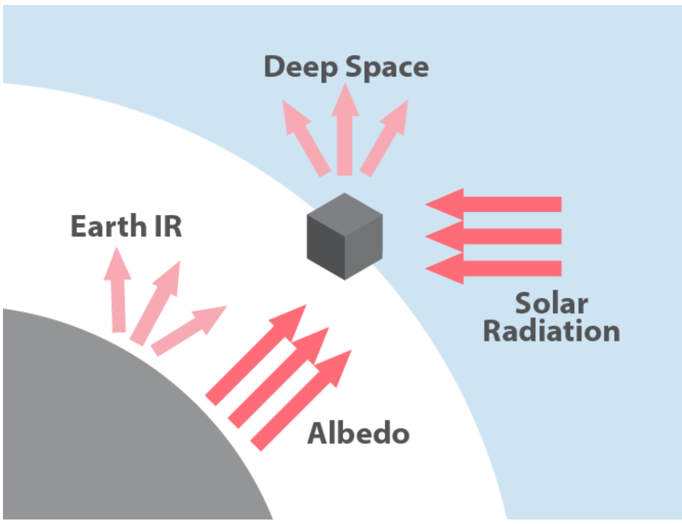


FIGURE 5.1: Thermal Environment

- Radiation to **Deep Space**: Space behaves as a universal sink, that dissipate all the heat fluxes coming from external bodies without changing is own temperature. Thermal analysts must be cautious because such a sink may eventually bring the spacecraft down to low critical temperatures. It's temperature as been fixed at: 2.73K [20, 21].

- **Direct Solar Flux**: Sunlight is the major source of environmental heating on most spacecrafts. Because the Earth's elliptical orbit, its intensity varies approximately around 3.5% in a year (summer solstice: 1414 W/m^2 to winter solstice: 1323 W/m^2). For the following analysis a mean value of 1368.5 W/m^2 has been used.

- **Albedo**: It's the fraction of sunlight reflected from a planet. Accounting for the terrestrial albedo it is not straightforward because it depends on the current aspect angle which in terms depends on Sun-Satellite-Earth position and on how far the satellite is from the termination line. Albedo will therefore vary from zero to a maximum value, that corresponds to the termination line at noon. In this point, and for Low Earth Orbits, the average overall albedo of Earth ranges between 30% and 35%. Moreover, such a value varies widely locally across the surface, depending on the geological and environmental features.

- **Earth Infrared Radiation**: The part of sunlight that is not reflected as albedo is absorbed by Earth and eventually re-emitted as IR energy. This emitted energy can be regarded as a blackbody radiation considering the planet with an effective average equivalent temperature of 250K [20, 21].

- **Equipment Power Dissipation** Energy dissipated by electronic components inside the satellite due to Joule effect.

TABLE 5.2: Equipment Power Dissipation

ADCS	0.61 W
OBDH	1.02 W
EPS	1.21 W
TXRX	0.74 W

The whole thermal analysis has been carried out using the Thermal Simulation Tool present in the software SolidWorks. Two model are used, a detailed one for Hot and Cold case analysis and a simplified one for the transient analysis. Both models have been modelled taking into account the main features that control the thermal behaviour of the satellite: materials and geometry.

5.3 Steady-State Thermal Analysis: Cold and Hot Case

The first analysis treat a simple steady state condition regarding the worst cases that the satellite can undergo while orbiting, named the hot and the cold case. As far as the thermal control is concerned, the only difference between these situations are the impinging flows and equipment power dissipation. For the analysis the satellite has been modelled with the following assumptions:

- For the *main structure*: an aluminium alloy 6061-T6 with a hard-anodized finishing is used.
- The *equipment* has been modelled as blocks composed by copper, aluminium and FR4,.
- *Solar Arrays* are modelled as a PCB board with triple junction GaAs solar cells.
- *PCB boards* assumed as composed by 30% of copper and 70% of FR4.

This modelization provide a mathematical representation of the physical surfaces of the satellite and its components, and is used to calculate the radiation couplings between all surfaces in the model, as well as heating rates to each surface from external flux sources such as solar, Earth IR, and albedo radiation. The thermal properties that characterize the thermal behaviour of the spacecraft in space are listed in the following table. For the equipment blocks (ADCS, OBDH, EPS and TXRX) mean values, weighted on each block structure/specifications based on technology available today on the market, has been used.

TABLE 5.3: Model Thermal Properties [12, 20, 22–24]

	k [W/mK]	**C** [J/kgK]	α	ε
Structure	167	896	0.88	0.88
Solar Cell	50	350	0.92	0.85
PCB Board	120.44	385	0.5	0.91
Copper	400	390	0.08 - 0.93	0.03-0.78

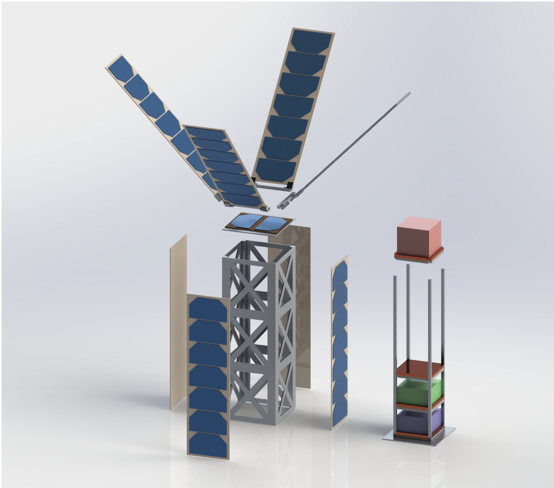


FIGURE 5.2: Steady-State Model

5.3.1 Hot Case

The satellite in the **hot case** receives the heat flux contribution from the Sun, the Albedo and the Earth IR, because it is steadily placed in between the imaginary line connecting the Sun and the Earth. In this case all the equipment is considered be active and dissipating power.

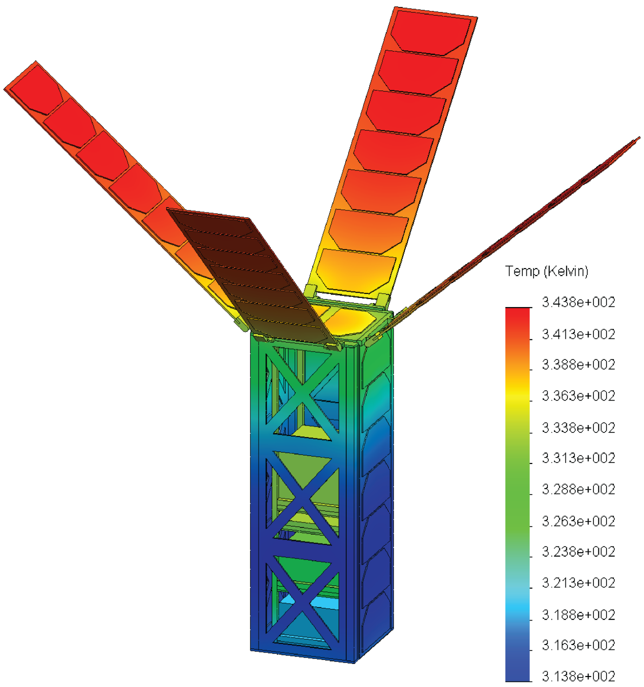


FIGURE 5.3: Hot Case

5.3.2 Cold Case

In contrast, in the **cold case** the satellite is hit only by the radiation due to the Earth infrared emission, because it is steadily placed in a diametrically opposed point with respect to the hot case, so that it is in the middle of the umbra cone of Earth. As a worst case scenario, in this case equipment are assumed to be powered off, so no power dissipation occurs.

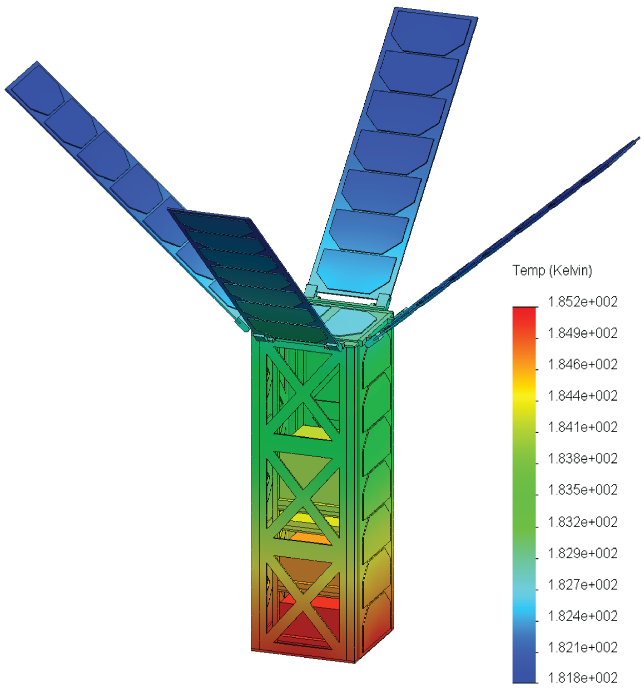


FIGURE 5.4: Cold Case

During eclipse the solar-array temperature drops dramatically. In this period, the temperature of the electronics boxes and other components also drops; however, because their thermal mass is high, they do not cool nearly as fast as the relatively lightweight solar array. The result is that the spacecraft can often go through the eclipse without falling below the minimum allowable operating temperature of the electronics, as it will seen in the transient analysis.

5.4 Transient Thermal Analysis

Since equatorial orbits are the one of interest, the model is build considering an orbit of zero inclination and an altitude of 1400km, at this altitude a satellite perform about 12 revolutions per day, with an orbit period of 114 minutes, where 30% of this time (about 35 minutes) is spent in umbra, while the remaining 70% in sunlight, the satellite will so experience alternate light and eclipse phases. Due to this periodicity and the thermal behaviour of spacecraft, temperatures will oscillate reaching eventually a steady state condition.

To perform this analysis a time dependant scenario has been built considering the different heat exchanges that vary and occur orbiting around Earth. The time dependency has been simulated calculating the variation of the heat flux for each satellite surface during orbit taking into account: eclipse periods, surface-to-surface shadowing, surface-to-Sun and surface-to-Earth perspective [20]. Due to the fact that the orbit considered is at zero inclination the implementation is quite straightforward without loosing too much accuracy.

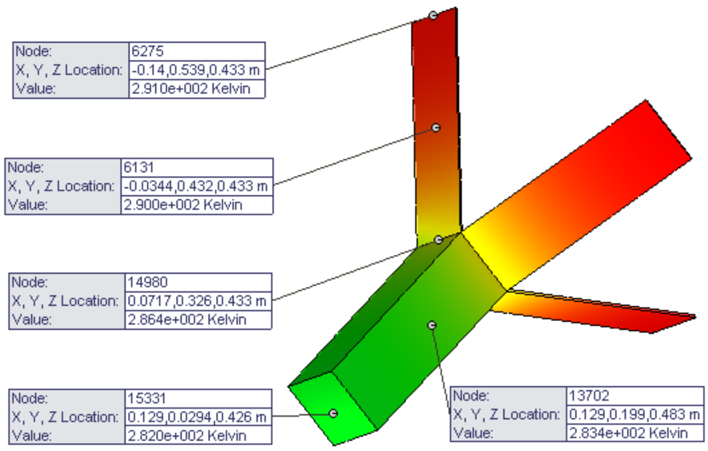


FIGURE 5.5: Transient Analysis Termperature Distribution

Here is reporter a instant-picture where the temperature of 5 nodes along the satellite are highlighted in order to obtain an idea of the temperature distribution. It is assumed that the satellite is at an initial temperature of 290K, from the transient results is clear how the temperature stabilize to a steady-state condition, with oscillation around 280K, after few orbits.

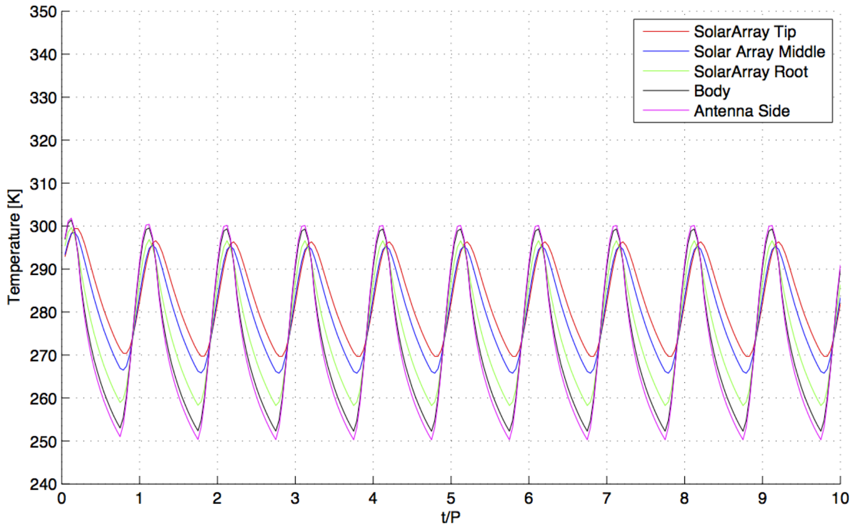


FIGURE 5.6: Transient Analysis

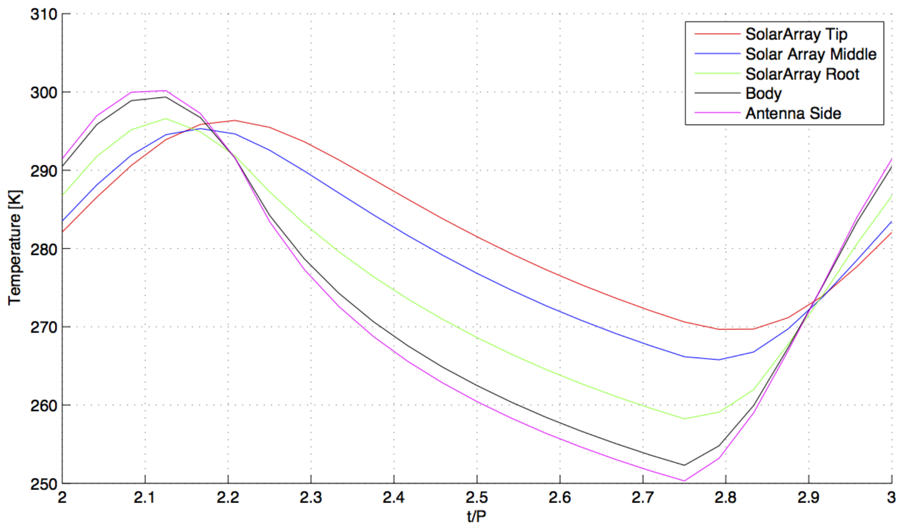


FIGURE 5.7: Transient Analysis 1 Period

Considering a single revolution it can be noticed, with this preliminary analysis, that almost all satellite components temperature remains in the required limits, special care must be taken towards the antenna side of the satellite, which taking into account a margin of ±10°C [20], it can go under the low temperature limits. A possible solution to avoid issues, is to apply an insulation layer on critical components, such as onboard computer, telecommunication system and batteries, for the latter one *ClydeSpace* offer an integrated battery heating system with thermostat that can maintain batteries above 0°C [8].

5.5 Conclusions

Although the current numerical simulation is based on a simplified model of the system, it's possible to state, particularly looking at the transient analysis, that the temperature requirements are well satisfied without the use of any specific passive control. If needed a passive one can easily keep the system in safety, e.g. changing the internal equipment positioning, using different surface finishes to improve heat emission or absorption where needed or covering equipment with insulation layers.

Despite best efforts and the sophistication of today's analytical codes and computer workstations, flight experience teaches that predicted temperatures are not always precisely accurate. For components that have no thermal control or have passive thermal control only, an uncertainty margin of at least $\pm 10°C$ should be included in all cases in determining the maximum or minimum expected flight temperature [20]. Further analysis and actual thermal testing are anyway needed, in order to provide more precise results that will lead to a well tailored thermal control system to ensure that the system can operate without failure.

Chapter 6

Attitude Determination and Control

As seen during the analysis of the telecommunication subsystem, BRIDGE requires an attitude control system due to pointing requirements coming from link budget margins. In the following section the design of such a system is carried out by modelling the system into the Simulink environment, a detumbling and pointing budget are analyzed.

6.1 Equipment: Actuators and Sensors

The MAI-101 reaction wheels system has been used for the ADCS model simulation.

Table 6.1: Reaction Wheel Specification

Reaction Wheel: *MAI-101*	
Maximum momentum storage	1.1 mNms
Maximum torque	0.635 mNm
Wheel Inertia:	1.5 10^{-5} kgm^2

Figure 6.1: MAI-101 Reaction Wheel

For the attitude determination the system is assumed to be equipped with a sun sensor system, it measures the two angles that sun radiation form relative to the face the hardware is mounted on, providing in this way a measure of the satellite attitude. Information on the reference sensor has been taken from [17]. To determine also a value for the angular velocity of the satellite it has been modelled also a gyroscope that evaluate it directly. On board are assumed to be installed a mechanical gyro system, capable of sampling angular velocities for each axis.

6.2 Dynamic and Kinematic Model

Supposing the satellite behaves as a rigid body, the angular velocity may be integrated exploiting Euler's moment equations, expressed in the principal axes of inertia reference frame, eventually taking into account the presence of rotors, assuming to have three reaction wheels aligned with the body axes, all with the same moment of inertia I_r. Euler's equations used to model the satellite are shown hereafter:

$$\begin{cases} I_x\dot{\omega_x} + (I_z - I_y)\omega_y\omega z + I_r(\omega_{rz}\omega_y - \omega_{ry}\omega_z) = M_x \\ I_x\dot{\omega_y} + (I_x - I_z)\omega_x\omega z + I_r(\omega_{rx}\omega_z - \omega_{rz}\omega_x) = M_y \\ I_x\dot{\omega_z} + (I_y - I_x)\omega_y\omega x + I_r(\omega_{ry}\omega_x - \omega_{rx}\omega_y) = M_z \end{cases} \tag{6.1}$$

Table 6.2: Satellite Inertial Properties

3RiDGE Inertial Properties	
I_x	0.0647 Kgm^2
I_y	0.0184 Kgm^2
I_z	0.0647 Kgm^2

Once the angular velocity is known, it's easy to reconstruct the attitude in terms of the Euler angles, by integrating the angular velocity vector by means of the kinematic equation for the derivatives of the Euler's angles themselves. Selecting a specific sequence of rotation, namely a 321 one, the corresponding equations are:

$$\begin{cases} \dot{\varphi} = \frac{\omega_z \, cos\psi + \omega_y \, sin\psi}{cos\vartheta} \\ \dot{\vartheta} = \omega_y \, cos\psi - \omega_z \, sin\psi \\ \dot{\psi} = \omega_x + (\omega_z \, cos\psi + \omega_y \, sin\psi) \, \frac{sin\theta}{cos\theta} \end{cases} \tag{6.2}$$

The correctness of the implementation has been checked by evaluating the angular momentum and the kinetic energy in the inertial reference frame for a torque-free motion, validating their conservation in time as predicted by theory [25, 26].

6.3 Environmental Disturbance Torques

For the analysis the following disturbance are considered, due to the altitude considered the atmospheric drag contribution is assumed as negligible.

- **Gravity gradient:** Due to the fact that Earth's gravity field is not uniform, there could be a torque acting on the satellite, especially for large or non regular satellites. Even if the this kind of disturb is small its effect can be considerable if a long period is considered. The gravity gradient depends on the relative position of the Earth and the satellite and satellite inertial properties.

- **Solar pressure:** This disturb is due to the pressure generated by the impinging solar radiation on the satellite surface, generally the center of mass and the solar pressure center are not coincident. This gives origin to a torque, which is function of satellite surface properties, geometry and relative position between Sun and satellite.

- **Magnetic torque:** Generally satellites present a residual dipole magnetic moment, which its interaction with Earth's magnetic field generates a torque. It is assumed for the satellite a residual dipole of [5.57 5.57 5.57] $10^{-4} Am^2$ with modulus of $0.001 Am^2$. To model the Earth magnetic field the built-in Simulink block is used.

Looking at the following images is clear that without any kind of control, disturbances tend to diverge the satellite orientation. Even though a control technique is needed due to telecommunication constraint, it's evident that it would have been needed also to perform a nadir pointing attitude.

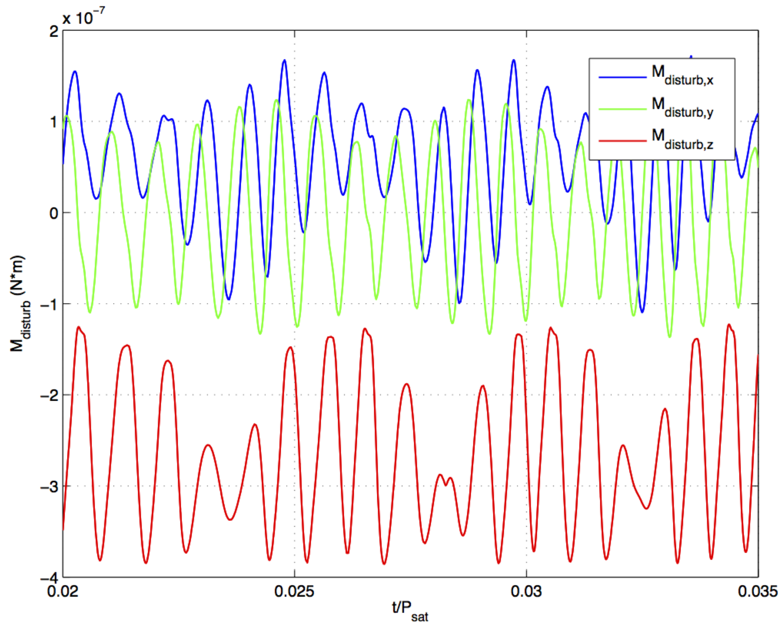


FIGURE 6.2: Uncontroled Motion: Disturbing Torques

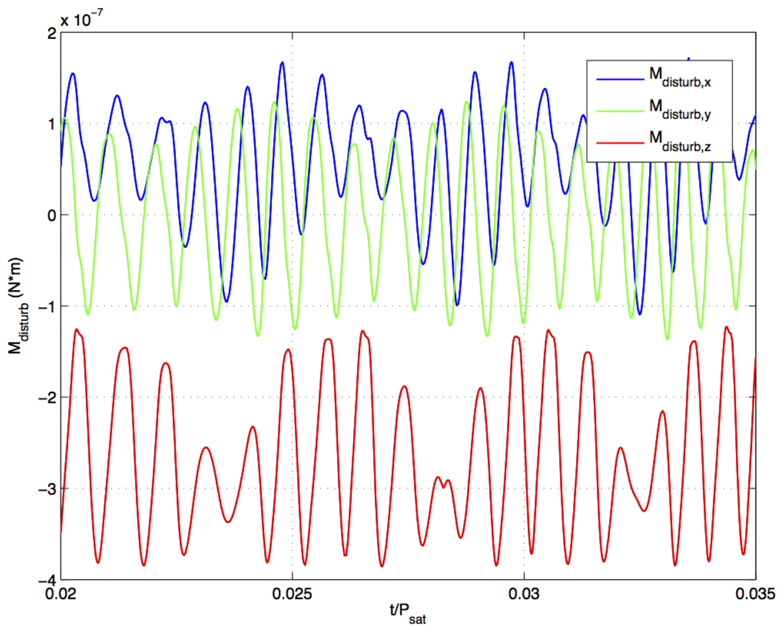


FIGURE 6.3: Uncontroled Motion: Euler Angles

6.4 Analysis

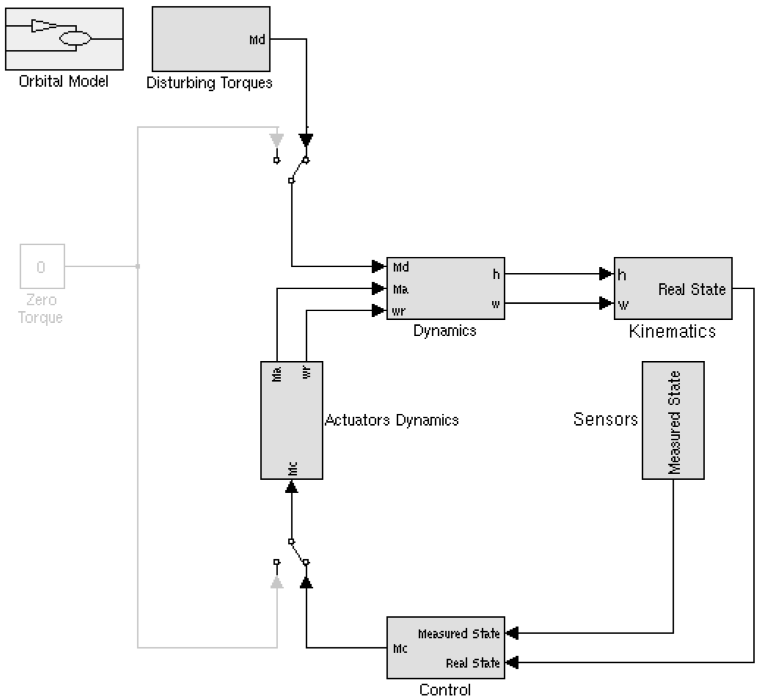


FIGURE 6.4: Simulink Scheme

Here are presented the results coming from the implementation of an optimal control law, which gain matrix is weighted by a cost function that takes into account attitude perturbation minimization and actuators control effort, using the Matlab linear quadratic regulator tool. Two different scenarios are considered:

Detumbling:	*Initial Condition*	$\varphi = \vartheta = \psi = 30°$	$\dot{\theta} = 5°/s$
	Target	$\varphi = \vartheta = \psi = 0°$	$\dot{\theta} = 0.1°/s$
Area Pointing:	*Initial Condition*	$\varphi = \vartheta = \psi = 5°$	$\dot{\theta} = 1°/s$
	Target	$\varphi = \vartheta = \psi = 0°$	$\dot{\theta} = 0.1°/s$

6.4.1 Detumbling

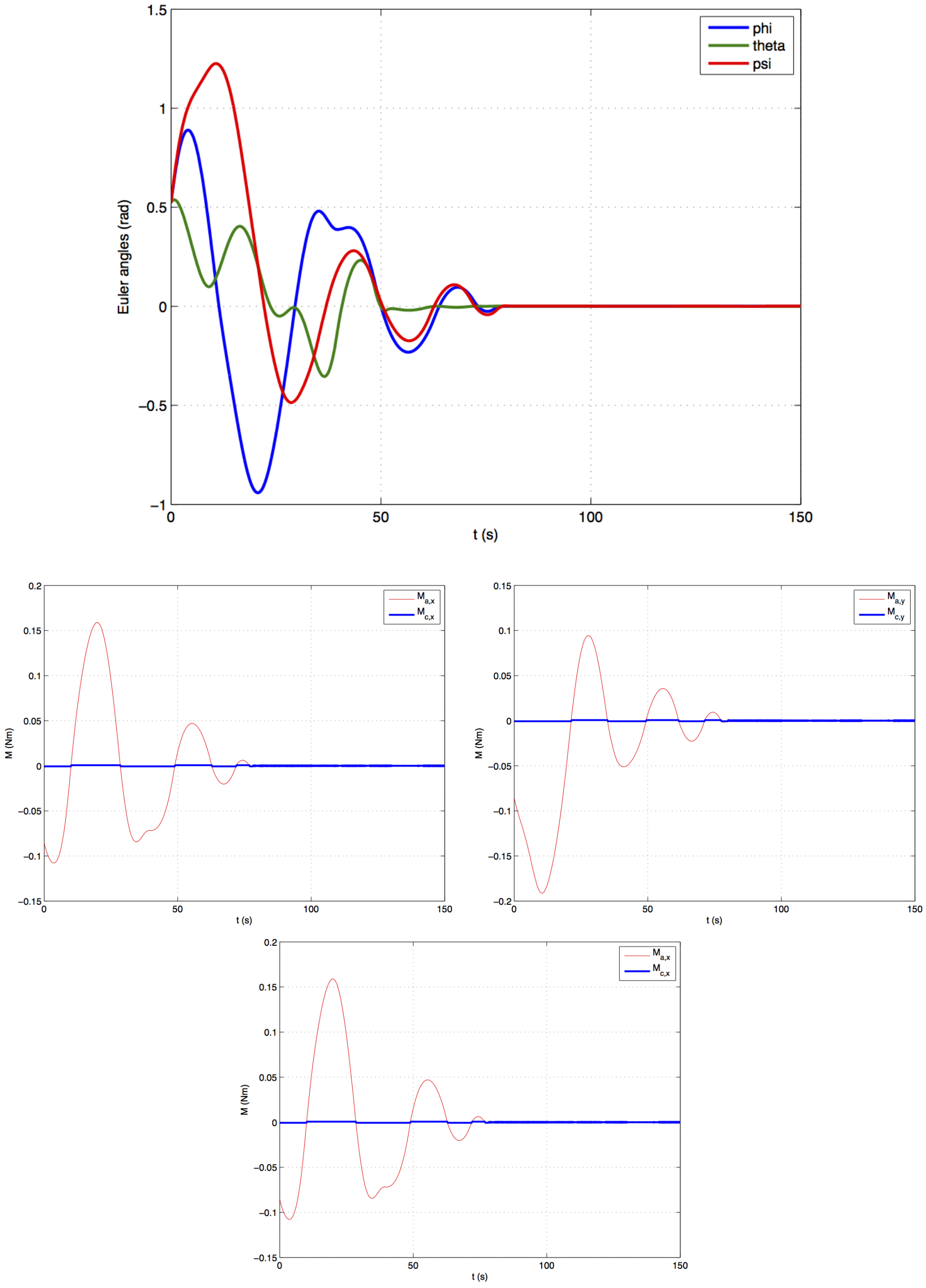


FIGURE 6.5: Detumbling Analysis, Euler Angles and Actuator Torques

From above results it can be seen that detumbling control can be successfully achieved with the implementation of a reaction wheel system, pointing the satellite to its operative attitude in a reasonably short time interval.

6.4.2 Area Pointing

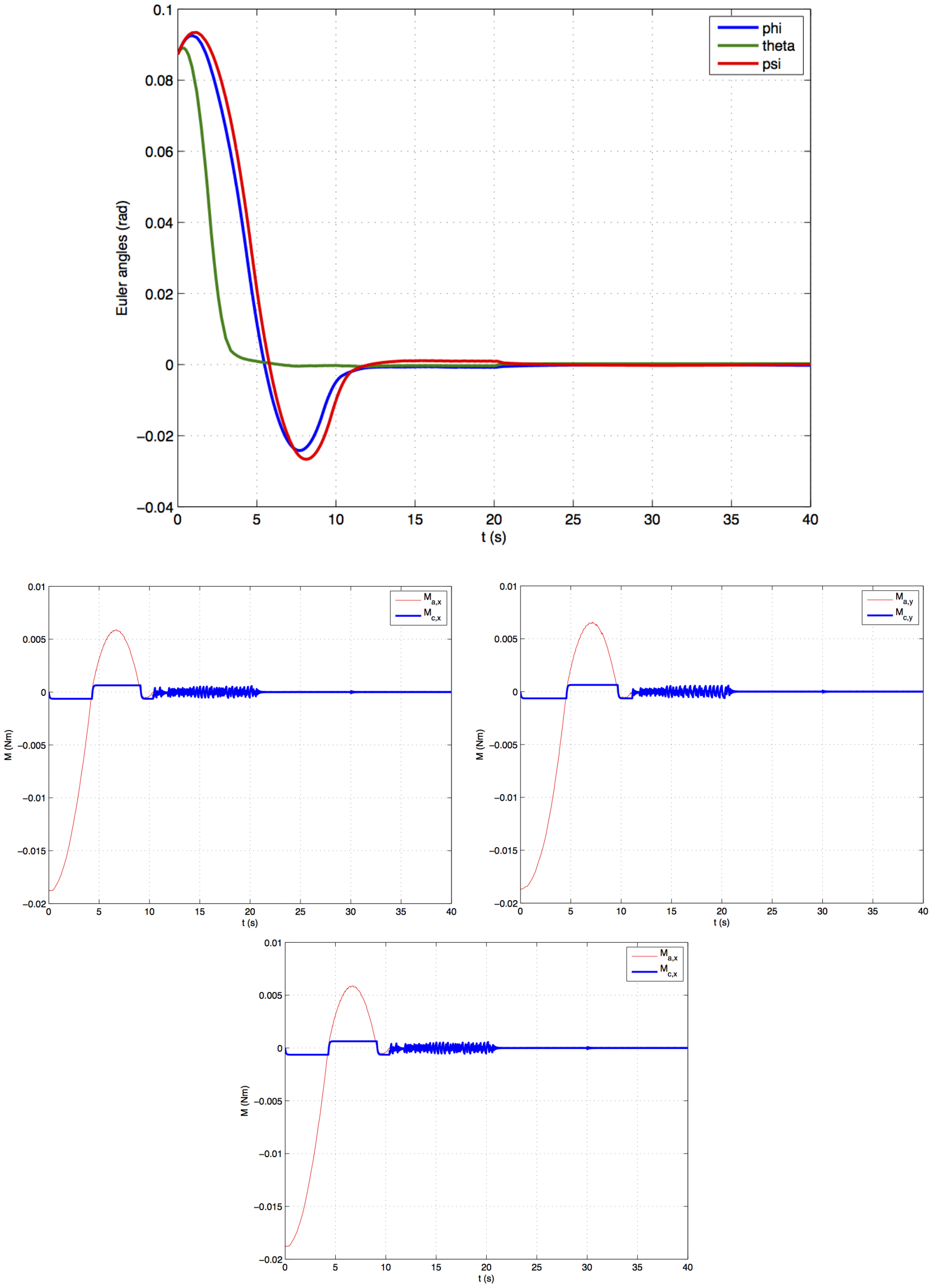


FIGURE 6.6: Area Pointing Analysis, Euler Angles and Actuator Torques

The area pointing simulation shows that the system can easily provide the desired orientation, also with a good order of promptness.

6.4.3 Momentum Damping

Because the system of spacecraft and reaction wheels conserves momentum, as the spacecraft loses momentum the wheel speeds must increase. As the wheels are spun up to provide attitude control they will eventually reach their saturation limit with regards to wheel speed if left unchecked. To eliminate momentum from the spacecraft/reaction wheel system, an external means of torquing is required. Magnetic torque bars can be used for this purpose. By creating a magnetic torque on the space craft in a controlled manner the wheels can be despun [25, 27].

As a preliminary sizing of a momentum damping system a COTS magnetorquer [17] is considered:

Table 6.3: CubeTorquer Specifications

Magnetorquer: *CubeTorquer*	
Nominal magnetic moment (D):	$\pm 0.2 Am^2$

Given an orbit period of 114 minutes and taking a worst-case disturbing torque of $2.8\ 10^{-7} Nm$, based on simulation results, every orbit a reaction wheel accumulate about $1.91\ 10^{-3} Nms$. Assuming a desaturation time of 10 minutes, the magnetorquer must provide a torque of about $3.19\ 10^{-6} Nm$.

$$\tau_{MTQ} = \frac{\tau_{disturb}\, t_{orbit}}{t_{desaturation}} \tag{6.3}$$

At an altitude of 1400 km in an equatorial orbit, the magnetorquer considered can provide a torque of about $6.77\ 10^{-6} Nm$, which can afford the desaturation task required at each orbit.

6.5 Conclusions

Given the analysis results illustrated, it's possible to state that:

- Thanks to the patch antenna large beamwidth no high pointing accuracy is required, even the simple attitude control analyzed is capable to satisfy telecommunication pointing requirements. Moreover the system is capable to overcome a tumbling motion, right after launcher release;
- due to the small momentum capacity of CubeSat reaction wheels, the attitude control system require a momentum damping technique, task that can be accomplished by a set of magnetorquers.

Chapter 7

Deorbit Analysis

The satellite is constrained to an altitude in which the atmosphere is so rarefied that any spacecraft is destined to remain in its orbit without any disposal strategy.
In the last decade a new sensibility evolved from the satellite population increase, and an EOL disposal strategy is now mandatory for any orbiting object [28].

Problem arises from the nature of the satellite system that is considered. A CubeSat is generally put into a low-LEO, typically under 700km, where it re-enter in the atmosphere naturally without any further mechanism. BRIDGE is put into a circular 1400km orbit, in which atmosphere is absent, hence a certain force is to be applied to lower its altitude. Several methods are analyzed in the following sections.

7.1 Strategies

The first two strategies employ the use of chemical propulsion, whereas the last one, deals with a new type of mechanism to perform an orbital maneuver. All type of maneuvers are considered impulsive.

7.1.1 Hohmann Transfer

The Hohmann transfer represent the optimal maneuver between two coplanar circular orbits, in the sense of minimum Δv. Fixing the initial orbit radius at 1400km, the total Δv increase in a quasi-linear fashion, as the final orbit radius decrease.
Since the maximum Δv obtainable by thruster available on market for CubeSat use is $\sim$35 m/s [29], the maximum decrease in height is in the range of $\sim$80 km.

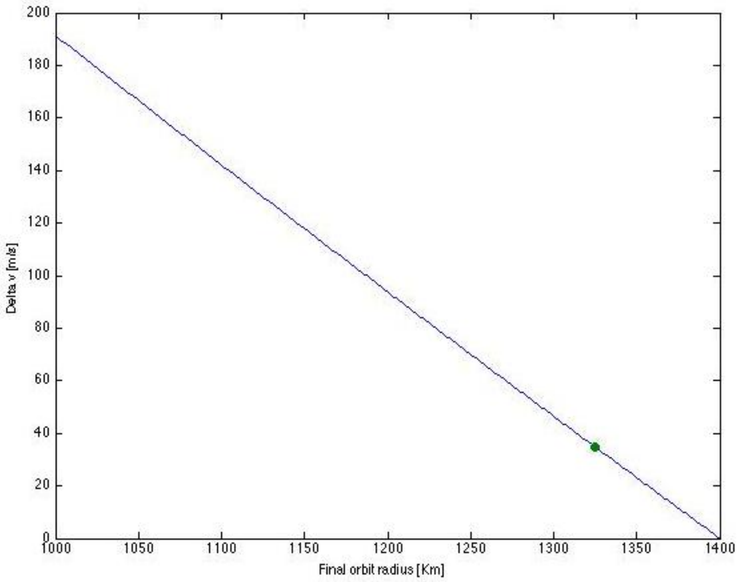


Figure 7.1: Hohmann Maneuver Budget

Such decrease isn't enough to catch the outer part of the atmosphere (although very rarefied), for which at least ∼190 m/s are needed.

7.1.2 Perigee Height Change

The maximum decrease in altitude is obtained by lowering the periapsis, keeping constant the apoapsis. It is clear that the total Δv to reach the atmosphere is lower than an Hohmann maneuver, at ∼95 m/s.

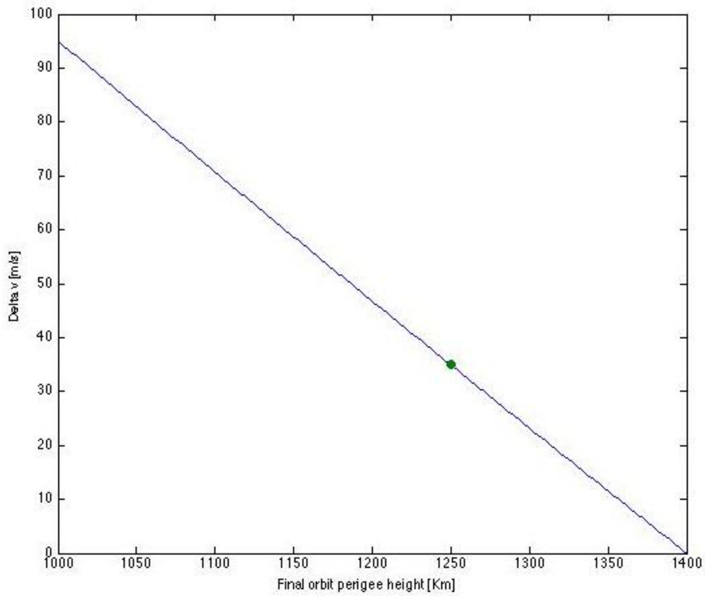


FIGURE 7.2: Perigee Change Budget

Having in mind to use off-the-shelf equipment[29], the ∼35m/s force the satellite to reach a minimum periapsis height of 1250 km.

7.1.3 Tether

Using electrodynamic drag to greatly increase the orbital decay rate, an electrodynamic space tether can remove spent or dysfunctional spacecraft from low Earth orbit rapidly and safely. Moreover, the low mass requirements of such tether devices make them highly advantageous compared to conventional rocket-based de-orbit systems. However, a tether system is much more vulnerable to space debris impacts than a typical spacecraft and its design must prove to be safe to a certain confidence level before being adopted for potential applications.

The electrodynamic drag concept is based on the exploitation of the Lorentz force due to the interaction between the electric current flowing in a conductive tether and the geomagnetic field. The decelerating Lorentz force F (electrodynamic drag) depends in a complex way on the design parameters of the system, the orbit and the characteristics of the local ionosphere [30].
Concern about the tether interaction with other objects in space and its correct deployment are still under investigation.

The main equation considered to evaluate the decay time is the following [31]:

$$\Delta t = \frac{M_{s/c} R}{12\, L^2\, B_E^2\, R_E^6\, cos^2\alpha\, cos^2\lambda} \tag{7.1}$$

Where:

- $M_{s/c}$ is the spacecraft mass;
- R is the end-to-end resistance value, computed as

$$R \;=\; \frac{\rho\, d\, L^2}{m_{tether}} \tag{7.2}$$

  with ρ as the tether resistivity, m_{tether} as the tether total mass, and L as the total length;
- B_E is the strength of the magnetic field;
- R_E is the Earth radius;
- α is the orientation of the tether with respect to the radial component of the orbit;
- λ is the orbit inclination.

A brief summary of the values adopted in the model is presented hereafter:

TABLE 7.1: Tether Deorbit

Parameter	**Value**
$M_{s/c}$	4 kg
R	58 Ω
L	250 m [32]
m_{Tether}	0.08 kg
B_E	31 μT
R_E	6378 km
α	0 deg
λ	10 deg

The use of such a system [32] guarantee a decay time in the order of 38 years.

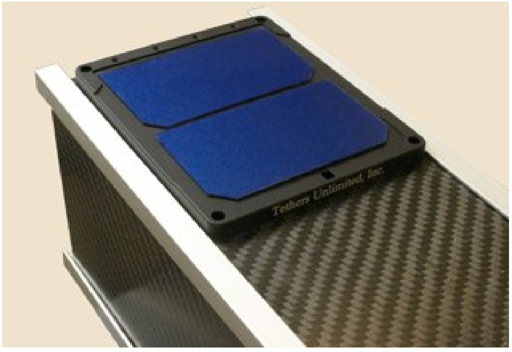


FIGURE 7.3: TermTape Tether

Combining an impulsive strategy, i.e. Hohmann maneuver, and exploiting electrodynamic drag, a decay time of 26 years is achieved. Although out of boundaries set by international regulations [28], it's a good starting point for further analysis.

7.2 STK Analysis

Using the lifetime tool installed into STK, several analysis are performed using the atmosphere model embedded, namely *Jacchia 1970*. *NRLMSISE 2000*, although more recent, was not used because its applicability range is 0 to 1000 km, hence not suitable for our purposes. On the other hand, *Jacchia 1970*, computes atmospheric density based on its composition, which depends on the satellite's altitude as well as a divisional and seasonal variation, whose valid range is a more appropriate 90 to 2500 km [13].

7.2.1 Impulsive Maneuvers Results

Simulating the two impulsive chemical maneuvers presented into the STK environment leads to the following results:

TABLE 7.2: STK Deorbit

Strategy	**Time to deorbit [years]**
Hohmann	>200
Perigee change	>200

Such deorbit times do not falls within the regulation limits set at 25 years. Therefore it's mandatory the use of the Tether system [32], although not fully space qualified, in terms of several tests.

7.3 Backup Strategies

Two other possible solutions deals with a synergy between the chemical propulsion and some drag enhancement mechanism.

7.3.1 AEOLDOS

The first drag enhancement mechanism taken into considerations is a sort of deployable sail, which area extend to $3m^2$. The AEOLDOS [8], acronym for Aerodynamic End-Of-Life DeOrbit System, module deploys an aerobrake membrane on command or as part of a pre-planned end-of-mission disposal sequence. Re-entry times from LEO are vastly reduced, enabling higher missions and ensuring compliance with debris mitigation recommendations and requirements.

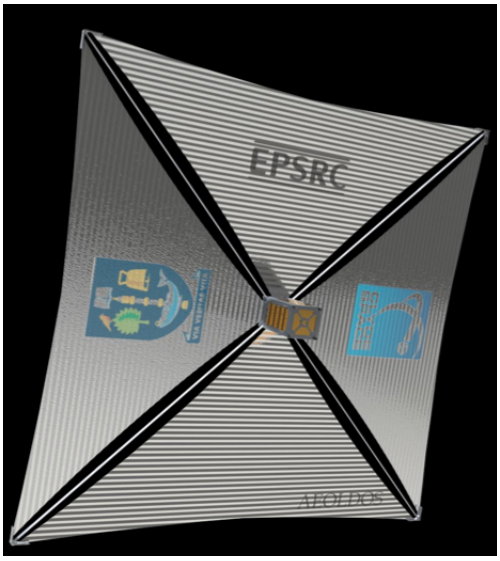


FIGURE 7.4: AEOLDOS System

AEOLDOS will be deployed after the impulsive maneuvers, and it's capable of reducing the orbit lifetime accordingly:

TABLE 7.3: AEOLDOS Deorbit

Strategy	**Time to deorbit [years]**
Hohmann	152
Perigee change	98

7.3.2 Inflatable Balloon

Another more advanced concept to deorbit a spacecraft make use of an inflatable Kapton balloon of 0.5 m diameter. It exploit both aerodynamic drag as well the atomic oxygen interaction with itself to produce a net force capable of slowing the spacecraft itself [33]. The possibility of increasing the balloon size will enhance the efficiency of deorbiting. The STK analysis is performed using the 0.5 m diameter, standard balloon. Deorbit times are far greater than the AEOLDOS solution due to the balloon reduced drag area. For this reason this solution is discarded for the time being.

7.4 Mass Budget

Table 7.4: Deorbit Mass Budget

Strategy	**Mass [g]**
Chemical only	500
Chemical + Sail	700 *estimated*
Chemical + Balloon	620
Tether	83
Chemical + Tether	583

7.5 Conclusions

Although results are preliminary they show the capability of the system to safely deorbit and meet international regulations. Further analysis shall be performed to better asses the requirements satisfaction.

Chapter 8

Structure and Configuration

The Structural subsystem has the task to withstand all the expected load and vibration during each phase mission, ensuring a safe support to all the integrated components. At this stage of development, since we are considering a feasibility design, the structural subsystem analysis has been limited to a survey of what is available today on the COTS market. Moreover the use of a commercial-off-the-shelf structure has different advantages:

- *Standardization*: They are already designed following CubeSat standards, meeting space qualification requirements.
- *Reliability*: They are successfully used and tested technology.
- *Deployment*: They support standard deployment system, P-POD, requirements.

It follows that the use of such a proven system let the designer to save time and resources. It will be anyway fundamental in subsequent design phases to quantify loads and vibrations of the satellite final configuration, since the internal disposition of equipment differ from mission to mission.

8.1 Commercial Off The Shelf Structures

Up to now on the market two COTS structure are available from two companies: *Pumpkin* and *ISIS*.

- The *Pumpkin* structure comes with two configuration: a lighter one with a skeletonized structure and a heavier one with a solid-walls structure.

Table 8.1: Pumpkin Structure Specification

Pumpkin Structure Main Specification	
Chassis	5052-H32 Aluminium
Cover Plates	6061-T64 Aluminium
Mass (3U Skeletonized)	166g
Mass (3U Solid-Walls)	250g
Thermal Range	-40/+85°C

- The *ISIS* structure comes just as a solid-walls configuration, it is entirely made of 6061-T6 aluminium with the side surfaces black-hard anodized while the ribs and shear-panels black alodyned.

Table 8.2: ISIS Structure Specification

ISIS Structure Main Specification	
Chassis	6061-T6 Aluminium
Cover Plates	6061-T6 Aluminium
Mass (3U Solid-Walls)	270g
Thermal Range	-50/+90°C

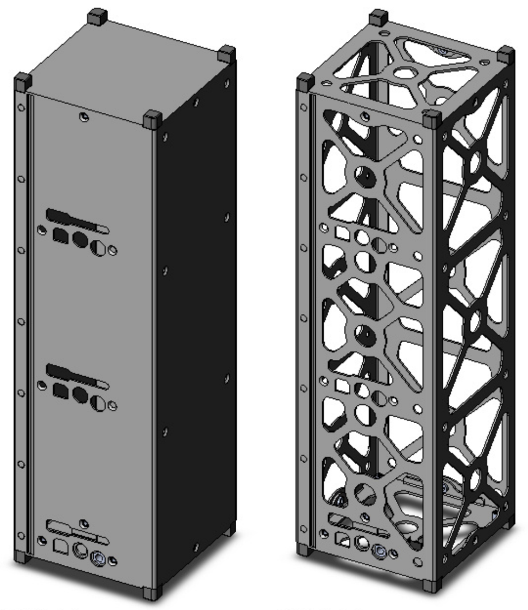


Figure 8.1: Solid-Wall and Skeletonized Structure Example [17]

8.2 Satellite Deployment System: P-POD

The *Poly Picosatellite Orbital Deployer*, P-POD), is a standard CubeSat deployment system. It is capable of carrying three standard CubeSats and serves as the interface between the CubeSats and launcher. The P-POD is a rectangular box with a door and a spring mechanism. Once the release mechanism of the P-POD is actuated by a deployment signal sent from the launcher, a set of torsion springs at the door hinge force the door open and the CubeSats are deployed by the main spring gliding on its rails and the P-PODs rails. It is made of anodized aluminium. CubeSats slide along a series of rails during ejection into orbit [3].

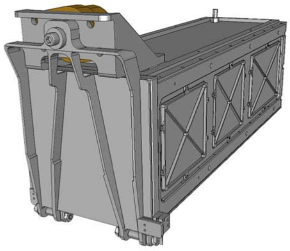



Figure 8.2: Deployment System

8.3 Solar Arrays Deployment System

Due to the small size of the satellite architecture and the power requested for the accomplishment of the mission, the system requires a large solar array area, which can be obtained with the use of a deployable system. The deployment mechanism generally consists in a spring system with travel limit and blocking, the deployment angle is typically controllable in function of power requirements, orbit orientation and payload, *Clyde Space* for example can provide: 45° or 90°.

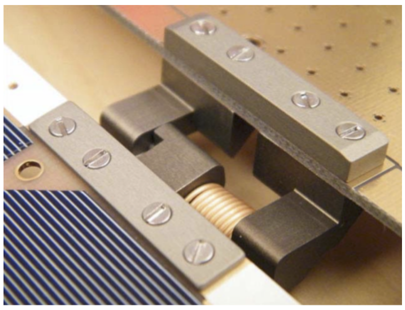


Figure 8.3: Pumpkin Solar Array Deployment System

From a requirements point of view it shall ensure a maximum envelope in order to avoid interference with the satellite deployer of about 6.5 mm from lateral surfaces [3].

8.4 Satellite Internal Configuration

CubeSat are designed aiming to a fast development and delivery, this is achievable also thanks to their slot-like internal structure, which allow an easy and flexible positioning of internal components.

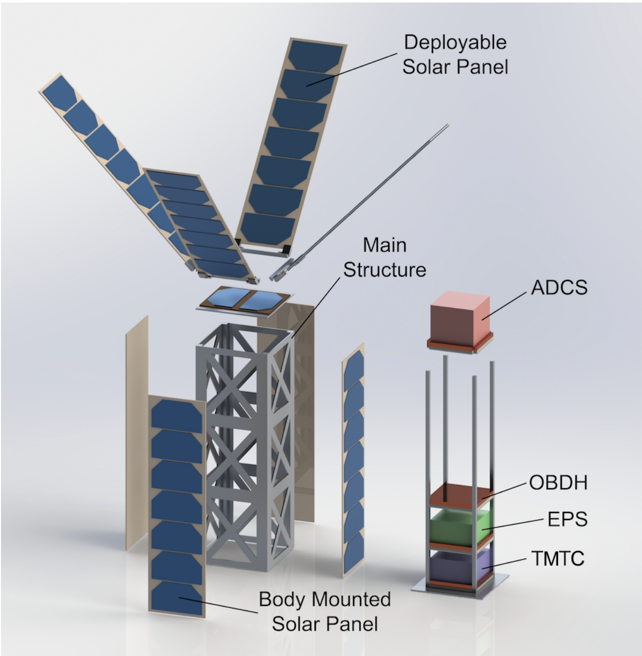


FIGURE 8.4: Stack Configuration

This internal configuration will change phase by phase during the system development, driven by mass distribution requirements, thermal and structural needs, and so on, reaching eventually its final configuration.

Chapter 9

Costs Analysis

At this point a preliminary costs analysis of the whole system is considered, from development phase to the decommissioning of the satellite.

Satellite systems costs estimation analysis are generally performed by using CER (Cost-Estimation-Relation) [9] which relationships are based on well known and already developed system. Unfortunately for nanosatellites system there is no cost estimating methodology, and the use of current small satellites costs model can lead to low fidelity results. This lack of a formal costing approach is also due to the immaturity of this kind of system, moreover cost information are usually not easy to obtain or available to the public [34]

BRIDGE design has been principally base on the employment of COTS components, for this reason cost estimation based on COTS has been considered. As reference, from [35] has been estimated that for the complete development of a 3U CubeSat, with all its subsystem, by an university would cost about: $1.5M. This estimation doesn't take into account the launch cost.

9.1 COTS: Commercial Off-The-Shelf

In the development process of a CubeSat, costs with major influence are the one from Integration and Testing, COTS and Softwares [34]. With this in mind a costs estimation can be performed with the following assumptions. Results of a first estimation are illustrated in table 9.1.

- Project management, Software engineering, Integration and Testing, are assumed to be performed and developed by students of Politecnico di Milano, therefore they aren't considered as a cost;
- flight system components and Ground facilities are bought as COTS,
- system launched as piggyback.

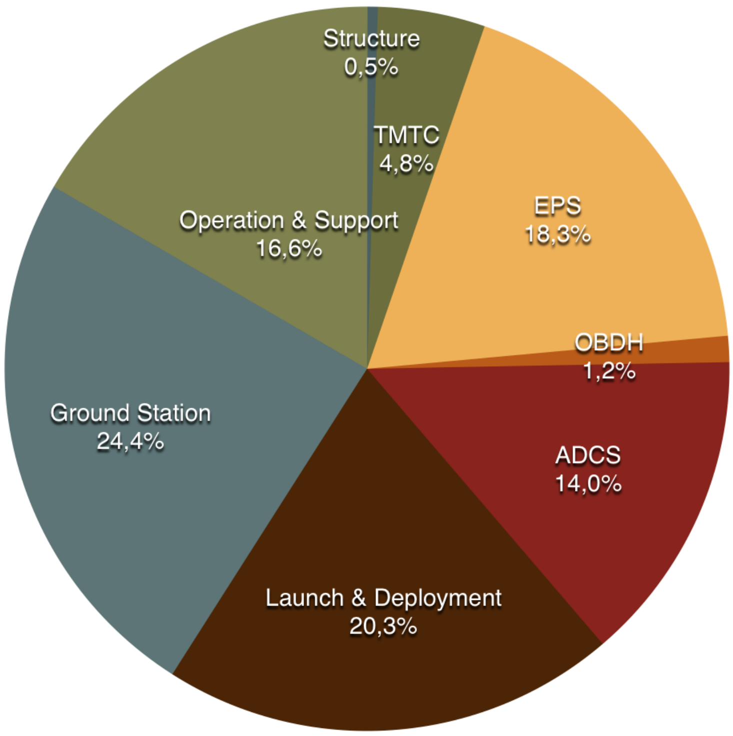


Figure 9.1: COTS Apporach Cost Distribution

9.2 Conclusions

Even if at first glance the cost seems high for a nonprofit organization as *Patologi Oltre Frontiera*, there are several non cost related returns that shall be considered:

- The development of a new satellite system will give students the possibility to gain spacecraft design experience.
- The possibility to share the service provided by BRIDGE system with other costumers. Moreover this allow the possibility of write off expenses, and achieve a return of investment over time.

Table 9.1: COTS apporach Cost Estimation

Item	**Brand**	**Details**	**Price**
Structure	Pumpkin	-	1667 €
Transmitter	ClydeSpace	S-Band TX	6590 €
Receiver	HISPICO	S-Band RX	7800 €
Antenna	ClydeSpace	S-Band Patch	3294 €
Solar Arrays	ClydeSpace	Body+Deployable	52570 €
Battery	ClydeSpace	Standalone 30Whr	2630 €
EPS	ClydeSpace	-	6627 €
Power Handling	ClydeSpace	-	5738 €
OBDH	ISIS	-	4300 €
ADCS	MAI	Full ADCS	51824 €
	1 Satellite Subtotal		**143040 €**
Ground Station	ISIS	Complete kit	45000 €
	2 Ground Station Subtotal		**90000 €**
Launch	Piggyback	-	50000 €
Deployer	ISIS	3U P-POD	25000 €
	Launch and Deployment Subtotal		**75000 €**
Operation and Support	-		61470 €
	TOTAL + 10% Margin		**406461 €**

Chapter 10

Conclusions and Future Development

The BRIDGE design study presented, allow to state that the project is feasible according to the input requirements. BRIDGE system can be a suitable solution for telemedicine purposes, giving *Patologi Oltre Frontiera* a potential alternative to existing services. Moreover it is shown that a CubeSat, thanks to state-of-the-art technology, is capable of providing a telecommunication service that can manage high data rates, while keeping a straightforward and simple design philosophy.
To enforce this position, the evaluation of the telecommunication link, power generation, thermal and attitude control, permit to confirm the robustness of the overall system.

On the other hand the analysis highlighted criticalities regarding deorbiting, even though results are preliminary and show the capability of the system to safely deorbit and meet international policies, a further analysis shall be performed to better asses the international requirements satisfaction.
The thesis development also pointed out the issue regarding the management of large images, which will need a dedicated large data processing research.

Finally a possible *Future Development and Integration Agenda* could be the following:

- Though is definitely not an easy step, and it will require university interdisciplinary collaboration, it would be nice to organize other teams at Politecnico di Milano and proceed to the next design phases. Deepening each subsystem characterization and eventually launch a demonstrator.

- Once the system capability has been proved, the development of the train concept, considering the results obtained, shall be set up aiming to enhance system performances.
- The final leap could be sharing the platform to other customers with the possibility to write off a part of the expenses, and verify that low cost satellite mission are equally performing with respect to traditional ones.

Abbreviations

COTS	**C**ommercial **O**f **T**he **S**helf
ONLUS	**O**rganizzazione **N**on **L**ucrativa (di) **U**tilità **S**ociale
WHO	**W**orld **H**ealth **O**rganization
CR	**C**ontest **R**ate
KISS	**K**eep **I**t **S**imple **S**tupid
LEO	**L**ow **E**arth **O**rbit
MEO	**M**edium **E**arth **O**rbit
GEO	**G**eostationary **E**arth **O**rbit
BOL	**B**egin **O**f **L**ife
EOL	**E**nd **O**f **L**ife
TBD	**T**o **B**e **D**efined
ACDS	**A**ttitude **C**ontrol (and) **D**etermination **S**ystem
TMTC	**T**ele**m**etry (and) **T**ele**c**ommands
TX	**T**ransmitter
RX	**R**eceiver
EPS	**E**lectric **P**ower **S**ystem
TCS	**T**hermal **C**ontrol **S**ystme
OBDH	**O**n **B**oard **D**ata **H**andling
ISL	**I**nter **S**atellite **L**ink
RAAN	**R**ight **A**scension (of the) **A**scending **N**ode
O/QPSK	**O**ffset / **Q**uadrative **Phase**-**S**hift **K**eying
RF	**R**adio **F**requency
EIRP	**E**ffective **I**sotropic **R**adiated **P**ower
BER	**B**it **E**rror **R**ate
PS	**P**ropulsion **S**ystem
DOD	**D**epth **O**f **D**ischarge
PCB	**P**rinted **C**ircuit **B**oard
P-POD	**P**oly-**P**icoSatellite **O**rbital **D**eployer
CER	**C**ost **E**stmating **R**elationship

Appendix A

Requirements

High Level Requirements		
ID	Requirement	Notes
HL-1	The system shall cover the sub-Saharan region.	-
HL-2	The main purpose of the mission shall be telemedicine.	-
HL-3	The system shall be cost-saving oriented.	-
HL-4	The system shall operate for at least 2 years.	-
HL-5	The system shall follow KISS philosophy.	-
HL-6	The system shall guarantee the minimum time access for a diagnosis.	-
HL-7	The system shall be capable to manage multiple access.	-
HL-8	The system shall be built with off-the-shelf equipment, even not space qualified.	-
HL-9	The system shall provide the quality needed to perform a diagnosis.	-
HL-10	The system shall be capable to stream contents from the servers to the end-users.	-
HL-11	The system shall operate without orbit correction.	-
HL-12	The system shall provide the correct antenna pointing.	-
HL-13	The system shall be operative right after launch.	-

HL-14	The system shall guarantee the connection during the whole mission life.	-
HL-15	The system shall foresee de-orbiting.	-
HL-16	The system shall not use a dedicated launch system.	Piggyback Launch
HL-17	The system shall manage contents coming from ground equipment	APERIO scanner.
HL-18	The system shall survive launch, eclipse and radiation environment.	-
HL-19	The connection shall be available at least during working hours.	Daylight
HL-20	The system shall be capable to manage multiple access.	-
HL-21	The system shall be able to reschedule upon ground commands.	-
HL-22	The system communication link shall be in radio frequency.	-
	Optional	
HL-23	It would be nice to have an european access.	-
HL-24	It would be nice to have the possibility to share the service with different users.	-
HL-25	It would be nice to add different purpose payloads.	-
HL-26	It would be nice to use a CubeSat configuration.	-
HL-27	It would be nice to manage contemporary multiple access.	-
HL-28	It would be nice to have a full-autonomous system.	-
HL-29	It would be nice to have an altitude higher than 700 Km to avoid drag perturbation.	-

Mission Analysis Requirements		
ID	Requirement	Notes
MA-1	The system shall operate for at least 2 years	-
MA-2	The system shall cover the sub-Saharan region	Based on customer needs
MA-3	The system shall operate without orbit correction	Retain KISS philosophy
MA-4	The system shall not use a dedicated launch system.	Cost saving
MA-5	The system shall foresee de-orbiting	Fulfil International regulations
MA-6	The system shall limit orbital lifetime in LEO after mission completion or maneuvering to a disposal orbit	Fulfil International regulations
MA-7	The system shall limit the human casualty risk from space system components surviving reentry as a result of postmission disposal	Fulfil International regulations
MA-8	The system shall limit the debris hazard posed by tether systems	Fulfil International regulations
MA-9	The system shall limit the probability of impact with other objects in orbit	Fulfil International regulations
Optional		
MA-10	It would be nice to have an european access	Quality control on diagnosis
MA-11	It would be nice to launch the system before the WHO-project starts.	-
MA-11	It would be nice to have an available window long enough to perform a diagnosis	Based on customer needs

Telecommunication Requirements		
ID	Requirement	Notes
TMTC-1	The system shall provide the quality needed to perform a diagnosis	-
TMTC-2	The system shall be capable to transfer contents	Based on customer needs
TMTC-3	The system shall be built with off-the-shelf equipment, even if not space qualified	Cost saving, retain KISS philosophy
TMTC-4	The system shall manage contents coming from ground equipment	APERIO scanner
TMTC-5	The connection shall be available at least during working hours (daylight)	Workability
TMTC-6	The system shall sustain a store-and-forward architecture	Ref. TMTC-2
Optional		
TMTC-7	It would be nice to have the possibility to share the service with different users	Cost sharing
TMTC-8	it will be nice to stream contents from the servers to the end-users	Customer primary need
TMTC-9	It will be nice to use all available time window	-

Electric Power System Requirements		
ID	Requirement	Notes
EPS-1	The system shall provide the power needed by the satellite in all its phases.	-
EPS-2	The system shall control and distribute electrical power to the spacecraft.	-
EPS-3	The system shall support power requirements for average and peak power loads.	-
EPS-4	The system shall convert for AC and regulated DC power buses.	-
EPS-5	The system shall provide power at spacecraft EOL.	-
EPS-6	The system shall provide telemetry and health statuses.	-
EPS-7	The system shall manage the battery charging.	-
EPS-8	The system shall provide power under eclipse phase.	-
Optional		
EPS-9	It will be nice to provide enough power to work under light, without using batteries.	-
EPS-10	It will be nice to have hinged solar panels.	-
EPS-11	It will be nice to have controllability of the orientation of the solar panels.	-
EPS-12	It will be nice to have an autonomous re-orientation of the solar panels, based on the Sun position.	-
EPS-13	It will be nice to respect CubeSat requirements	-

Thermal Control Requirements		
ID	Requirement	Notes
TC-1	The system shall be maintained inside equipment survival temperature ranges	During all life phases
TC-2	The system design shall prefer passive over active means	Ref. HL-5

Atittude Control and Determination Requirements		
ID	Requirement	Notes
ADC-1	The system shall be capable to detumble after launcher release	-
ADC-2	The system shall provide the correct pointing accuracy	Telecommunication driven
ADC-3	The system shall monitor its attitude	-

Deorbit Requirements		
ID	Requirement	Notes
DO-1	The system shall limit orbital lifetime in LEO after mission completion	-
DO-2	The system shall limit the human casualty risk from space system components surviving reentry as a result of post-mission disposal	- -
DO-3	The system shall limit the debris hazard posed by tether systems	-
DO-4	The system shall limit the probability of impact with other objects in orbit	-

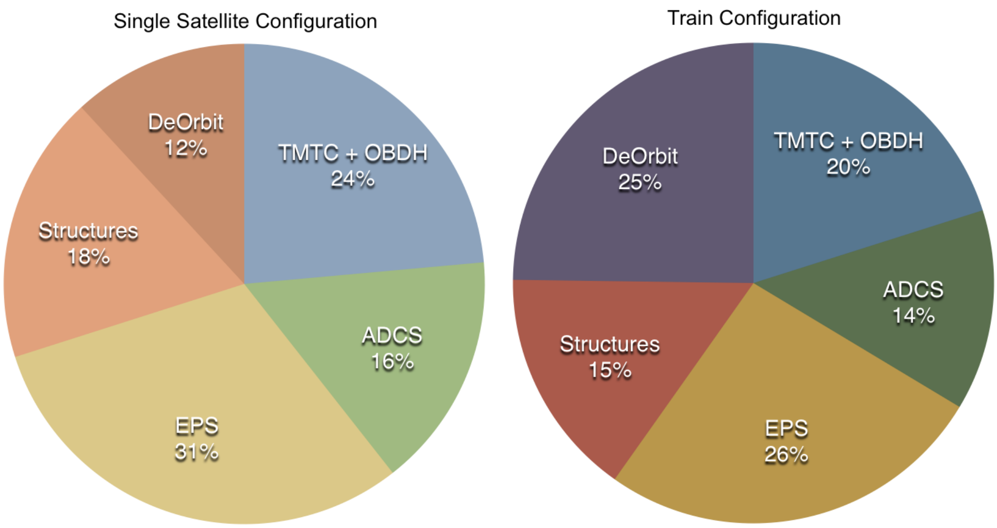
Structure Requirements		
ID	Requirement	Notes
7cm ST-1	The system shall respect the geometrical constraints given by the launcher	-
ST-2	The system shall be capable to withstand loads given by the launcher	-
ST-3	The system shall sustain the accelerations given by disposal maneuver	-
ST-4	The system shall withstand the thermal loads during launching	-
ST-5	The system shall be designed to accommodate the antenna pointing toward Nadir	-
ST-6	The system shall be capable to deploy the solar arrays	-
ST-7	The system shall be compatible with the P-POD deployment system	-
ST-8	The system shall be able to communicate deployment success or failure	-
ST-9	The System shall be capable to ensure a smooth deployment	ADCS capability
Optional		
ST-7	It would be nice have processor, data bus and other control component close together	Minimize harnesses
ST-8	It would be nice to accommodate the reaction wheels with axis aligned to the principal inertia axis of the satellite	Ref. ADC-3

Appendix B

Mass Budget

A mass budget for the system can been performed combining data coming from datasheets [8], derived from specific (with respect to mass) figures of merit, and from literature [9, 19] for what concern harnesses. A summary table is shown hereafter. Margins applied are taken from literature [11], namely 4%.

Subsystem	**Single Mass [kg]**	**Train Mass [kg]**
TMTC + OBDH	1.16	1.16
EPS	1.51	1.51
ADCS	0.78	0.78
Structures	0.89	0.89
DeOrbit	0.583	1.43
Thermal	0	0
Total +4%	4.51	5.99



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