Université de Liège

OUFTI-1 Phase A: Mission definition Space and ground systems description

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Abbreviated Terms

ADC	Analog to Digital Converter
AOS	Acquisition of Signal
ASI	Agenzia Spaziale Italiana (Italian Space Agency)
AVUM	Attitude Vernier Upper Module
BER	Bit Error Ratio
BOL	Beginning of Life
CCSDS	Consultative Committee on Space Data System
COG	Center of Gravity
COM	Telecommunication subsystem
COTS	Component Off-The-Shelf
CSG	Centre Spatial Guyanais (Guiana Space Center)
CSK	CubeSat Kit
EOL	End of Life
EPS	Electrical Power Supply
ESA	European Space Agency
FEA	Finite Element Analysis
FOS	Factor of Safety
GS	Ground Station
I/O	Input/Output
IARU	International Amateur Radio Union
IR	Infrared
ISIL	Institut Supérieur Industriel Liégeois
ITU	International Telecommunication Union
LARES	Laser Relativity Satellite
LEO	Low Earth Orbit
LEOP	Launch and Early Operations Phase
LOS	Loss of Signal
LV	Launch Vehicle
MCC	Mission Control Center
MECH	Mechanism Subsystem
MOS	Margin of Safety
MPPT	Maximum Power Point Tracker
OBC	On-Board Computer
OS	Operating System
OUFTI	Orbital Utility For Telecommunication Innovation

PCB	Printed Circuit Board
PCC	Poste Central de Commande
P-POD	Poly-PicoSatellite Orbital Deployer
PUS	Packet Utilization Standard
RAAN	Right Ascension of the Ascending Node
RBF	Remove Before Flight
RF	Radio-frequency
SEB	Single Event Burnout
SEL	Single Event Latch-up
SEP	Single Event Phenomena
SEU	Single Event Upset
STK	Satellite Tool Kit
TBC	To Be Confirmed
TBD	To Be Defined
TC	Telecommand
TM	Telemetry
TNC	Terminal Node Controller
UHF	Ultra High Frequency
VEGA	Vettore Europeo di Generazione Avanzata (European Advanced Generation Carrier)
VHF	Very High Frequency
xEPS	Experimental Electrical Power Supply

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Chapter 1

Introduction

The OUFTI-1 project started in November 2007. The main objective of the nanosatellite project is to provide hands-on experience to students in the design, construction, and control of complete satellite systems that will ultimately serve as the basis for a variety of space experiments. Increasing interest both in space studies and in engineering studies is also part of the initial motivations.

OUFTI-1 will be the first satellite of the University of Liege. The project began with a huge enthusiasm, but without real experience in project management. A substantial amount of work was performed and after some time, the crucial need for a complete document summarizing the phase A appeared. This document summarizes the work performed and the results achieved during the OUFTI-1 phase A study. The choice was made of reflecting the historical background of the project trough this document.

This document includes the status, the main results of analysis, and the conceptual baseline design for:

- Objectives and requirements analysis,
- Mission analysis and operations,
- Space system configuration and trade-offs,
- Ground system design.

About 15 students from the Aerospace and Mechanical Engineering and from the Electrical Engineering and Computer Science Departments of the University of Liege, from the ISIL engineering school, and from the Gramme engineering school participated in work. The project's organization is described in the "OUFTI-1 Project: Hypothèses de management" document. The students framework was Master Theses, corresponding to the work of a student for half of its academic time (September to June). It is of primary importance to note that students had no prior experience in space projects before being involved in OUFTI-1.

The project engineering team would like to thank the students who participated, the laboratories involved, the project's partners, and the ham-radio community, who are, all together, making this project possible.

The OUFTI-1 project also wishes to thank the SwissCube team for their support, and for being an example in this challenging task. The two visits the team made to Lausanne were really interesting and enjoyed by the participants [12, 24].

Chapter 2

Analysis of mission objectives and requirements

This chapter presents the mission and payload objectives. Top-level requirements are then derived from those objectives.

2.1 Mission objectives of OUFTI-1

Below are listed the different objectives of the OUFTI-1 mission by order of importance. For each objective, there is one success criterion that will allow the team to measure whether an objective has been reached or not. It is obvious that the order in which the objectives are listed is significant, in the sense that an objective will not be considered if the objectives listed above are not encountered. The list is voluntarily detailed, so that all objectives are clear in the mind of every member of the team.

1. **Fun and education.** It results from contacts with various teams that having fun while working on a project is the basis of success. However, the OUFTI-1 project aims at providing hands-on experience, mostly to students. Therefore, both fun and education will be considered as the first objective. There are three criteria to evaluate whether this objective has been achieved. First, we should have a number of students getting good grades for their work on the project. Second, there should be a significant amount of students willing to work on the project each year. Third, the success of a year's work can be measured by the amount of graduate students who, after their graduation, continue to participate, actively or not, in the project.

2. **Design of the OUFTI-1 system.** Besides the fun and educational objective, the first goal of OUFTI-1 project is to design and build a satellite, which implies designing and building both a space system and a ground system. The success criterion is to have a functional satellite and its corresponding ground system working properly.

3. **Launch of OUFTI-1.** Once the satellite is built, the next step is to launch it. Having a proper launch is the success criterion.

4. **Having the satellite alive in space.** After launch, the satellite's life starts. The success criteria is to receive a signal from the satellite, indicating that it is alive.

5. **Operating the satellite.** The success criterion is to be able to upload telecommand and to download telemetry.

6. **Having the D-STAR system functional.** The D-STAR payload will be the first to be operated, as it is the main payload. The goal is to demonstrate the use of the D-STAR protocol in space. A single D-STAR contact via the satellite is enough to consider the D-STAR payload as a success.

7. **Operating the secondary payloads.** The two secondary payloads, which are high-efficiency solar cells and a digitally-controlled power system, are then considered. The success criterion is to receive telemetry from those two payloads.

2.2 Payloads

There will be three different payloads aboard OUFTI-1, the D-STAR protocol, a digital power supply, and new solar cells. They are briefly described in this section.

2.2.1 D-STAR in space

The first payload is D-STAR, which is a recent amateur-radio digital communication protocol that allows the simultaneous transmission of voice and data. More details can be found in [10, 15, 21]. The objective of the main payload of OUFTI-1 is to prove the suitability of the emerging D-STAR technology for space use. While this telecommunication system already proved itself to be highly beneficial for routine ham-radio operations, it also demonstrated many efficient dedicated crisis management supports to various safety organizations worldwide.

The D-STAR network on Earth is currently developing, mainly in the USA and in Europe (Fig. 2.1). Users contact other ham-radio operators thanks to their off-the-shelf transceivers and through repeaters that are connected to each other via the Internet. These ground-to-ground communications are described in Section 5.4.

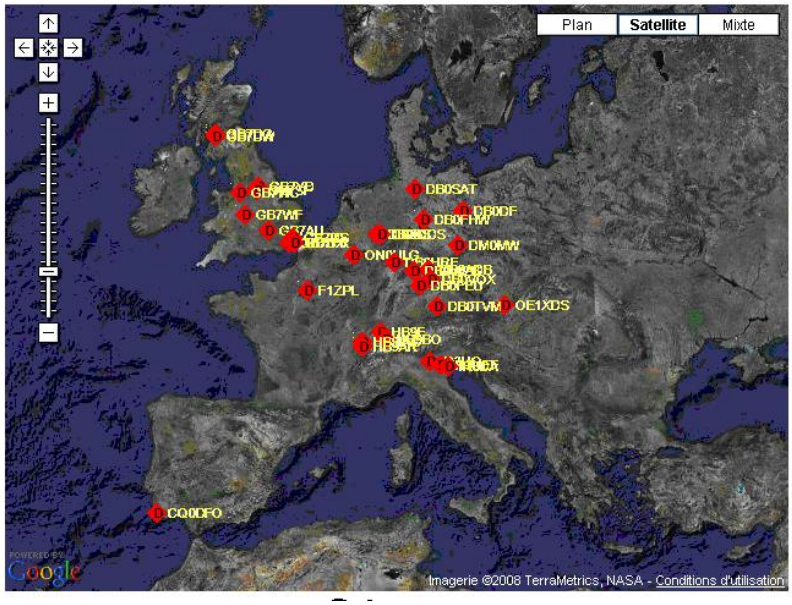


Figure 2.1: D-STAR network in Europe (source: www.jfindu.net) .

To prove the viability of the D-STAR technology in space, the satellite has to be compatible with the existing ground D-STAR network, so that users can use the satellite as a repeater in space. The two different modes of communication are presented here, at a high level. Users will be able to communicate directly via the satellite, as described in Fig. 2.2. They will also be able to enter the D-STAR network, as described in Fig. 2.3. Details about these communication modes and the configuration that will be used by the ham-radio operators can be found in [15, 21].

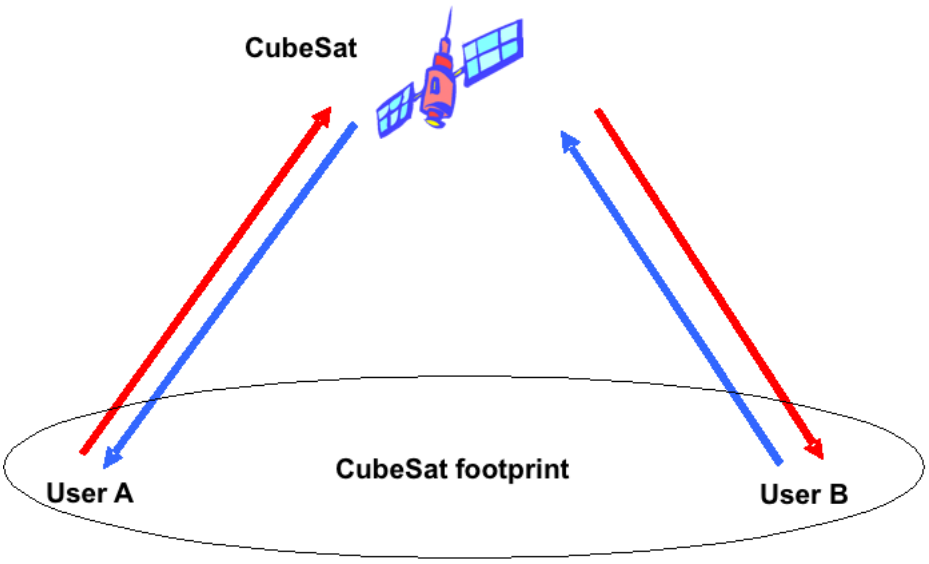


Figure 2.2: Direct D-STAR communication between two users via the satellite.

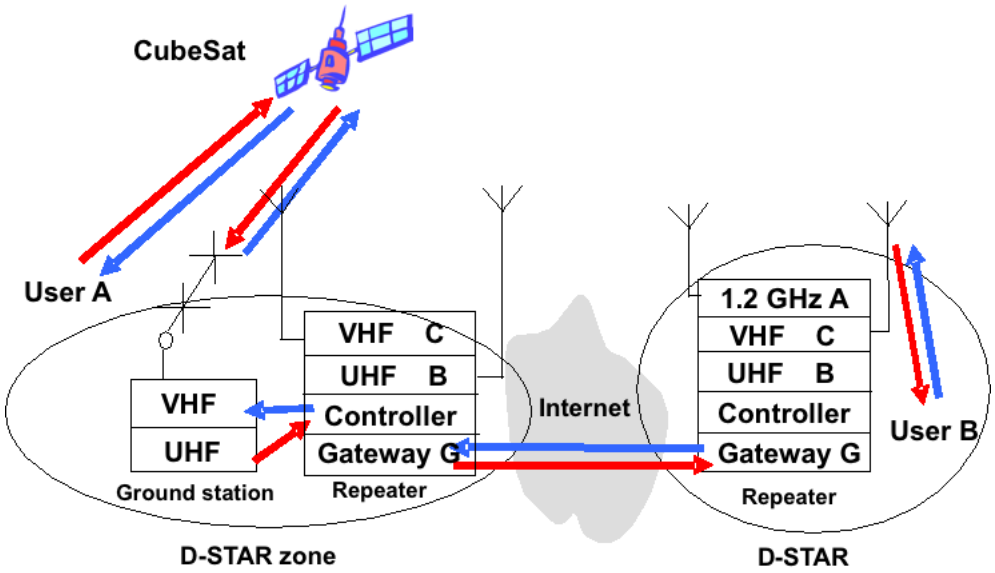


Figure 2.3: D-STAR communication between two users via the satellite and repeaters.

The use of D-STAR in space implies specific solutions to be found to take care of known additional problems typically encountered with space missions, e.g. instantaneous Doppler shift compensation, intelligent use of the limited available visibility period for LEO orbits, limited power on-board the satellite, etc.

The D-STAR payload flying on OUFTI-1 provides a kind return to the ham-radio community for the friendly use of the available ham-radio spectrum.

2.2.2 Digitally-controlled power supply

The second payload is a digitally-controlled electrical power supply developed in collaboration with Thales Alenia Space ETCA, Charleroi, Belgium. The objective is to prove that such a power supply can reliably operate in space, based on the telemetry downloaded (Section 4.6.6).

2.2.3 New solar cells

The third payload of OUFTI-1 is meant to test in space high-efficiency (30%) solar cells furnished by AzurSpace. These solar cells will be the only ones aboard OUFTI-1. Based on the telemetry received, the solar cells will be characterized in the real space environment.

2.3 Mission requirements

Below are listed the different requirements of the project. Even though this list could appear as non-optimized, it takes the history of the project into account, including the choices of the VEGA launcher and of its orbit. There are three types of requirements: process requirements, indicating how the work should be done; non-functional requirements which state some choices made by the project, and functional requirements which take the functionalities of the mission into account.

2.3.1 Process requirements.

- PR1: All work directly valuable to a student must be done by a student.
- PR2: Due to the education objective, commercial off-the-shelf (COTS) components or custom solutions will be preferred to space-qualified materials.
- PR3: When compliant with the educational objective, flight heritage will be considered.

2.3.2 Non-functional requirements.

- MR1: The project shall use the organizational frame and requirements of the CubeSat Standard developed by the California Polytechnic State University. This implies further requirements detailed in [23].
- MR2: The spacecraft will be designed to comply with the VEGA Maiden Flight specifications [2].
- MR3: The system shall be designed for a 9-month mission in space.
- MR4: The ground system shall be designed to last 48 months.
- MR5: The PC/104-Plus standard shall be respected aboard the spacecraft.
- MR6: Once proved feasible, the ADF7021 shall be used aboard the spacecraft.

2.3.3 Functional requirements.

- MR7: All links must be located within the agreed ham band specific space allocations (see the latest IARU band planning for reference).
- MR8: The project shall provide continuous communication capability between the space segment and the ground segment after antennas deployment.
- MR9: TM format will be known to all users while TC format will not be disclosed to the public for safety reasons as agreed by IARU.

- MR10: The project shall establish a way to recover the TM downloaded by the ham-radio community, anywhere in the world.
- MR11: The spacecraft will, independently of the mode it has entered, permanently provide a way of performing an immediate shutdown or an immediate transition to safe mode, at any time, by any authorized ground control station, possibly following an immediate request by the legal authorities.
- MR12: The OUFTI-1 system shall be entirely compliant with the D-STAR network.
- MR13: The ground system shall be capable of operating with other external amateur radio satellites than the project's and possible networks of amateur radio ground stations.
- MR14: The spacecraft shall always be in any case, in one of the following modes:
  - Initial mode
  - Safe mode
  - Default mode
  - Payload mode.

These modes are further investigated in Section 4.5.

2.4 Payloads requirements

2.4.1 D-STAR payload

- PL1_1: The whole system must be able to provide open cross-band D-STAR communications to any duly authorized ham radio user.
- PL1_2: Uplink to the satellite will be performed in the 70-cm band while downlink will be done in the 2-m band. This will certainly increase the potential number of ham stations able to receive the D-STAR signal, even probably into a mobile station.
- PL1_3: The use of the D-STAR system must be managed by a dynamic allocation scheme.
- PL1_4: D-STAR communications must only be provided, one user transmitting at a time, between stations located within the footprint of the satellite, following any user's case enumerated below :

  - between two defined geographical areas,

  - between one geographical area and a suitably equipped D-STAR terrestrial repeater, potentially providing worldwide roaming,

- between a specific area designated as the crisis area and a crisis management center located somewhere within the footprint to provide remote crisis management. This specific case is foreseen to efficiently demonstrate the use of the D-STAR system applied to crisis situations,

- any other station elsewhere but having to build its entire system to cope with the necessary additional steps to be successful with the system.

- PL1_5: The instantaneous Doppler shift must remain unnoticed to the various users (but the later case) thanks to a by-region automatic Doppler compensation applied on- board the spacecraft on both the up and down link frequencies.

- PL1_6: D-STAR operations will only take place after meaningful checks have been successfully completed by a control station in order to ensure satisfactory operations. The use of the D-STAR mode will be started by specifying an activation period that will cover, at a maximum, a whole orbit. For efficiency and legal aspects, this activation time could possibly be restricted to an entire pass always compatible with a control station that will be able to in spacecraft during the complete pass.

- PL1_7: D-STAR operational status has to be reported to the interested users in real-time when applicable.

- PL1_8: All Wednesdays will be reserved for scientific use of the satellite and therefore no D-STAR operation but for testing purpose will be provided on these specific days.

- PL1_9: The orientation of the satellite must be designed in order to perform correct communications.

2.4.2 Experimental digitally-controlled electrical power supply (xEPS)

- PL2_1: The xEPS shall be activated upon request only.

- PL2_2: When activated, the xEPS shall supply either a resistive load, or a non-critical bus.

- PL2_3: Measurements will be performed to evaluate the performance of the xEPS.

2.4.3 Solar cells

- PL3_1: The high-efficiency solar cells from Azur Space will be used to supply the spacecraft with power.

- PL3_2: Measurements will be defined in agreement with Azur Space and performed to evaluate the performance of the solar cells.

Chapter 3

Mission analysis

The choice of the orbit is an important step in every space mission as it strongly influences the final performance. The first task of the satellite mission analysis is to select the optimum orbit that best enables the satellite to perform its mission. It is usually driven by the mission and satellite design requirements.

However, in the case of OUFTI-1, the CubeSats are secondary payloads on the Vega maiden flight. The orbit is thus imposed by the main payload, the LARES satellite. We cannot therefore choose the orbit that would be convenient for our mission. The first task of the mission analysis is then not to choose the orbit, but to analyze the influence of the imposed orbit on the mission of the satellite.

This chapter first presents the influence on OUFTI-1 mission of the launch vehicle, the orbit, and the space environment. A mission timeline can then be deduced. The last step is to define mission operations and characterize data flows.

Details about mission analysis can be found in [3].

3.1 Launch vehicle

3.1.1 The Vega launcher

Vega, which stands for Vettore Europeo di Generazione Avanzata, is the new European small launcher designed to put into orbit satellites between 300 and 2000 kg. This project is funded by Belgium, France, Italy, Spain, Sweden, Switzerland, and The Netherlands. As shown in Fig. 3.1, the Vega launcher is a single-body vehicle composed of three solid-propulsion stages (P80, Zefiro-23 and Zefiro-9), and an additional liquid propulsion restartable upper module known as AVUM, which stands for Attitude Vernier Upper Module. The AVUM is used for attitude control, orbit control, and satellite release. At lift-off, the launcher is 30.2 m high, has a

maximum diameter of 3 m and weighs 139 tonnes. It will be launched from the Guyana Space Center (known as CSG, which stands for Centre Spatial Guyanais, in French).

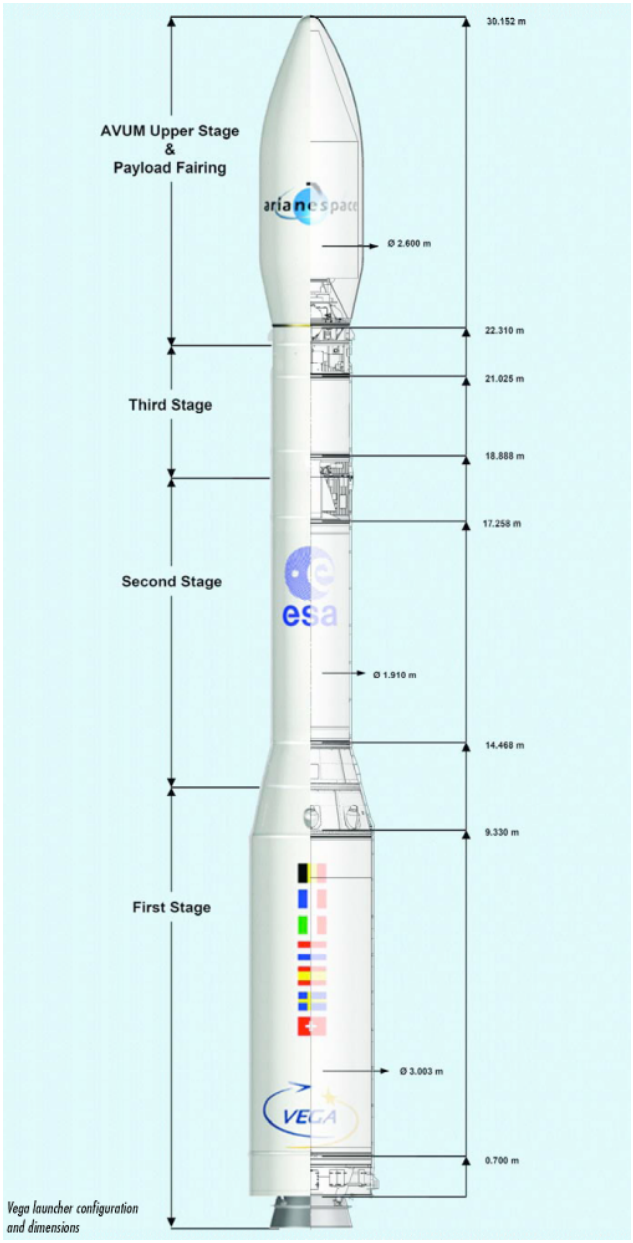


Figure 3.1: Configuration of Vega launcher.

3.1.2 Typical mission profile

The typical mission profile of VEGA consists of three phases (Fig. 3.2) :

1. Phase I: Ascent of the first three stages of the launch vehicle into a low elliptic trajectory (sub-orbital profile),

2. Phase II: Transfer of upper stage (including payload) to the initial parking orbit by first AVUM burn, orbital passive flight and orbital maneuvers of the AVUM for payload delivery to final orbit,

3. Phase III: Deorbitation of AVUM or orbit disposal maneuvers.

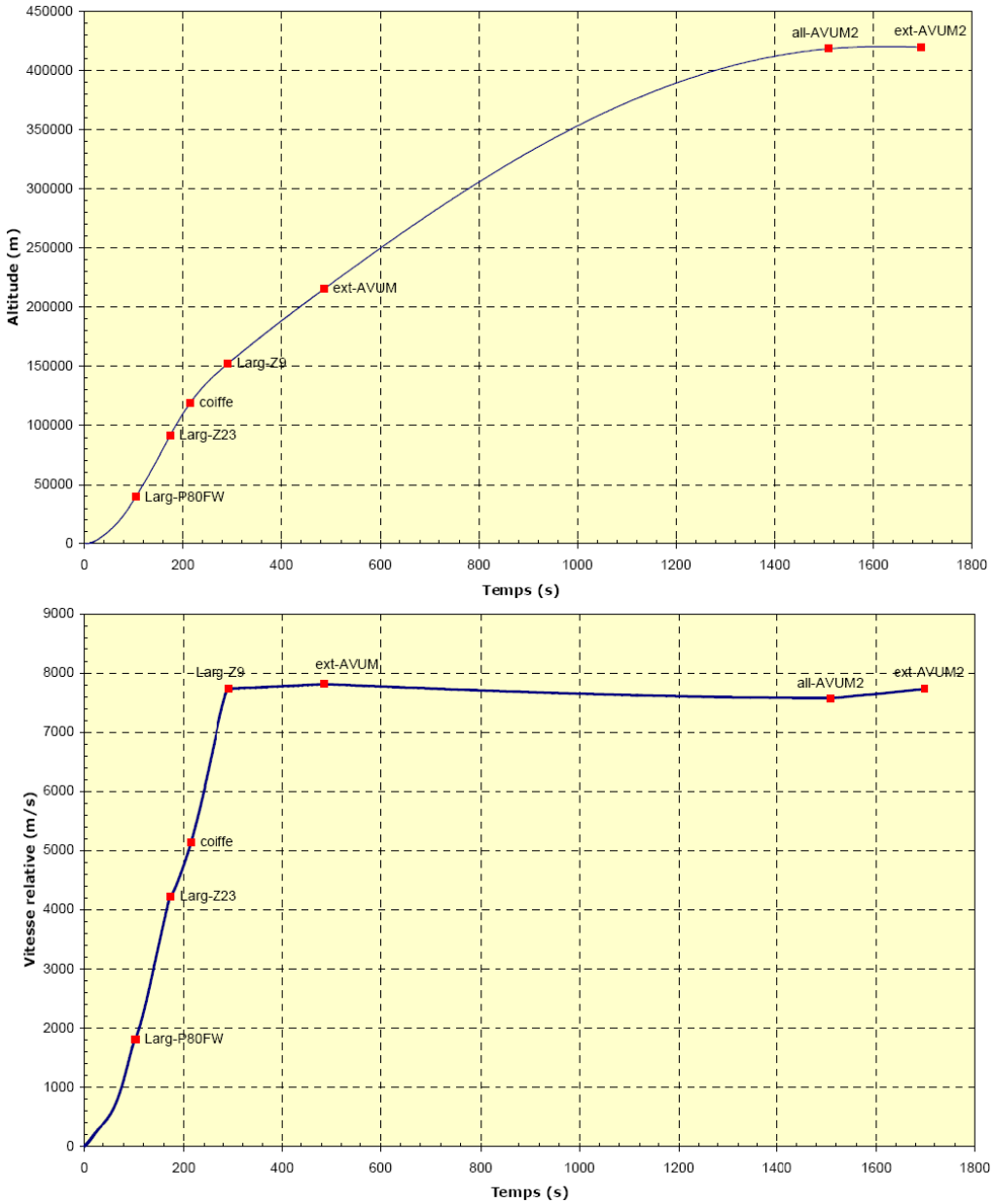


Figure 3.2: Typical mission profile of Vega: altitude and relative speed as a function of time after lift-off.

3.1.3 Performances

As shown in Fig. 3.3, Vega is designed to launch a wide range of payload configurations and to place them on various orbits, from equatorial to sun-synchronous and

polar. In particular, its design points to place into orbit a 1500-kg satellite into a 700-km polar orbit.

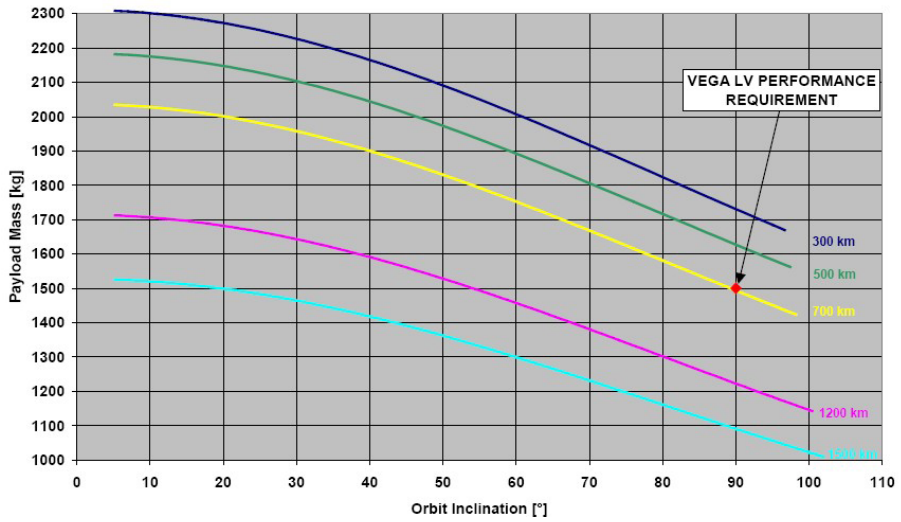


Figure 3.3: Vega performance: mass of payload as a function of orbit inclination for different desired altitudes.

3.1.4 The Vega maiden flight

The main feature of Vega is that, unlike most small launchers, it is able to place multiple payloads in orbit. Indeed, the AVUM is restartable up to five times, allowing the release of satellites in different orbits.

In the case of the Vega maiden flight, the AVUM will maneuver to an orbit of 1450 X 1450 km altitude with an inclination of 71° and deploy the LARES payload. LARES stands for LAser RElativity Satellite and consists in a 400-kg tungsten sphere. It is an Italian satellite, designed by the Italian Space Agency (ASI) in cooperation with the University of Rome. LARES will test a prediction following from Einstein's theory of General Relativity, the so-called "frame-dragging" or "Lense-Thirring" effect. Furthermore, an adaptation of the Upper Composite test dummy used during the campaign of mechanical tests will measure the actual launch loads experienced by a typical payload.

The AVUM will then perform a final maneuver in order to lower the perigee altitude to 354 km and deploy the nine CubeSats, placed in three Picosatellite Orbital Deployers (P-POD procured from CalPoly), into an orbit of 354 x 1447 km altitude with 71° inclination. The time delay between lift-off and CubeSats release shall be up to 5000 sec.

3.2 Orbit

3.2.1 Orbital parameters

As mentioned above, the foreseen orbit is elliptical with a perigee at 354 km altitude, an apogee at 1447 km altitude, and an inclination of 71°. The other parameters are still to be set by ESA, and are therefore considered as free parameters.

Parameter	Value
Type	LEO
Perigee altitude	354 km
Apogee altitude	1447 km
Inclination	71°
Argument of perigee	TBD
RAAN	TBD
True anomaly	TBD

Table 3.1: OUFTI-1's orbital parameters.

For the simulations presented later, the free parameters have been arbitrary set to values so that the apogee is over the GS in Liège at the beginning of the mission. This guarantees a passing time in view of the GS as long as possible, hence the longest time to communicate with the satellite. It also favors the Belgian amateur radio operators, which seems logical as the CubeSat is Belgian.

Argument of perigee	230°
RAAN	10°
True anomaly	90°

Table 3.2: OUFTI-1's orbital parameters for simulations.

Figures 3.4 and 3.5 respectively show a typical ground track and a 3D view of the orbit of OUFTI-1.

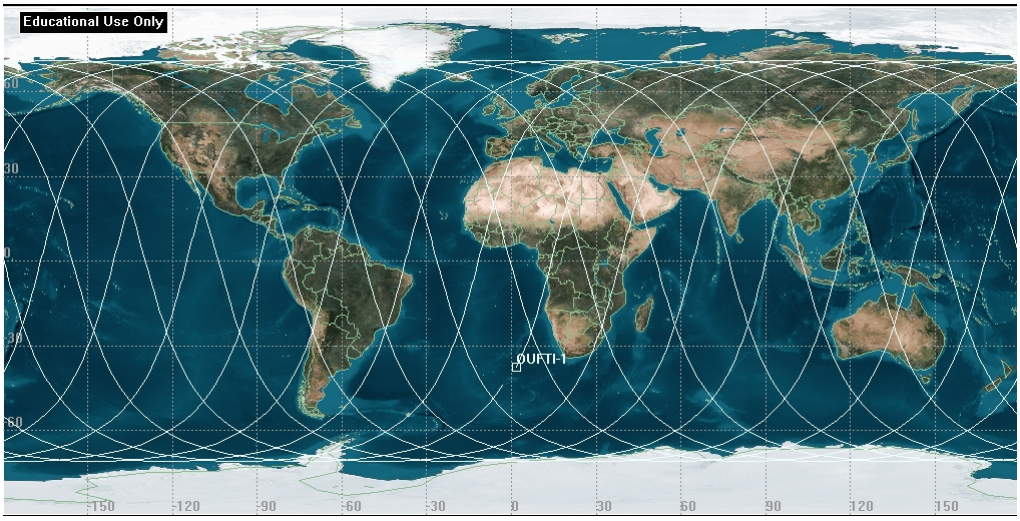


Figure 3.4: Ground track of OUFTI-1.

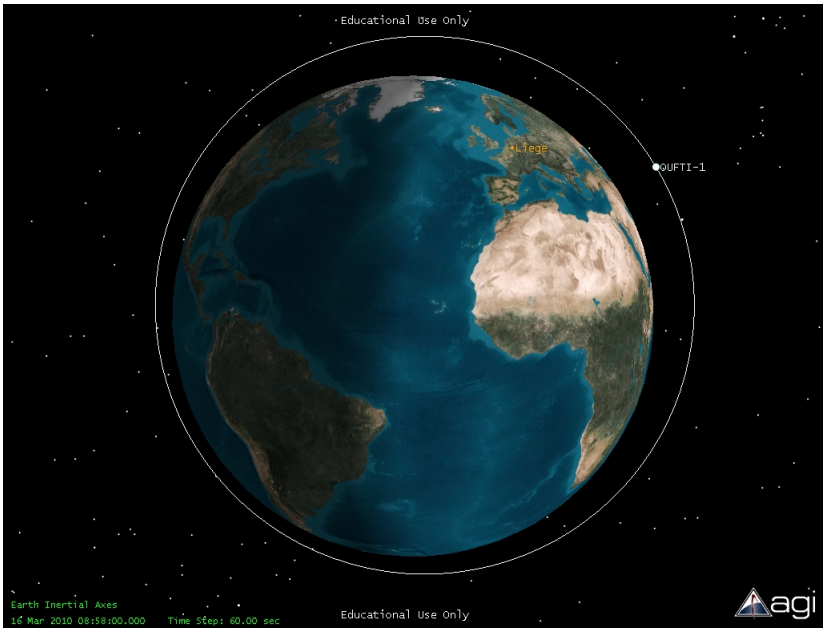


Figure 3.5: Orbit of OUFTI-1.

3.2.2 Lifetime of the orbit

The main cause of the orbital decay is the drag caused by the residual atmosphere at low altitudes. The perigee of the orbit of OUFTI-1 is at 354 km, and, at this altitude, the atmosphere exerts a significant effect. The result of this friction is

a decrease of the apogee altitude. To keep a constant apogee altitude, a regular altitude boost is needed to counteract orbital decay. Since OUFTI-1 is completely passive, there is no way to control the orbit. In order to estimate the lifetime, the STK software was used, which calculates when the satellite enters the atmosphere, thereby ending its life. STK takes into account the drag coefficient and the drag area used in drag calculations, as well as the reflection coefficient and the area exposed to Sun used in solar radiation pressure calculations. It also uses predicted values of the monthly mean 10.7 cm solar radiation flux influencing the atmospheric density.

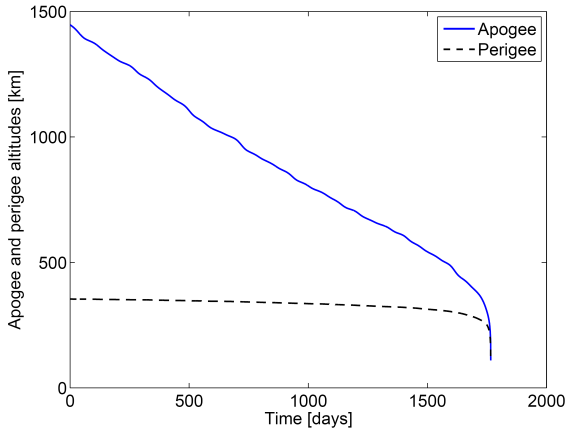


Figure 3.6: OUFTI-1 lifetime.

STK estimates a lifetime of 4.8 years. This orbit guarantees a lifetime of much less than 25 years, compliant with the international requirement related to space debris. The evolutions of the perigee and apogee altitudes as a function of time are shown in Fig. 3.6.

It should be emphasized that, although this lifetime computation is based upon sophisticated orbital theory and accurate environment models, the result is still only an estimate. Due to the difficulty in accurately predicting variations of solar and geomagnetic activities, errors inherent in models of atmospheric density, and uncertainties in cross-section areas and drag coefficients, satellite lifetimes cannot be determined with accuracy better than 10% of the actual lifetime. Furthermore, this lifetime estimate is to be carefully considered because the radiation environment will considerably reduce the lifetime of the electronic components, as COTS will be used.

3.2.3 Durations of eclipses

The eclipse duration is an important parameter for the design of a space system, especially for the power subsystem. For a satellite, the time spent in darkness, i.e. when Earth is blocking the sunlight, appears much more frequently than on Earth because of the high frequency at which the satellite rotates around the Earth.

Indeed, in the case of OUFTI-1, the satellite will perform 14 revolutions per day, which means that there will be a fast alternance between periods of sunlight and periods of darkness. Knowing the orbit of OUFTI-1 , we can calculate, using STK, the duration the satellite will be in sunlight and the duration it will be in darkness.

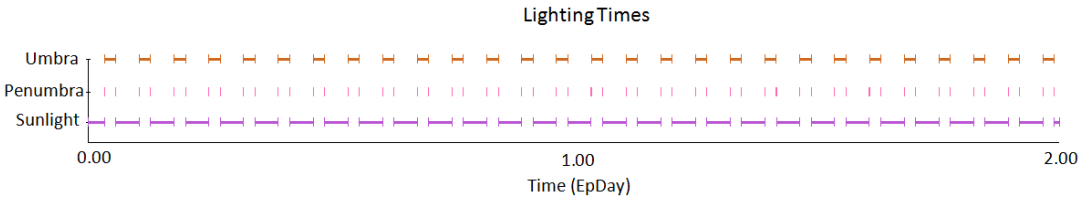


Figure 3.7: Periods of sunlight, umbra, and penumbra.

Concerning sunlight periods, during one year, OUFTI-1 will be 75.8% of its time in sunlight, 23.8% in umbra, and 0.4% in penumbra. As expected, the penumbra periods are very short due to the low height of the orbit. During one orbit, the maximum eclipse duration is 37.03 min, which represents 35.9% of the orbit. This means that during this rather long fraction of the orbit in the Earth's shadow, the satellite must operate with batteries because solar panels receive no light.

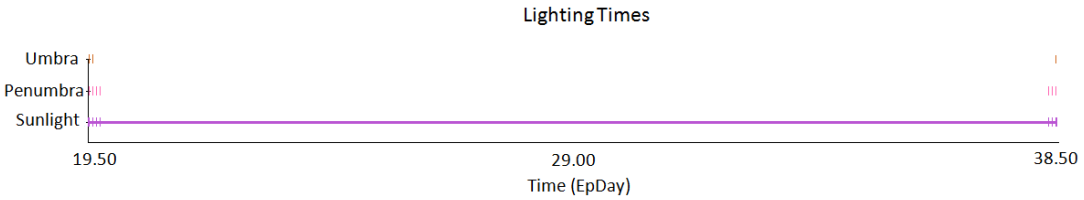


Figure 3.8: Constant light.

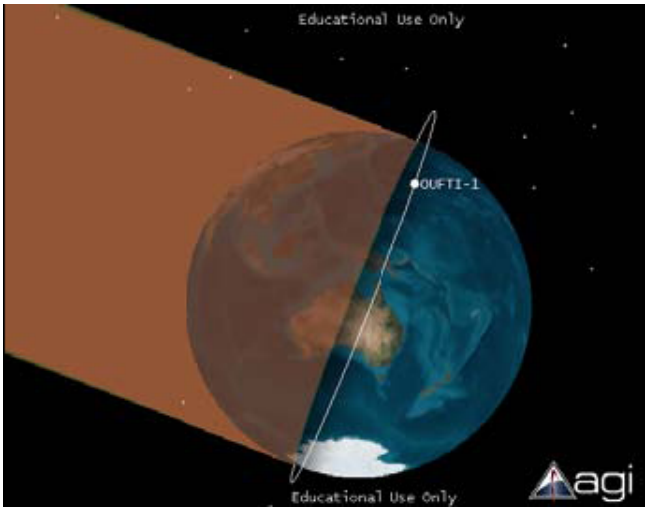


Figure 3.9: Configuration of the orbit during long constant light times.

Note that during the whole year, there will be a non-negligible duration of time for which the satellite will be in constant sunlight, and there will be no eclipse (Figs. 3.8 and 3.9). This time, which will last approximately 18 days, corresponds to the moment of the year when the orbit plane is perpendicular to the Sun direction. It has greatly influenced the thermal design of OUFTI-1, as it is considered to be the hot case for thermal calculations.

3.2.4 Access

Determination of the footprint

For an omnidirectional antenna, one can assume that the whole area the satellite can see at an instant is in the access zone of the antenna. The orbit altitude will affect the coverage area. To determine the field of view, we consider a minimal elevation of the spacecraft over the horizon, ϵ, of 5°.

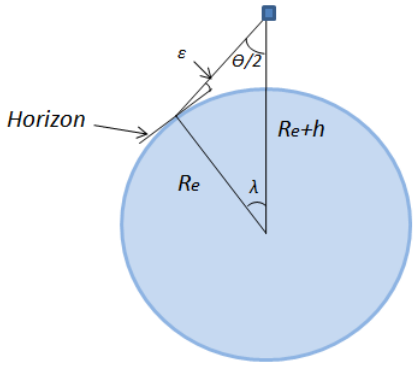


Figure 3.10: Graphical notations used in the determination of the footprint.

	Perigee (354 km)	Apogee (1447 km)
$\theta[°]$	141.40	108.58
$\lambda[°]$	14.30	30.71
Footprint length$[km]$	3183	6837
Footprint area$[km^2]$	7.918 E6	3.584 E7

Table 3.3: Field of view of OUFTI-1.

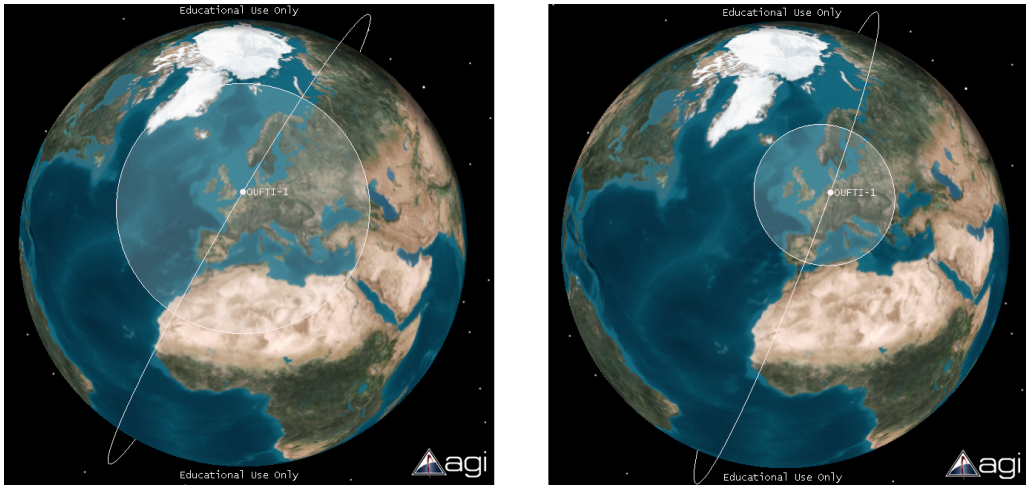


Figure 3.11: Footprint of OUFTI-1 in two cases: apogee over Europe (left), and perigee over Europe (right).

Footprints are displayed in Fig. 3.11 for two different cases. In the best case, corresponding to the apogee, the footprint has a length of 6837 km. OUFTI-1 can then cover a large piece of populated areas, hence having a broad group of users. The satellite passing over Liège can keep a ham-radio operator in the north of Norway in contact with one in the south of Morocco. At the perigee, OUFTI-1 still covers the entire western Europe, with a footprint length of 3183 km.

Access to ground station

This analysis aims at estimating the frequency and duration of access with the ground station, hence the time available for communication with the ground station in Liège. In order to take into account the obstacles around the ground station and the refraction by the atmosphere, two constraints must be met by those access periods:

- Duration of visibility from AOS at 5° to LOS at 5°,
- One only considered orbit with a minimum elevation reached of 25°.

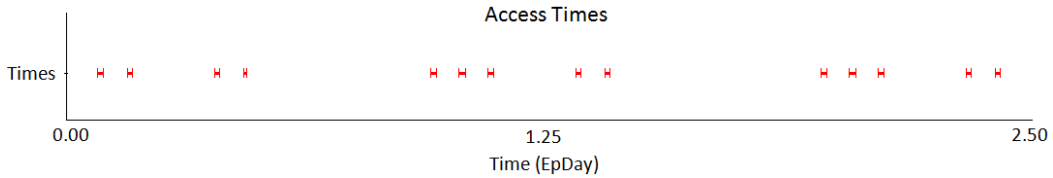


Figure 3.12: GS access times.

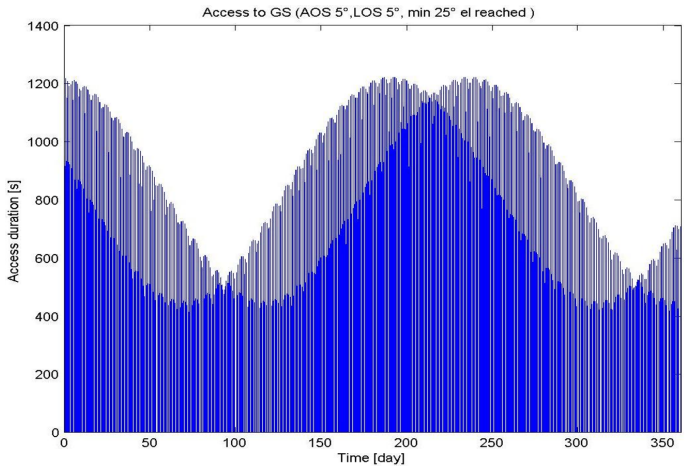


Figure 3.13: Access to the ground station .

Figure 3.13 shows the duration of access with respect to time during one year. The variations of duration are due to the change of the argument of perigee of the orbit during the year. The periods when the access duration are maximum correspond to the moment of the year when the argument of perigee is near 230°, which corresponds to the apogee above Liège. The duration of access is then longer because of the low speed of the satellite when passing above the GS and the larger covered area of the satellite.

The mean duration of the access will be 14.33 min, and there will be 3.3 accesses per day. Therefore, on average, the satellite will be visible 47 min/day. Table 3.4 shows that there will be at least one access per day which ensures a daily communication link between the GS in Liège and the satellite.

	Duration of access	Nbr of access per day
Mean	14,33	3,30
Max	20,35	6
Min	7,42	1

Table 3.4: Access to the ground station.

The shortest duration of access is when the perigee is over Belgium. This is a result of the combined effects of the extremely high speed of the satellite and of its low altitude. The communication time is consequently really short. In this worst case the access time is of 7.42 min.

3.3 Space environment

Satellites operate in an environment that is significantly different from what we know on Earth. It is thus important to understand the nature of space environment

(which is quite hostile) and of its effects on the satellite hardware and on its capacity to operate. In some specific cases, it is also useful to perform tests in chambers reproducing the conditions of space, e.g. in terms of pressure (vacuum), temperature, and radiation. Such understanding and tests are the basis for properly design the satellite and predict its performance via simulations.

The environment encountered by a satellite depends strongly on its orbit, and this section will focus on aspects corresponding to typical Earth missions. The section describes the influence of the environmental conditions on the design of the satellite and the choice of appropriate materials and technologies.

3.3.1 Neutral atmosphere

Atomic oxygen

The atomic oxygen atoms impact the satellite with high velocities, about 7.5 km/s, and their high chemical reactivity causes rapid reactions with impacted materials. To reduce this erosion, the satellite faces are covered with a protective layer resistant to atomic oxygen.

3.3.2 Plasma

Particles in plasma around the spacecraft are not neutral, therefore possibly leading to charging of the spacecraft, and hence to subsequent electric discharges. These discharges may cause electrical problems, deterioration of the thermal shielding and of the solar panels. Electrical charges are created not only by the impact of charged particles, but also by the photo-ionization due to solar radiation. The face exposed to the sun will be charged positively compared to the one in the shadow. Therefore, the outer surfaces of the satellite will be electrically connected and will be recovered by a conducting layer.

3.3.3 Radiations

Figure 3.14 shows the four sources of radiation and the two effects they contribute to.

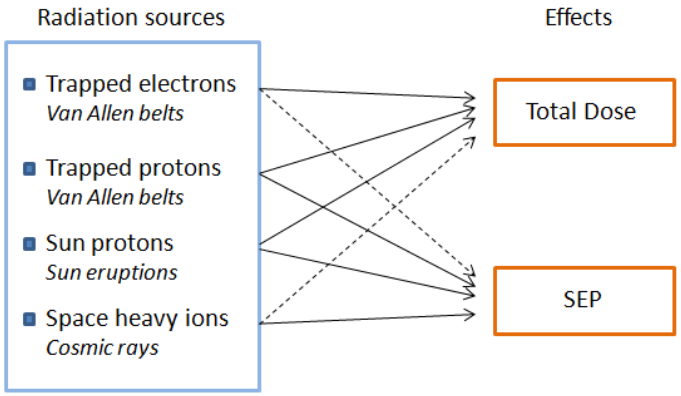


Figure 3.14: Four sources of radiation and the two effects they contribute to.

Radiation environment

The population of charged particles stably trapped by the Earth magnetic field consists mainly of protons with energies between 100 keV and several hundreds of MeV and electrons with energies between a few tens of keV and 10 MeV (Fig. 3.15).

- The trapped proton population: The energetic (above 10 MeV) trapped proton population is confined to altitudes below 20000 km, while lower energy protons cover a wider region, with protons below 1 MeV reaching geosynchronous altitudes.
- The trapped electron population: The population distribution is characterized by two zones of high intensity, below altitudes of one Earth radius and above two Earth radii in the magnetic equatorial plane, respectively, which are separated by a region of low intensities. The location of the inner and outer belts depends on electron energy, with higher energy electrons confined more to the inner belt.

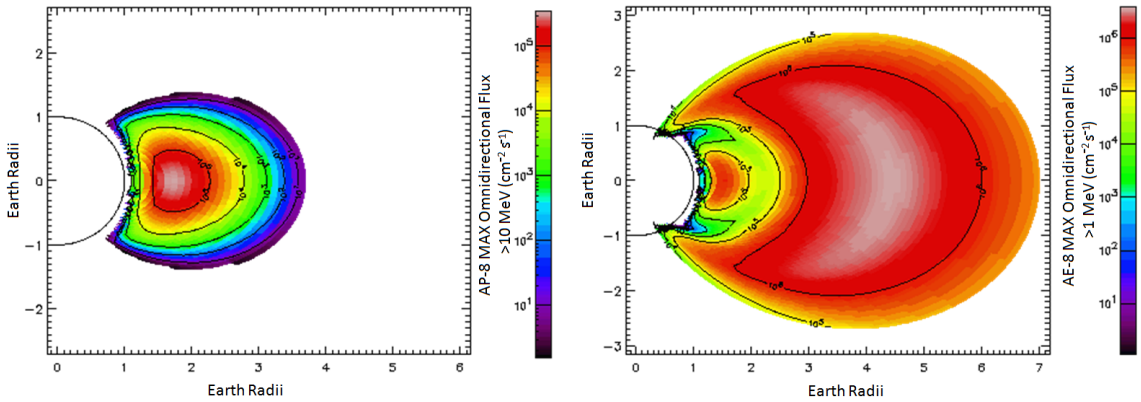


Figure 3.15: Trapped radiation (Left: Distribution of trapped protons; Right: Distribution of trapped electrons).

The sources of radiation have two main effects on the satellite, referred to as total dose and single event phenomena (SEP). The first effect is the result of accumulation of radiation over time. The second effect can be the result of sudden effect of a single particle hitting an electronic component.

Total dose

OUFTI-1 will have its apogee inside the Van Allen belt, and, it will thus be confronted to a very intense radiation environment. Energetic particles, particularly from the Van Allen radiation belts, cause radiation damages to electronic components, solar cells, and materials. These particles can easily penetrate typical spacecraft walls and deposit doses of hundreds of kilorads during missions in some orbits.

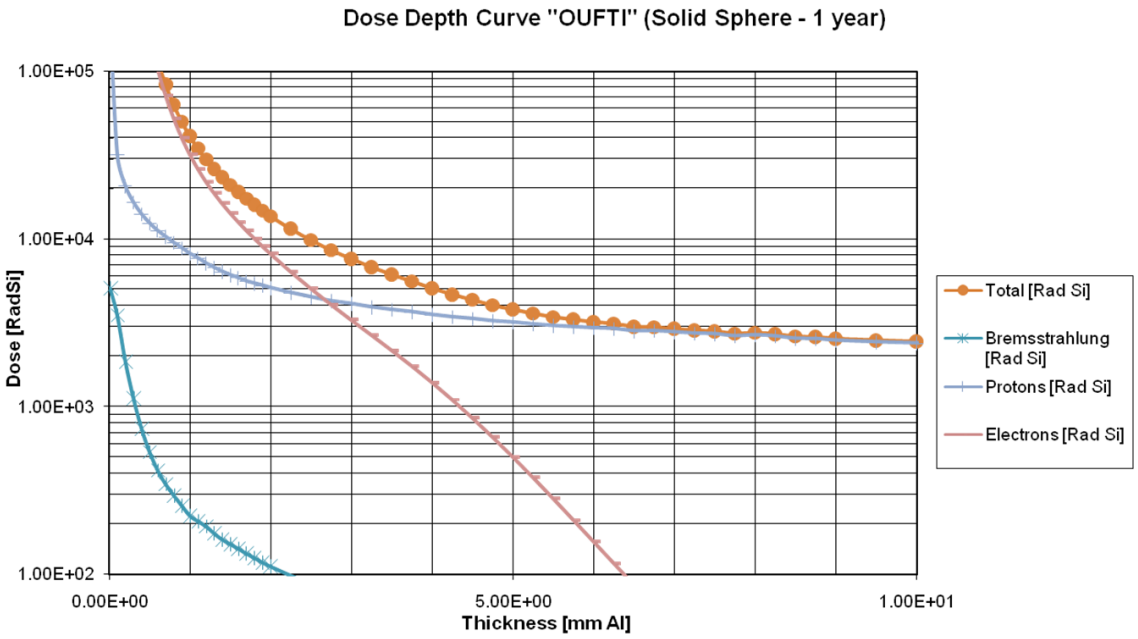


Figure 3.16: Dose depth curve for OUFTI-1's orbit.

In figure **3.16**, the radiation dose as a function of equivalent aluminum shielding thickness is represented. The unit for the radiation dose is the rad, which is the amount that deposits 100 ergs (6.25˜E7MeV) per gram of target material. These values were calculated for a total mission duration of one year, with the hypothesis of solar maximum and a spherical shell shield configuration. The total radiation dose is the addition of the effects of the trapped electrons and protons, the solar protons, and the Bremsstrahlung effect, which is the electromagnetic radiation produced by the deceleration of a charged particle, such as an electron, when deflected by another charged particle, such as an atomic nucleus.

In order to protect the electronic components from this radiation dose and its effects, we need to harden the sensible parts. OUFTI-1's lifetime will be determined by its capacity to harden against the radiation environment much more than by its orbital decay into the atmosphere.

Single Event Phenomena (SEP)

The trapped particles in the radiation belt, as well as solar flare protons and galactic cosmic rays, can cause an instantaneous effect on electronics. The effect is known as Single Event Phenomena (SEP). There are three different types of SEP:

- The Single-Event Upset (SEU) occurs when an incident charged particle deposits a short but intense trail of charge in the sensitive volume of a component. This trail of charge is e.g. capable of reversing the logic state of a memory element.
- The Single-Event Latch-up (SEL) causes destructive latch-up where a parasitic current path is created, allowing large currents to destroy the device. The device no longer operates until the power is turned off and then back on.
- The Single-Event Burnout (SEB): is the most critical situation because the device fails permanently.

3.3.4 Out-gassing

Out-gassing is due to the vacuum conditions and is the slow release of a gas that was absorbed or adsorbed in some material. For OUFTI-1, the major contamination problem from out-gassing is the deposit on solar cell surfaces, resulting in a decrease of their efficiency by obscuring them. This phenomenon can be minimized by the proper selection of materials.

3.3.5 Thermal environment

Space vacuum means that heat transfers can only take place by radiation and conduction. The absorbed external fluxes come from the Sun, the Earth albedo, and IR radiation. Typically, the outer surfaces of the CubeSat, e.g. the solar cells, may experience temperatures ranging from -30°C to +60°C, whereas the inner parts, e.g. the electronic components, may experience temperatures ranging from -10°C to +40°C. The thermal cycles create structural constraints leading to the degradation of the structure. These constraints can be reduced by using materials having the same expansion coefficients.

3.4 Mission timeline

3.4.1 Timeline drivers

Three parameters will have an impact on the timeline of the mission:

- Orbital parameters,
- Location of ground station(s),
- CubeSat requirements of CalPoly (Appendix A).

3.4.2 Mission phases

The subsequent paragraphs provide a short description of the different phases, starting from the final acceptance of the satellite until its disposal. The "orbital time" should be understood as being the time elapsed since the moment the satellite separates from the launcher.

Pre-launch

In the pre-launch phase, planned final launch site tests are performed and the satellite is prepared for launch, which includes the charging of batteries and the verification of the satellite subsystems. This action should be performed once OUFTI-1 will already be integrated into its launch POD, if possible.

Launch and early operations (LEOP)

The LEOP phase will start after satellite separation from its launch POD. The following operations will occur:

- A 30-minutes "off" time duration before transmission and antenna deployment,
- Then, OUFTI-1 can be switched on and deploy its antennas,
- Operators can initiate the satellite acquisition (CW beacon acquired somewhere on Earth and brought back to OUFTI-1's ground system),
- Operators end up by validating the correct operation environment on-board the satellite (decision at ground station).

This phase will end once the validation steps above have been performed. This phase will be terminated earliest on orbital day 10 [TBC].

Commissioning phase

During the satellite commissioning phase, the on-board systems of the OUFTI-1 satellite will be tested and their operational performances confirmed. The results will be used to correct and calibrate the on-ground satellite models (simulators). Commissioning will end after the validation of all systems and will be concluded the earliest on orbital day 10 [TBC]. It should be well-understood that the commissioning phase should take place when the satellite will be acquired by its own ground station.

Nominal phase

During OUFTI-1's nominal phase, the satellite will be operational and shall perform the defined program to demonstrate the different payloads objectives. This phase will end after the third orbital month [TBC].

Extended phase and disposal

If the satellite is still operational after this phase, the mission will be extended by maintaining the D-STAR payload active so that it can be used by the ham-radio community. The satellite degradation will be monitored up to a mission critical failure.

Recovery phases

For each phase, possible failure scenarios and recovery plans will be elaborated. The recovery plans for the nominal and extended phases will be identical.

3.4.3 LEOP tentative mission timelines

This section will present a proposed mission timeline for LEOP for the launch site. The launch site will be the Centre Spatial Guyanais (CSG) at Kourou, French Guiana. A 400-km altitude launch towards north has been considered so far. Launch during daylight has been assumed. A reference ground track will be provided as soon as enough parameters from ESA will be available. At this point, too many assumptions must be made to consider any simulation to be significant. The first contact between OUFTI-1 and the Liège ground station is foreseen to be made 20 hours after satellite injection. This value is TBC and is taken as a working hypothesis.

Up to date, it is foreseen to have more than OUFTI-1's own ground station located in Liège, Belgium. Participate in a ground station network or building a ground station for OUFTI-1 somewhere else are options under study.

As far as the mission is concerned, the launcher performance and the launch site will influence the time after injection at which the acquisition of OUFTI-1 will be acquired by OUFTI-1's ground station. A proposition of a timeline for the launch will also be provided when enough parameters will be known

In conclusion, the first contact with OUFTI-1's ground station might occur as late as 20 hours [TBC] after satellite injection. We thus need to know perfectly the orbital parameters, which is something not always well-known in the case of CubeSats. These simulations drive the following requirements:

- Capability to perform autonomous operations after injection for 20 hours [TBC],
- It is highly possible that the first signal (CW beacon) will be received by the ham-radio community. Therefore, a need to establish contacts with the ham-radio throughout the world is necessary.
- If possible, building a second ground station elsewhere or establishing partnership with other ground stations could result in the decrease of the LEOP duration.

3.5 Operations and data flows

3.5.1 Operations

OUFTI-1 will be operated by students. The different operation scenarios and the way students will be trained to operate a satellite will be established later. Developing a test bench should be considered.

As described earlier in this document, three operations scenarios should be considered:

- Satellite commissioning (checkup of the system),
- Nominal satellite operations,
- Fault handling (how to resolve a failure).

3.5.2 Space and ground data system

We will have two different possible data rates: 9.6kbits/sec for the TC/TM communication, and 4.8 kbits/sec for the D-STAR payload. These low data rates are the design drivers for our data system, both in the spacecraft and for our ground system. This should be considered in the system architecture of OUFTI-1 as a major requirement.

We briefly describe the telecommand and telemetry exchange. The ground-to-space data link will be implemented using the AX.25 protocol. In a first time, we will define our own format of telecommand and telemetry exchange. In a second time, we could consider using CCSDS and the ESA packet utilization standard (PUS) [1, 16, 17].

The various measurement systems on-board will generate the telemetry. We plan to have a centralized TM processing, which is to say that each subsystem/payload will be able to generate its own telemetry, but the on-board computer will centralize all the information before sending it through the RF transceiver to the ground station. There should also be a buffer system in OUFTI-1 so that TM will be stored before being sent to the ground station.

The OUFTI-1 mission control center will be responsible for generating the telecommand packets, and uplinking it to the satellite through the RF transceivers. The on-board computer system will be in charge of performing the right commands and getting back the telemetry from the right subsystem/payload.

Even though D-STAR is a payload and will not be used for TC/TM exchanges, it must be considered here, as it will interfere in the link budget and the system architecture, both on board and on ground.

3.5.3 Link budgets

As mentioned earlier, there are three link budgets of importance in the case of OUFTI-1, one for the TC/TM link, one for the CW beacon, and the third one for the D-STAR payload. The "AMSAT/IARU Annotated Link Model System" developed by J. A. King has been used to compute our link budgets. Those link budgets are guidelines for further design and will be re-computed as the system design goes further. The different values used in those budgets will either be confirmed or changed if necessary [21].

Uplink data link budget (AX.25)

OUFTI-1 Uplink Command Budget:		NOTE:	
Parameter:		*Value:*	*Units:*
Ground Station:			
Ground Station Transmitter Power Output:		100.0	watts
	In dBW:	20.0	dBW
	In dBm:	50.0	dBm
Ground Stn. Total Transmission Line Losses:		2.9	dB
Antenna Gain:		12.8	dBi
Ground Station EIRP:		29.9	dBW
Uplink Path:			
Ground Station Antenna Pointing Loss:		0.3	dB
Gnd-to-S/C Antenna Polarization Losses:		3.0	dB
Path Loss:		153.4	dB
Atmospheric Losses:		1.1	dB
Ionospheric Losses:		0.7	dB
Rain Losses:		0.0	dB
Isotropic Signal Level at Spacecraft:		-128.6	dBW
Spacecraft (Eb/No Method): ***------- Eb/No Method -------***			
Spacecraft Antenna Pointing Loss:		1.5	dB
Spacecraft Antenna Gain:		2.2	dBi
Spacecraft Total Transmission Line Losses:		1.1	dB
Spacecraft Effective Noise Temperature:		241	K
Spacecraft Figure of Merrit (G/T):		-22.8	dB/K
S/C Signal-to-Noise Power Density (S/No):		75.7	dBHz
System Desired Data Rate:		9600	bps
	In dBHz:	39.8	dBHz
Command System Eb/No:		35.9	dB
Demodulation Method Seleted:		G3RUH FSK	
Forward Error Correction Coding Used:		None	
System Allowed or Specified Bit-Error-Rate:		1.0E-05	
Demodulator Implementation Loss:		1.0	dB
Telemetry System Required Eb/No:		18	dB
Eb/No Threshold:		19	dB
System Link Margin:		16.9	dB
Spacecraft Alternative Signal Analysis Method (SNR Computation): ***--------- SNR Method ----------***			
Spacecraft Antenna Pointing Loss:		1.5	dB
Spacecraft Antenna Gain:		2.2	dBi
Spacecraft Total Transmission Line Losses:		1.1	dB
Spacecraft Effective Noise Temperature:		241	K
Spacecraft Figure of Merrit (G/T):		-22.8	dB/K
Signal Power at Spacecraft LNA Input:		-129.1	dBW
Spacecraft Receiver Bandwidth:		19,200	Hz
Spacecraft Receiver Noise Power (Pn = kTB)		-161.9	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:		32.9	dB
Analog or Digital System Required S/N:		19.0	dB
System Link Margin		13.9	dB

Figure 3.17: Data uplink budget.

Downlink data link budget (AX.25)

OUFTI-1 Downlink Telemetry Budget:		NOTE:	
Parameter:		*Value:*	*Units:*
Spacecraft:			
Spacecraft Transmitter Power Output:		0.7	watts
	In dBW:	-1.9	dBW
	In dBm:	28.1	dBm
Spacecraft Total Transmission Line Losses:		1.1	dB
Spacecraft Antenna Gain:		2.2	dBi
Spacecraft EIRP:		-0.8	dBW
Downlink Path:			
Spacecraft Antenna Pointing Loss:		1.5	dB
S/C-to-Ground Antenna Polarization Loss:		3.0	dB
Path Loss:		143.9	dB
Atmospheric Loss:		1.1	dB
Ionospheric Loss:		0.7	dB
Rain Loss:		0.0	dB
Isotropic Signal Level at Ground Station:		-151.1	dBW
Ground Station (EbNo Method): ***------- Eb/No Method -------***			
Ground Station Antenna Pointing Loss:		0.5	dB
Ground Station Antenna Gain:		18.5	dBi
Ground Station Total Transmission Line Losses:		2.4	dB
Ground Station Effective Noise Temperature:		480	K
Ground Station Figure of Merrit (G/T):		-10.7	dB/K
G.S. Signal-to-Noise Power Density (S/No):		66.4	dBHz
System Desired Data Rate:		9600	bps
	In dBHz:	39.8	dBHz
Telemetry System Eb/No for the Downlink:		26.5	dB
Demodulation Method Seleted:		G3RUH FSK	
Forward Error Correction Coding Used:		None	
System Allowed or Specified Bit-Error-Rate:		1.0E-05	
Demodulator Implementation Loss:		1	dB
Telemetry System Required Eb/No:		18	dB
Eb/No Threshold:		19	dB
System Link Margin:		7.5	dB
Ground Station Alternative Signal Analysis Method (SNR Computation): ***---------- SNR Method ----------***			
Ground Station Antenna Pointing Loss:		0.5	dB
Ground Station Antenna Gain:		18.5	dBi
Ground Station Total Transmission Line Losses:		2.4	dB
Ground Station Effective Noise Temperature:		480	K
Ground Station Figure of Merrit (G/T):		-10.7	dB/K
Signal Power at Ground Station LNA Input:		-135.4	dBW
Ground Station Receiver Bandwidth (B):		19,200	Hz
G.S. Receiver Noise Power (Pn = kTB)		-159.0	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:		23.5	dB
Analog or Digital System Required S/N:		16.0	dB
System Link Margin		7.5	dB

Figure 3.18: Data downlink budget.

Uplink D-STAR payload link budget

OUFTI-1	NOTE:	
Uplink D-STAR budget:		
Parameter:	*Value:*	*Units:*
Ground Station:		
Ground Station Transmitter Power Output:	25.0	watts
In dBW:	14.0	dBW
In dBm:	44.0	dBm
Ground Stn. Total Transmission Line Losses:	2.9	dB
Antenna Gain:	12.5	dBi
Ground Station EIRP:	23.5	dBW
Uplink Path:		
Ground Station Antenna Pointing Loss:	0.1	dB
Gnd-to-S/C Antenna Polarization Losses:	0.2	dB
Path Loss:	154.8	dB
Atmospheric Losses:	2.1	dB
Ionospheric Losses:	0.7	dB
Rain Losses:	0.0	dB
Isotropic Signal Level at Spacecraft:	-134.4	dBW
Spacecraft (Eb/No Method):		
------- Eb/No Method -------		
Spacecraft Antenna Pointing Loss:	0.0	dB
Spacecraft Antenna Gain:	0.0	dBi
Spacecraft Total Transmission Line Losses:	1.1	dB
Spacecraft Effective Noise Temperature:	240	K
Spacecraft Figure of Merrit (G/T):	-24.9	dB/K
S/C Signal-to-Noise Power Density (S/No):	69.3	dBHz
System Desired Data Rate:	4800	bps
In dBHz:	36.8	dBHz
Command System Eb/No:	32.5	dB
Demodulation Method Seleted:	GMSK	
Forward Error Correction Coding Used:	None	
System Allowed or Specified Bit-Error-Rate:	1.0E-05	
Demodulator Implementation Loss:	1.0	dB
Telemetry System Required Eb/No:	9.6	dB
Eb/No Threshold:	10.6	dB
System Link Margin:	21.9	dB
Spacecraft Alternative Signal Analysis Method (SNR Computation):		
---------- SNR Method ----------		
Spacecraft Antenna Pointing Loss:	0.0	dB
Spacecraft Antenna Gain:	0.0	dBi
Spacecraft Total Transmission Line Losses:	1.1	dB
Spacecraft Effective Noise Temperature:	240	K
Spacecraft Figure of Merrit (G/T):	-24.9	dB/K
Signal Power at Spacecraft LNA Input:	-135.5	dBW
Spacecraft Receiver Bandwidth:	6,000	Hz
Spacecraft Receiver Noise Power (Pn = kTB)	-167.0	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:	31.5	dB
Analog or Digital System Required S/N:	14.4	dB
System Link Margin	17.1	dB

Figure 3.19: D-STAR uplink budget.

Downlink D-STAR payload link budget

OUFTI-1 Downlink D-STAR Budget:		NOTE:	
Parameter:		*Value:*	*Units:*
Spacecraft:			
Spacecraft Transmitter Power Output:		0.8	watts
	In dBW:	-1.2	dBW
	In dBm:	28.8	dBm
Spacecraft Total Transmission Line Losses:		1.1	dB
Spacecraft Antenna Gain:		0.0	dBi
Spacecraft EIRP:		-2.3	dBW
Downlink Path:			
Spacecraft Antenna Pointing Loss:		0.0	dB
S/C-to-Ground Antenna Polarization Loss:		0.2	dB
Path Loss:		145.3	dB
Atmospheric Loss:		2.1	dB
Ionospheric Loss:		0.7	dB
Rain Loss:		0.0	dB
Isotropic Signal Level at Ground Station:		-150.7	dBW
Ground Station (EbNo Method):			
------- Eb/No Method -------			
Ground Station Antenna Pointing Loss:		0.5	dB
Ground Station Antenna Gain:		10.9	dBi
Ground Station Total Transmission Line Losses:		0.4	dB
Ground Station Effective Noise Temperature:		551	K
Ground Station Figure of Merrit (G/T):		-16.9	dB/K
G.S. Signal-to-Noise Power Density (S/No):		60.5	dBHz
System Desired Data Rate:		4800	bps
	In dBHz:	36.8	dBHz
Telemetry System Eb/No for the Downlink:		23.7	dB
Demodulation Method Seleted:		GMSK	
Forward Error Correction Coding Used:		None	
System Allowed or Specified Bit-Error-Rate:		1.0E-05	
Demodulator Implementation Loss:		1	dB
Telemetry System Required Eb/No:		9.6	dB
Eb/No Threshold:		10.6	dB
System Link Margin:		13.1	dB
Ground Station Alternative Signal Analysis Method (SNR Computation):			
---------- SNR Method ----------			
Ground Station Antenna Pointing Loss:		0.5	dB
Ground Station Antenna Gain:		10.9	dBi
Ground Station Total Transmission Line Losses:		0.4	dB
Ground Station Effective Noise Temperature:		551	K
Ground Station Figure of Merrit (G/T):		-16.9	dB/K
Signal Power at Ground Station LNA Input:		-140.6	dBW
Ground Station Receiver Bandwidth (B):		6,000	Hz
G.S. Receiver Noise Power (Pn = kTB)		-163.4	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:		22.8	dB
Analog or Digital System Required S/N:		10.0	dB
System Link Margin		12.8	dB

Figure 3.20: D-STAR downlink budget.

Downlink CW beacon link budget

There are two ways to compute a link budget for the beacon, the FRIIS formula or the same model as for the previous link budgets.

When using the FRIIS formula [5, 27]:

$$P_{received} = P_{emitted} \times G_1 \times G_2 \times \left(\frac{\lambda}{4\pi}\right)^2 \times \frac{1}{D^2}$$

- $P_{received}$ is the power received at the ground station from the beacon,
- $P_{emitted}$ the power emitted by the beacon,
- G_1 the isotropic gain of the beacon antenna,
- G_2 the isotropic gain of the ground station antenna,
- $\left(\frac{\lambda}{4\pi}\right)^2$ the frequency factor,
- $\frac{1}{D^2}$ the distance factor.

As assumptions, we suppose that $P_{emitted} = 0.1\,W$, $G_1 = 0\,dB$, $G_2 = 10.5\,dB$, $f = 145.940\,MHz$, and $D = 1447\,km$.

After all computations and transformations in dB units, we get

$$P_{received} = -10 + 0 + 10.5 - 15.73 - 126.42 = -141.65\,dB$$

This is a very low power, but it should be high enough to be received by the ground station.

OUFTI-1 Downlink beacon budget:		NOTE:	
Parameter:		*Value:*	*Units:*
Spacecraft:			
Spacecraft Transmitter Power Output:		0.1	watts
	In dBW:	-10.0	dBW
	In dBm:	20.0	dBm
Spacecraft Total Transmission Line Losses:		0.4	dB
Spacecraft Antenna Gain:		-8.0	dBi
Spacecraft EIRP:		-18.4	dBW
Downlink Path:			
Spacecraft Antenna Pointing Loss:		0.0	dB
S/C-to-Ground Antenna Polarization Loss:		0.2	dB
Path Loss:		145.3	dB
Atmospheric Loss:		2.1	dB
Ionospheric Loss:		0.7	dB
Rain Loss:		0.0	dB
Isotropic Signal Level at Ground Station:		-166.7	dBW
Ground Station (EbNo Method):			
------- Eb/No Method -------			
Ground Station Antenna Pointing Loss:		0.5	dB
Ground Station Antenna Gain:		10.9	dBi
Ground Station Total Transmission Line Losses:		2.4	dB
Ground Station Effective Noise Temperature:		480	K
Ground Station Figure of Merrit (G/T):		-18.3	dB/K
G.S. Signal-to-Noise Power Density (S/No):		43.1	dBHz
System Desired Data Rate:		20	bps
	In dBHz:	13.0	dBHz
Telemetry System Eb/No for the Downlink:		30.1	dB
Demodulation Method Seleted:		User Defined	
Forward Error Correction Coding Used:		None	
System Allowed or Specified Bit-Error-Rate:		1.0E-05	
Demodulator Implementation Loss:		1	dB
Telemetry System Required Eb/No:		9.6	dB
Eb/No Threshold:		10.6	dB
System Link Margin:		19.5	dB

Figure 3.21: Beacon downlink budget.

Based on this rapid calculation, we can get a more precise result by using the model (Fig. 3.21).

Chapter 4

Design of space system

4.1 Overview

A few steps led to the current design of the space system.

1. Some system level trade-offs were carried out. These involve several areas, such as communication, on-board data handling and configuration.
2. Some technical orientations were chosen and permitted to establish system block-diagrams. There constituted a first step in defining the interfaces between subsystems.
3. Budgets (mass, power, and data) were then allocated. This whole process ended by the consideration of subsystems. For each of them, the design drivers were identified. Technical solutions were then proposed trough a baseline design.

This chapter presents the different steps briefly described above. Some analysis are still to be made, as well as detailed design. They will be presented in the Phase B documentation.

4.2 System level trade-offs

Different trade-offs must be made at the system level. They are presented in this section. The trade-offs to be made at a subsystem level are presented in the subsystem reports.

4.2.1 Communication

The choice must be made between a radio system working in a simplex mode, both for uplink and downlink, or in a duplex mode. A simplex mode requires the radio to switch between the transmit mode and the receive mode. There are two major issues against the use of a simplex mode.

The first issue is the D-STAR payload. Indeed, the D-STAR payload could not be working as described in Section 2.2.1 if OUFTI-1 was in a simplex mode.

The second issue concerns the ITU radio regulations, which state that "Space stations shall be fitted with devices to ensure immediate cessation of their radio emission by telecommand, whenever such cessation is required under the provisions of these regulations" [22]. We must therefore be able to control the state of the radio, and in particular its transmitter at all time. A simplex radio might not be considered to fulfill this requirement, especially if the system is blocked in a transmit state. In addition, a separate receive channel must be considered for telecommands.

For the reasons stated above, a full-duplex radio system will be built. The frequencies will be in the 145 MHz band (VHF) for downlink and 435 MHz band (UHF) for uplink. Even though the UHF band is not a primary radio amateur band (radars are the primary users of this band), the choice has been made considering the latest IARU recommendations. If our uplink frequency was in the same band as our downlink frequency it could cause interferences in our uplink and downlink signals.

Another recommendation is to limit the bandwidth of signals to a minimum. To use of a low-power beacon, and a data transceiver, in addition of our D-STAR transceiver has been decided. The data transceiver can be shutdown.

OUFTI-1 team has been in contact with Graham Shirville, the frequency coordinator for the IARU Region 1. He has given his authorization for the chosen communication system.

4.2.2 On-board data handling

Details about the on-board data handling can be found in [25].

On-board computer architecture

Two general trends are to be considered: centralized architecture and distributed architecture (Figure 4.1).

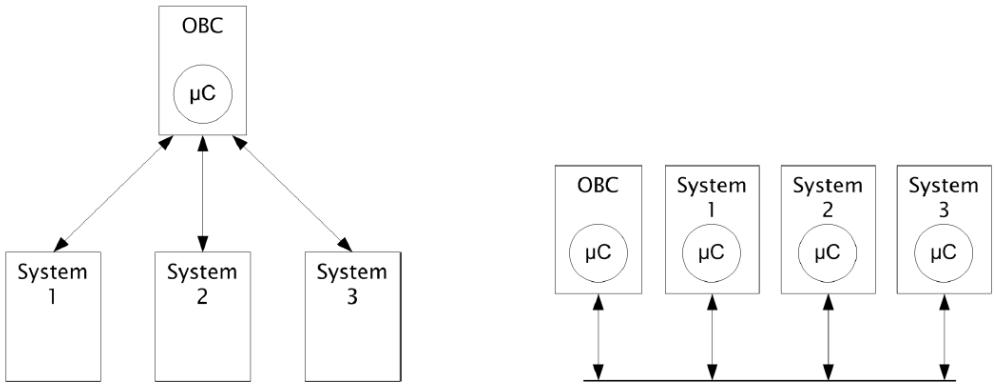


Figure 4.1: Centralized architecture (left) and distributed architecture (right).

In a centralized architecture, the processing power is concentrated in one node, the on board computer (OBC). This OBC is connected to all the subsystems it must interact with. These connections can use either direct lines, or a bus-based system, but this choice will often be dictated by the nature of the components the OBC needs to communicate with. Some advantages and disadvantages are listed in Table 4.1. Let us mention that a centralized architecture is fairly simple to develop and to debug, but may lack some flexibility and, more importantly, redundancy.

In a distributed architecture, each subsystem is typically equipped with its own processor or micro-controller. Information transfer can occur between two subsystems without going through the OBC. The role of the computer is typically to handle and supervise the tasks that involve several subsystems. Such an architecture can use point-to-point connections, but it generally uses a bus-based system. The advantages and disadvantages of this type of architecture are also listed in Table 4.1. The main advantage is that this architecture is more flexible and scalable than a centralized architecture. The different parts of the software may be individually simpler, and they can be developed by independent teams, but a rigorous coordination between them is needed. Moreover, the additional inter-module communication code may add significant development time. Note that the two types of architectures can be combined in various ways, and that hybrid solutions are perfectly possible.

Given the above considerations, we decided to choose a centralized architecture in the case of OUFTI-1.

Signal and data flow paths

To design an appropriate electrical architecture for our satellite, it is crucial to determine where data must be exchanged in the satellite. As indicated before, using

Centralized architecture	Distributed architecture
⊕ Less complex, thus probably more reliable.	⊕ Modular and scalable.
⊕ Communication problems between subsystems less likely.	⊕ Many possible communications paths.
⊕ Simpler hardware.	⊕ Independent testing of subsystems more practical.
⊖ Large single point of failure.	⊖ More complex hardware and software.
⊖ Neither flexible nor scalable.	⊖ One faulty system could perturb the bus and thus all the other subsystems.
⊖ Central processing power shared between subsystems.	⊖ As a whole, trickier to debug.

Table 4.1: Pros and cons of centralized architecture and distributed architecture.

a centralized architecture allows one to simplify the communication means between the subsystems inside the satellite. The following solutions were adopted.

- Data flow paths related to the EPS subsystem

It is clear that the OBC must be able to read the status of the various subsystems, and to send them enabling/disabling commands. It appears clear that this information would be best exchanged through single digital lines, especially for reliability (as opposed to bus-based communication). However, we must keep in mind that this solution requires a significant number of I/O lines on the processor.

- Data flow paths related to the COM subsystem

The modulation/demodulation integrated circuits are controlled via a set of dedicated I/O lines. It is important to note that the digital circuits used in the COM subsystem need a large number of digital I/O lines, on the order of 20 (see [Henrard09, Mahy09]).

- Data flow paths related to the EEPROM and ADCs

We choose these peripherals with an I2C interface. This makes their physical placement on different boards in the satellite easier (which is desirable for the ADCs).

Figure 4.2 gives a high level view of the information transfer between all the subsystems. A more detailed diagram will be produced during the phase B work.

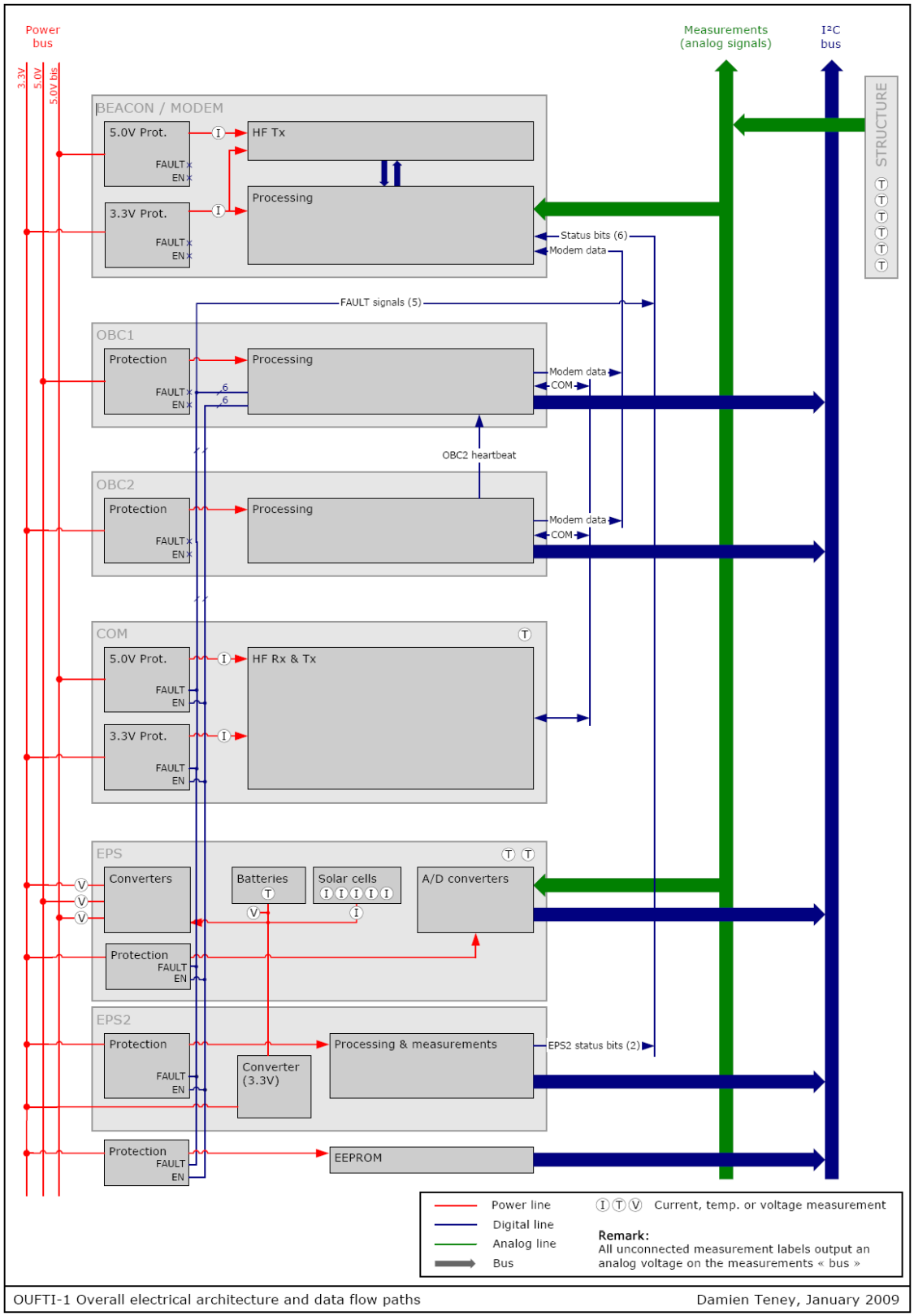


Figure 4.2: Data flow block diagram.

All housekeeping data and payload data (digital data) acquired by the different subsystems through their own sensors will be sent to the OBC, except for 12 important parameters (analog signals) that will be converted and sent to the beacon. The beacon will directly send the data, while the OBC will store information until it gets a chance to transmit information through the COM subsystem.

Processing power

A major criterion for designing a computer for an application as complex as a CubeSat is probably the processing power. Even though other characteristics such as power consumption are important, a processor would be of little interest if it cannot handle all the real-time tasks that need to be accomplished.

In the case of OUFTI-1, the critical real-time operations are related to the EPS and COM subsystems.

First, the OBC needs to manage the utilization of electrical power available from the EPS. The OBC must enable and disable the various subsystems depending in particular as a function of the available electrical power. The OBC must turn off the different subsystems, typically those that draw too much current because of a short circuit. One can easily determine that a trivial implementation of these tasks would need very little processing time on any common micro-controller.

Second, the OBC needs to process in real time data from the COM subsystem. Integrated circuits are used for the modulation and the demodulation of the radio signals, but the tasks of encoding and decoding of the data are left to the OBC.

For the D-STAR, the OBC must involve processing, for each communication channel, a 4800 bit-per-second (bps) flow with a number of complex algorithms ((de)scrambling, (de)interleaving, (de)convolution, etc.).

For AX.25 the OBC must handle a data rate of 9600bps to be taken into account. In the OUFTI-1 project, the design and the programming of the COM part of the software was chosen to be the responsibility of the COM team, and the corresponding analysis of the required processing power was therefore performed as part of its work. The team also evaluated the size in memory of the required data structures, such as software buffers. Details are available in [Henrard09, Mahy09].

4.2.3 Structure and configuration

According to the CubeSat standards, the possibility exists to build a one unit, two units, or three units CubeSat. The analysis of the OUFTI-1 mission indicates that a one unit CubeSat is sufficient we do not need additional surface to accommodate more solar panels. The structure will therefore be a one unit CubeSat bought from Pumpkin. The internal configuration will be established, based on the placement of the different electronic cards.

4.3 Various block diagrams of the Satellite

4.3.1 Functional block diagram of Satellite

Figure 4.3 shows the interactions between subsystems. The main functionalities of each subsystem are also highlighted. Electrical interfaces are represented by red dotted lines and data flows by blue lines. Electrical interfaces and dataflows will be discussed in the following sections. The functionalities of the various subsystems and preliminary designs for them are in Section 4.6.

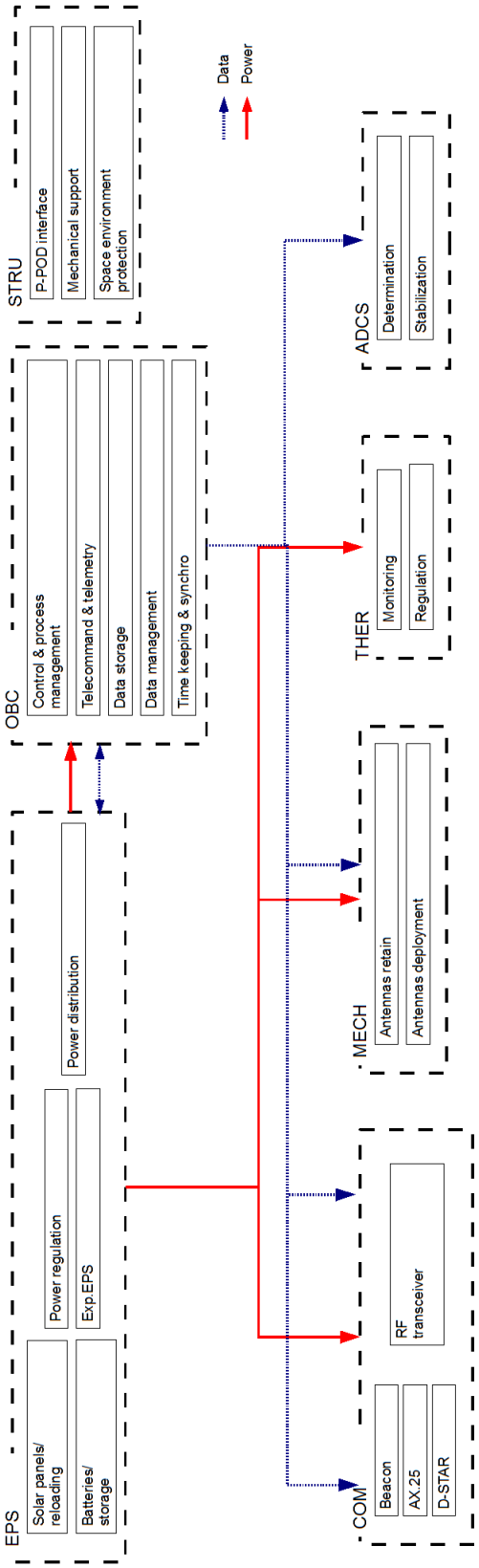


Figure 4.3: System functional block-diagram.

4.3.2 Block diagram of electronics

A block diagram for the entire hardware of the satellite must be made. It will represent the internal information transfer between the different parts of the satellite. This functional diagram will be done as soon as precise interfaces between the different subsystems will be defined.

4.3.3 Block diagram of electrical system

Figure 4.4 shows the block diagram of the electrical system. This diagram shows all the main elements of the electrical architecture and the distribution of power to these elements at the various voltages (3.3V, 3.0V, and 7.2V) The block diagram is draw for the hypothesis where the RF amplifiers uses 7.2V but this may change in the future. A more detailed diagram will be produced during the phase B work. More details can be found in [13, 26].

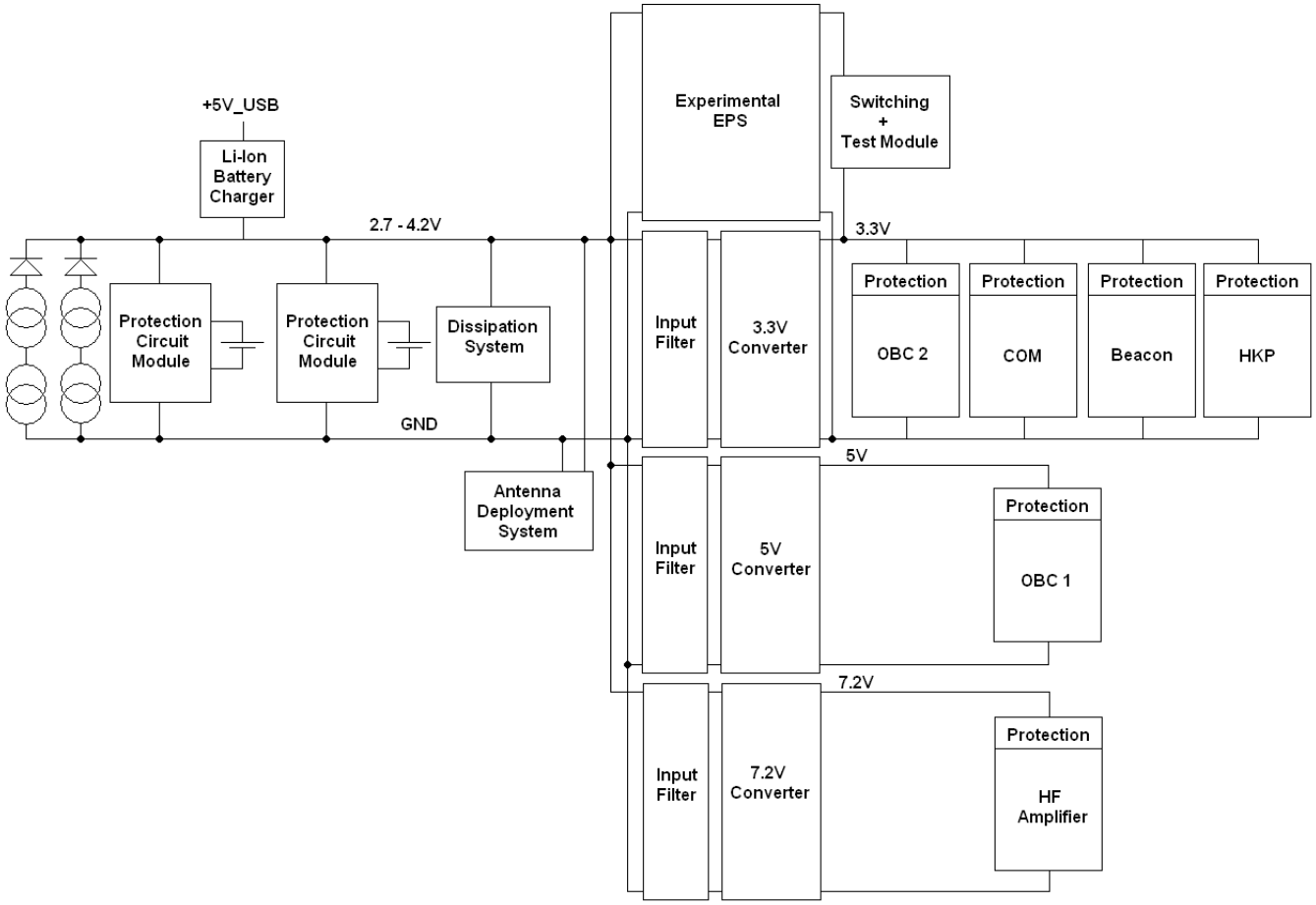


Figure 4.4: Block diagram of electrical system

4.4 Budgets

4.4.1 Mass budget

The CubeSat standard specifies a maximum mass of 1kg. Table 4.2 shows the individual assemblies as well as the current total. We thus have a current mass margin of about 10%.

Assemblies	Mass [g]
Structure	153
Side panels (included solar cells)	216
Antennas (included panel)	33
FM430	66
OBC2	51
EPS	81
Batteries (included PCB)	100
xEPS	61
COM	60
ADCS	14
Others (adhesive, cabling, sensors)	60
Total	895

Table 4.2: Mass budget.

The detailed mass budget can be found in [18].

4.4.2 Power budget

Whether we use one or more units, the electrical power available is fixed by the external surface of the structure (since we do not envision deployable solar panels). The electrical power is then fixed and limited. Therefore, we have to pay a close attention to power budget. The power budgets available with different subsystems on and off will impose specific constraints on the use of the satellite. The power budget that has been established still relies on assumptions. These assumptions will either be confirmed or infirmed.

Assumptions

- Hypotheses were regarding the time each subsystem of the satellite is on or off. These are shown in Table 4.3.
- For the case considered, an orbit lasts 1.72 hours, i.e. 103.2 minutes. The total generated power for an orbit is summarized in Table 4.4.
- The efficiencies of the three converters used in the EPS (further discussed later in Section 4.6.5) are shown in Table 4.5.

- The power needed by the antenna's deployment system is not included here, since it is a one time operation. Its consumption is negligible.

Subsystem	Mode	Time share (%)
EPS	On	100
xEPS	Supply	99
	Test	1
OBC1	On	100
OBC2	On	100
Telemetries	On	100
COM	RX	90
	RX+TX	10
Beacon	On	100

Table 4.3: Time each subsystem is on (in % of the total orbit time).

Min	Mean	Max
1491.80	2365.50	3247.10

Table 4.4: Solar cells input per orbit (in mWh).

Voltage [V]	3.3	5	7.2
Efficiency (%)	90	70	90

Table 4.5: Converters' efficiency (in %).

Energy consumption

The energy consumption is then computed. It is shown in Table 4.6. The total energy consumed is the sum of all individual energy consumed. Then the remaining energy is the difference between the total energy consumed and the energy provided by the solar cells.

The Table 4.7 shows three figures for the remaining energy for three cases. The "minimum" case correspond to all subsystems on, the "maximum" case correspond to all subsystems off and the "mean" case correspond to all subsystems on. Of course a negative value corresponds to a physical impossibility.

Since the energy shown is the Table 4.7 are per orbit, one can convert then in power (mW) by dividing the energy by the orbit duration

Subsystem	Mode	Consumption per orbit [mWh]			
		On 3,3V	On 5V	On 7,2V	On batt. bus
EPS	On	0.00	0.00	0.00	37.84
xEPS	Supply	0.00	0.00	0.00	561.92
	Test	0.00	0.00	0.00	11.35
OBC1	On	0.00	24.57	0.00	0.00
OBC2	On	18.92	0.00	0.00	0.00
Telemetries	On	31.53	0.00	0.00	0.00
COM	RX	141.90	0.00	0.00	0.00
	RX+TX	28.38	0.00	275.20	0.00

Table 4.6: Consumption per orbit (in mWh).

Remaining electrical energy (mWh)		
Minimum	Mean	Maximum
-1116.73	-243.03	638.57

Table 4.7: Remaining electrical energy (in mWh).

Battery capacity

To dimension the battery, two parameters must be taken into account. The battery capacity must be higher than the remaining power corresponding to the "minimum" case. The second parameter is the depth of discharge (DOD) of the battery. It corresponds to the percentage of the capacity of the battery used at each cycle. The analysis of the battery was performed and can be found in [13].

4.4.3 Data budget

Preliminary data budgets were established [6]. Table 4.8 shows the data budget for the housekeeping parameters, while Table 4.9 shows the measures that we plan to be taken for the xEPS. If we end up not having the time to take all measurements, we will ignore the less critical ones. We made the following choices:

- We use 8 bits for all measurements.
- We use 32 bits for the clock of the satellite.

Measurements	**Sampling rate**	**Number of bytes**	**Number of bytes for clock data**	**TOTAL**
Current solar cell 1-5	Once every 2 minutes	$\frac{5\times(24\times60)}{2} = 3,600$	Every 16 minutes $\Rightarrow \frac{24\times60\times4}{16} = 360$	3960
Current solar cells - total	Once every 2 minutes	$\frac{(24\times60)}{2} = 720$	Every 16 minutes $\Rightarrow 360$	1080
Temperature of faces 1-6	Once every minute	$6 \times 24 \times 60 = 8,640$	Every 16 minutes $\Rightarrow 360$	9000
Battery voltage	Once every 10s	$\frac{24\times3600}{10} = 8,640$	Every 2 minutes $\Rightarrow \frac{24\times60\times4}{2} = 2,880$	11520
Temperature of battery	Once every 30s	$\frac{24\times3600}{30} = 2,880$	Every 4 minutes $\Rightarrow \frac{24\times60\times4}{4} = 1,440$	4320
Temperature of MSP430	Once every 2 minutes	$\frac{24\times60}{2} = 720$	Every 16 minutes $\Rightarrow 360$	1080
Temperature of 7.2V converter (EPS)	Once every minute	$24 \times 60 \times 4 = 1,440$	Every 16 minutes $\Rightarrow 360$	1800
Voltages of each of the 3 buses	Once every 5 minutes	$\frac{3\times24\times60}{5} = 864$	Every 15 minutes $\Rightarrow \frac{24\times60\times4}{15} = 384$	1248
Temperature & current on BCN	Once every 5 minutes	$\frac{3\times24\times60}{5} = 864$	Every 15 minutes $\Rightarrow \frac{24\times60\times4}{15} = 384$	1248
Temperature of dissipation transistor	Once every 30s for maximum 6h	$6 \times 60 \times 2 = 720$	For each measurement $\Rightarrow 6 \times 60 \times 2 \times 4 = 2,880$	3600
Current on COM	Once every 5 minutes	$\frac{24\times60\times2}{5} = 576$	Every 15 minutes $\Rightarrow \frac{24\times60\times4}{15} = 384$	960

Table 4.8: Data budget for the housekeeping parameters.

xEPS	**Sampling rate**	**Number of bytes**	**Number of bytes for clock data**	**TOTAL**
Output current & voltage of converter	Once every minute (note: 24h log even if xEPS not continuously supplied)	$2 \times 24 \times 60 = 2,880$	Every 16 minutes $\Rightarrow \frac{24\times60\times4}{16} = 360$	3,240
Current & voltage of digital circuit	Once every 10 minutes	$\frac{2\times24\times60}{10} = 288$	Every 20 minutes $\Rightarrow \frac{24\times60\times4}{20} = 288$	576
Temperature of converter and digital circuit	Once every 5 minutes	$\frac{2\times24\times60}{5} = 578$	Every 20 minutes $\Rightarrow \frac{24\times60\times4}{20} = 288$	866
Current solar cells 1-5	2 Hz during 1,5 minute every 50 minutes	$90\times2\times5\times20 = 18,000$	Every 50 minutes $\Rightarrow \frac{1000\times4}{50} = 80$	18,080

Table 4.9: Data budget for parameters concerning xEPS.

4.5 Satellite state modes

At the present time, we envision, and use as a works hypothesis, that our satellite will be, at any times, in exactly one of the following(exclusive) modes (also called states):

- Initial mode: This is the mode the satellite will be in during the 30-minutes period that follows the separation from the P-POD.
- Safe mode: Beacon Tx on, AX.25 Rx on.
- Default mode: Beacon Tx on, AX.25 Rx and Tx on.
- D-STAR mode: Beacon Tx on, AX.25 Rx on, D-STAR Rx and Tx on.
- xEPS mode: Beacon Tx on, AX.25 Rx and TX on, xEPS on.

Figure 4.5 shows these modes and gives a preliminary look at the transition between modes.

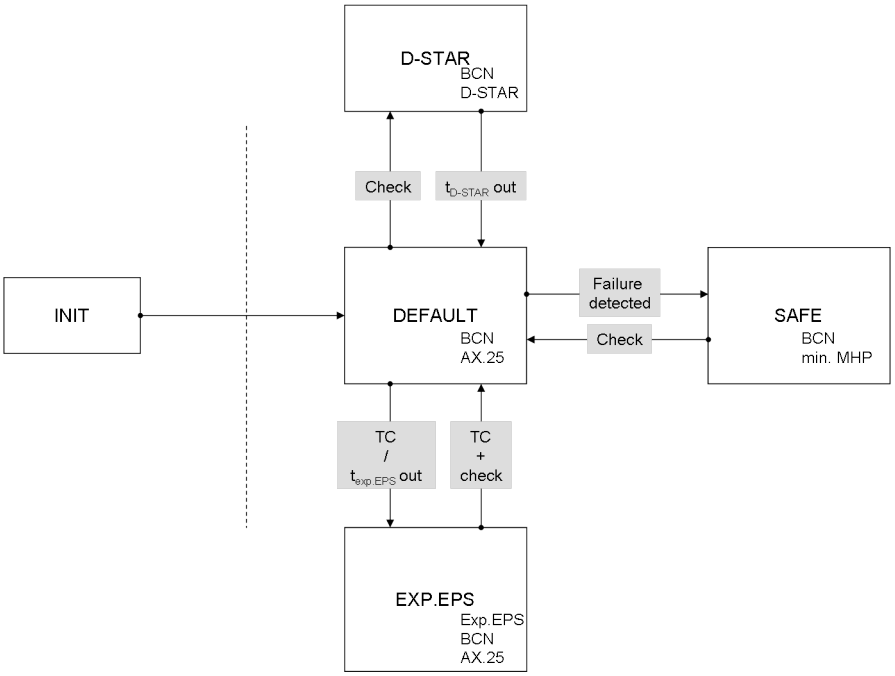


Figure 4.5: Satellite state modes.

Due to their influence on the design of the whole system, hardware and software, the study of the modes will be a high priority of phase B.

4.6 Subsystems

This section describes the different subsystems. For each subsystem, we give the principal design drivers and a preliminary baseline design.

4.6.1 Structure and configuration subsystem

Reference point and coordinates frame of satellite

A coordinates frame must be defined to allow us to give the exact location of each component of our satellite. Its origin (the OUFTI-1 reference point) is chosen at the geometric center of the satellite. The frame is oriented using a right rule, and in such a way that the face with the communication ports is normal to the X-axis and at a negative X coordinates, and that the base plate (face with feet and switches) is normal to the Z axis at a negative Z coordinates. Figure 4.6 shows the reference point and the system of X,Y, Z axes.

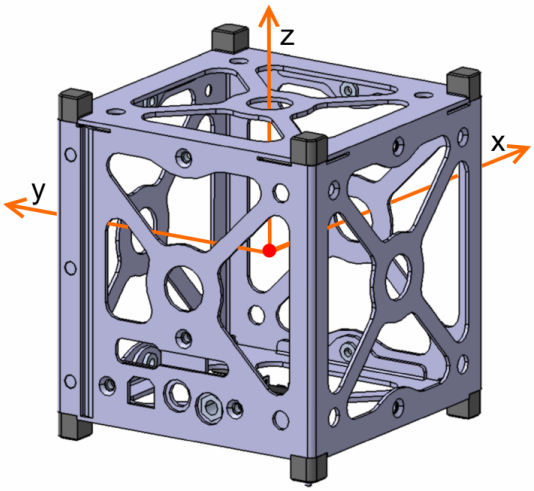


Figure 4.6: OUFTI-1 reference frame.

Overview of functionality

The mechanical structure must contain and support all other subsystems. The structure also plays a major role during the launch,it must withstand the static and dynamic loading as well as the vibrations during the flight and the shocks during the separation, the ignition and the jettisoning of the fairing. Until the CubeSat is ejected in space, the structure is the only interface with the P-POD. Once in orbit, the structure will protect the main payload against the harsh space environment, including radiations, debris, and thermal variations.

The structure can be divided in three parts. The primary structure is the exterior cubical frame. The secondary structure represents any part that will not directly

sustain the structural loads from Launch Vehicle (LV), e.g. solar panels, screws, Printed Circuit Boards (PCBs). Finally, the interface with the P-POD includes the rails, the feet, a Remove Before Flight pin (RBF). Figure 4.7 shows the functionalities assigned to each of these three parts.

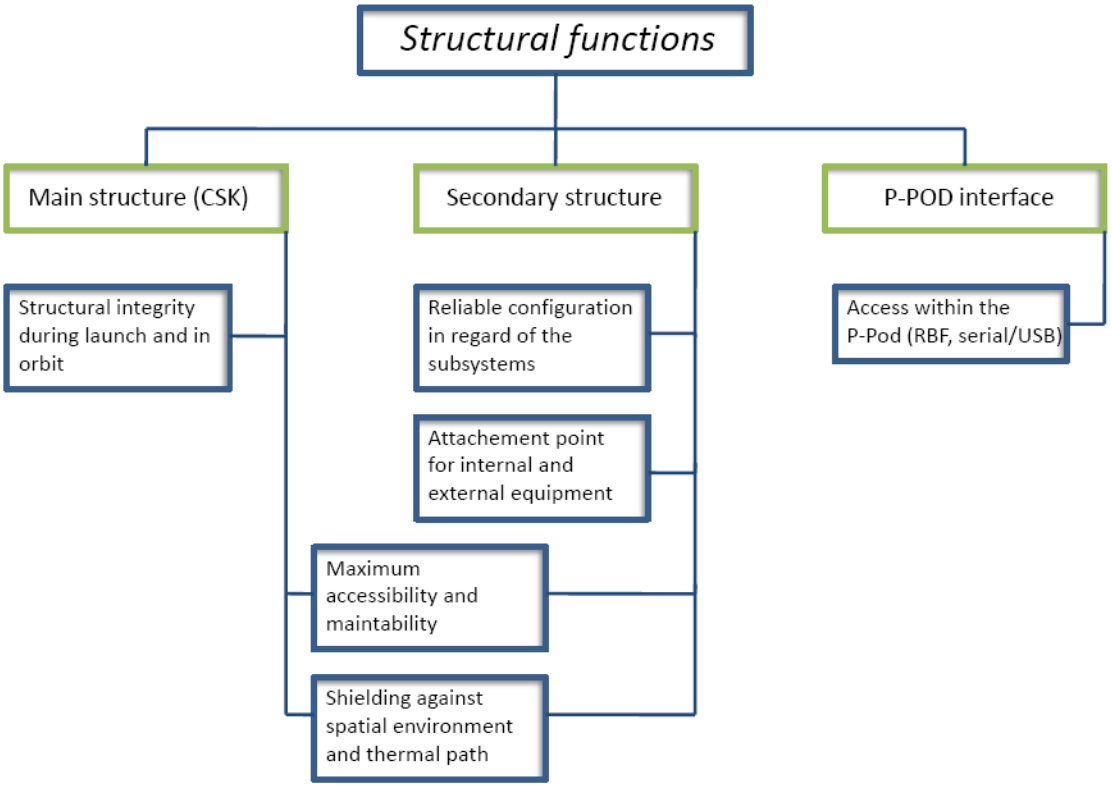


Figure 4.7: Functionalities of each of the three parts of the structure.

Principal design drivers

The structure must be compatible with the CubeSat standard. Reliability and simplicity of development will be considered. The configuration must accommodate all platform and payload elements.

Baseline design

Structure A COTS structure was chosen in order to be compatible with the standard and to gain time in the development. The Pumpkin structure was chosen due to its flight heritage.

Configuration The five electronic cards will follow the PC-104 format and will be stacked on top of each other. The card will be held in place by four vertical endless screws and midplane standoff components from Pumpkin. The batteries which are the heaviest elements will be placed on a sixth dedicated PCB, near the center of

the structure in order to fulfill the CubeSat requirement for the gravity center. Five (of the six) faces of the cube will be covered by solar cells fixed on aluminum panels. These panels will be useful to protect electronic components against radiations. The sixth face will be dedicated to external ports and antennas deployment mechanisms.

This preliminary configuration is shown in the figure 4.8.

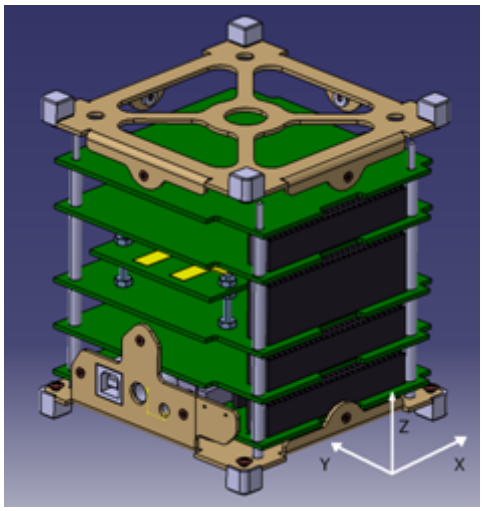


Figure 4.8: Preliminary configuration of OUFTI-1.

Analysis

Finite element analysis was performed to investigate the static loading case and the modal analysis. All simulations were performed with Samcef and Samcef Field.

Detailed analysis, models, and hypotheses can be found in [18].

Static analysis The spacecraft is subjected to quasi-static loads during flight. The aim of this study is to define whether or not the three main structural parts are able to withstand the steady-state acceleration, and to calculate the factor and margin of safety.

The current accommodation of the nine CubeSat on the LV is represented in Figure 4.9. P-Pods are mounted horizontally and radially on the launcher interface. The only case considered in our analysis is that where the P-Pod is in the horizontal position.

The quasi-static accelerations are of $6.3g$ in compression along the launcher longitudinal axis, and $1.2g$ in compression along the lateral axis. With a factor of safety (FOS) of 1.8 (ultimate loading), these accelerations become $11.5g$ and $2.25g$, respectively. The structure modeling will be loaded with three static accelerations in the three spatial directions. Two representative cases will be simulated:

1. The X-axis of the CubeSat correspond to the X-axis of the launcher implying the Y and the Z axis of the CubeSat correspond to the lateral direction of the launcher. This is called the "X case".

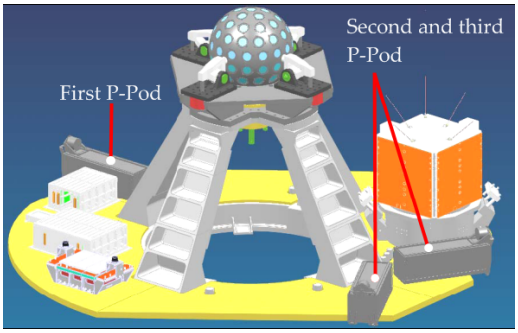


Figure 4.9: P-Pods configuration inside Vega launcher.

2. The Y-axis of the CubeSat corresponds to the X-axis of the launcher implying the X and Z axis of the launcher correspond to the lateral axis of the launcher. This is called the "Y case".

An additional safety way is used in both cases: the lateral acceleration resultant is $\sqrt{2 \times 2.25} = 3.2g$ instead of only $2.25g$. Moreover, the aforementioned steady-state accelerations occur during different phases of the flight, i.e. the longitudinal load is maximum during the third stage acceleration while the lateral loads are maximum during lift-off. The resulting model is therefore conservative.

The results of our analysis are summarized in Table 4.10.

Loading	Max. displacements (mm)	Max. Von-Mises stress (MPa)
X case	0.025	1.66
Y case	0.017	1.73

Table 4.10: Static finite element analysis results.

The maximal displacements are limited to $0.025mm$ on the PCB holding the batteries (see Fig. 3.13), while the maximum Von Mises stress is $1.73MPa$ on the aluminum chassis. We can define the margin of safety (MOS) :

$$MOS = \frac{allowable\,load}{applied\,load \,.\, FOS} - 1$$

With a yield stress of $150MPa$ and a FOS of 1.8, the MOS in the X and Y cases are 49 and 47, respectively. This means that, OUFTI-1 can support a load higher than the one it will experience through all phases of flight before entering the plastic zone.

Modal analysis The model used only includes the external structure. In particular, the deployment switch and all others protruding elements were neglected. Linear structural behavior was assumed, and damping was not included in the FEA.

At least six modes are included in the modal basis under 1,000 Hz. First numerical results are shown in Table 4.11.

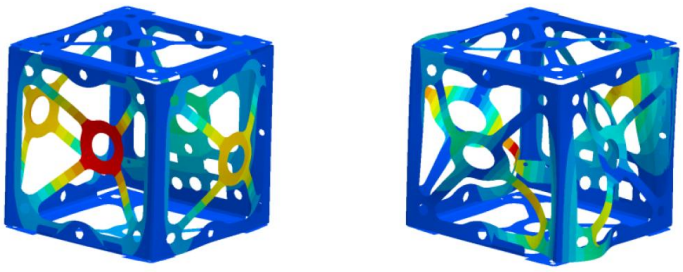
The CSK structure presents two typical modes :

Mode	Frequency (Hz)
1	522.6
2	591
3	604.3
4	643.2
5	709.5
6	796

Table 4.11: Modal parameters under 1,000 Hz.

1. Under 1,000 Hz, each mode involves a vibration in phase of the panel's arms,
2. Above 1,200 Hz, these vibrations change into a more complex deformation and the arms can vibrate out of phase.

Both kinds of vibrations are shown in Fig. 4.10.



(a) Below 1,000Hz (b) Above 1,200Hz

Figure 4.10: Typical vibration patterns of the side panels of the main structure.

Center of gravity and moments of inertia

Location of the COG We decided to test three cases :

1. The upper face of the PCB supporting the batteries is located at 10 mm from the xEPS (case 1) ,
2. The upper face of the PCB supporting the batteries is located at 18 mm from the xEPS (case 2) ,
3. The upper face of the PCB supporting the batteries is located at 5.5 mm from the xEPS (case 3).

The first case corresponds to the height used on the flight model, the second case to the lower battery on the EPS while the third case to the upper battery on xEPS. The results are shown in Table 4.12.

Axis	Case 1 (mm)	Case 2 (mm)	Case 3 (mm)
x	0.61	0.61	0.61
y	-2.36	-2.36	-2.36
z	0.24	-0.66	0.75
Geometric center	2.45	2.52	2.55

Table 4.12: COG location.

The results in the Table 4.12 show that the COG of our satellite is separated from its geometric center by less than $2.6mm$ in the three cases. The COG varies by almost 1.1 mm between the extremal cases. This can be interpreted as a uniform distribution of mass along the direction Z. The COG location along the Y axis can be explained by the unsymmetrical properties of the electronic cards and, particularly, by the effect of the PC-104 connector. If necessary, this effect can be counteracted by moving the PCB holding the batteries in +Y direction: the COG gets closer to the geometric center with $1.2mm$ for each centimeter of the batteries' PCB. Along X axis, the slight difference can be explained with the mass difference between solar and antennas' panel.

From this result, we can conclude that the requirement related to the COG is largely fulfilled, never mind the configuration.

Moment of inertia The moment of inertia can be calculated from any point in the structure, but it is generally preferred to define it from the COG. Thus, the tension of inertia linked to the COG and the frame XYZ is pointing:

$$I_G = \begin{pmatrix} 2.082 & -0.057 & -0.012 \\ -0.057 & 1.765 & -0.096 \\ -0.012 & -0.096 & 2.018 \end{pmatrix} . 10^6 g\, mm^2$$

Moreover, principal moments and vectors can be found to define completely the system :

$$A_1 = \begin{pmatrix} -0.16 \\ 0.94 \\ 0.31 \end{pmatrix}, \quad M_1 = 1.723 \cdot 10^6 g\, mm^2$$

$$A_2 = \begin{pmatrix} 0.15 \\ -0.29 \\ 0.95 \end{pmatrix}, \quad M_2 = 2.049 \cdot 10^6 g\, mm^2$$

$$A_3 = \begin{pmatrix} 0.98 \\ 0.2 \\ -0.1 \end{pmatrix}, \quad M_3 = 2.092 \cdot 10^6 g\, mm^2$$

These values show that the mass distribution is quite uniform. Finally, note that the COG and inertia should be verified experimentally when a complete spacecraft will be available.

Assembly of our satellite

The PCBs will be stacked up on the base plate. They will be held in place by four endless screws and vertically separated by spacers. The chassis is then slid down around the stack. Endless screws are maintained by midplane standoffs screwed to the chassis. The top plate is then assembled. Finally, solar panel and the antennas panel are bonded to the structure. A first procedure for satellite integration can be found in [19].

The structure and the configuration are discussed in details in [18].

4.6.2 Thermal subsystem

Design drivers

The thermal management system has to ensure that all on-board units will be working within their own allowable temperature range. Thermal requirements are summarized in Table 4.13. Batteries have distinct requirements when (1) they are discharged and are in the processes of charging, and (2) they are charged. The most demanding case correspond to the case of charge : the temperature of the batteries must be between 0°C and 45°C.

Component	T_{min}[°C]	T_{max}[°C]	Note
Main structure	-40	+85	
Solar cells	-100	+100	
Electronics	-40	+85	
LiPo batteries	0	45	charge
	-20	60	discharge

Table 4.13: Thermal requirements.

Because of the size and weight constraint, passive means will then be preferred. We have limited flexibility in closing the means to control the temperature inside the CubeSat.

Analysis

Preliminary analysis were performed. Both the hot-case and the cold-case were considered in a single-node transient model of the CubeSat, implemented in the Matlab Simulink environment. The cold-case and the hot-case assumptions are summarized in Table 4.14.

Parameter	Hot case	Cold case
Orbital parameters	permanently illuminated	max eclipse time
Solar constant	1414 [W/m^2]	1322 [W/m^2]
Albedo coefficient	0.35	0.25
Earth temperature	250K ($\Leftrightarrow$220[W/m^2])	260K ($\Leftrightarrow$260[W/m^2])
Internal dissipation	full	none

Table 4.14: Assumptions for hot-case and cold-case.

Hot case Figure 4.11 shows the influence of the orbit orientation on the evolution of the temperature in the hot case temperatures evolution.

One can see that the lower the angle between the line of nodes and the sun direction, the lower the temperature reached. But there is an angle near 70° for which the satellite still experiences no eclipse and yet has a slightly higher albedo. This case

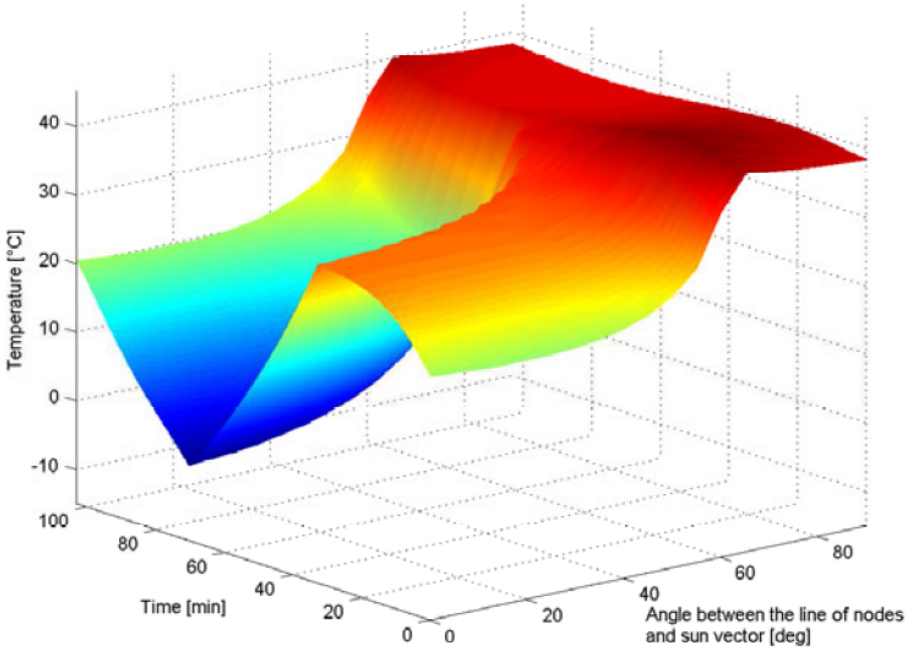


Figure 4.11: Hot case influence of orbit orientation.

would be slightly worst than the assumed one but the difference is small and the hot case will still consider an orbit whose line of nodes is perpendicular to the sun's direction. Recalling that the maximal specified temperature of the batteries of 45°C, the mean temperature of the CubeSat in this preliminary hot-case analysis is already close to it and will thus be carefully looked at in the future analysis during Phase-B work.

Cold case Variation of the temperature as a function of (1) time along a complete orbit and (2) the orientation of the orbit with respect to the orbital flow. Under the cold-case assumptions of Table 4.14.

The temperature falls down to -20°C. This would represent the mean temperature of the CubeSat and one can imagine that external parts could still be cooler and internal parts slightly hotter. When the satellite comes out of eclipse, the batteries begin to charge and this analysis let us think that the batteries could be below their required minimal temperature of 0°C during the charge.

Baseline design

A heating system should be designed for the batteries in order to ensure that their temperature will remain within their allowable ranges. We should plan for using heaters and temperature sensors.

Appropriate coatings should be chosen. However, the available surface area for thermal control coating is small and the effect of coating on the extreme reached

temperatures is quite limited.

Thermal management is discussed in details in [11].

4.6.3 Mechanism subsystem

Design drivers

Our satellite has two mechanisms for the retention and the deployment of the two antennas (VHF and UHF). We refer to these as the "MECH" subsystem.

Besides dimensional requirements, several standard specifications for CubeSats are directly applicable to the MECH subsystem:

2.2.1.3: *Absolutely no pyrotechnics are allowed inside the CubeSat.*

3.2.9: *Deployables must be constrained by the CubeSat. The P-POD rails and walls are NOT to be used to constrain deployables.*

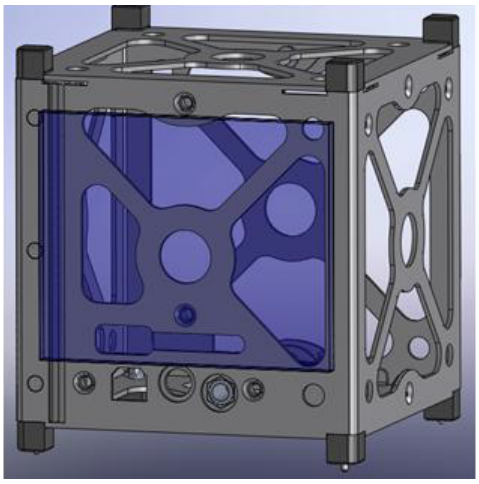
3.4.2: *To allow adequate separation of CubeSats, antennas may be deployed 15 minutes after ejection from the P-POD (as detected by CubeSat deployment switches). (...)*

High reliability is required,both for the retention mechanism and for the deployment mechanism.

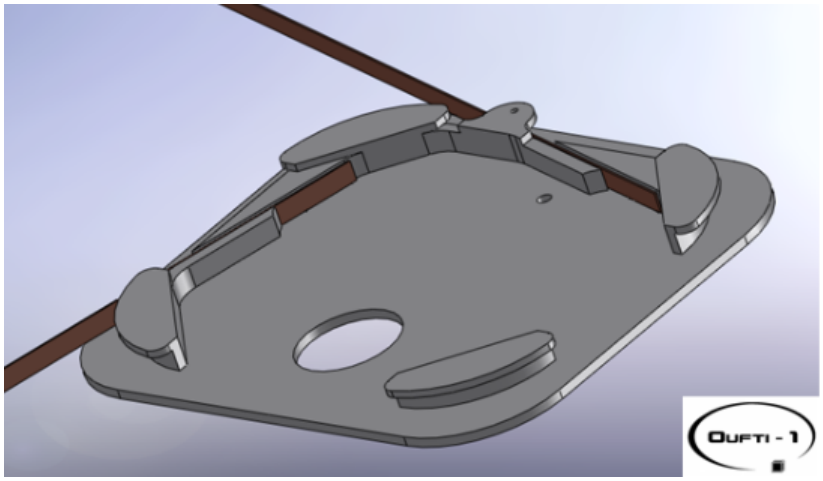
Baseline design

Each antennas is a quarter-wavelength monopole. This means that our antennas will be 17cm and 50cm in length. They will be wound around a support and held in place by a piece of Dyneema wire. To deploy each antenna, a current in a simple thermal knife will melt the wire. These mechanism will be installed on the -X face, without obstructing communication ports, as illustrated in Fig.4.12.

Mechanisms are deeply presented in [28].



(a) Mechanism are situated on the -X face, on the side containing various ports.



(b) Preliminary designs of two deployment mechanisms.

Figure 4.12: Mechanism for deploying our pair of antennas.

4.6.4 Attitude determination and control subsystem

Design drivers

Quarter-waves antennas can be thought of as having a donut-shaped radiation pattern. They could be considered as quasi-omnidirectional antennas, so no precise attitude control is needed. However, the rotation rate should not be too high (because of modulation issues) or too low (because of thermal issues). The precise acceptable range remains to be defined.

The ADCS system should comply with the CubeSat space, weight and power requirements.

Baseline design

The ADCS system will be passive. A permanent magnet will align the spacecraft along the Earth magnetic field, while hysteretic materials will damp the rotation (Fig. 4.13).

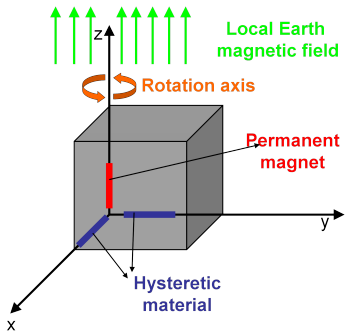


Figure 4.13: Baseline design for ADCS.

The ADCS is discussed in details in [8].

4.6.5 Main electrical power (sub)system

There will be two electrical power systems (EPS) aboard OUFTI-1, i.e. a main EPS and an experimental EPS. They are respectively referred as EPS and xEPS. This section describes the EPS.

Overview of functions and design drivers

The electrical power system has to supply other subsystems of the CubeSat, as required by the mission timeline. Power will be collected from solar panels. Solar cells produce power when they are exposed to direct solar radiations or to indirect radiations from albedo. Batteries will be used to store energy. The collected power will be distributed to the various subsystems needing it: OBC (OBC1 and OBC2), the COM, MECH, THER, EPS and xEPS.

Analysis ([13])

Due to the small size of the satellite and the power required,we tried to find the most efficient type of solar cells. We use triple junction GaAs-based solar cells from AzurSpace which have an efficiency of 30% at begin of life (BOL).This will be the first time this type of solar cells is tested in space environment.

The analysis was performed under the following assumptions [7]:

- Cell degradation factor, at end of life (EOL) values of 0.25,
- Simulation performed at 28°C,
- Solar cells placed on only 5 of the 6 faces of the CubeSat,
- Limit angle of total reflection of 5°,
- Earth Albedo of 30%.

The calculation shows that a mean power of 1.3 W is generated during an orbit for the worst period of the year. This value will be used to dimension the entire system.

Baseline design

Figure 4.14 shows the architecture of the EPS. It consists of three converters providing currents at three different voltages of 3.3V, 5V, and 7.2V, and supplying various subsystems with the voltage required. The need of 7.2V output (and thus the corresponding converter) remains to be confirmed. The (electronic) switches controlling the current for the antenna deployment mechanisms are on the EPS. Note that no MPPT (maximum power point tracker) is used in the system, since the additional achievable power is low compared to the increasing complexity [14].

Protecting the power outputs against overcurrents is another function of the EPS. The choice was made to place such protection circuits, as well as the switches, on the power input of each subsystem. The main reason for this is one of modularity. If changes are required to a client, no changes will be required on the EPS. It is conceptually clear that each subsystem has to take care of its on/off switch and protection.

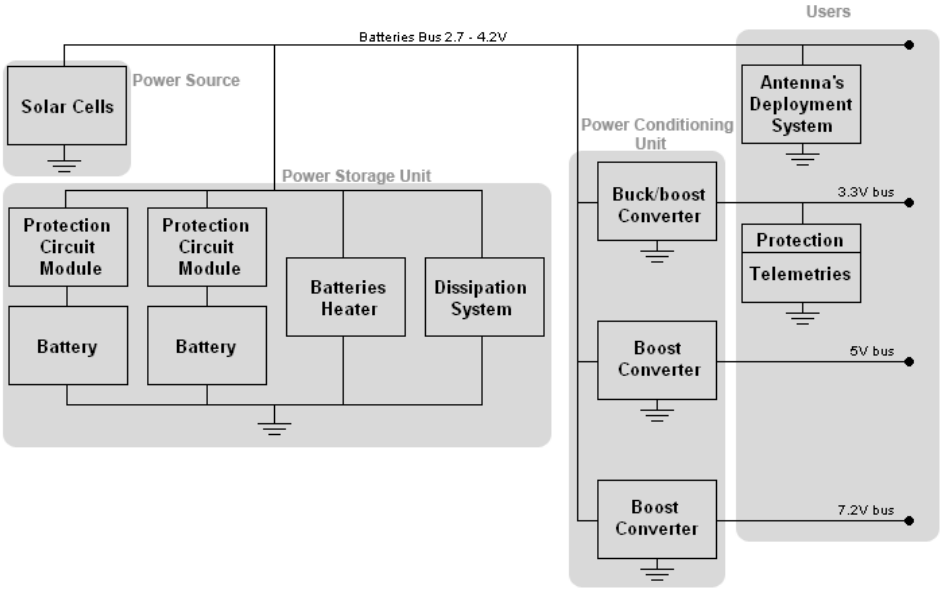


Figure 4.14: Block diagram of EPS .

We chose Li-polymer batteries as storing devices due to their high energy density. We chose a Kokam model as tests performed on them turned out to be satisfactory.

The main electrical power system is discussed in details in [26].

4.6.6 Experimental electrical power (sub)system

This section describes the experimental EPS, which is one the payloads of our satellite. Even though xEPS is considered as a payload, it has to be connected to the main EPS, since xEPS will also partially supply the 3.3V bus.

Overview of functions and design drivers

xEPS consist in a digitally-controlled converter. This digital control is the most innovative part of the xEPS. xEPS will be in charge of supplying either a resistive load or the 3.3 V bus. A switch will be used to determine which output will be chosen. The role of the resistive load is to test the xEPS before the xEPS is connected to 3.3V bus in parallel with the output of the (main) EPS. A protection will be put at the input of the xEPS in order to protect it and to be able to turn it on and off. The on-board computer will be in charge of turning it on and off, upon request from the ground operator.

Baseline design

Figure 4.15 shows a block diagram of the xEPS. It consists in a flyback converter digitally controlled. The input of the xEPS is connected to the batteries and the solar cells with the unregulated bus. xEPS supplies either a load or the 3.3 V bus.

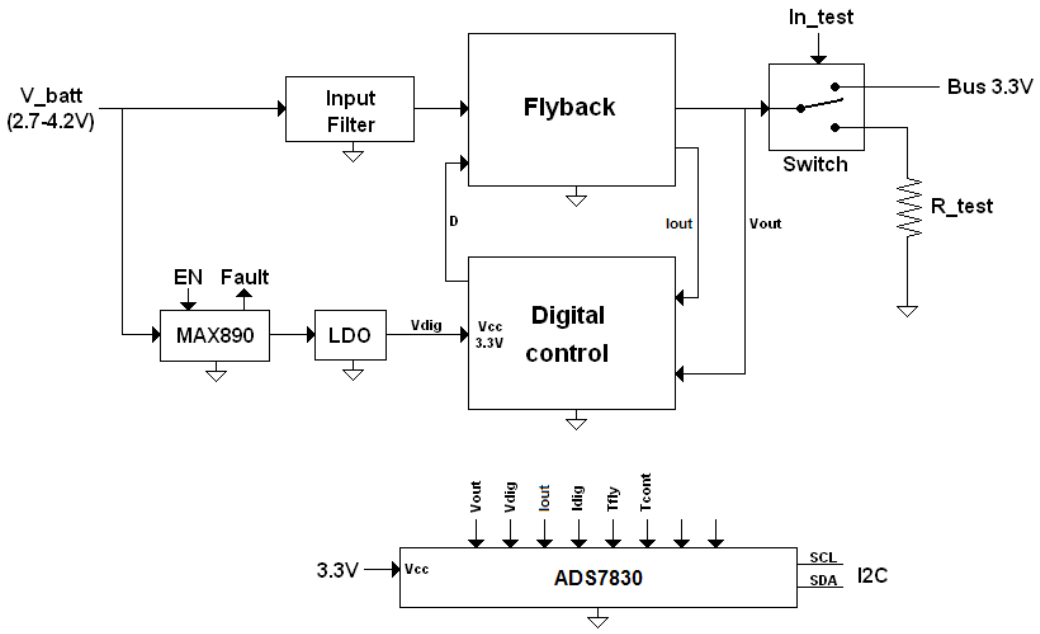


Figure 4.15: Block diagram of the experimental EPS (xEPS).

Interface with main EPS

Both power subsystems will supply the 3.3 V bus. Although some design have been proposed, the details of the interface between the main EPS and the xEPS remains to be determined.

The experimental power system is discussed in details in [13].

4.6.7 On-board computer subsystem

This section focuses on the hardware aspects of the on-board computer (OBC). Such software will be discussed in Subsection 4.6.8.

Design drivers

Beyond the evident criterion of processing power, some other concerns must be taken into account in the design of OBC:

- reliability: achieved through flight-heritage, minimization of single point of failures (e.g via redundancy), and limitation of interdependence between the various subsystems,
- power consumption,
- protection against radiations,
- mass memory.

Baseline design

Processor Amongst numerous available processors and COTS solutions, one seems particularly attractive in our situation: the flight computer FM430 from Pumpkin FM430 flight computer. Its very low power consumption, and the fact that it has already flown several times in space, make it very attractive. As stated in requirements PR2 and PR3, it will be preferred not to start from scratch, but rather from a proved and off-the-shelf solution. Therefore, the FM430 flight computer, or at least one of the MSP430 microcontrollers from Texas Instrument, become the strongest candidate for our baseline design.

Before making a final choice regarding the FM430 and the MSP430, we must check that they have sufficient processing capabilities and memory resources to process the D-STAR and the AX.25. A preliminary analysis led to a positive conclusion, (including a processor occupation of 15%) allowing the use of a centralized architecture.

As indicated before, the main drawback of a centralized architecture is that it presents a large single point of failure, i.e. the processor itself. In our case we take care of this issue, by adding a second, fully-redundant processor to the first one. Our strategy is the following: as long as one computer, called the default OBC is working properly, the other one, called the backup OBC stays silent but monitors the activity of the other one. In the event where the default OBC becomes ineffective, the backup OBC detects this situation, and immediately takes over by starting, by itself, to control the normal sequence of operations.

Indeed, at least conceptually and at the hardware level, all the corresponding I/O pins of both processors can be tied together. Unfortunately, Pumpkin's FM430

boards use a predefined bus standard, namely the CubeSat bus, which includes various signals that cannot be directly connected together between two FM430 board. Simple approaches were proposed, such as fitting a custom board between two FM430s, and interrupting the problematic signals at that level (e.g. by removing the corresponding physical pins of the connector).

Another solution was however chosen. It consists in developing a custom computer board, in place of one of the FM430s. This board would basically be a simplified replica of a FM430, but it would give more flexibility, as well as the expertise and the experience needed to avoid being limited to use commercial boards (FM430s) in our future CubeSats. This custom board, named OBC2, is chosen to be the default OBC, whereas the FM430, considered more reliable, and named OBC1, is chosen to be the backup OBC.

Peripherals Since the MSP430 offers very little internal memory (of 55 Kb at the maximum), we decided to include an external storage memory. The FM430 includes a socket and electrical interface for an SD card, but these cards are relatively power hungry, and their capacity is well beyond our needs. We therefore decided to include a small EEPROM chip, the capacity of which will be tailored to our needs as the design of the system has further progressed.

The MSP430 includes a 10-bit, 8-channel, analog to digital converter (ADC), but, in our application, we need much more than 8 analog inputs. For uniformity of design, we decided to use only external ADCs. Therefore, since the internal ADC of the MSP430s are not used at all, we plan to use its input pins as digital I/O, rather than as analog inputs.

The on-board computer is discussed in details in [25].

4.6.8 Software

Design drivers

The on-board computer (OBC) , consisting in OBC1 and OBC2, will provide the following services on-board:

- overall management,
- monitoring and control,
- telecommand and telemetry processing,
- data storage,
- data management,
- time keeping and synchronization.

These high level services can be converted into the following software functionalities:

1. Perform the initial satellite operations (antennas deployment, first activation of the other subsystems) according to a predefined sequence.
2. Perform AX.25 and D-STAR encoding and decoding.
3. Handle telecommands received on the uplink channel.
4. Perform measurements of housekeeping and science parameters aboard the satellite.
5. Store relevant measurements until they can be sent to the ground station.
6. Respond to telemetry requests by sending to ground present or past (stored) measurements.
7. Provide a time reference.
8. Perform power supply management, by enabling and disabling other subsystems in predefined conditions (e.g. a low battery voltage).
9. Perform power cycling in case of latch-up in a subsystem.
10. Manage the experimental electrical power supply (xEPS), by enabling and disabling it in predefined conditions.
11. Manage the D-STAR system, by configuring it (e.g. for Doppler compensation) according to data received via specific telecommands.
12. Keep a log of meaningful events happening aboard the satellite, and send to ground the log entries requested by specific telecommands.

13. Monitor, for the backup processor (OBC1), the activity of the default processor (OBC2) and detect when it stops functioning.

Concerning the TM/TC format, there is no requirement stating that OUFTI-1 should follow the standards applied by ESA and NASA, i.e. the CCSDS (Consultative Committee for Space Data Systems). However, this standard will be used as a starting point for the software layout. The CCSDS is based upon the 7-layer OSI model. This standardization of data exchange between layers makes the development of the communication easier between our satellite and the ground.

Baseline design

Operating system We will use an operating system (OS), in order to make the software safer and more easily maintainable. This OS should be available, off-the-shelf, and ideally with an architecture port for the MSP430.

Two general purpose OS have ports for the MSP430: i.e. Salvo and FreeRTOS. Each has pros and cons, listed in Table 4.15. After some measurements of the memory requirements (RAM and code memory), FreeRTOS was chosen.

Salvo	FreeRTOS
⊕ lightweight ⊕ written in C ⊕ very low RAM usage ⊕ implementation of queues, mutexes, and semaphores ⊕ used in other CubeSats ⊖ cooperative (thus not very safe)	⊕ lightweight ⊕ written in C ⊕ preemptive mode possible ⊕ implementation of queues, mutexes, and semaphores ⊕ open source ⊕ free (no cost) ⊕ known to be reliable

Table 4.15: Pros and cons of Salvo and FreeRTOS

Organization of on-board software In the traditional space industry, the on-board software of a satellite system is typically divided relatively to two dimensions. A similar, but simpler, organization is considered for OUFTI-1. This organization is illustrated in Fig. 4.16.

For the vertical division, two layers can be identified. The bottom one, called *driver layer*, provides drivers for the physical devices we have to interact with (e.g. the external ADCs, and some internal modules of the microcontroller). These drivers consist in functions that can be called from another layer to interact with these devices. The top layer, called *application layer*, includes all the main functionalities of the system listed above.

Horizontally, the application layer is itself divided into different modules. These modules allow one to handle all functionalities defined above, and make the software very modular.

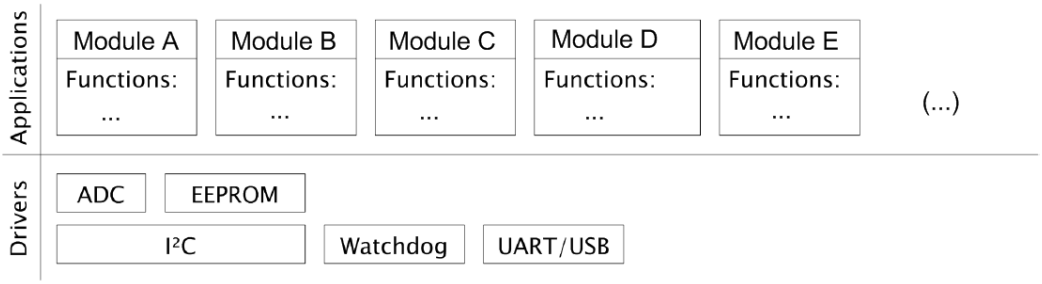


Figure 4.16: Baseline design for the organization of the on-board software.

The on-board software is discussed in details in [25, 6].

4.6.9 Communication subsystem

Three different communication systems will be used in OUFTI-1, the beacon, AX.25 for data exchanges, and D-STAR as the main payload. The three types are presented together since the requirements are the same for the three of them.

Design drivers

There are two main design drivers for the telecommunication subsystem. First, it has to comply with the low power available. Second, the possibility to perform an immediate shutdown (or an immediate transition to the safe mode) of the satellite, at any time, by any authorized ground station operator must exist.

Baseline design of the COM subsystem

Figure 4.17 shows the block diagram of the communication subsystem and table 4.16 illustrates the functionalities of the COM subsystem . The beacon consists of only a transmitter that will be on permanently. It will send information about 12 critical parameters in Morse code at 12 words per minute (WPM). The AX.25 data transmitter will send both housekeeping and "science" telemetry at 9,600 bits/sec. The beacon and the AX.25 data transceiver will both use FSK modulation. The D-STAR payload will perform ham-radio communications between two users at a rate of 4800 bits/sec, using GMSK modulation. The RF chains, both at emission and reception, have still to be studied. The major issue concerns the interconnection of the three RF signals. It will be a high priority concern of phase B. The downlink frequency will be in the 145-MHz band, while the uplink frequency will be in the 435-MHz band. Both frequency bands are in the amateur bands.

Beacon	AX.25	D-STAR
Always on	Rx always on, Tx on after TC reception	Rx on and Tx on after TC reception
12 WPM	9600 bps	4800bps
FSK	FSK	GMSK
Send 12 critical parameters in Morse code	Used for TC/TM	Payload, used to perform ham-radio communication

Table 4.16: Overview of the COM subsystem functionalities

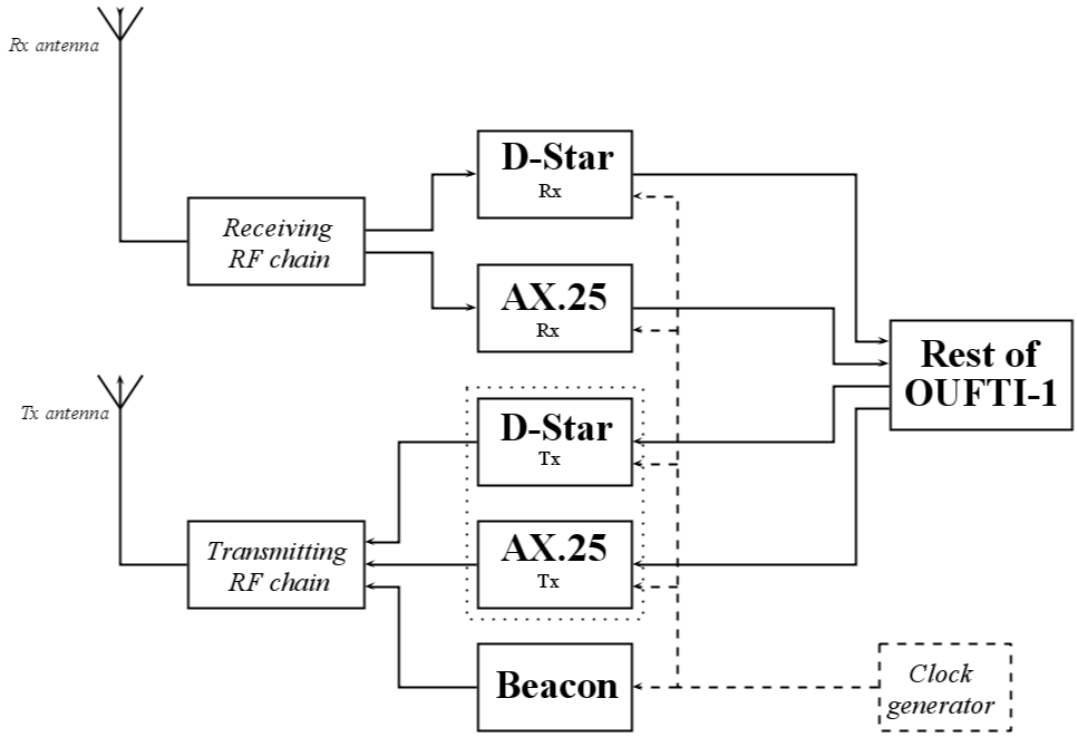


Figure 4.17: Block diagram of the communication subsystem.

Beacon

Figure 4.18 shows the block diagram of the beacon. The analog signals corresponding to the 8 critical parameters are the main input to the beacon coming directly from the sensors. The two micro-controllers perform the encoding operations, each one on 8 parameters, with 4 parameters being redundant and encoded twice. The 8 digital parameters are successively sent, in an endless loop, to the correspondent ADF7021 which output an FSK-modulated, low power RF signal. The RF FSK signals from the two ADF7021 are combined (in a way that remains to be determined) and then sent to the RF power amplifier.

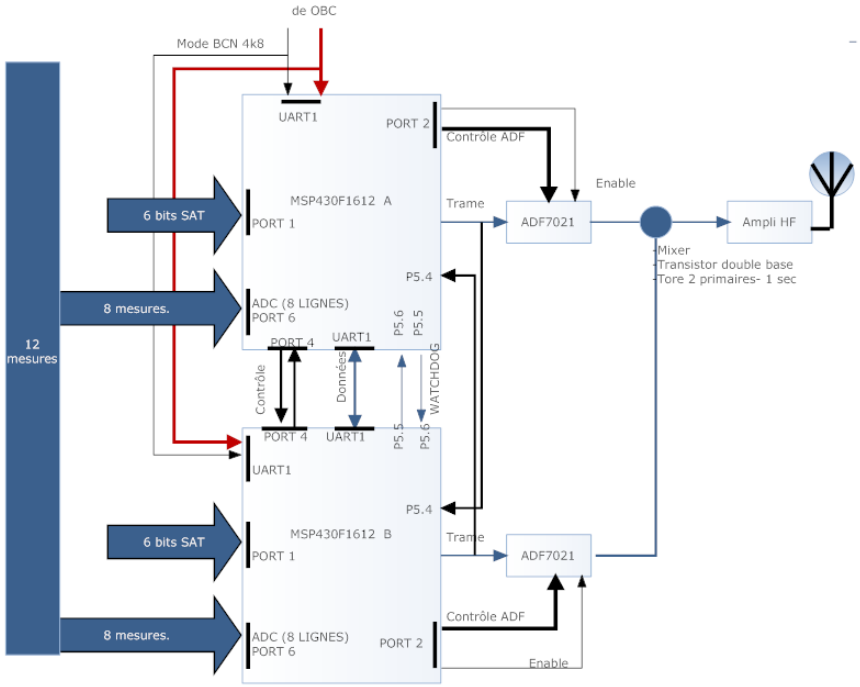


Figure 4.18: Block diagram of the beacon.

AX.25 and D-STAR transceivers

The AX.25 and D-STAR transceivers are both implemented using the same technology and are therefore presented in the same section. Figure 4.19 shows the block diagram for the two transceivers.

At reception, a signal is directed to the appropriate demodulator, according to the frequency of the signal. It is then demodulated (FSK or GMSK) to give a sequence of bits. In the micro-controller, an MSP430, either the D-STAR decoder or the AX.25 decoder interprets this sequence according to the right protocol. Data are sent to the on-board computer to be interpreted in terms of telecommands, while D-STAR communications are sent to the emitter. Information about data signals is interpreted both in terms of science data and in terms of data used to drive the ADF7021 (frequency change, and Doppler-compensation).

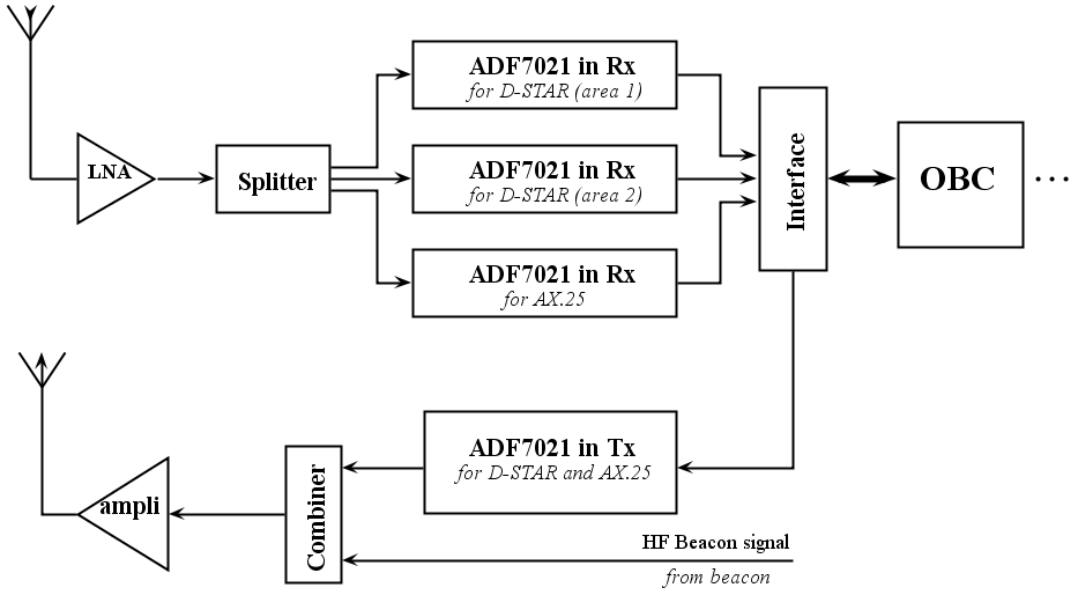


Figure 4.19: Block diagram of the AX.25 and D-STAR transceivers.

The OBC send the telemetry to the communication subsystem. The AX.25 encoder is in charge of converting these TM bits into frames according to the AX.25 protocol. The resulting digital sequence is FSK-modulated by an ADF7021. The D-STAR digital frames coming from the receiver are encoded by the D-STAR encoder, and then GMSK-modulated by an ADF7021. Both the AX.25 and the D-STAR signal coming out of the ADF7021s pass through the TX part of the RF stage connected to the antenna.

The TX part of the RF stage is still under study. Its requirements are:

- It must satisfy the link budget for a bit error rate (BER) of 10^{-5},
- The transmit power for the AX.25 and the D-STAR signal must be of 750mW,
- Transmit power for the beacon signal must be of 100mW.

At this point, we still need to settle two issues. First we must decide how many ADF7021 we need (three or four). Second, we must decide whether the COM subsystem will have its own MSP430 or whether the OBC micro-controller will also execute the COM software. These two issues will be solved once the ADF7021's behavior will be precisely known.

It is foreseen all receivers and emitters on-board the spacecraft will be getting their frequency reference from a unique, high-precision master oscillator to ease as much as possible remote frequency determination to provide accurate real-time prediction of ground transmit frequencies.

Antennas

We chose to use two quarter-wavelength monopole antennas one for the 145 MHz band and one for the 435 MHz band. The configuration of the antennas is discussed in Section 4.6.3.

The different hardware and software elements of the communication system are discussed in [15, 10] for the D-STAR routines and the modulation stage, and in [9] for the AX.25 routines.

Chapter 5

Design of ground system

5.1 System overview

OUFTI-1 will be monitored and controlled from a ground station. The ground station will also be capable of contacting other satellites operating on the ham-radio bands.

The requirements of the ground station are as follow:

- Compatibility with the ham-radio systems,
- Full compliance with AMSAT Standards,
- Ability to collaborate with other ground stations in order to reduce the time for Launch and Early Operations Phase (LEOP) and to have a wider area to control the satellite i.e. to receive its telemetries and to send telecommands,
- Full compliance with the existing D-STAR network,
- Ability to operate in the following two ways.
    1. Manual operations where basic TC are send to the satellite, when the satellite is in LEOP mode and safe mode
    2. Autonomous, pre-programed operations with little or no operator involvement

The contents of Sections 5.2 and 5.3 are discussed in greater details in [4], and the content of Section 5.4 is deeply discussed in [21].

5.1.1 Architecture of the ground system

To satisfy the above requirements, we have decided to structure the architecture of the ground station of OUFTI-1 as shown in Fig.5.1. The four key elements of the architecture of the ground system are:

- The ground station (GS),
- The mission control center (MCC),
- The D-STAR repeater,
- The D-STAR Satellite communication (SatCom) module.

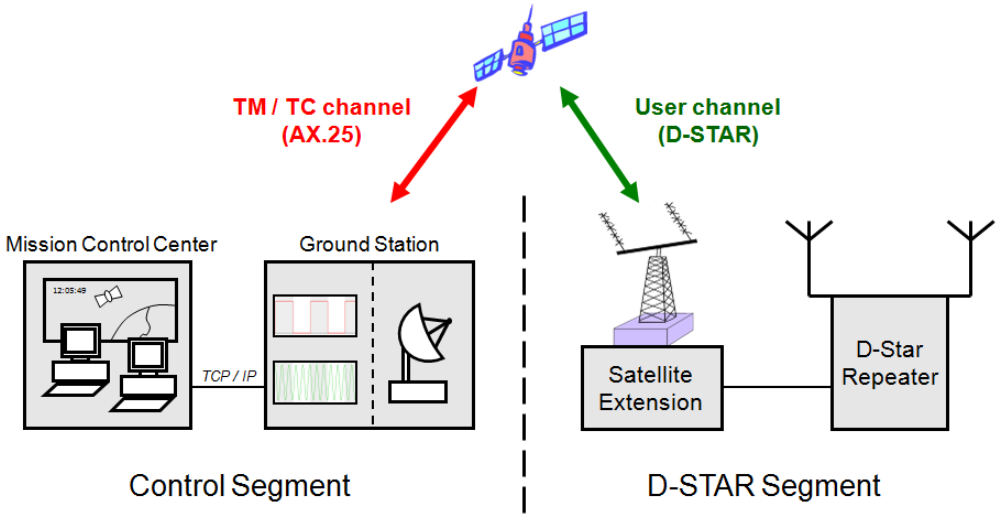


Figure 5.1: Architecture of the ground system.

The overall ground system is entirely controlled from the mission control center (MCC). More specifically the MCC controls the ground station (GS), the standard D-STAR repeater (for ground D-STAR communications), and the D-STAR SatCom module that must be added to allow D-STAR communications to be routed between OUFTI-1 and the standard D-STAR repeater. As suggested in Fig.5.1, the connection between the MCC and the other three elements is made via Internet. This means that the MCC, which could simply be a laptop computer could be placed anywhere in the world; it only requires access to the Internet.

5.2 Ground station

5.2.1 Overview

The ground station is responsible for the RF links between the satellite and the ground system. It controls the antenna rotors, the terminal node controller (TNC), and the transceivers used. It is also responsible for the link between MCC and any ground station or any ground station network, OUFTI-1 will be in contact with. It is based on commercial off-the-shelf components. Figure 5.2 shows the architecture of the ground station.

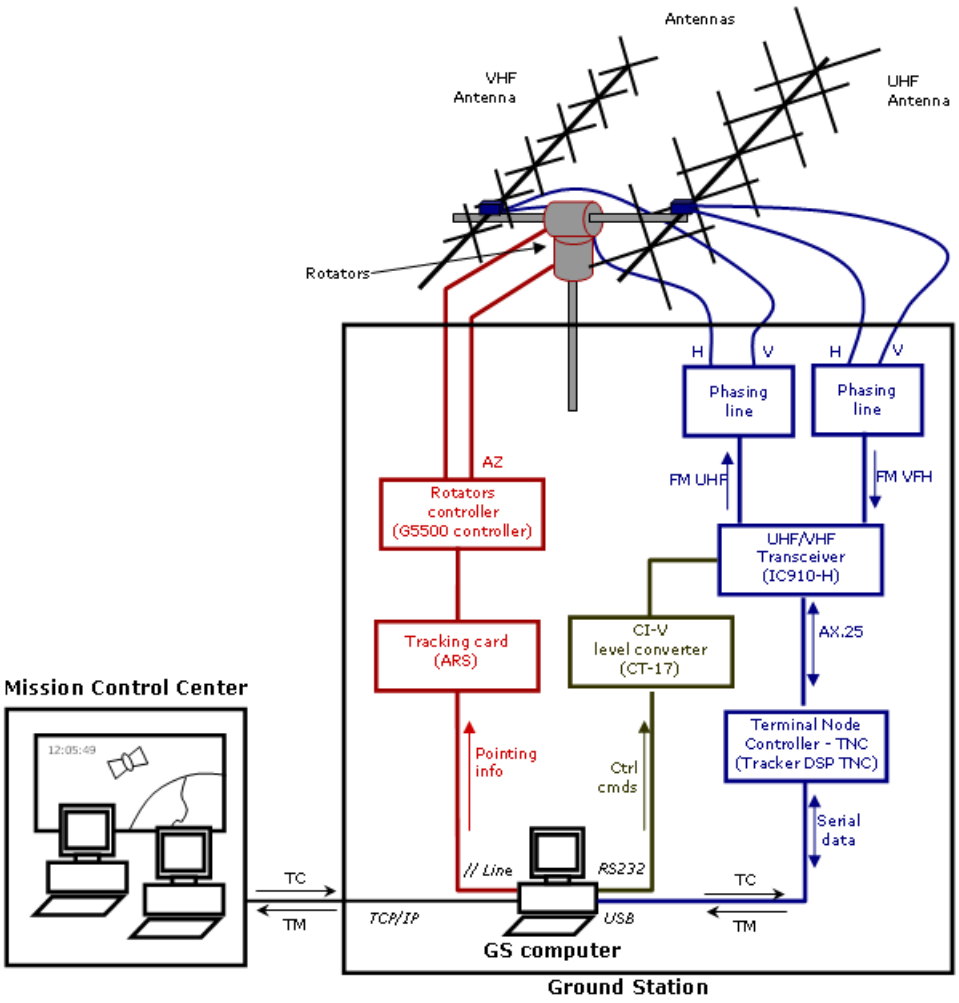


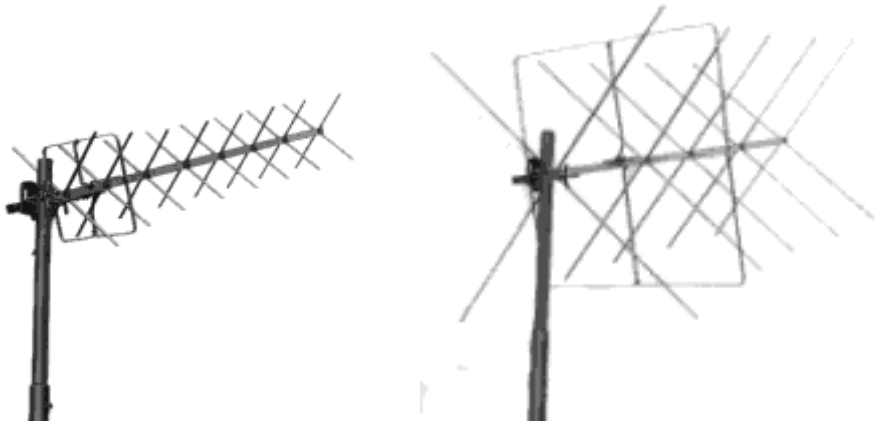
Figure 5.2: Ground station architecture.

Two ground stations will be built. The main one will be built in two parts at the University of Liege. The AX.25 part will be located in the Montefiore Institute and the D-STAR part will be located in the "Poste Central de Commande" (PCC). In the PCC, the radio transceivers will be installed on the top floor in the attic of the building, just below the antenna, it allows us to minimize the length of our RF cables. A backup ground station will be built at the EuroSpace Center, Redu, Belgium. The detailed description and plans will be the subject of a separate document.

5.2.2 Primary installation

For educational reasons, we decided to place the primary installation of the ground station in the department of Electrical Engineering and Computer Science of the

University of Liège, the Montefiore Institute. 2-m and 70-cm X-Quad antennas (Fig. 5.3) are mounted on Az/El rotor (Fig. 5.4) , and transceivers (Fig. 5.5) are placed in the students' office. It allows students to test their work in real situations, and to gain experience working on real amateur satellites. In the next sections, we will only discuss the main installation at the University of Liege.



(a) 70-cm band (b) 2-m band

Figure 5.3: X-Quad antennas to be used for the satellite coordination

Figure 5.4: Yaesu G-5500 antenna rotator.

Figure 5.5: On the left, the IC-910H used to communicate with satellites, and, on the right, the IC-E2820 used for D-STAR communications.

5.2.3 Antennas and antenna rotors

The final choice of the antennas and the antenna rotors has still to be made. It will be based upon a trade-off between the gain, the polarization, and the size of the different antennas. The following types of antennas will be considered : crossed Yagi, the helix, and the parabolic dish. Various types of rotors will be considered.

5.2.4 Transceiver

The transceiver has to meet with the following requirements:

- It must be adapted to the amateur radio frequency bands: 145 MHz and 435 MHz. The 1.2 Ghz band is an asset.
- It must allow reception on one of the frequency bands (145 MHz), and transmission on the other (435 MHz).
- It must be able to compensate the Doppler effect, i.e. one must be able to modify the synthesizing in step of 1 kHz.
- It must work with as many modulations used by amateur satellites as possible: FM, AM, CW, FSK, AFSK, 1200 bauds and 9600 bauds packet...
- The possibility of being controlled by a computer is an asset.

Both the ICOM IC-910H and the Kenwood TS-2000 fit the above requirements. Due to its simplicity of use, the IC-910H was chosen.

5.2.5 Ground station network

Three different possibilities of coverage were envisioned.

- Liège: There would be only one ground station for OUFTI-1. The major drawback is its small coverage.
- Ham operator shack: This is a ground station operated by a ham-radio somewhere in the world. It should be an interesting way of recovering OUFTI-1's beacon telemetry, which would the forwarded to Liège. However, due to the uncertainty abort the hardware used (manual pointing...), it is believed that this should not be considered to bring the telemetry back to Liege.
- Ground station network (GSN): It consists of a number of interconnected ground stations. It allows a user to operate remotely a satellite. The appropriate ground station network software should be used, and operation time shall be given to external satellite operators that are members of this network.

In conclusion, the baseline for the safe recovery of telemetry and LEOP operations is to be part of a GSN. This will increase dramatically the coverage zone and time. Ham-radio operators interesting in satellite communications will further be included for the recovery of the beacon data and possibly of the downlink of telemetry.

5.3 Mission control center

OUFTI-1's mission control center (MCC) will be designed to perform the following operations:

- Preparation and transmission of telecommands, both manually and automatically,
- Reception and processing of telemetry , both manually and automatically,
- Archiving and retrieval of data,
- Displays of data,
- Real-time updates.

The MCC will allow operators to command and control the satellite from terminals via the operation server.

5.4 D-STAR repeater system

The main payload aboard OUFTI-1 is the D-STAR system. The D-STAR telecommunication system has already proved itself to be highly beneficial for ham radio routine operation, and has also demonstrated many efficient dedicated crisis management supports to various safety organizations worldwide. To get a practical, hands-on understanding of the D-STAR protocol and D-STAR communications, a D-STAR repeater was installed at the University of Liège. Figure 5.6 shows the tower and the antenna of the repeater, it consists of a bi-band antenna (145.625 MHz, and 439.525 MHz) mounted on a 12-m mast. The antenna is connected to the VHF and UHF modules of the D-STAR repeater (Fig. 5.7) through bandpass filter and duplexer. The controller manages the repeater and the gateway links the repeater to the worldwide D-STAR network. Note that the D-STAR repeater is an independent part of the OUFTI-1 system. It is also a service offered to the ham-radio community.

Figure 5.6: D-STAR repeater's antenna.

Figure 5.7: D-STAR ground repeater.

Figure 5.8: Direct communication between two users that are close enough.

In regular terrestrial D-STAR, there are three possibilities of establishing a communication between two users. The simplest one is a direct communication from a user to another one (Fig. 5.8). Of course, users must be close enough and in good visibility of each other to establish the link.

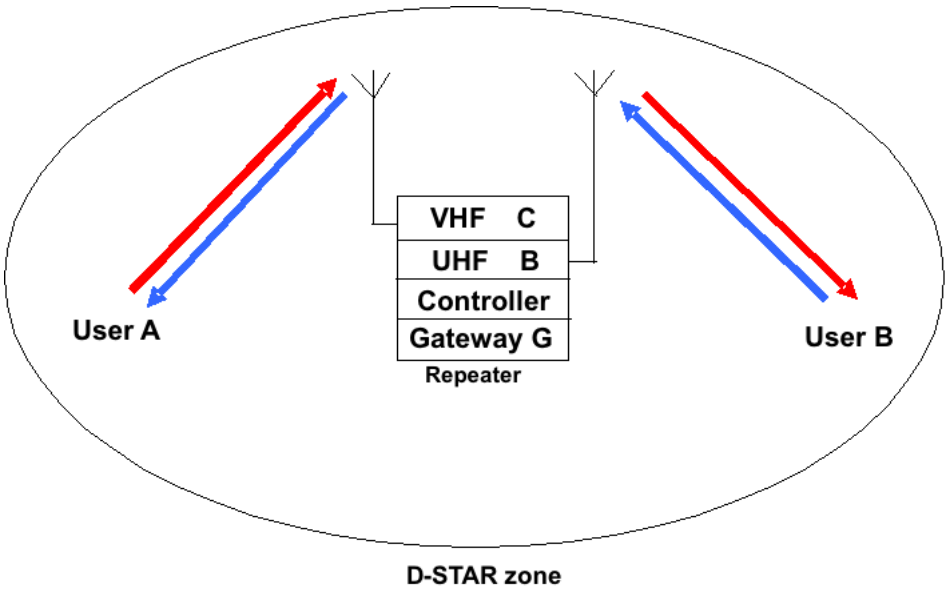


Figure 5.9: Local communication through repeater.

When they are too far away for a direct path to be set up, users can use a local repeater. In the case they are both in the footprint of a single local repeater, i.e. the geographic zone inside which any user can reach the repeater, the signals sent by any of these users will transit by the local repeater. This is shown in Fig. 5.9.

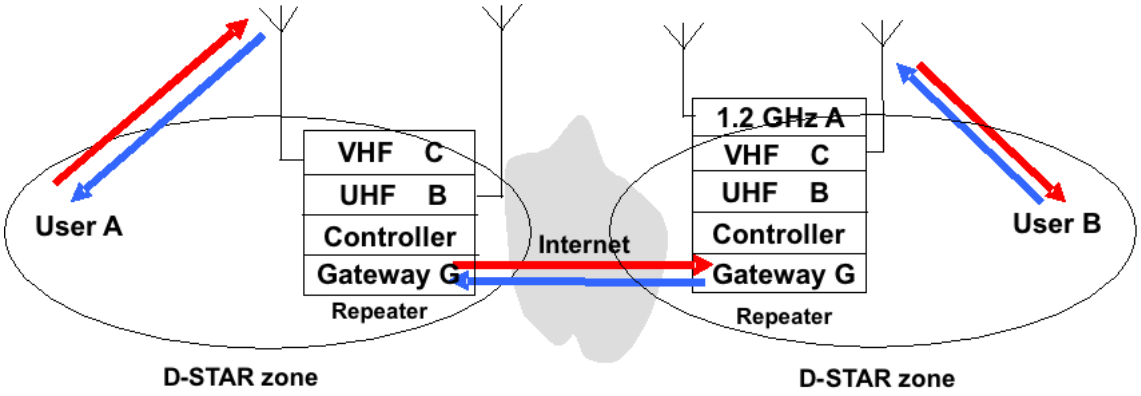


Figure 5.10: Worldwide communication through gateway.

If users are in the footprint of different repeaters being each connected to the Internet by a gateway, they will be able to communicate, as depicted in Fig.5.10. More details about the use of the D-STAR network can be found in [20].

5.5 Module for communications via OUFTI-1 via D-STAR

Users will be able to communicate using the D-STAR protocol through OUFTI-1. The main requirement is that OUFTI-1 shall be entirely compatible with the existing D-STAR network. To be fully compliant with the D-STAR network, a module has to be added between the D-STAR repeater and the satellite. It will perform the RF link between the repeater's controller and the satellite's D-STAR payload. This module will consist in a tracking system with its proper antennas and rotors, and in a D-STAR module linked to the rest of the D-STAR repeater. The different types of D-STAR communications via the satellite and through the D-STAR repeater will be described in a later document.

Chapter 6

Conclusion

This document presents the main results achieved during the phase A study of the OUFTI-1 project. The work was performed during the academic year 2008-2009. It shows that significant progress was made by all the members of the team in the understanding of a space system design, and the issues related to it.

The mission and payload objectives were defined, leading to mission and payloads requirements. The mission was analyzed in terms of launch vehicle, orbit, space environment, and on-board operations.

A preliminary design of the space system was performed. Various trade-off were performed. These led to system block diagrams, system budgets, and state modes. A first design of each of the different subsystems of the CubeSat was presented.

A preliminary design of the ground system was also performed. It consists of a D-STAR part (constituted of a classical D-STAR repeater and of an additional module dedicated to satellite D-STAR communications) and of a control part (constituted of the mission control center and the ground station).

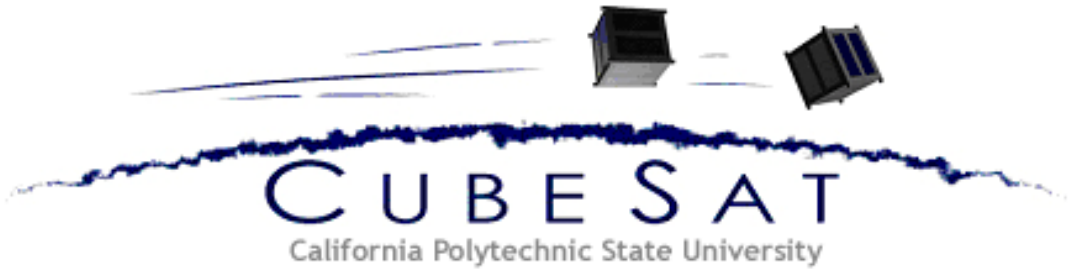
The next step in the project is to write a complete interface control document (ICD) and to carry out the work that will lead to the phase B document.

Appendix A: Standard CubeSat

	Document Classification
X	Public Domain
	ITAR Controlled
	Internal Only

CubeSat Design Specification

(CDS)



CHANGE HISTORY LOG

Effective Date	Revision	Author	Description of Changes
N/A	8	Simon Lee	N/A
5/26/05	8.1	Amy Hutputanasin	Formatting updated.
5/15/06	9	Armen Toorian	Information and presentation revised.
8/2/07	10	Wenschel Lan	Information updated.
10/02/08	11	Riki Munakata	Format, Design specification and Mk.III P-POD compatibility update.

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List of Acronyms

CAC	CubeSat Acceptance Checklist
Cal Poly	California Polytechnic State University, San Luis Obispo
CDS	CubeSat Design Specification
cm	Centimeters
FCC	Federal Communication Commission
GSE	Ground Support Equipment
HPTM	High Power Transmit Mode
IARU	Internation Amateur Radio Union
in	Inches
kg	Kilograms
LPTM	Low Power Transmit Mode
LV	Launch Vehicle
mm	Millimeters
MTP	Mission Test Plan
P/N	Part Number
P-POD	Poly Picosatellite Orbital Deployer
RBF	Remove Before Flight
Rev.	Revision
SSDL	Space Systems Development Lab

1. Introduction

1.1 Overview

Started in 1999, the CubeSat Project began as a collaborative effort between California Polytechnic State University (Cal Poly), San Luis Obispo, and Stanford University's Space Systems Development Laboratory (SSDL). The purpose of the project is to provide a standard for design of picosatellites to reduce cost and development time, increase accessibility to space, and sustain frequent launches. Presently, the CubeSat Project is an international collaboration of over 100 universities, high schools, and private firms developing picosatellites containing scientific, private, and government payloads. A CubeSat is a 10 cm cube with a mass of up to 1 kg. Developers benefit from the sharing of information within the community. If you are planning to start a CubeSat project, please contact Cal Poly. Visit the CubeSat website at http://cubesat.calpoly.edu for more information.

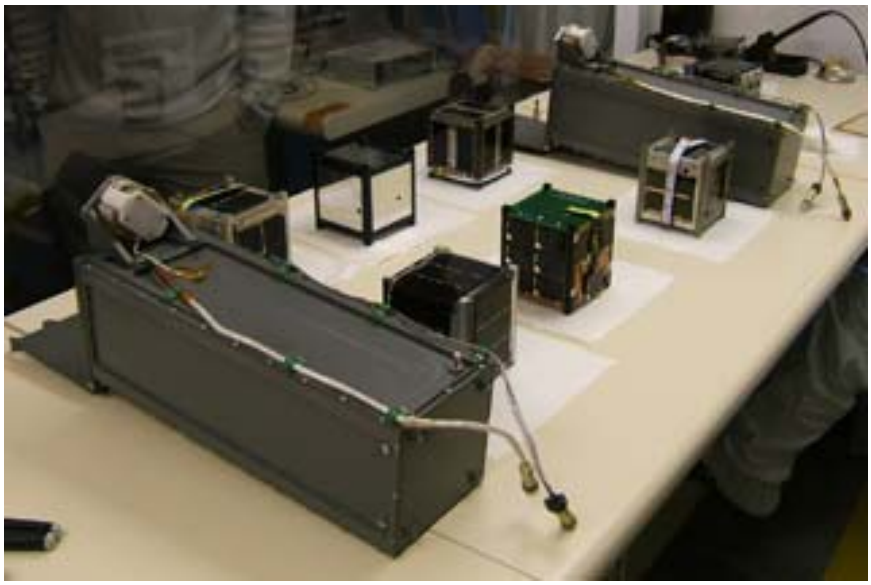


Figure 1: Six CubeSats and their deployment systems.

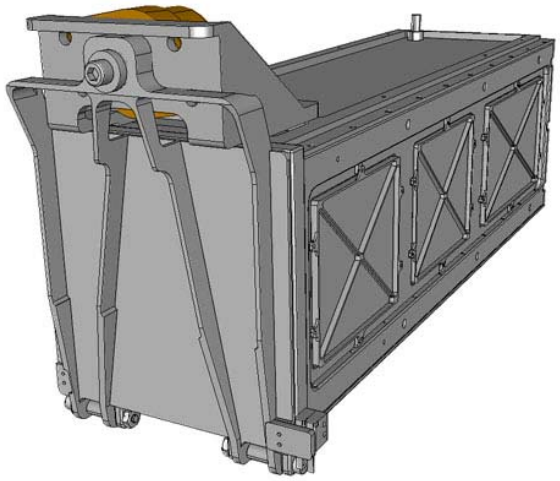
1.2 Purpose

The primary mission of the CubeSat Program is to provide access to space for small payloads. The primary responsibility of Cal Poly, as a launch coordinator and the developer of the Poly Picosatellite Orbital Deployer (P-POD), is to ensure the safety of the CubeSat and protect the launch vehicle (LV), primary payload, and other CubeSats. CubeSat developers should play an active role in ensuring the safety and success of CubeSat missions by implementing good engineering practice, testing, and verification of their systems. Failures of CubeSats, the P-POD, or interface hardware can damage the LV or a primary payload and put the entire CubeSat Program in jeopardy. As part of the CubeSat Community, all participants have an obligation to ensure safe operation of their systems and to meet the design and minimum testing requirements outlined in this document. Requirements in this document maybe superseded by launch provider requirements.

2. Poly Picosatellite Orbital Deployer

2.1 Interface

The Poly Picosatellite Orbital Deployer (P-POD) is Cal Poly's standardized CubeSat deployment system. It is capable of carrying three standard CubeSats and serves as the interface between the CubeSats and LV. The P-POD is a rectangular box with a door and a spring mechanism. The P-POD is made up of anodized aluminum. CubeSats slide along a series of rails during ejection into orbit. CubeSats must be compatible with the P-POD to ensure safety and success of the mission, by meeting the requirements outlined in this document. The P-POD is backward compatible, and any CubeSat developed within the design specification of CDS rev. 9 and later, will not have compatibility issues. Developers are encouraged to design to the most current CDS to take full advantage of the P-POD features. Additional unforeseen compatibility issues will be addressed as they arise.



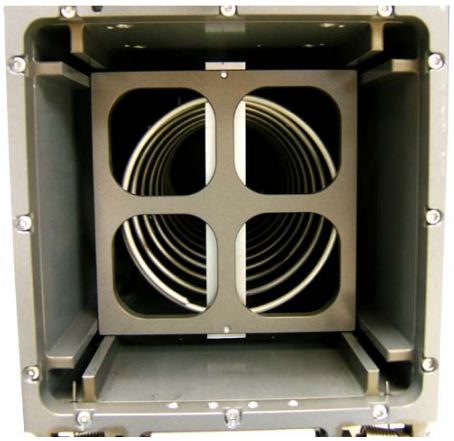


Figure2a and 2b: Poly Picosatellite Orbital Deployer (P-POD) and cross section

2.2 General Responsibilities

2.2.1 CubeSats shall not present any danger to neighboring CubeSats in the P-POD, the LV, or primary payloads:

2.2.1.1 All parts shall remain attached to the CubeSats during launch, ejection and operation. No additional space debris shall be created.

2.2.1.2 CubeSats shall be designed to minimize jamming in the P-POD.

2.2.1.3 Absolutely no pyrotechnics are allowed inside the CubeSat.

2.2.2 NASA approved materials should be used whenever possible to prevent contamination of other spacecraft during integration, testing, and launch.

2.2.3 The newest revision of the CubeSat Design Specification is always the official version

2.2.3.1 Developers are responsible for being aware of changes.

2.2.3.2 Changes will be made as infrequently as possible bearing launch provider requirements or widespread safety concerns within the community.

2.2.3.3 Cal Poly will send an update to the CubeSat mailing list upon any changes to the specification.

2.2.3.4 CubeSats using an older version of the specification *may* be exempt from implementing changes to the specification on a case-by-case basis.

Cal Poly holds final approval of all CubeSat designs. Any deviations from the specification must be approved by Cal Poly launch personnel. Any CubeSat may be pulled from the launch if it is deemed a safety hazard by Cal Poly launch personnel.

3. CubeSat Specification

3.1 Dimensional and Mass Requirements

CubeSats are cube shaped picosatellites with a nominal length of 100 mm per side. Dimensions and features are outlined in the CubeSat Specification Drawing (Appendix A). General features of all CubeSats are:

3.1.1 Each single CubeSat may not exceed 1 kg mass.

3.1.2 Center of mass must be within 2 cm of its geometric center.

3.1.3 Double and triple configurations are possible. In these cases masses 2 kg or 3 kg respectively are allowable. Only the dimensions in the Z axis change (227 mm for doubles and 340.5 mm for triples). X and Y dimensions remain the same.

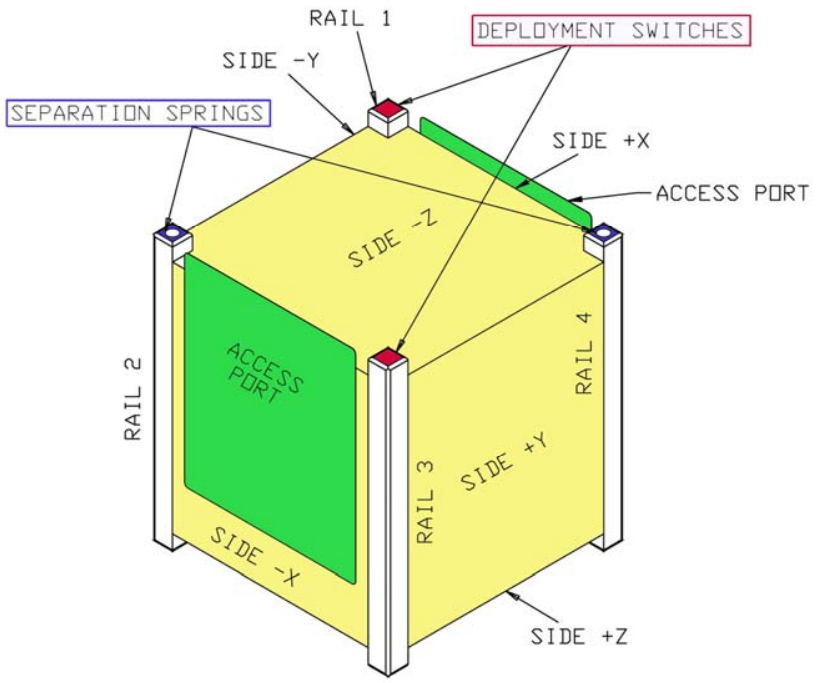


Figure 3: CubeSat isometric drawing

3.2 Structural Requirements

The structure of the CubeSat must be strong enough to survive maximum loading defined in the testing requirements and cumulative loading of all required tests and launch. The CubeSat structure **must** be compatible with the P-POD.

3.2.1 The CubeSat shall be 100.0±0.1 mm wide (X and Y) and 113.5+0.1 mm tall (Z).

3.2.2 Rails must be smooth and edges must be rounded to a minimum radius of 1 mm.

3.2.3 No external components other then the rails shall be in contact with the internals of the P-POD.

3.2.4 Each rail shall be a minimum of 8.5 mm wide.

3.2.5 At least 75% (85.125 mm of a possible 113.5mm) of the rail must be in contact with the P-POD rails. 25% of the rails may be recessed and **NO** part of the rails may exceed the specification.

3.2.6 All rails must be hard anodized to prevent cold-welding, reduce wear, and provide electrical isolation between the CubeSats and the P-POD.

3.2.7 Separation springs must be included at designated contact points (Appendix A). Spring plungers are highly recommended (McMaster-Carr P/N: 84985A76 available at http://www.mcmaster.com). A custom separation system may be used, but must be approved by Cal Poly launch personnel.

3.2.8 The use of Aluminum 7075 or 6061-T6 is suggested for the main structure. If other materials are used, the thermal expansion must be similar to that of Aluminum 7075-T73 (P-POD material) and approved by Cal Poly launch personnel.

3.2.9 Deployables must be constrained by the CubeSat. The P-POD rails and walls are **NOT** to be used to constrain delpolyables.

3.2.10 The CubeSat shall meet all other requirements noted in Appendix A.

Figure 4: Spring plunger

3.3 Electrical Requirements

Electronic systems must be designed with the following safety features.

3.3.1 No electronics shall be active during launch to prevent any electrical or RF interference with the launch vehicle and primary payloads. CubeSats with rechargeable batteries shall be fully deactivated during launch or launch with discharged batteries.

3.3.2 One deployment switch is required (two are recommended) for each CubeSat. The deployment switch should be located at designated points (Appendix A).

3.3.3 Deployment switch shall be compatible with +Z contact point(s).

3.3.4 Developers who wish to perform testing and battery charging after integration must provide ground support equipment (GSE) that connects to the CubeSat through designated access ports (Appendix A).

3.3.5 A remove before flight (RBF) pin is required to deactivate the CubeSats during integration outside the P-POD. The pin will be removed once the CubeSats are integrated into the P-POD. RBF pins must fit within the designated data ports (Appendix A). RBF pins should not protrude more than 6.5 mm from the rails when fully inserted.

3.4 Operational Requirements

CubeSats must meet certain requirements pertaining to integration and operation to meet legal obligations and ensure safety of other CubeSats.

3.4.1 CubeSats with rechargeable batteries shall have the capability to receive a transmitter shutdown command, as per Federal Communications Commission (FCC) regulation.

3.4.2 To allow adequate separation of CubeSats, antennas may be deployed 15 minutes **after** ejection from the P-POD (as detected by CubeSat deployment switches). Larger deployables such as booms and solar panels may be deployed 30 minutes **after** ejection from the P-POD.

3.4.3 CubeSats may enter low power transmit mode (LPTM) 15 minutes **after** ejection from the P-POD. LPTM is defined as short, periodic beacons from the CubeSat.

3.4.4 CubeSats may activate all primary transmitters, or enter high power transmit mode (HPTM) 30 minutes **after** ejection from the P-POD.

3.4.5 Operators shall obtain and provide documentation of proper licenses for use of frequencies. For amateur frequency use, this requires proof of frequency coordination by the International Amateur Radio Union (IARU). Applications can be found at www.iaru.org.

3.4.6 Developers shall obtain and provide documentation of approval of an orbital debris mitigation plan from the FCC. Contact Robert Nelson at rnelson@fcc.org

3.4.7 Cal Poly will conduct a minimum of one fit check in which developer hardware will be inspected and integrated into the P-POD. A final fit check will be conducted prior to launch. The CubeSat Acceptance Checklist (CAC) will be used to verify compliance of the specification (Appendix B).

3.4.8 Periodic teleconferences, videoconferences, and progress reports may be required.

4. Testing Requirements

Testing must be performed to meet all launch provider requirements as well as any additional testing requirements deemed necessary to ensure the safety of the CubeSats and the P-POD. All flight hardware will undergo qualification and acceptance testing. The P-PODs will be tested in a similar fashion to ensure the safety and workmanship before integration with the CubeSats. At the very minimum, all CubeSats will undergo the following tests.

4.1.1 Random vibration testing at a level higher than the published launch vehicle envelope outlined in the Mission Test Plan (MTP).

4.1.2 Thermal vacuum bakeout to ensure proper outgassing of components. The test cycle and duration will be outlined in the MTP.

4.1.3 Visual inspection of the CubeSat and measurement of critical areas as per the CAC (Appendix B).

4.2 Qualification

All CubeSats must survive qualification testing as outlined in the MTP for their specific launch. Qualification testing will be performed at developer facilities. In some circumstances, Cal Poly can assist developers in finding testing facilities or provide testing for the developers. A fee may be associated with any tests performed by Cal Poly. CubeSats must **NOT** be disassembled or modified after qualification testing. **Additional testing will be required if modifications or changes are made to the CubeSats after qualification testing.**

4.3 Acceptance

After delivery and integration of the CubeSats, additional testing will be performed with the integrated system. This test assures proper integration of the CubeSats into the P-POD. Additionally, any unknown, harmful interactions between CubeSats may be discovered during acceptance testing. Cal Poly will coordinate and perform acceptance testing. No additional cost is associated with acceptance testing. After acceptance testing, developers may perform diagnostics through the designated P-POD diagnostic ports, and visual inspection of the system will be performed by Cal Poly launch personnel. The P-PODs **WILL NOT** be deintegrated at this point. If a CubeSat failure is discovered, a decision to deintegrate the P-POD will be made by the developers, in that P-POD, and Cal Poly based on safety concerns. The developer is responsible for any additional testing required due to corrective modifications to deintegrated P-PODs and CubeSats.

5. Contacts

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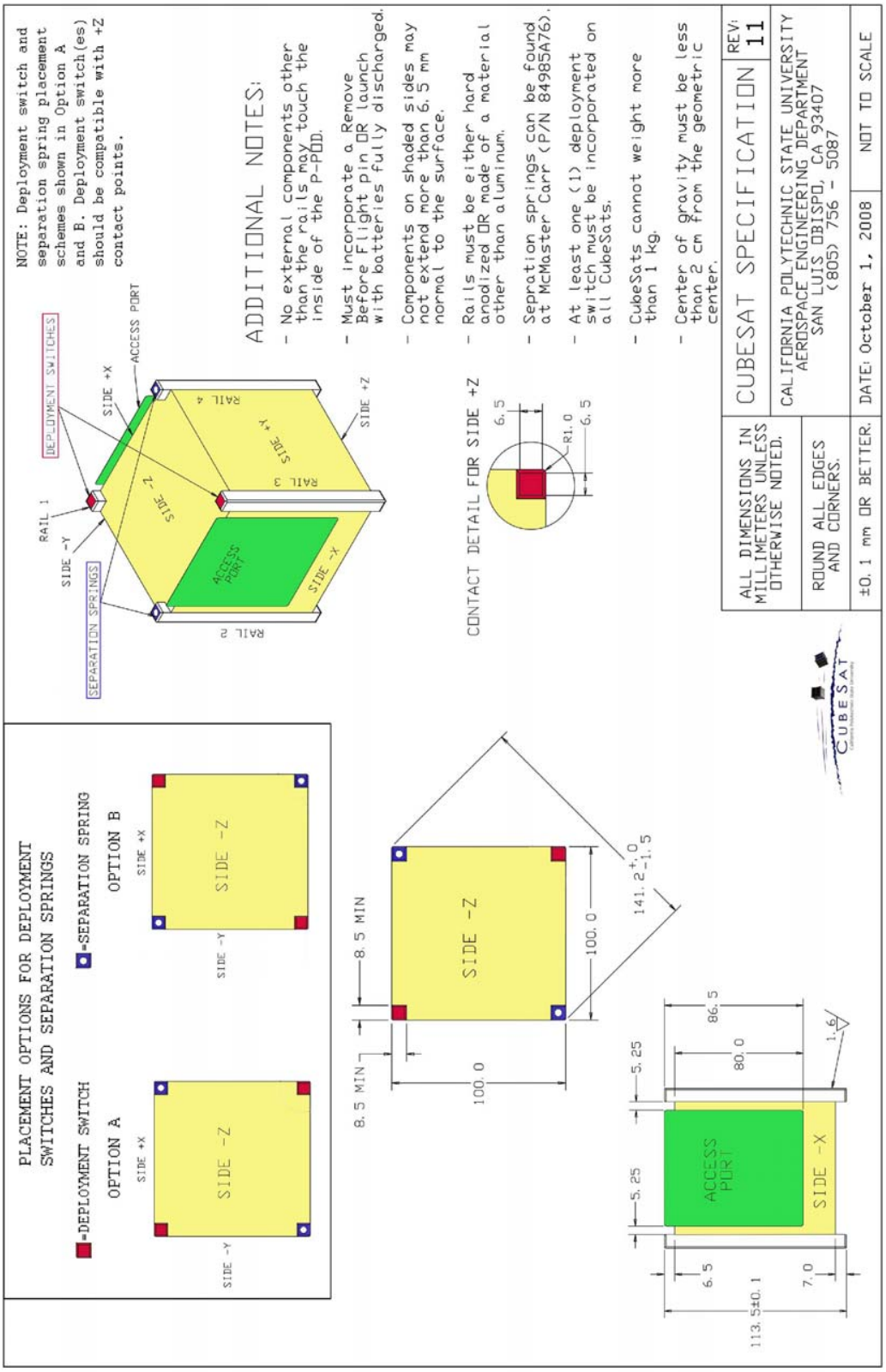
Appendix A: CubeSat Design Specification Drawing

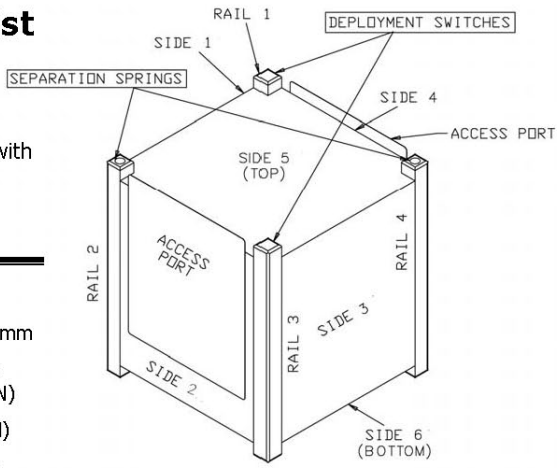


Appendix B: CubeSat Acceptance Checklist

CubeSat Acceptance Checklist

Revision Date: August 1, 2007
Author: Wenschel Lan

This document is intended to be used concurrently with the CubeSat Integration Procedure (CIP).



List Item	Actual	Required
Mass	_____	$\leq 1000g$
Remove Before Flight	_____	Protrudes $\leq$ 6.5 mm
Spring Plungers	_____	Option (A/B) Functional (Y/N)
Rails	_____	Anodized (Y/N)
Deployment Switches	_____	Option (A/B) Functional (Y/N)
Width [x-y], Top		
Side 1	_____	$100.0 \pm 0.1mm$
Side 2	_____	$100.0 \pm 0.1mm$
Side 3	_____	$100.0 \pm 0.1mm$
Side 4	_____	$100.0 \pm 0.1mm$
Width [x-y], Middle		
Side 1	_____	$100.0 \pm 0.1mm$
Side 2	_____	$100.0 \pm 0.1mm$
Side 3	_____	$100.0 \pm 0.1mm$
Side 4	_____	$100.0 \pm 0.1mm$
Width [x-y], Bottom		
Side 1	_____	$100.0 \pm 0.1mm$
Side 2	_____	$100.0 \pm 0.1mm$
Side 3	_____	$100.0 \pm 0.1mm$
Side 4	_____	$100.0 \pm 0.1mm$

Authorized By:	Testing Info:
IT #1: _____	Date: _____
IT #2: _____	Passed: **Y / N**

List Item	Actual	Required
Height [z]		
Rail 1	_____	$113.5 \pm 0.1mm$
Rail 2	_____	$113.5 \pm 0.1mm$
Rail 3	_____	$113.5 \pm 0.1mm$
Rail 4	_____	$113.5 \pm 0.1mm$
Diagonal [x-y]		
Top 1&3	_____	$141.2^{+0}_{-1.5}mm$
Top 2&4	_____	$141.2^{+0}_{-1.5}mm$
Bottom 1&3	_____	$141.2^{+0}_{-1.5}mm$
Bottom 2&4	_____	$141.2^{+0}_{-1.5}mm$
Protrusions		
Side 1	_____	$6.5 + 0.0mm$
Side 2	_____	$6.5 + 0.0mm$
Side 3	_____	$6.5 + 0.0mm$
Side 4	_____	$6.5 + 0.0mm$
Side 5	_____	$6.5 + 0.0mm$
Side 6	_____	$6.5 + 0.0mm$

Bibliography

[1] *Ground systems and operations – Telemetry and telecommand packet utilization.* Space Engineering, ECSS-E-70-41A. Noordwijk: ESA Publications Division, January 2003.

[2] Arianespace. Vega overview. Technical report, 2009.

[3] V. Beukelaers. From mission analysis to space flight simulation of the oufti-1 nano-satellite. Technical report, ULg, Liege, Belgique, May 2009.

[4] L. Chiarello. Design and implementation of the control and monitoring system for the ground station of nanosatellite oufti-1 of university of liege. Master's thesis, University of Liege (Montefiore), 2009.

[5] M. Van Droogenbroeck. *Principes des télécommunications analogiques et numériques, version 5.18.* University of Liège, June 2005.

[6] N. Evrard. Conception et realisation du systeme de mesures du nanosatellite oufti-1. Technical report, Institut Gramme, Jun 2009.

[7] S. Galli. Mission Design for the CubeSat OUFTI-1. Master's thesis, University of Liège, 2008.

[8] S. Hannay. Attitude determination and control system of student nanosatellite oufti-1 of university of liege. Technical report, ULg, May 2009.

[9] J. Hardy. Implementation du protocole ax.25 a bord du nanosatellite oufti-1. Technical report, Institut Gramme, Jun 2009.

[10] R. Henrard. Design et implementation du systeme de telecommunications du nanosatellite oufti-1. Technical report, Institut Superieur Industriel Liegeois, May 2009.

[11] L. Jacques. Thermal design of the oufti-1 nanosatellite. Technical report, ULg, May 2009.

[12] R. Krpoun, M. Noca, and N. Scheidegger. *Mission, Space and Ground System Description – Phase A.* École Polytechnique Fédérale de Lausanne (EPFL), June 2006. SwissCube Projet.

[13] P. Ledent. Design and implementation of on-board digitaly controled electrical power supply of student nanosatellite oufti-1 of university of liege. Technical report, ULg, May 2009.

[14] P. Ledent. Rapport de stage: Architecture et design de l'alimentation electrique d'oufti-1. Technical report, University of Liege, Jun 2009.

[15] F. Mahy. Design and implementation of on-board telecommunication system of student nanosatellite oufti-1 of university of liege. Technical report, ULg, May 2009.

[16] M. Merri. *ECSS E-70-41: Telemetry & Telecommand Packet Utilisation*. European Space Agency, 2001. Presentation at OMG meeting, Paris.

[17] M. Merri, B. Melton, S. Valera, and A. Parkes. *The ECSS Packet Utilization Standard and its support tool*. European Space Agency and Nova Space Associates Ltd, 2002. Presentation at SpaceOps, Houston.

[18] G. Pierlot. Oufti-1 : structural integrity, system configuration and vibration testing. Master's thesis, University of Liege (LTAS), 2009.

[19] G. Pierlot. Oufti-1's integration procedure. Technical report, ULg, Mar 2009.

[20] J. Pisane. Master's thesis.

[21] J. Pisane. Design and implementation of the terrestrial and space telecommunication elements of the student nanosatellite of the University of Liège. Master's thesis, University of Liège, 2008.

[22] J. Pisane. Oufti-1 frequency coordination. Technical report, ULg, Aug 2009.

[23] The CubeSat Program Cal Poly. Cubesat design specification. Technical report, 2008. `http://cubesat.atl.calpoly.edu/media/CDS_rev11.pdf`.

[24] SwissCube. Swisscube website. `http://swisscube.epfl.ch/`.

[25] D. Teney. Design and implementation of on-board processor and software of student nanosatellite oufti-1 of university of liege. Technical report, ULg, May 2009.

[26] P. Thirion. Design and implementation of on-board electrical power supply of student nanosatellite oufti-1 of university of liege. Technical report, ULg, May 2009.

[27] M. Van Droogenbroeck. *Analyse et conception des systemes de telecommunications*, 2005. Course note.

[28] J. Wertz. Conception et realisation du systeme de deploiement des antennes du nanosatellite oufti-1. Technical report, Institut Gramme, Jun 2009.