Phase B/C

Delta-PDR

Project, Mission, Space and Ground System Overview

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PROJECT SPONSORS

Sponsor type	Company/Institution
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TABLE OF CONTENT

RECORD OF REVISIONS

ISS/REV	Date	Modifications	Created/modified by
1/0	13/08/07	Initial Issue	M. Noca
1/1	22/08/07	Integrated subsystems updates	M. Noca
1/2	23/08/07	Integrated English and format review - Release	M. Noca

I INTRODUCTION

This document summarizes the work performed and the results achieved during the SwissCube Phase B study. Phase B overlapped two semesters, from July 06 till August 07. A Phase B design review (PDR) was held on March 6-7, 2007 and concluded that although some subsystems had satisfied the requirements for Phase B at the end of the first semester, some had to continue developments during the second semester. This document thus provides a summary of the entire Phase B.

About 50 students (2 batches of 25) from 10 different laboratories at the EPFL, 1 laboratory at the University of Neuchâtel and 5 laboratories from the Haute Ecole Specialisée de Suisse Occidentale (HES-ARC, EIVD, HEVs, HES-Fr) participated in this work. The Project's organization is described in the SwissCube Project Management Plan. The students worked within the frame of semester and master projects, meaning from ~4-12 hours per week up to 40 hours per week depending on the student's year for about 4 months per semester. The students have various backgrounds including Electrical, Software, Mechanical, Material, Communication and Micro-mechanical Engineering.

This document consists of an overview of the project engineering, mission, space and ground systems. Topics include:

- Project and mission objectives,
- Requirement analysis,
- Assembly, integration and test plan,
- Mission and operations,
- Space system configuration and design,
- Ground segment.

An assessment of the project development status is also provided in the Project Overview section.

II PROJECT OVERVIEW

1 Mission and Science Objectives

The motivation for the overall SwissCube project development is primarily to educate students in space technologies and space system engineering. This motivation has several impacts:

1) The project involves undergraduate and postgraduate students and young engineers through its whole life cycle;
2) The project cost is relatively low, in accordance with a university type of development;
3) Compared to an industry type space project, decisions are taken to simplify the design or design for low-cost and thus might not comply with the usual standards.

Keeping these aspects in mind, the mission and science objectives for the project are summarized in the following requirements. These requirements are the basis for the design provided in the rest of this document.

1.1 Mission Objective 1

The project shall design, build, and test a satellite. The success criterion is: deliver a fully tested satellite to the launch site.

This objective assumes the development of both a ground and space system.

1.2 Mission Objective 2

The project shall launch the satellite and communicate with it using the ground and space systems. The success criterion is: establish a radio connection with the developed ground system and download telemetry.

1.3 Mission Objective 3

The project shall operate a scientific or technology demonstration payload. The success criterion is: receive data from the payload and confirm operations.

The approach taken in regard to the nature of the scientific and/or technology demonstration payload is described in the next section. Note that the science requirements were defined to fit a system that is primarily designed for success of telecommunication (Mission Objective 2) and therefore represent a fine balance between the science desires and the ***capability*** of the space system.

1.4 Science Objectives

After discussions with several partners of the project, it was decided that the SwissCube mission should focus on the observation of the airglow phenomena. The motivation for these observations is to demonstrate the feasibility of using the airglow as basis for development of a low cost Earth Sensor (ES). A model of the airglow emissions as a function of intensity, latitude, longitude and time has been established and the objective the science mission is to collect data that will validate, or at least bring additional information to the model. The development of the Earth Sensor is a separate activity to SwissCube led by the EPFL-LMTS laboratory.

In addition, at the project level and as a technology demonstration, it was decided to develop a payload that has the most commonality/synergy as possible with the Earth Sensor. This decision impacts the design of the payload and the requirements to this effect can be found in the Project (level 2) requirements.

The nightglow is a photoluminescence of the atmosphere at night, occurring at approximately 100 km altitude (see Figure II-1). It is principally due to the recombination of the atomic oxygen, which is dissociated during the day. To study variations of the emissions as a function of time, the minimum science duration is 3 months, with an extended science mission of duration up to 1 year.

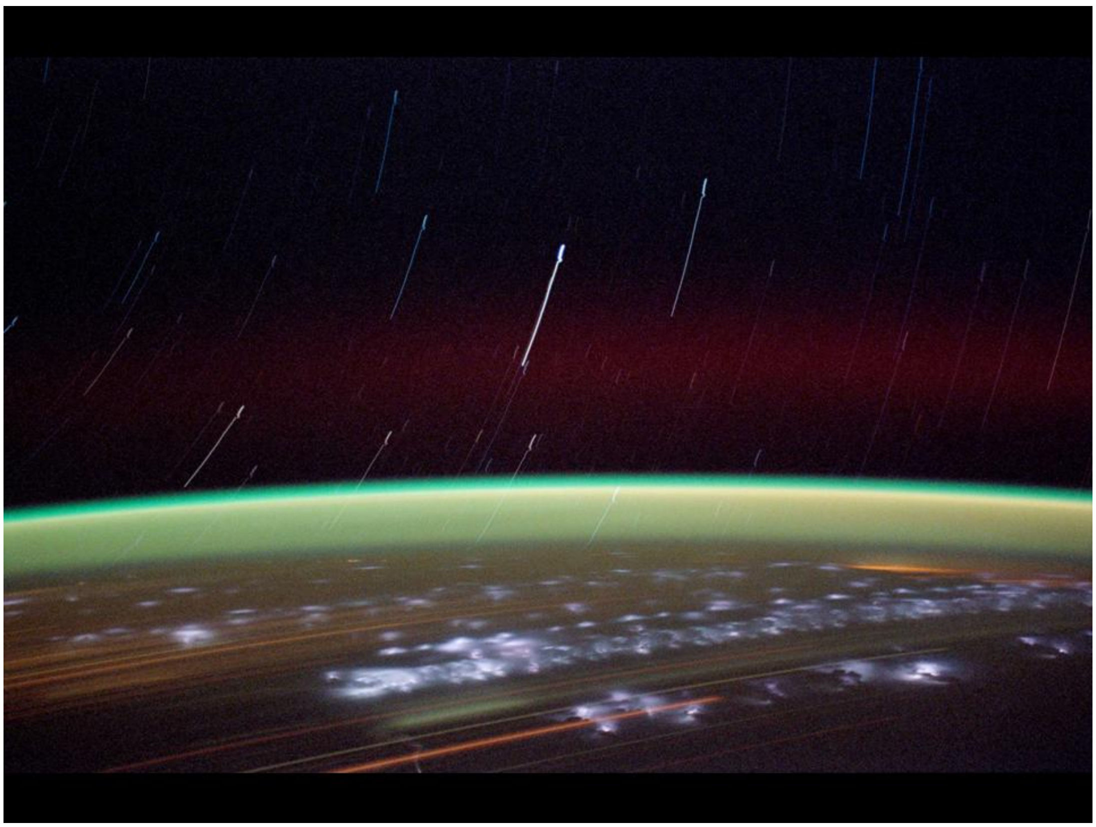


Figure II-1: NASA Photo of the nightglow.

At the mission level, the main science requirements are listed below. The full suite of science requirements can be found in the "SwissCube Mission, Science and Project Specifications" [1].

1_SR_01 **Science mission duration**

The science mission duration shall be at least [3] months. This primary duration shall be called the scoping phase, during which the primary science objectives shall be achieved. An extended science mission phase of up to [1] year is desirable.

The scientific goal is to observe the Airglow phenomena at least over 3 months, preferably over a year. Observing it over a 3 months period should be sufficient to determine geographical variation. A year of observations would ensure the understanding of seasonal variations.

1_SR_02 **Primary science objective**

The primary science objective is to observe the airglow emissions both at night and during day. Both limb and zenith measurements shall be made.

Definition of science objective.

1_SR_03 **Science observations altitude**

The Project shall take measurements in the [50 - 120] km altitude band of the Earth's atmosphere.

Airglow emission occurs between 50 - 120 km at day, respectively 80 - 120 km at night.

1_SR_04 **Spatial resolution**

The required spatial resolution at limb shall be less or equal to 20 km.

Minimum resolution to ensure usefulness of data.

1_SR_05 **Science observations coverage - Scoping phase**

Over the duration of the scoping phase of the science mission, a minimum of [20] images shall be taken. These images can include at least [5] limb images during day, [5] limb images at night, [5] zenith images at night and [5] during the day.

To ensure minimum coverage for science and technology demonstration.

1_SR_06 **Science observations coverage - Extended phase**

Over the duration of the extended phase of the science mission, the following observations shall be conducted: day glow at latitude in the ranges [> 80°N], [40°N-50°N], [5°S-5°N], [40°S-50°S],[>80°S] and nightglow at the same ranges as the day glow.

The first 3 months provide a first idea of the intensities of airglow during both day and night. Observations after 3 months allow to measure variations of emission intensity depending on latitude and seasonal variations.

1_SR_07 **Spectral range**

The project shall measure at least 1 band of emission in the spectral range of 550 – 880 nm.

Airglow emissions occur within this wavelength band. One band shall be measured to ensure minimum science.

1_SR_10 **Science data products**

The science data products (data needed for each observation) shall include the measurement of the airglow intensity, the position of the observed area, and the local solar time of the area of observation.

The data shall provide the results of measurements, the local solar time and latitude of measurements. The local solar time is the local time in reference to the position of the sun.

2 Project Development Assessment

2.1 Phase B activities

At this time in the development of the space and ground systems, it is necessary to highlight the expectations at the end of the Preliminary Definition Phase (Phase B) and make an assessment of the progress of the project with respect to these expectations. The expectations listed in Table II-1 follow the definitions of the European Standards ECSS M-30-A and ECSS-10-Part1B. Table II-1 also shows the progress in terms of "Achieved" (green), "In Progress" (orange) and "Not started" (red). Although not an industry project, care is nevertheless given to follow as much as possible the standards. It is however very unlikely that project level assessments such as Reliability, Safety and Environmental Impact will be fully analyzed.

Note that the current design has taken into account comments from the Phase A and PDR.

This evaluation shows that most expectations have been filled for Phase B and it is now proposed that the project advances to Phase C. The current critical areas of the development and the critical paths are:

- At the subsystem level: thermal design of the batteries and communication subsystem; tests of the integrated communication system is under way but not completed; verification of the determination and control algorithms with the appropriate disturbance models remains to be done;
- At the system level: the failure analysis has started but needs to be completed;
- As previously mentioned, it is very unlikely that a full Reliability, Safety and Environmental Impact assessment will be done. An orbital debris analysis assessment has been done and further analysis will continue in this area. A semester projects will be dedicated to quality assurance and will be done as part of the project requirements.

2.2 Schedule and cost assessment

Quarterly reviews with the SwissCube Advisory board allow for the review of the technical, financial and schedule status of the project.

Compared to original plans, the CDR has been moved from end of June 07 to the end of December 07 (the overall schedule can be found in the SwissCube Project Management Plan). This change has been integrated in the master AI&V schedule.

In addition, a grass-roots cost analysis was performed for the fabrication of the test and flight models. This analysis also included the workforce needed by the project. The estimated budget required at this point is around 500 kCHF for a flight end-2008.

Delta-PDR Sept 3, 2007	At System			At Subsystem Level												
Phase B Expectations	Launch system	Space System	Ground System	PAYLOAD	EPS	CDMS	COM Data	COM Beacon	Mechanical	ADCS HW	ADCS SW	Flight SW	Thermal	Mechanisms	G Station RF	GS SW
Selection of a technical solution																
Design or technical solution																
Confirmation of feasibility of technical solution																
Functional/Characterization tests of components																
Functional tests of assemblies/subsystems																
Assessment of critical techniques or technologies																
Assessment of pre-development work																
Make or Buy																
Assessment of manufacturing, production & operating cost																
Subsystem development																
Integration and Test																
Operations																
Assessment of manufacturing, production & operating schedule																
Subsystem development																
Integration and Test																
Operations																
Specifications																
Specifications documented and reviewed																
Specifications tree/Requirements traceability matrix																
Start interface documents																
Mechanical																
Electrical																
Data																
Thermal																
Fabrication Guidelines and Plan																
Integration and Test/Verification Plan																
Plannification																
Logistics requirements																
Elaboration of Design Justification File																
Reports																
Assessment of Reliability and Safety																
Assessment of environmental impact																

Table II-1: Project assessment against Phase B expectations.

3 Requirement Analysis

3.1 Status on the development of the specifications

The SwissCube specifications have been classified into levels and categories, as explained in the next section. At this point requirements are being written at 5 different levels:

- Mission and Science (Level 1)
- Project (Level 2)
- System (Level 3)
- Subsystem (Level 4)
- Assembly (Level 5).

The Mission/Science and Project specifications were elaborated during Phase A and updated during phase B. The approach taken during Phase B to establish lower level requirements was to have the students of each main subsystem write the specifications for their subsystem. Additional system support at the end of Phase B started the unification and consolidation of the system level requirements. The subsystem specifications were initially written based on the analysis performed and the baseline system design and updated as further analysis was performed.

About 90% of the specification documents have been written (most of them belong to the Space System). The missing documents are the ADCS software requirements and the flight software requirements. For the flight software, the specifications do exist but they have not been integrated in a document yet. All other specification documents have been internally reviewed. They are still subject to minor changes.

The reader is encouraged to have a careful look at the specification documents. The driving requirements at the system and subsystem levels are highlighted in this report.

3.2 Specifications Documentation and Hierarchy

The SwissCube requirements are currently gathered in several Specifications documents. The highest level document includes requirements at the Mission (level 1) and Project (level 2) levels [1]. System (level 3) and subsystem (level 4) levels specifications are documented separately [2, 3, 4]. Figure II-2 shows the specification document tree. Separate guidelines were written to facilitate the elaboration of the requirements. These guidelines include description and examples of the categories used to classify the requirements [5].

Project level requirements have an impact at least on two of the following main systems: Mission and Science, Launch System, Space System, Ground System, System Integration and Test or Mission Operations. System level requirement have an impact at least on two subsystems.

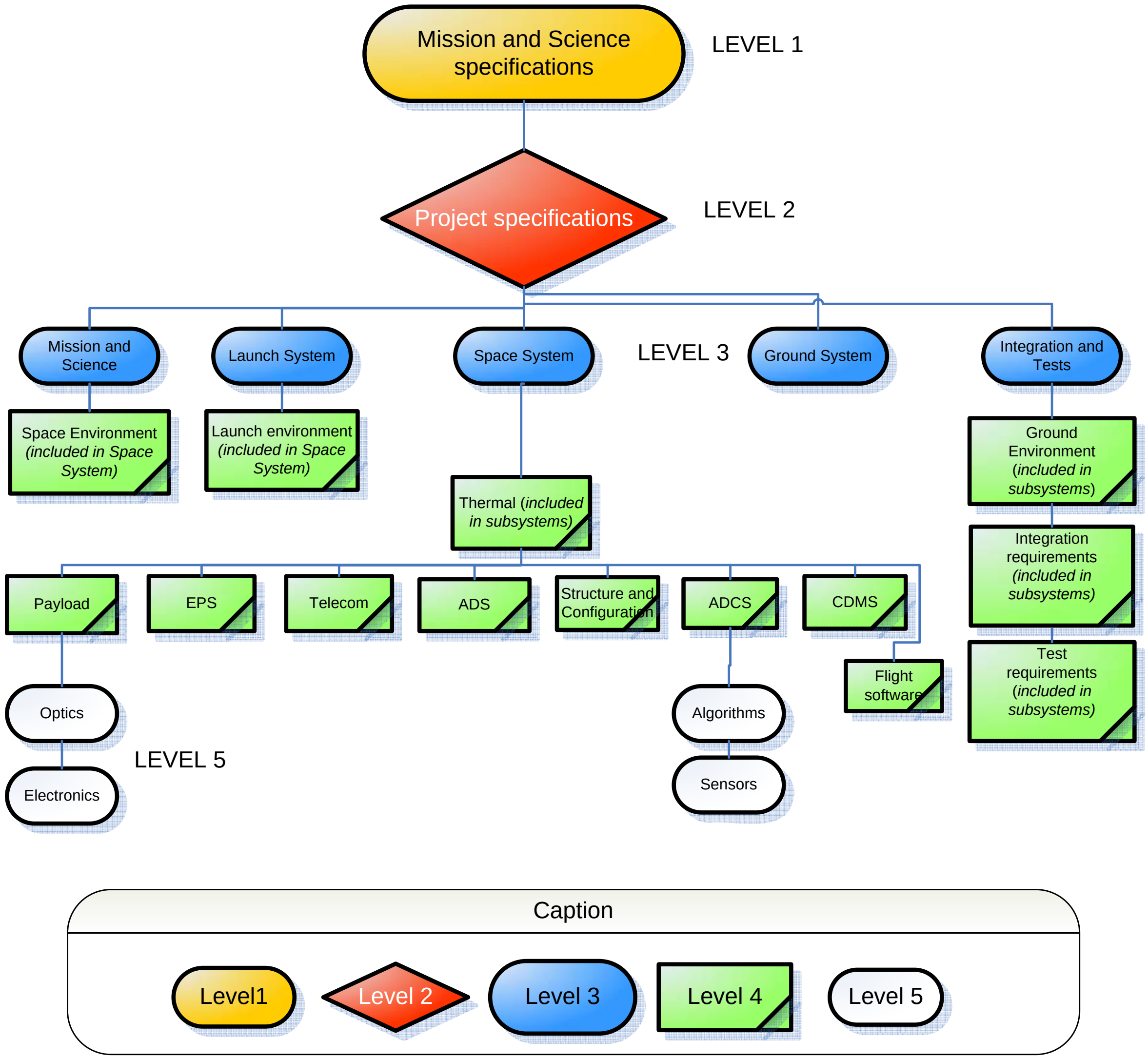


Figure II-2: SwissCube specification document tree.

At system and subsystem levels, the requirements are organized in categories as shown in Table II-2.

A requirement database and friendly interface was developed. This tool facilitates the verification of the requirement flow between each level.

A traceability matrix will be established along with the AI&V plan.

Subtopic	Definition/Question	Example
Functional Requirement		
System Functions	What function does the "system" need to perform?	
Mission & Performance requirements		
Modes	Operation modes of the element	Ex. Modes ON & OFF
States	Possible states of the element in each mode	Ex. In mode ON the system turns between 5000 and 10000 rpm
H/W performance	What performances does the system exhibit in each state?	Power
S/W performance		Software capacity
Reliability & Redundancy	What is the reliability of the system? What redundancies should be added?	
Design requirements		
Constraints	Constraints are a characteristic, result or design feature which is made compulsory or has been prohibited for any reason. Constraints are generally restrictions on the choice of solution in a system.	Materials, Marking, Tribology, operational conditions, law, standards…
Thermal	How does the system regulate its thermal environment? Thermal design rules.	Passive or active thermal control. Heaters
Maintainability	Does the system need to be maintained during ground life? How?	
Interface		
Structural	How does the system connect to other structural elements?	Volume, Shape, Attachment, Location
Thermal		Heat generation, thermal resistance of interfaces, Heat capacity
Electrical		Voltage, current, connectors (pin definition), including electrical part of data I/F
Data interfaces		Data messages (content), format (protocols)
Physical properties		Size, Mass, CoG, MoI
Other Interfaces		
Environments		
Thermal		Qualification temperature range, Operational temperature range, non-operational range, Thermal test gradients
Static and dynamic loads		
Vacuum		
Radiation		
Operations		
Autonomy		
Control		
Failure Management		

Table II-2: System and subsystem requirement categories.

4 Assembly, Integration and Verification (AI&V) Plan

A separate document [6] describes the approach and plan for the SwissCube AI&V. It describes:

- the verification approach (consistent with the ECSS),
- the model philosophy,
- the verification matrices (still to be refined),
- the test plan as a function of the type of tests (mechanical, thermal, functional, qualification…) and as function of models (integration models, EQM…),
- the ground support equipment,
- the required verification documentation (tests specifications, procedures, report formats),
- the test schedule,
- and the space system assembly procedure.

We summarize here the short term considerations for the AI&V plan.

4.1 Model philosophy

The following model philosophy has been defined for the SwissCube project:

- Integration (Electrical) Model (IM)
- Satellite Structural Model (SM)
- Satellite Engineering Qualification Model (EQM)
- Satellite Flight Model (FM)
- Satellite Flight Spare (FS)

Figure II-3 shows the model flow diagram.

The current plan (deliverables) for the space subsystem models is summarized in Table II-3.

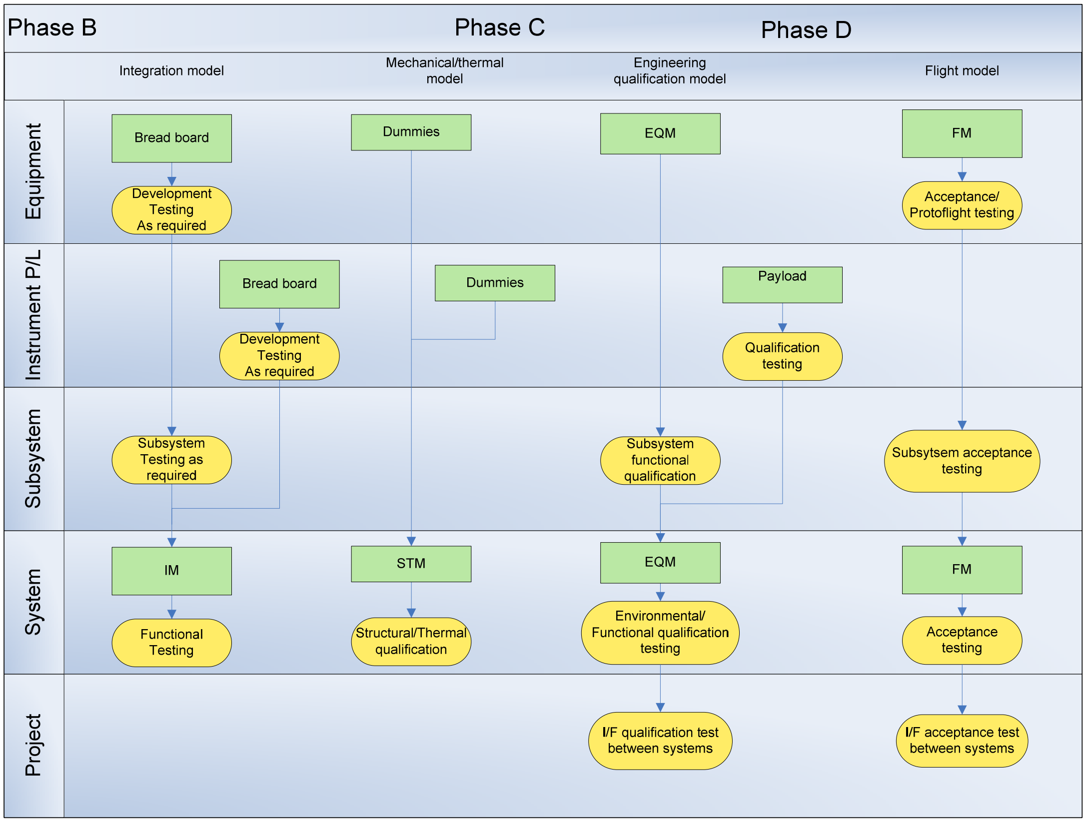


Figure II-3: SwissCube model flow diagram.

Subsystems	PDR, March 07	Delta-PDR, Sept 07	CDR, Dec.07	AR, June 08
EPS	Analogic functional	Analogic + digital (micro-controller) functional – tested prototype	EQM Board	2 Flight Models
COM Beacon		Analogic functional	EQM Board	2 Flight Models
COM Data	Receiver and transmitter functional	Receiver, transmitter, micro-controller integrated functional	EQM Board	2 Flight Models
CDMS	Analogic + Digital (Micro-controller) functional	EM Board	EQM Board	2 Flight Models
Structure	Structural model	Structural thermal model	EQM structural and thermal	2 Flight Models
Mechanisms		Structural/functional	Integrated into EQM structural/thermal	
PAYLOAD		Analogic + digital (micro-controller) functional Structural model	EQM Board	2 Flight Models
ADCS HW	Digital functional	Analogic + digital functional – tested prototype	EQM Board	2 Flight Models

Table II-3: Deliverable models for the space subsystems.

4.2 Verification matrix

A verification matrix has been established for each model but still needs to be consolidated with the requirements at each level. Table II-4 shows an example of the verification matrix for the EQM.

Requirement Category	Equipment Level	Instrument Level	Subsystem Level	System Level
Functional	T	T	T	T
Performance	T	T	T	T/A
Solar cells soldering	T	-	-	-
Launcher Interfaces	-	-	-	T
Ground Segment Interfaces	T/R	-	T/R	T/R
Physical Properties (mass, CoG)	T/R	T	T	T/I
Vibration	-	-	-	T
Shock	-	-	-	T
Thermal Verification	T	T	-	T
EMC/ESD	T	-	T	-
Radiation Environment	T*/A	T*	-	-
Outgassing	T*	T	-	T
Acoustic	-	-	-	A

Table II-4: EQM Verification Matrix.

4.3 Test flow

General test flows have been established for each model. Figure II-4 and Figure II-5 give an example of the test plan for the Structural and Thermal Model and for the Engineering Qualification Model.

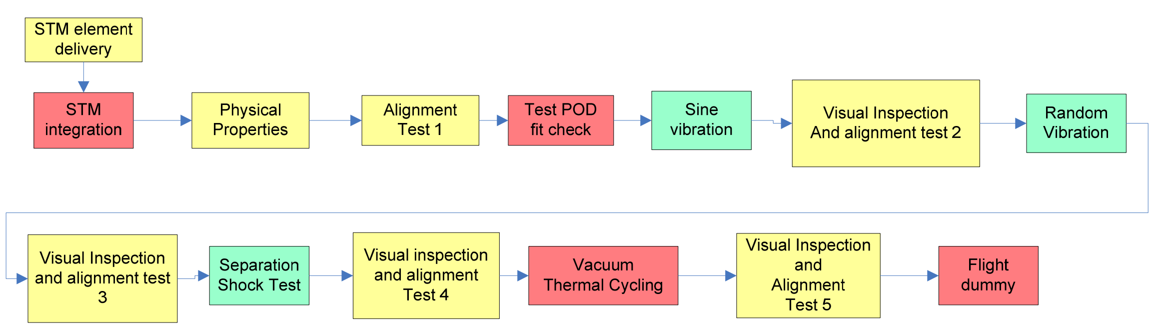
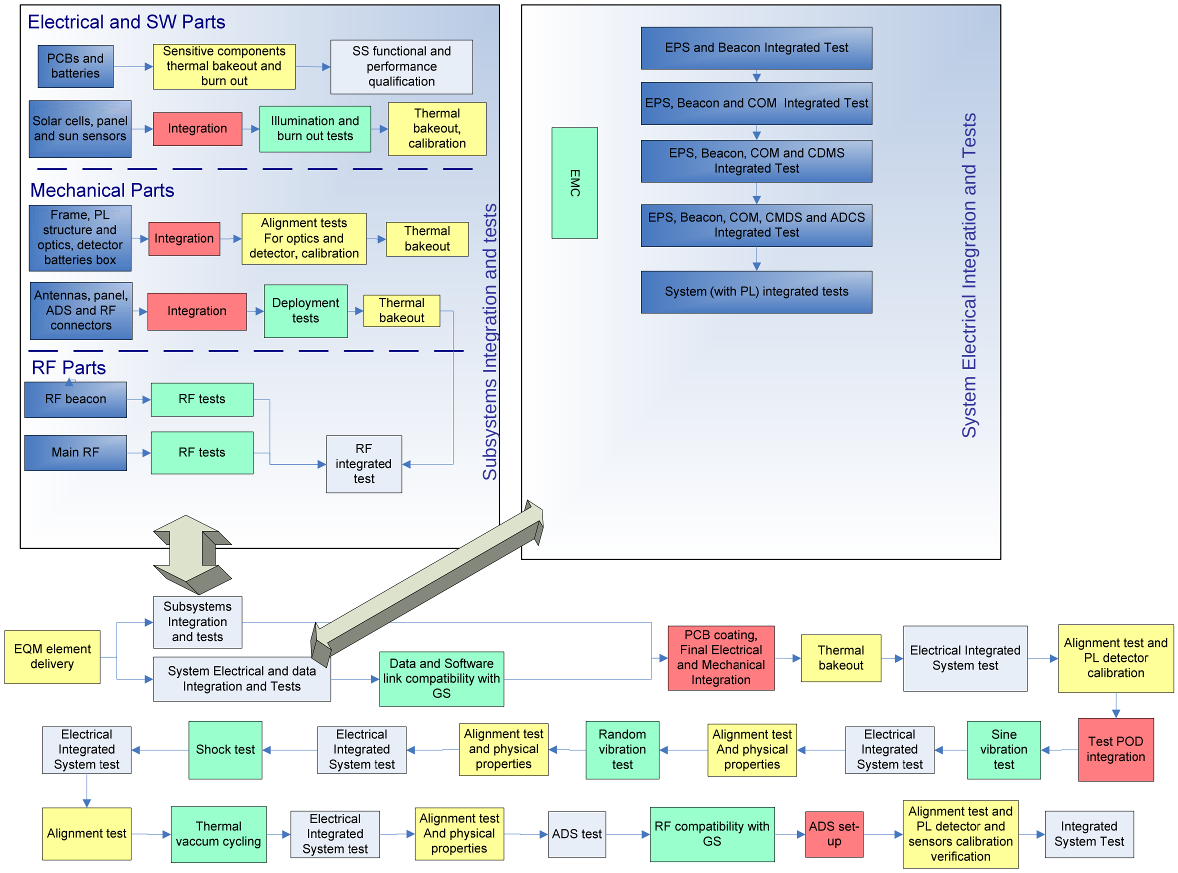


Figure II-4: STM test flow.



ADS: Antenna Deployment System
PL: Payload
CDMS: Command and Data Management System
ADCS: Attitude Determination and Control System
COM: Communication
EPS: Electrical Power System
RF: Radio-frequency
GS: Ground Station
EQM: Engineering Qualification Model

Figure II-5- EQM test flow.

III MISSION OVERVIEW

This chapter summarizes the elements related to mission design and mission planning. It also describes the assumptions and analysis regarding the space environment characterization.

1 Mission driving requirements

2_PR_11_01 **CubeSat Pico-satellite**

The project shall use the organizational frame and requirements of the Cubesat Standard developed by the California Polytechnic State University (Cal Poly).

Cubesats are a standard that allows for an easier access to space and access to an experienced university community. It is assumed that the ESA VEGA launch process will be similar to the CalPoly one.

2_PR_11_03 **Space to ground frequencies and protocols**

The ground to space communication link shall comply with the Amateur Radio Satellites services.

For a student satellite this is the easiest solution to implement.

2_PR_14_02 **Launch vehicle**

The project shall be able to launch on a [DNEPR, VEGA, Rockot, Kosmos, MV-8, Indian PSLV, SOYUZ or Ariane 5] launch vehicle.

Possible options of launchers that have been used so far for Cubesats or that are plausible solutions for the project.

2_PR_14_04 **Launch Date**

The satellite shall be ready for launch by [end]-2008.

This date allows for a reasonable development schedule and the most probable launch opportunities.

2_PR_15_03 **Communication Availability**

The project shall provide downlink capability between the space system and the ground at every opportunity after antennas deployment.

This requirement ensures communication capability between the space system and a ground station available and in view.

2 Mission Design

SwissCube will be an auxiliary payload. The mission design will be limited to a range of possible orbits.

2.1 Orbit Design Drivers

There are different types of design drivers on the SwissCube orbit design, namely:

- The location of the ground stations at EPFL and at HES-Fr requires an inclination of at least 45° to achieve reasonable pass durations. The most probable orbits that will be flown are thus sun-synchronous. However, due to orbital perturbation, the satellite will drift out of the sun-synchronous state. The drift has not yet been assessed. For the analysis, the assumption was made that the orbit would remain sun-synchronous.
- Analysis of past sun-synchronous launches showed that the most likely range for altitude heights is between [400 – 1000] km.
- The space system has no trajectory station keeping or maneuvering capability.

For the SwissCube mission other types of orbits could be considered below 1000km and above 45° inclination. Due to the enormous range of possibilities other orbits will only be studied if required by the launch provider.

2_PR_14_05 **Orbit Altitude**

The project shall operate at Earth distances between [400] and [1000] km altitudes.

Analysis shows that for Sun-synchronous orbits, the expected altitude range is [400-1000] km (Ref. Analyse de mission d'un satellite, N. Scheidegger)

2_PR_14_01 **Operational orbit**

The project shall select an orbit which has an inclination above [45] degrees. For mission analysis a sun-synchronous orbit shall be considered.

To have direct access with the Ground Segment in Switzerland. This requirement implies a very high probability of flying on a sunsynchronous orbit.

2.2 Sun-synchronous Orbits (SSO)

In Sun-synchronous orbits (SSO), the nodal regression caused by the J2 (non-spherical Earth) is matched with the angular rotation of the Earth around the Sun. Thus the plane of a Sun-synchronous orbit keeps a constant angle alpha with the Earth-Sun vector (see Figure III-1). As shows Figure III-2 these orbits are almost polar and cover therefore all latitudes.

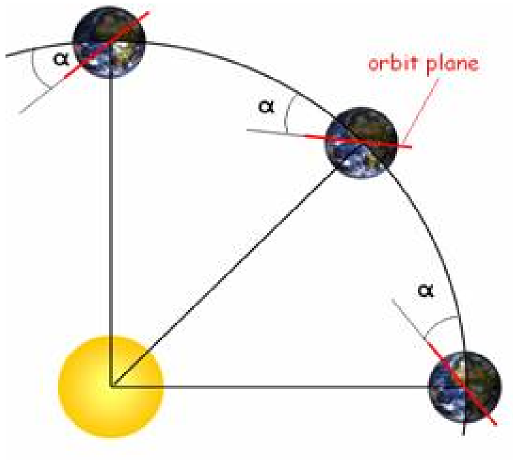


Figure III-1: Constant Sun/Orbit plane angle alpha for sun-synchronous orbits.

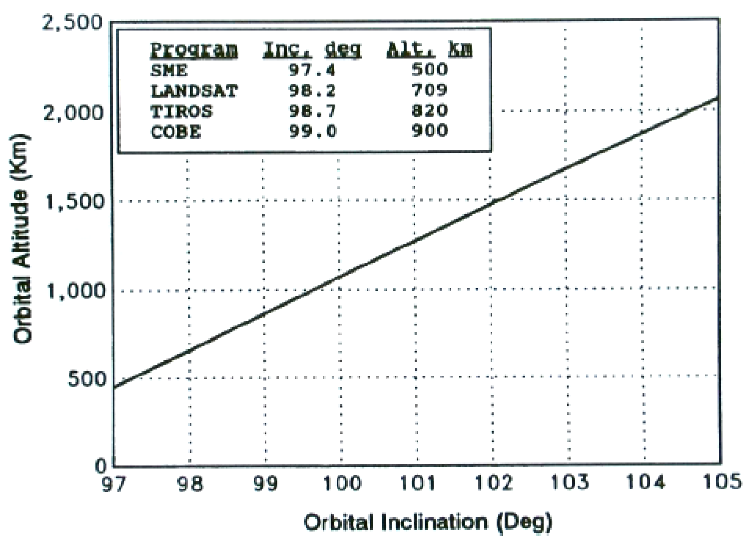


Figure III-2: Orbital inclination and altitudes for sun-synchronous orbits.

SSO orbits are ideal for Earth observation missions since the satellite always crosses the Equator at the same local time. Furthermore, they simplify the satellite design since eclipse durations are almost constant.

2.2.1 Eclipse durations

The eclipse duration is an important parameter for the design of the space system. An analysis was performed that calculates the minimum, mean and maximum eclipse duration as a function of the Sun /orbit plane angle alpha and the altitude (using STK). Figure III-3 shows that:

1. For $\alpha=0$, the orbit plane coincides with the terminator, the satellite is in constant light, and there are no eclipses.

2. While eclipse duration varies between 18 min and 36 min for orbits with 20 deg < α <80 deg at 400 km altitudes, it ranges between 22 min and 34 min for orbits at 1000 km altitude and 40 deg < α <80 deg.

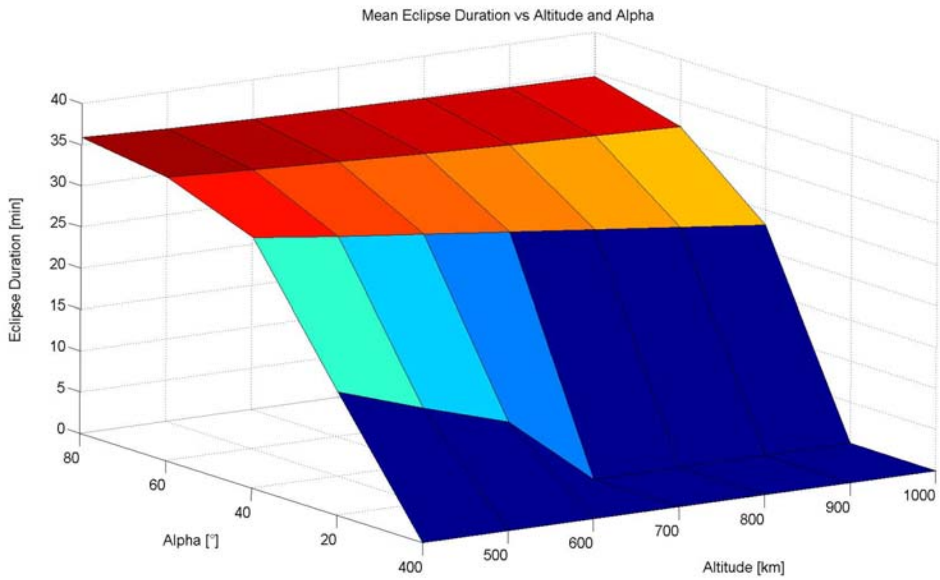


Figure III-3: Mean eclipse duration as a function of alpha and altitude.

2.2.2 Orbit disturbance, drag and lifetime

The solar cycle is expected to peak in 2011. Thus in 2008, the solar environment can be expected to be relatively high. As the solar activity influences the Earth's atmospheric density profile at high altitude significantly, disturbance forces due to atmospheric drag will cause the satellite to loose altitude. The expected lifetime and analysis of the orbital disturbances remain to be done in detail.

2.3 Mission Operation Considerations

2.3.1 Time in view of a ground station

The time in view analysis aims at estimating the access frequency and duration with the ground station. The following analysis assumes a single ground station located in Lausanne. Time in view duration was computed with respect to the altitude, the elevation angle ε and α-angle. Figure III-4 shows the mean access time as a function of altitude for a minimum elevation of 10°. Mean access times between 5 and 10 minutes can be expected.

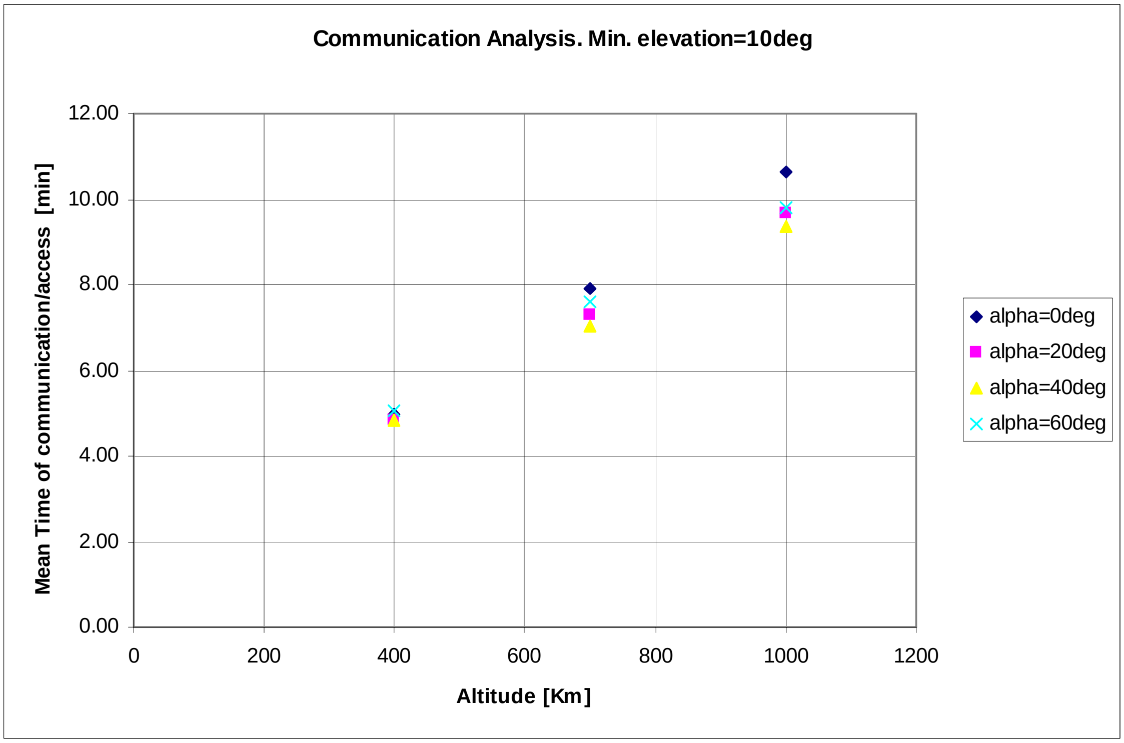


Figure III-4: Mean access time as a function of altitude for 10° min. elevation.

2.3.2 Sample timelines

Based on the analyses of the precedent section, it is possible to establish a time plan over a fixed period ranging from one (typical) day to the whole mission duration. Timelines offer an overview of the mission course and help design a schedule for the different tasks such as maintenance, scientific measurement, data exchange, etc.

Sample timelines presented in Figure III-5 and Figure III-6 span over the periods of one day for an altitude of 400 km and α=20 deg and a second one at an altitude of 1000 km and α=60 deg.

<u>Case1: Altitude=400 km, α=20 deg</u>

This orbit has an average eclipse time of 18 min with a frequency of 15 eclipses per day. For communicating there are at most five possible passes with the maximum possible duration (time in view) equal to 8 minutes per pass. The number of passes and their duration decrease under the $\varepsilon_{\min}$ constraints.

<u>Case2: Altitude=1000 km, α=60 deg</u>

In contrast to the first orbit this one has an average eclipse duration of 31 min and 14 eclipses per day. Since the altitude is higher there are more and longer possible access windows. In fact there are 8 possible communication windows with pass durations of at most 12 min.

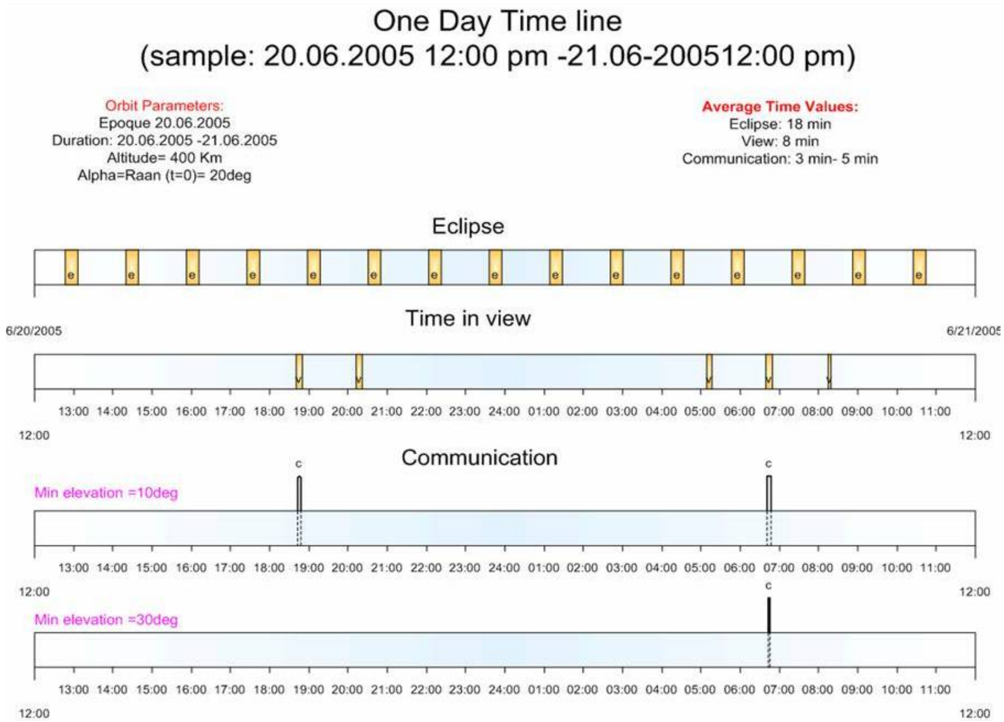


Figure III-5: Sample timeline for 400 km, α = 20 deg.

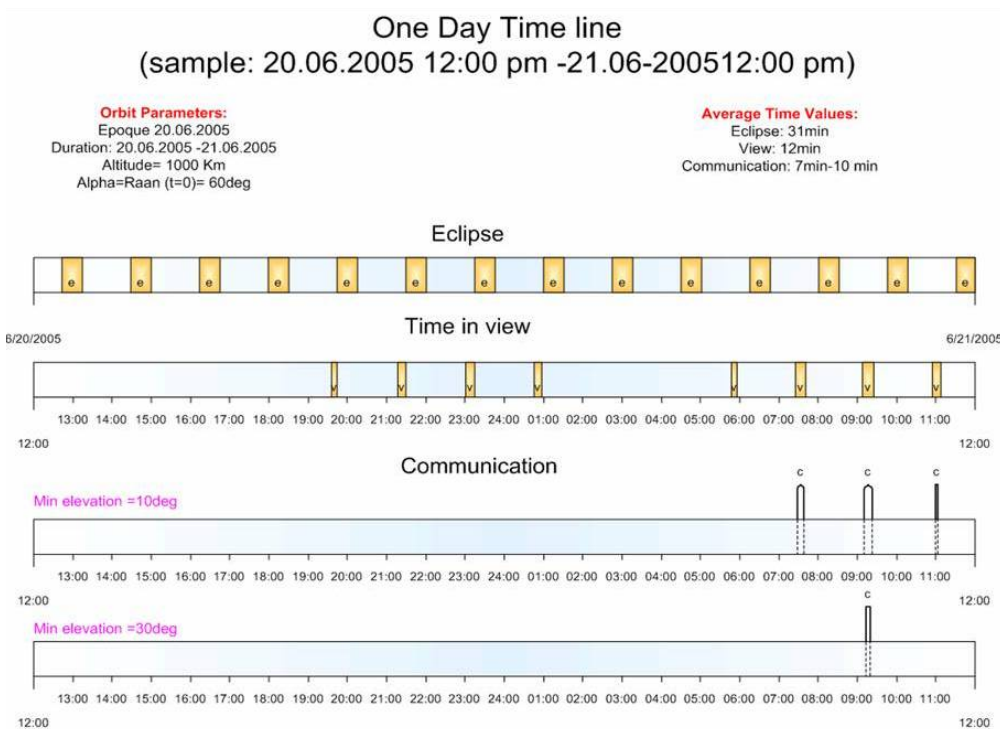


Figure III-6: Sample timeline for 1000 km, α = 60 deg.

3 Mission Environments

The space environment is far from benign in its effects on space systems. Understanding of the space environment and its interactions is the first step in mitigating these effects.

The principal interactions of concern for SwissCube are: cumulative radiation effects; single event upsets; latch up; surface and internal charging; contamination; space debris and micrometeoroid impacts. Thermal effects and torques were taken into account and were included in the space system design.

Although each interaction should be assessed, a few have been considered so far in the design of the space system. Given the relatively short mission duration, the major concerns assessed for SwissCube are listed below. A debris and meteoroid assessment has started (as included in the requirements).

3.1 Cumulative Radiation Effects

Table III-1 summarizes the trapped radiation dose cumulated for 4 months and 1 year behind 40 mils of Al (1mm) for maximum and minimum solar activity. The analysis assumes a spherical shell shield configuration. The analysis was done using the ESA Spenvis Tool.

Configuration	Cumulated Dose (krad[Si], 1 mm Al, Solar Max)	Cumulated Dose (krad[Si], 1 mm Al, Solar Min)
400 km 4 months	0.9	0.5
400 km 1 year	4	2
1000 km 4 months	5	4
1000 km 1 year	22	15

Table III-1: Trapped radiation cumulated dose worst and best cases behind Aluminum.

Further analysis needs to be done to confirm the dose and the effects on the PCB components given the new panels design. However, the cumulated dose for solar maximum and the required 4 months of operation is sufficiently small to confirm that COTS can be used without major shielding.

3.2 Single Event Upsets (SEUs)

SEUs will be mitigated by hardware and software design practices.

3.3 Latch-up

A separate latch-up protection circuit has been designed and will be implemented in each electronic subsystem.

3.4 Debris Environment

The debris hazard has been calculated with the NASA90 model. The model gives the flow of particles that a satellite will encounter during one year of operation as a function of particle size. The effect of a debris impact on the SwissCube depends on velocity and weight of the impacting object. As the SwissCube lacks redundant systems any penetrating hit can be expected to result in the loss of the satellite. The energy required for penetrating the solar panel surfaces has been very roughly estimated to be equivalent to a BB gun bullet (~7 J). The impact velocity is approximated with the satellite velocity (~7.5 km/s). This gives a critical object weight of 0.25mg. This has then been approximated to the nearest weight category in the NASA90 model which is 0.18mg, anything bigger than this is likely to cause the loss of the satellite.

The likely of such an impact over the satellite's extended lifetime (15 months) has been calculated for the different orbits (Figure III-7). The risk is very low and is lowest for low orbits (Table III-2).

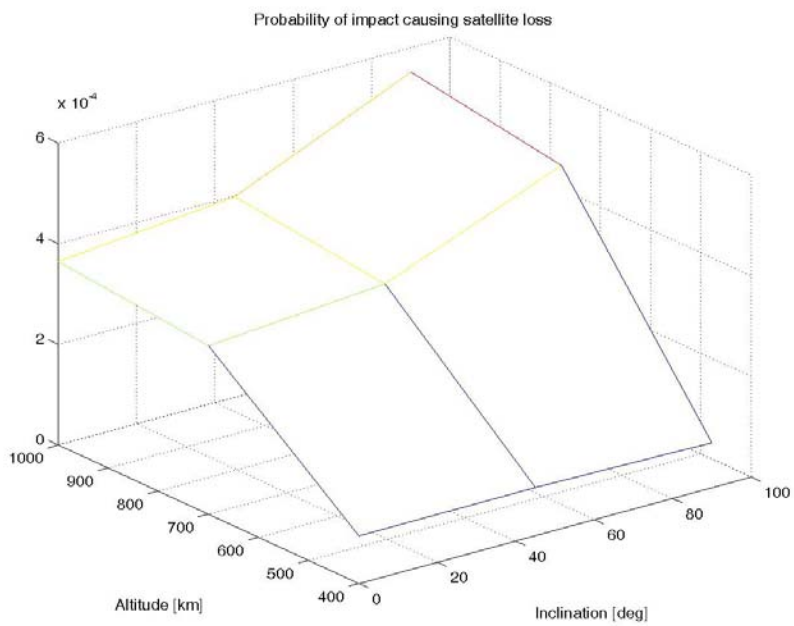


Figure III-7: Probability of impact causing satellite loss over 1.25 yr.

	Inclination [°]		
Altitude [km]	0	45	90
400	9.4E-05	9.6E-05	9.0E-05
700	3.3E-04	3.6E-04	5.0E-04
1000	3.7E-04	4.0E-04	5.5E-04

Table III-2: Probability of impact causing satellite loss over 1.25 yr.

4 Launch Vehicle

The launch interface for a CubeSat is defined by the Calpoly Design Specifications [7]. CubeSats are not attached to a separation ring as do "classical" satellites but are placed inside a closed pod. Once in orbit the pod's lid is opened and the CubeSats inside are released. The fundamental goal of the design is therefore to fit the interface requirements of the CubeSat launch pod. This standardization allows for the SwissCube to be placed on virtually any launcher capable of carrying the CubeSat launch pods.

4.1 Candidate Launch Vehicles

CubeSats have been launched from many different launchers. The list hereafter is not exhaustive:

- STS-113, Cape Canaveral, November 2001
- Rockot, Plesetsk, June 2003
- Kosmos-3M, Plesetsk, Octobre 2005
- M-V-8, Uchinoura, February 2006
- Dnepr, Baikonour, Summer 2006

The most plausible options identified so far for launch in 2008 are the following:

1) **VEGA Maiden Flight and demonstration flights**

   The VEGA maiden flight is scheduled for launch in 2008, with a launch date to be released. Following the maiden flight, a series of 5 qualification flights is planned in a sequence of one every 6 months. It is assumed that the same structure as the CalPoly P-POD will be used if CubeSats are to be launched on VEGA, as expected. Launch is also expected to be free (or close to).

2) **CalPoly launch services**

   CalPoly offers about one launch per year. The project registered to the “interested list” for flight on DNEPR mid 2009. Launch cost are about $50000.

3) **KAP on Ariane 5 or Soyouz**

   The Kaiser-Threde Arianespace Platform (KAP) is an experimental platform for in-orbit demonstration and/or verification of technology and scientific experiments. Arianespace as launcher authority as well as ESA and DLR are supporting KAP as future auxiliary payload for Technology In-Orbit Demonstration. A first flight opportunity will occur on Ariane 5 in 2008. It is unclear if CubeSats will be accepted on that flight. Another opportunity is on Soyuz (launched from Baikonur Cosmodrome) mid-2008, but details still need to be provided. To launch CubeSats, the KAP would most probably be equipped with the “Single PicoSat Launcher” (SPL) developed by Astro Feinwerktechnik, Berlin, Germany (see Figure III-8). Interfaces are expected to be similar to the P-POD, but still need to be defined.

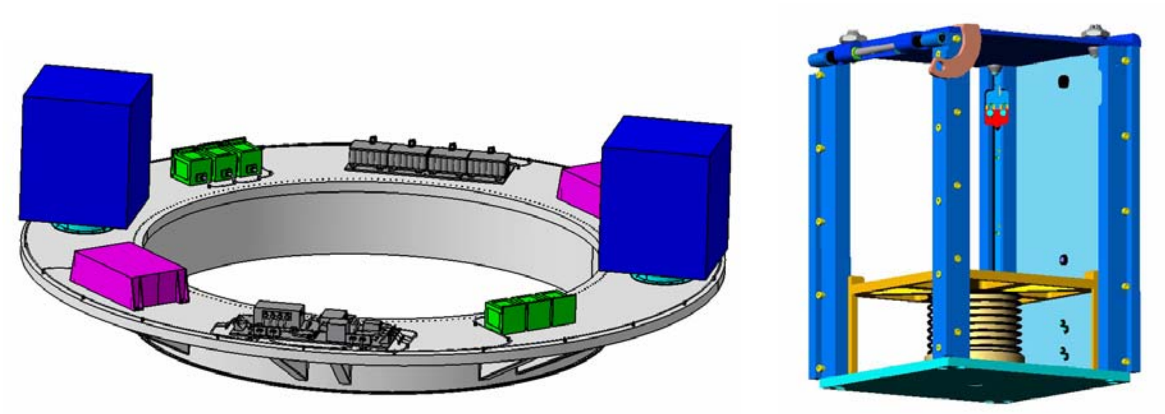


Figure III-8: KAP on Ariane 5 (in green) and Single PicoSat launcher from Astro Feinwerktechnik.

4.2 Launch environments

The launch environments are defined according to the specifications given in Calpoly Design Specifications [7] and the chosen launcher. The launch environment specification document [8] summarizes the launch environment for several launch vehicles. SwissCube should be compatible with any of the launch vehicles listed above; therefore the most stringent launch environments shall be met. Those environments are specified by the various launch providers.

4.3 Accommodation & Mechanical Interface

Mechanical interface between the satellite and the launcher is achieved through the P-POD or T-POD launcher.

Figure III-9: P-POD during Rokot launch campaign .

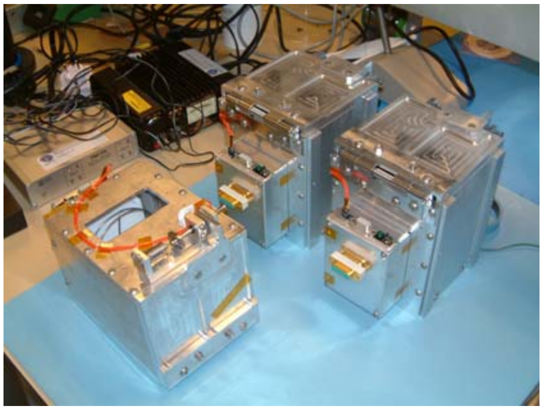
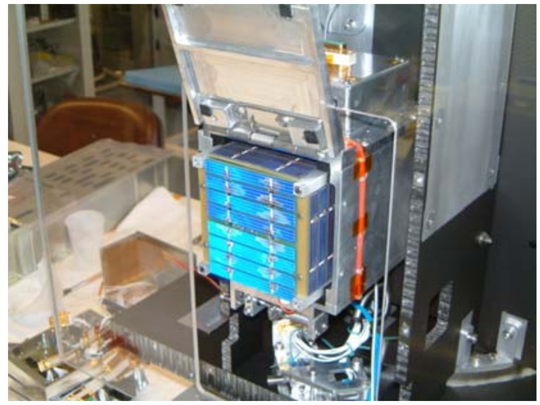



Figure III-10: T-POD's during integration of SSETI-Express.

4.4 Electrical Interface

2_PR_31_01 **Ground electrical interface**

During ground operations an umbilical link to the satellite will be available through the CubeSat's launch pod's access port.

Calpoly spec.

2_PR_31_02 **Launch electrical interface**

During launch there will be no electrical interfaces to the launch vehicle.

Calpoly requirements

2_PR_15_07 **Space System maintenance after launcher integration**

The space system shall be capable of being stored inside the P-POD without maintenance for at least [7] days.

Ensure autonomy of the satellite once integrated on the launch vehicule.

5 Mission Timeline

5.1 Timeline Drivers

The following external parameters will influence the mission timeline:

- Orbital parameters.
- Ground stations location.
- Calpoly CubeSat requirements.

The drivers above will influence the system performance parameters. Hereafter the major mission phases will be defined.

5.2 Mission Phases

The following paragraph summarizes the various phases after final satellite acceptance until the disposal of the satellite.

5.2.1 Pre-Launch

In the pre-launch phase the final launch site tests are performed and the satellite is prepared for launch. Activities include the charging of batteries and the check-out of the satellites subsystems. This action will be performed while the satellite is already integrated into its launch pod.

5.2.2 Launch and early operations (LEOP)

LEOP phase will start after satellite separation from its launch container. It will include the following:

- 15 minutes transmission and antenna deployment dead time.
- Switch-on and antenna deployment.
- Initial satellite acquisition (RF Beacon).
- Validation of the correct operation environment on-board the satellite.
- Validation of the space-ground data link (RF Transceiver).

This phase will end once the validation steps above have been performed. This phase will be terminated within [4] days after ejection from the launch container.

5.2.3 Commissioning phase

During satellite commissioning the on-board systems of the SwissCube satellite will be tested and their operational performances confirmed. The results will be used to correct and calibrate the on-ground satellite models. Commissioning will end after the validation of all systems and will be terminated within [20] days after ejection from the launch container.

5.2.4 Nominal phase

During nominal phase SwissCube will be fully operational and shall perform the defined science program. This phase will be over 3 months after the end of the commissioning phase.

5.2.5 Extended phase & disposal

If the satellite is still operational after this phase, the mission will be extended. No active disposal is foreseen. Science operations will be performed as long as possible and subsystem degradation will be monitored up to a mission critical failure, or up to [1] year, whichever comes first.

5.2.6 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.

5.3 LEOP Tentative Mission Timelines

This paragraph will present a tentative mission timeline for LEOP for the various launch site options and highlight subtleties that might occur for the various launch sites.

For this study a participation in a ground station network is assumed with stations in Stanford, (USA), Tokyo (Japan) and Lausanne (Switzerland).

The final launch site will have an impact on the satellite's early operation scenarios. The following have been considered for a 400 km altitude SSO launch towards north. A launch during daylight is assumed.

- CSG Kourou;
- Baikonour;
- Plesetsk.

5.3.1 CSG Kourou

One option is to launch from the Centre Spatial Guyanais (CSG) in Kourou. A reference ground track for a SSO orbit is given in Figure III-11.

From a mission point of view the launcher performance and launch site will affect how soon after ejection the satellite will be acquired by a SwissCube partner ground station. Taking into account that for a daytime SSO launch towards North the ascending part of the orbit will occur at daylight whereas the descending at night.

A tentative timeline for a CSG Kourou launch is given in Table III-3. The assumption is a 10° minimum elevation angle. The first pass over the Lausanne ground station occurs only 7:20 hours into the mission. This will have an impact on the design and shows the necessity of several partner ground stations and radio amateur surveillance of the satellite.

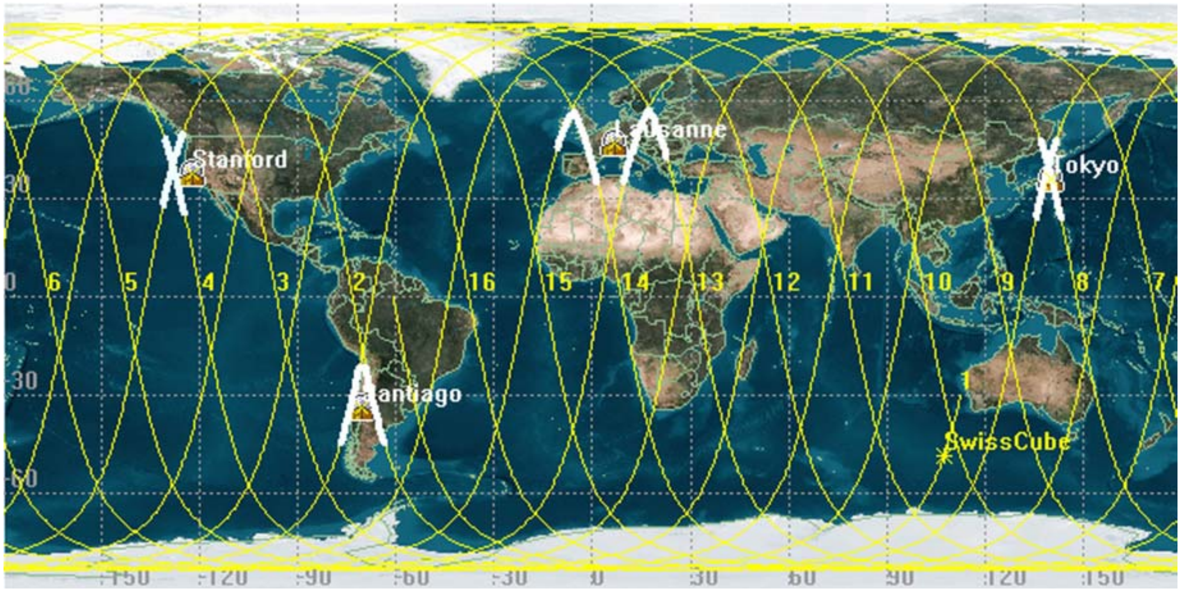


Figure III-11: Satellite ground track after injection by Vega from CSG Kourou.

Orbital time	Location	Ascending Descending	Event
0:00	West Australia		Satellite injection
0:15	Antarctica		Antenna deployment and beacon turned-on
0:37	Argentina/Chile	A	First landfall, possible radio amateur contacts
3:50	Stanford	A	GSN pass (355 sec)
7:20	Lausanne	D	First main GS satellite pass (355 sec)
9:55	Lausanne	D	Second main GS satellite pass (170 sec)
10:00	Tokyo	A	GSN pass (374 sec)
16:30	Stanford	D	GSN pass (341 sec)
17:40	Lausanne	A	Third main GS satellite pass (233 sec)
19:20	Lausanne	A	Fourth main GS satellite pass (336 sec)

Table III-3: Tentative timeline for SSO launch from CSG Kourou.

5.3.2 Baïkonour Cosmodrome

As for Kourou a tentative timeline was established for the Baïkonour Cosmodrome situated in Kazakhstan. For this option the launch would be most likely with a Dnepr launch vehicle [9]. For the present simulation a 98° Sun-synchronous launch has been assumed.

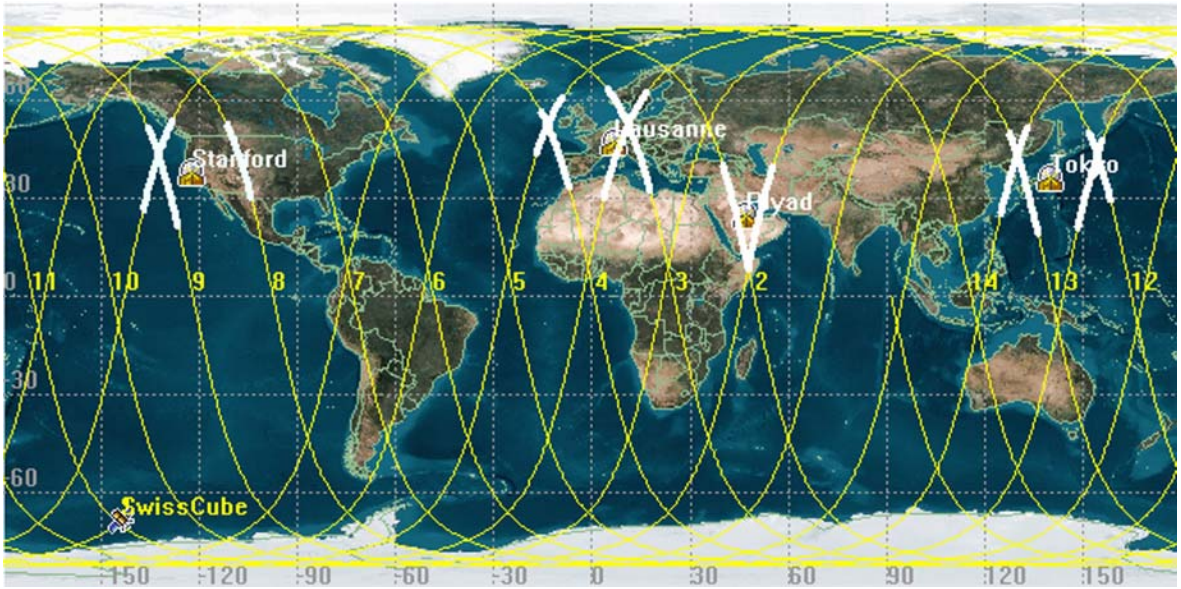


Figure III-12: Satellite ground track after injection by Dnepr from Baikonour Cosmodrome.

Orbital time	Location	Ascending Descending	Event
0:00	Around Antarctica		Satellite injection
0:15	Indian ocean	A	Antenna deployment and beacon turned-on
0:33	Yemen/Somalia	A	First landfall, possible radio amateur contacts
1:04	Stanford	D	GSN pass (457 sec)
2:16	Lausanne	A	First main GS satellite pass (525 sec)
3:54	Lausanne	A	Second main GS satellite pass (386 sec)
6:00	Tokyo	D	GSN pass (446 sec)
7:36	Tokyo	D	GSN pass (446 sec)
…			
15:42	Lausanne	D	Third main GS satellite pass (544 sec)
17:20	Lausanne	D	Fourth main GS satellite pas (291 sec)

Table III-4: Tentative timeline for a 98° SSO launch from Baïkonour Cosmodrome.

5.3.3 Plesetsk Cosmodrome

The third assessed launch site is the Plesetsk Cosmodrome situated in northern Russia. Two CubeSat launches have been carried out, one with a Kosmos-3M and the other with a Rockot [10] launcher.

Rockot places the satellites and the upper stage first into a 96.1° parking orbit. The satellites are then placed into their final orbit using the Breeze upper stage. At this stage, it is therefore not possible to predict any LEOP.

For Kosmos an analysis for a 98° Sun-synchronous launch has been carried out. Due to the geographic proximity to Baïkonour the results are very similar.

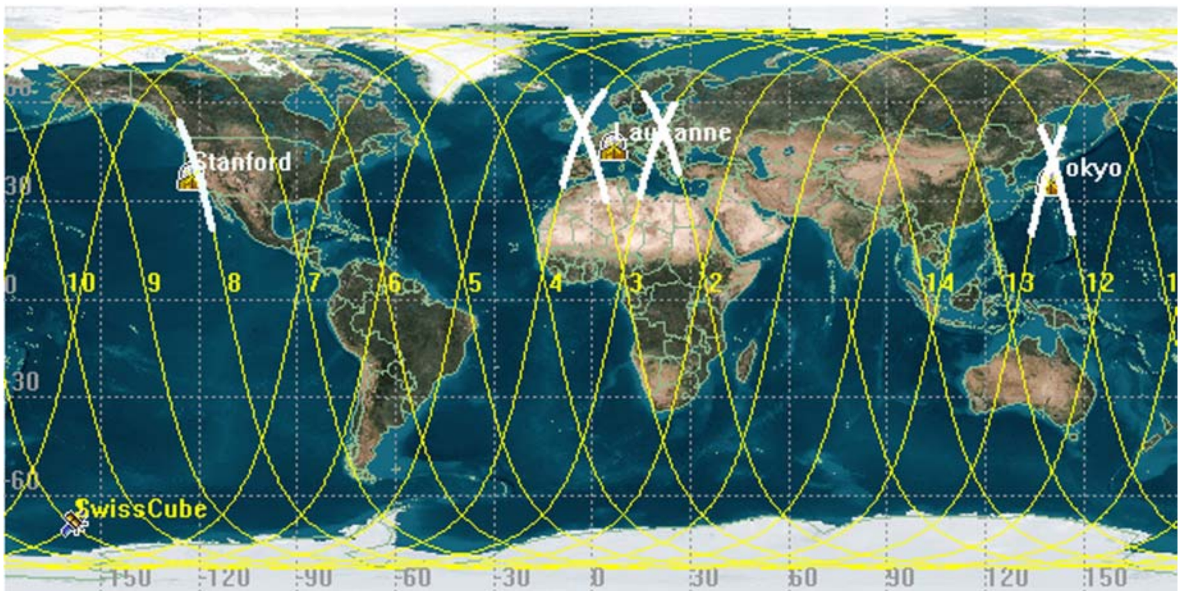


Table III-5: Tentative timeline for a 98° SSO launch from Plesetsk Cosmodrome.

5.3.4 Conclusion

From the carried out analyses it can be concluded that first ground contact with the principal ground station might occur more than 7 hours after satellite injection. This assumes a good knowledge of the orbital parameters of the satellites, which might not be true for a CubeSat launch. The following recommendations/requirements should be taken into account during the system design:

- Capability to perform autonomous operations after injection for at least [48] hours.
- Emission of a status signal (beacon) easily receivable by the amateur radio community.
- Work with partner ground stations to decrease LEOP duration.

6 Operations and Data Flow

6.1 Operations

SwissCube operations will be performed by a team of students trained by means of a satellite operations system. Operation scenarios will be elaborated and trained by using one of the two SwissCube test benches. The following scenarios will be considered:

- Satellite commissioning
- Nominal satellite operations
- Fault handling.

Training sessions will be performed in an operative environment. This will include real-time simulations where the operations team will try to resolve a system failure by fault assessment and debugging.

6.2 Space & Ground Data System

Design Drivers

The small data rate between space and ground (1.2 kbits/sec) requires the elaboration of a "light" data exchange protocol. Further the design should consider the distributed system architecture of SwissCube.

Approach

For telecommand and telemetry exchanges with SwissCube and mission control a packet service has been elaborated, inspired by the CCSDS [11, 12] and the ESA Packet Utilization Standard [13].

Telemetry data will be generated by the various on-board applications. SwissCube foresees a distributed TM processing where each subsystem/payload will be able to generate TM and route it through the RF system directly to the ground station or it might be buffered in the satellite's CDMS for later retrieval. All telemetry will be stored and processed at mission control for analysis and display.

Telecommand packet data generated at the Mission Operations centre will be uplinked to SwissCube, received by the RF receiver and directly routed to the destination subsystem or placed in the CDMS scheduler.

For the RF ground-to-space data link, the AX.25 has been selected. The end-to-end packet protocol is defined in [14].

6.3 Link Budget

The communication architecture between the ground and space systems as currently defined features a low data rate continuous downlink transmission (beacon) and a high data rate switched on by command from the ground uplink and downlink transmission (data link). Both communication budgets are discussed in this section.

To generate the link budget, the "AMSAT/IARU Annotated Link Model System" by Jan. A. King was used.

6.3.1 Downlink 437.5 MHz (Data)

The following assumptions were made [15]:

- For the slant range a worst case orbital altitude of 1000 km was assumed with a minimum elevation of 10°.
- A transmit power of 0.7 Watts was assumed with an antenna gain of 3dBi on the satellite side (power amplifier ~25% efficient).
- The downlink antenna being a dipole there is a theoretical zero in the axis of the antenna. For the antenna pointing loss, 3dB were assumed.
- The satellite's antennas being linearly polarized a circular polarization was assumed on ground, the worst case loss is 3dB.
- The chosen modulation scheme is FSK: For a bit error rate of 10^{-5} an Eb/No of 13.8dB is required.
- Maximum data rate is 1'200 bit/sec.
- A stack with 4 Yagis for downlink was assumed at the ground station yielding a conservative gain of 19dBi.

6.3.2 Uplink 145.8 MHz (Data)

The same assumptions were made for the Uplink with the following exceptions:

- Transmitted power: 20W.
- Double Yagi stack: 15.4 dBi.
- On the satellite side, a monopole with a gain of 2.2 dBi was assumed.

Table III-6 summarizes the generated TM/TC budget. For a 400 km the path losses decrease by about 6dB. The downlink budget show close to 10 dB margin, but further refinements need to be done in this area.

SwissCube Project
Downlink Telemetry Budget

Parameter:	*Value:*	*Units:*
Spacecraft:		
Spacecraft Transmitter Power Output:	0.7	watts
In dBW:	-1.5	dBW
In dBm:	28.5	dBm
Spacecraft Total Transmission Line Losses:	1.5	dB
Spacecraft Antenna Gain:	3.2	dBi
Spacecraft EIRP:	0.1	dBW
Downlink Path:		
Spacecraft Antenna Pointing Loss:	3.0	dB
S/C-to-Ground Antenna Polarization Loss:	3.0	dB
Path Loss:	155.4	dB
Atmospheric Loss:	1.1	dB
Ionospheric Loss:	0.4	dB
Rain Loss:	0.0	dB
Isotropic Signal Level at Ground Station:	-162.8	dBW
Ground Station (EbNo Method):		
------- Eb/No Method -------		
Ground Station Antenna Pointing Loss:	0.6	dB
Ground Station Antenna Gain:	19.0	dBi
Ground Station Total Transmission Line Losses:	1.8	dB
Ground Station Effective Noise Temperature:	557	K
Ground Station Figure of Merrit (G/T):	-10.3	dB/K
G.S. Signal-to-Noise Power Density (S/No):	54.9	dBHz
System Desired Data Rate:	1200	bps
In dBHz:	30.8	dBHz
Telemetry System Eb/No for the Downlink:	24.1	dB
Demodulation Method Seleted:	Non-Coherent 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:	13.8	dB
Eb/No Threshold:	14.8	dB
System Link Margin:	**9.3**	dB
Ground Station Alternative Signal Analysis Method (SNR Computation):		
---------- SNR Method ------------		
Ground Station Antenna Pointing Loss:	0.6	dB
Ground Station Antenna Gain:	19.0	dBi
Ground Station Total Transmission Line Losses:	1.8	dB
Ground Station Effective Noise Temperature:	557	K
Ground Station Figure of Merrit (G/T):	-10.3	dB/K
Signal Power at Ground Station LNA Input:	-146.3	dBW
Ground Station Receiver Bandwidth (B):	2,000	Hz
G.S. Receiver Noise Power (Pn = kTB)	-168.1	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:	21.9	dB
Analog or Digital System Required S/N:	14.8	dB
System Link Margin	**7.1**	dB

SwissCube Project NOTE:
Uplink Command Budget

Parameter:	*Value:*	*Units:*
Ground Station:		
Ground Station Transmitter Power Output:	20.0	watts
In dBW:	13.0	dBW
In dBm:	43.0	dBm
Ground Stn. Total Transmission Line Losses:	6.0	dB
Antenna Gain:	15.4	dBi
Ground Station EIRP:	22.4	dBW
Uplink Path:		
Ground Station Antenna Pointing Loss:	0.5	dB
Gnd-to-S/C Antenna Polarization Losses:	3.0	dB
Path Loss:	145.8	dB
Atmospheric Losses:	1.1	dB
Ionospheric Losses:	0.7	dB
Rain Losses:	0.0	dB
Isotropic Signal Level at Spacecraft:	-128.7	dBW
Spacecraft (Eb/No Method):		
------- Eb/No Method -------		
Spacecraft Antenna Pointing Loss:	3.0	dB
Spacecraft Antenna Gain:	2.2	dBi
Spacecraft Total Transmission Line Losses:	1.8	dB
Spacecraft Effective Noise Temperature:	268	K
Spacecraft Figure of Merrit (G/T):	-23.9	dB/K
S/C Signal-to-Noise Power Density (S/No):	79.0	dBHz
System Desired Data Rate:	1200	bps
In dBHz:	30.8	dBHz
Command System Eb/No:	48.2	dB
Demodulation Method Seleted:	AFSK/FM	
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:	23.2	dB
Eb/No Threshold:	24.2	dB
System Link Margin:	**24.0**	dB
Spacecraft Alternative Signal Analysis Method (SNR Computation):		
---------- SNR Method ------------		
Spacecraft Antenna Pointing Loss:	3.0	dB
Spacecraft Antenna Gain:	2.2	dBi
Spacecraft Total Transmission Line Losses:	1.8	dB
Spacecraft Effective Noise Temperature:	268	K
Spacecraft Figure of Merrit (G/T):	-23.9	dB/K
Signal Power at Spacecraft LNA Input:	-131.3	dBW
Spacecraft Receiver Bandwidth:	2,000	Hz
Spacecraft Receiver Noise Power (Pn = kTB)	-171.3	dBW
Signal-to-Noise Power Ratio at G.S. Rcvr:	40.0	dB
Analog or Digital System Required S/N:	15.8	dB
System Link Margin	**24.2**	dB

Table III-6: Link Budget for the SwissCube TM/TC data link.

6.3.3 Downlink (Beacon)

Again, the Beacon link budget used the "AMSAT/IARU Annotated Link Model System" by Jan. A. King. This budget assumes 0.09 W of RF transmitted power and a spacecraft antenna gain of 3.2 dBi. This budget takes into account of different losses in transmission lines and the misalignment of the antennas.

The beacon modulation is the standard CW Morse code.

Table III-7 shows at least a 10dB margin for the beacon link, with the stated assumptions.

SwissCube Project Downlink Telemetry Budget		
Parameter:	*Value:*	*Units:*
Spacecraft:		
Spacecraft Transmitter Power Output:	0.09	watts
In dBW:	-10.5	dBW
In dBm:	19.5	dBm
Spacecraft Total Transmission Line Losses:	1.5	dB
Spacecraft Antenna Gain:	3.2	dBi
Spacecraft EIRP:	-8.8	dBW
Downlink Path:		
Spacecraft Antenna Pointing Loss:	3.0	dB
S/C-to-Ground Antenna Polarization Loss:	3.0	dB
Path Loss:	155.4	dB
Atmospheric Loss:	1.1	dB
Ionospheric Loss:	0.4	dB
Rain Loss:	0.0	dB
Isotropic Signal Level at Ground Station:	-171.6	dBW
Ground Station (EbNo Method): ***------- Eb/No Method -------***		
Ground Station Antenna Pointing Loss:	0.6	dB
Ground Station Antenna Gain:	19.0	dBi
Ground Station Total Transmission Line Losses:	1.8	dB
Ground Station Effective Noise Temperature:	5704	K
Ground Station Figure of Merrit (G/T):	-20.4	dB/K
G.S. Signal-to-Noise Power Density (S/No):	35.9	dBHz
System Desired Data Rate:	14	bps
In dBHz:	11.5	dBHz
Telemetry System Eb/No for the Downlink:	24.4	dB
Demodulation Method Seleted:	CW OOK	
Forward Error Correction Coding Used:	None	
System Allowed or Specified Bit-Error-Rate:	-	
Demodulator Implementation Loss:	0	dB
Telemetry System Required Eb/No:	9.6	dB
Eb/No Threshold:	9.6	dB
System Link Margin:	**14.8**	dB

Table III-7: Beacon link budget.

IV SPACE SYSTEM DESIGN

1 System Architecture Choices and Threads

The current space system design is the result of a few concept iterations, system and subsystem trade-off options, done during Phase A, and of the more detailed design performed during Phase B. The rationale for the system decisions are summarized here and can be found in more details in referenced SwissCube reports.

The Space System is designed to have an operating lifetime greater or equal to 4 months, including the commissioning and nominal science operations.

1.1 Critical Functions and Reliability

The philosophy in the development of SwissCube is to implement functional redundancy in the basic systems that satisfy the Mission Objective 2, and a robust design but not necessarily redundant (to save mass and reduce complexity) to the systems necessary to satisfy the Mission Objectives 3 (defined in Section II-1). This philosophy has driven architecture and design choices. A criticality criterion was elaborated, which definition and implementation in the requirements follow.

Table IV-1 shows the criticality of different on-board functions. A compromise between robustness and redundancy has been chosen.

Criticality	Function	Criteria
High	Downlink	Satisfies primary mission objective
Medium	Uplink	Required for satellite operation & debugging
Low	Platform & payload functions	Required for secondary mission objective

Table IV-1: Mission critical function of the space system.

These design constraints lead to the two following requirements at the Space System level:

3_SSR_10_01 **Mission critical functions**

The Space System critical functions shall be:

1) Launch survival;
2) Basic housekeeping data transmission.

This requirement is the basis for fulfilling the 2nd mission objective

3_SSR_10_02 **Non mission critical functions**

The Space System shall ensure success of the following mission non-critical functions:

1) Science payload operation;
2) Advanced housekeeping data transmission.

This is required to fulfil the 3rd mission objective

The implementation of the mission's critical functions is especially applicable to the Electrical Power System (EPS), Beacon Signal, fabrication processes and structural design to survive launch.

Basic reliability considerations start with EPS for which partial redundancy and robustness have been implemented to maximize reliability. Redundancy is achieved by having separate batteries, charge and discharge circuits and solar cells. Robustness is achieved by the simplicity of the system that does not require any programmable controller.

The downlink communication architecture relies on three different paths from simplest to most complex:

1) A basic hardware beacon residing on the EPS board generates a simple signal which allows identifying the satellite from ground. This signal is then directed to the Beacon board where it is amplified and transmitted. This mode will be operated in case of an EPS microcontroller failure;

2) A more complex beacon signal message is generated by the EPS microcontroller (software). Again the signal is amplified and transmitted by the Beacon board. This signal can include status parameters of the satellite, such as bus voltages and temperatures. A hardware switch selects between both signals.

3) In case of a failure of the RF Beacon hardware, the signal can be sent via the COM board using the satellite's receiver and transmitter system.

However, the RF switch, used for downlink, still remains as single point of failure. By default this RF switch allows transmissions of the Beacon signal.

Figure IV-1 shows the reliability block diagram of the satellite's most critical functions, i.e. downlink of information.

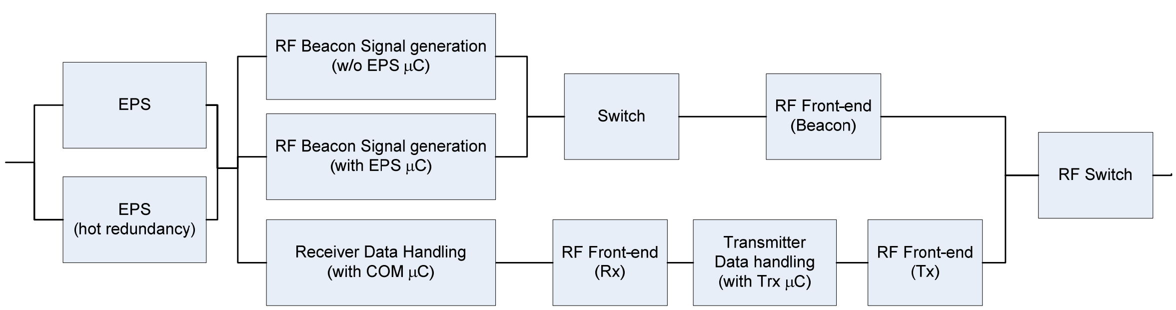


Figure IV-1: Critical function reliability block diagram.

1.2 On-board Data Distribution and Processing Thread

Mostly due to the project's organization, where responsibility of the development of each subsystem resides in laboratories spread over Switzerland, a distributed architecture was chosen for data processing. Thus each subsystem has its own board and own micro-controller. This architecture is well adapted for fabrication and test of each subsystem independently. The CDMS has an additional on-board computer (OBC) with greater capabilities. The micro-controller is the same for each board.

Figure IV-2 shows a simplified schema of the information flow between the various subsystems (functions). CDMS as the main data source/sink is identified. The CDMS provides various services to the other subsystems and contains the on-board scheduler. A peripheral information connection exists between the RF receiver and the EPS allowing for recovery in case of the failure of the CDMS system.

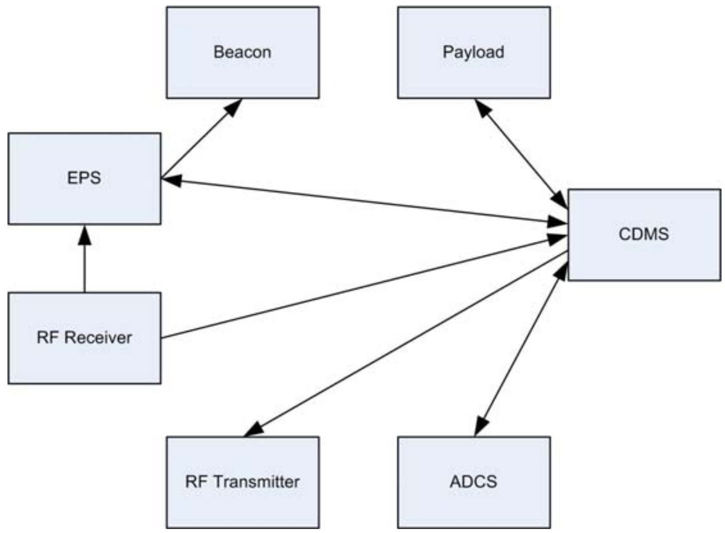


Figure IV-2: Information flow between the different subsystems.

Science and housekeeping data generated by each subsystem is sent upon request to the CDMS OBC via an I2C bus. The data is sorted, dated and stored or used for further processing.

As soon as the RF receiver receives a signal (a command) from the ground, the EPS microcontroller "turns off" the beacon, switches the main RF transmitter on and commands are received by the CDMS and telemetry is sent by CDMS (in nominal mode).

Figure IV-3 shows the baseline bus layout which is based on a distributed bus system. This architecture simplifies the software development and the cabling inside the satellite. The COM, Payload, ADCS, and EPS subsystems all have a MSP430F1611 microcontroller, while the CDMS has an ATMEL ARM AT91M55800A OBC. However, since this microprocessor has no hardware I2C capability, it was decided that it would be linked by an SPI data bus to a MSP430F1611 that would be used as an I2C-SPI bridge.

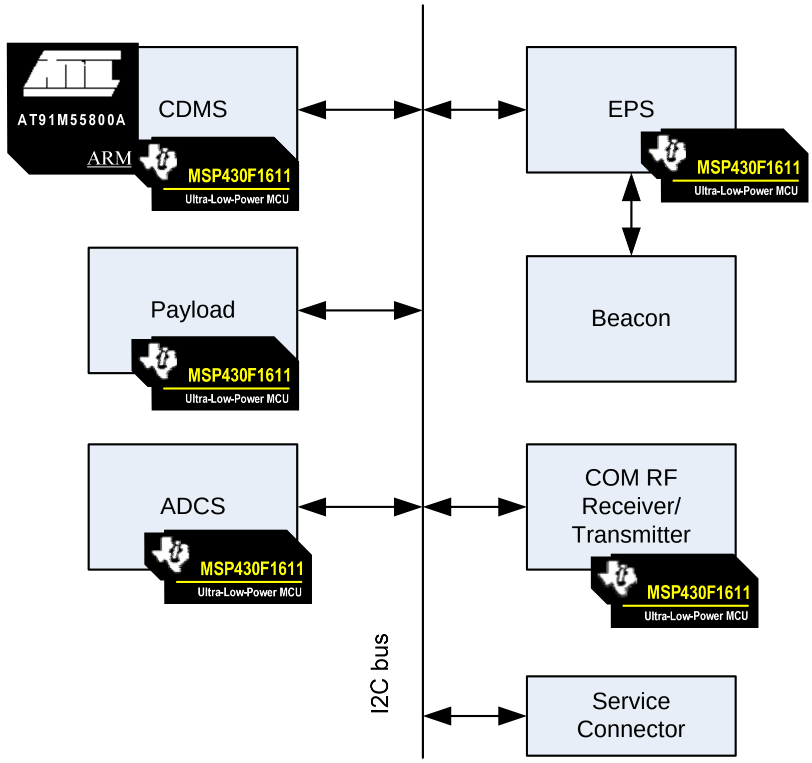


Figure IV-3: Baseline bus layout.

1.3 ADCS Thread

The requirements for the ADCS subsystem are driven by the science requirements (listed in II1.4).

The baseline design for the instrument is an optical sensor with 18.8° x 29.4° field of view (FOV) and 188*120 pixels giving each pixel a field of view of 0.16°. The sensor is mounted on the +x-axis of the satellite which is in orbit at 400-1000 km altitude. The attitude determination algorithm of the satellite has to determine the position of the captured image with a precision of ±[5]° in latitude, ±[7.5]° in longitude (which can be correlated to the solar local time at zenith) and ±[700] km in altitude to guarantee that the limb is within the FOV. If an accuracy of 5° is assumed on the orbital position of the satellite, the required accuracy for attitude determination is therefore listed as ±[12]° along the satellites pitch and yaw axis, and no requirement along its roll axis (which is aligned with the sensor).

The integration time ("shutter speed") is listed as 0.05 s for limb shots and [4] s for zenith shots which provide a good SNR with acceptable smearing (1 pixel for limb measurements, 5° for zenith measurements). This requires a pointing stability of the satellite during science observations of ± 3°/s for limb measurements and ± [1.25]°/s for zenith measurements. In both cases this requirement is valid for all three axes.

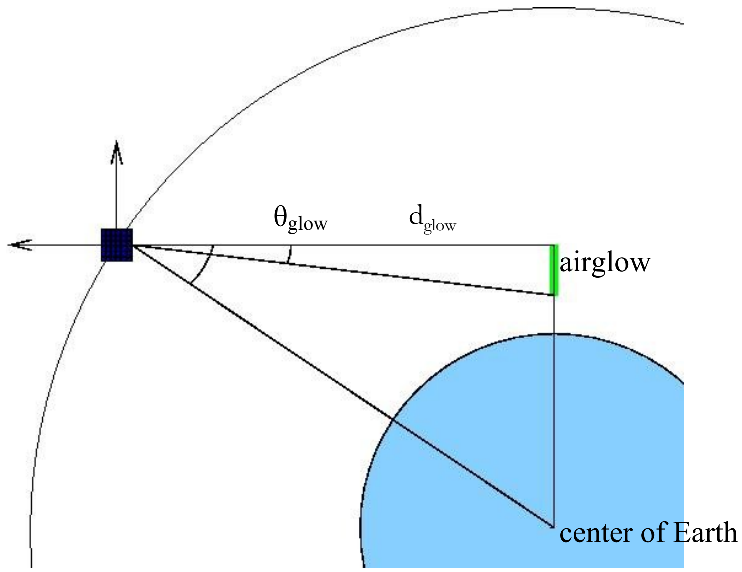


Figure IV-4: Illustration of limb measurements.

The design implementation of the ADCS system to achieve the pointing accuracy and stability requirements is shown in Figure IV-5. The determination is the most important point, as knowledge of the location where the payload is pointing is needed to characterise the nightglow phenomenon. The major issue of the control is to reduce the spinning rate of the satellite after the launch and maintain a relatively low spinning rate.

The Determination and Control algorithms as well as the Orbit Propagator and Earth Magnetic Field Model run on the CDMS OBC (which is the main computer of the SwissCube). The ADCS microcontroller itself is in charge of the sensor readings and actuators control.

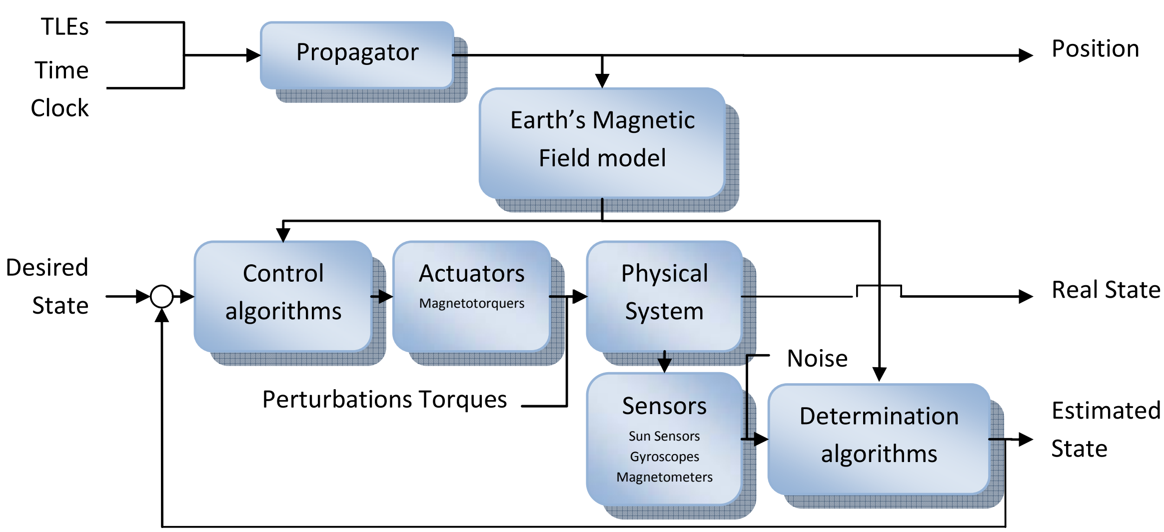


Figure IV-5: ADCS functional diagram.

The sensors used for the ADCS system are:

- A 3-axis magnetometer (MM) to measure the Earth's magnetic field (EMF) intensity and direction
- A 3-axis gyroscopes (GYR) to measure the spinning rate for each axis
- 6 Sun sensors (SS) to find the direction of the sun
- Temperature sensors to compensate the temperature drift of the other sensors

The actuators are:

- 3 magnetotorquers (or coils; MT) to produce a torque thanks to their interaction with the Earth's magnetic field.

Current simulations do not take into account the sun sensors' information and show that the control and determination algorithms maintain the required attitude, but the simulations do not yet take into account all perturbations. Further work is needed in this area.

The estimated error budget at this point is summarized in Table IV-2. The coordinate frame is a geocentric Cartesian coordinate with X along the velocity vector, Z pointed toward the centre of the Earth, and Y the complement. These errors still need to be translated into actual errors on the latitude and longitude of the satellite at the time of the picture.

Error source	Source	Axis 1	Axis 2	Axis 3
Orbital position	Propagator	ΔLat < +/- 5° after 24hr	ΔLon < +/-2° after 24hr	ΔH < ?
Magnetic field	Magnetic Field Model	X: 200nT/2000nT	Y: 200nT/20000nT	Z: 200nT/2000nT
Magnetometers	3 axes Hall sensor (AK8970N)	X: 650nT/60000nT	Y: 650nT/60000nT	Z: 650nT/60000nT
Sun' Reference	Sun Reference Model	TBD	TBD	TBD
Sun Sensors	DTU	< 1 deg.	< 1 deg.	< 1 deg.
Gyroscopes	InvenSense IDG-300	0.305 deg/s	0.305 deg/s	0.305 deg/s

Table IV-2: Estimated errors on the ADCS hardware and models.

The information from each sensors at the time at which the picture was taken will be transmitted back to Earth along with each image. This will provide data on the ground to reconstruct the knowledge of the attitude of the satellite at the moment the picture was taken.

1.4 Satellite Modes of Operation

The reference for the satellite mode will be the EPS microcontroller, which will communicate the current mode of operation to the other subsystems regularly through an I2C ping. The modes are summarized in Table IV-3.

Mode		Occurrence
Initialization		Satellite initialization. Occurs every time the EPS subsystem is powered up.
Safe		Mode that occurs after Initialization, after a critical failure or when the power is low. The SwissCube needs a telecommand from the Ground Station to exit this state. This mode will also be used during the commissioning phase.
Recovery		Mode that occurs only when power is extremely low. It stays in this mode until the satellite has enough power to go in safe mode.
Nominal	Nominal Science	This mode is used when the satellite is taking science data in nominal operations.
	Nominal COM	This mode is used when the satellite is communicating with the Ground Station in nominal operations.
	Nominal Idle	This mode is used when the satellite is in nominal operations but not taking science data or communicating with the Ground Station.

Table IV-3: Swisscube modes description.

The status of all subsystems during each mode is given in Table IV-4. HBM and SBM stand respectively for Hardware Beacon Message and Software Beacon Message, and are detailed in [16].

Mode		EPS	Beacon	COM Rx	COM Tx	CDMS	ADCS	PL
Initialization			**HBM**					
Safe			**SBM**					
Recovery			**SBM to 0**					
Nominal	Nominal Science		**SBM**					
	Nominal COM		**SBM**					
	Nominal Idle		**SBM**					

Table IV-4: Subsystem state during Swisscube modes.

Active: powered up and executing commands.
Standby: powered up but not doing anything.
Inactive: not powered up or disabled.

1.5 Satellite Reference Point (SRP) and Frame (SRF)

The satellite reference frame (SRF) is provided in Figure IV-6. In this right handed frame the payload aperture shall be oriented towards +X. The Z axis shall be parallel to the structure rails with the motherboard perpendicular to +Z, and the antenna deployment system shall be located in -Y. The satellite reference point (SRP) shall be in the geometrical center of the "cube".

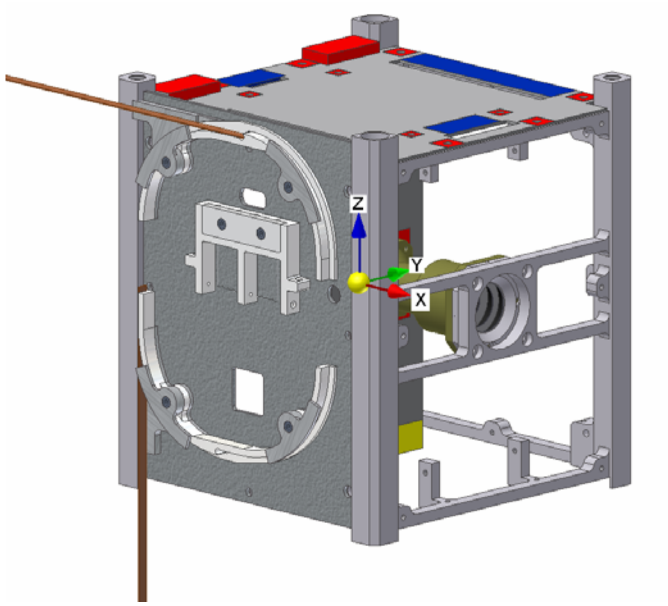


Figure IV-6: Satellite reference frame.

2 System Block Diagrams

2.1 Product Tree

The analyses of the required functionalities of the space system have led to the product tree, arranged by subsystem, shown in Figure IV-7.

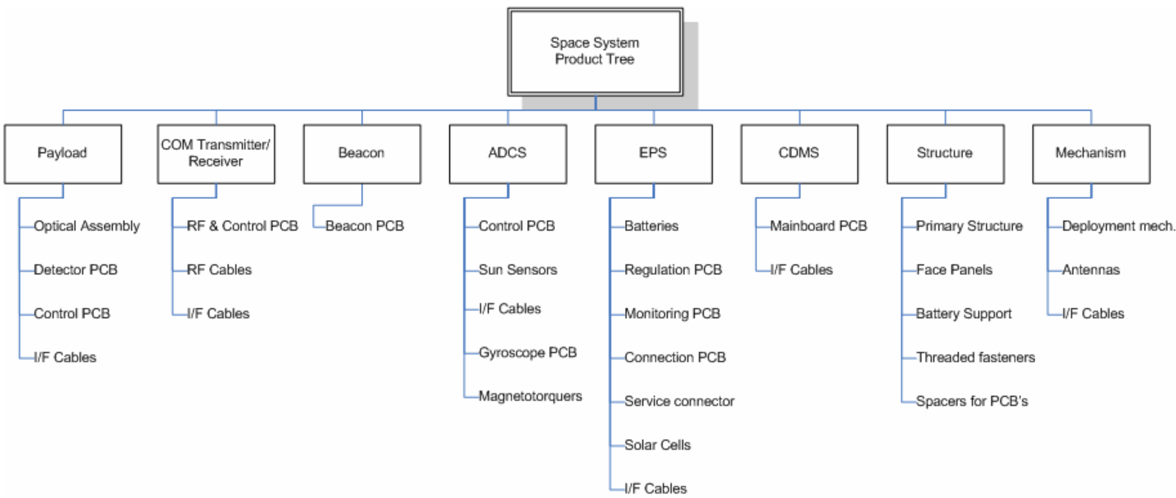


Figure IV-7: Space System Product Tree.

2.2 Electrical Block Diagram

Figure IV-8 shows the electrical block diagram of the satellite with the electrical interfaces. For a detailed version please consult the reference [17].

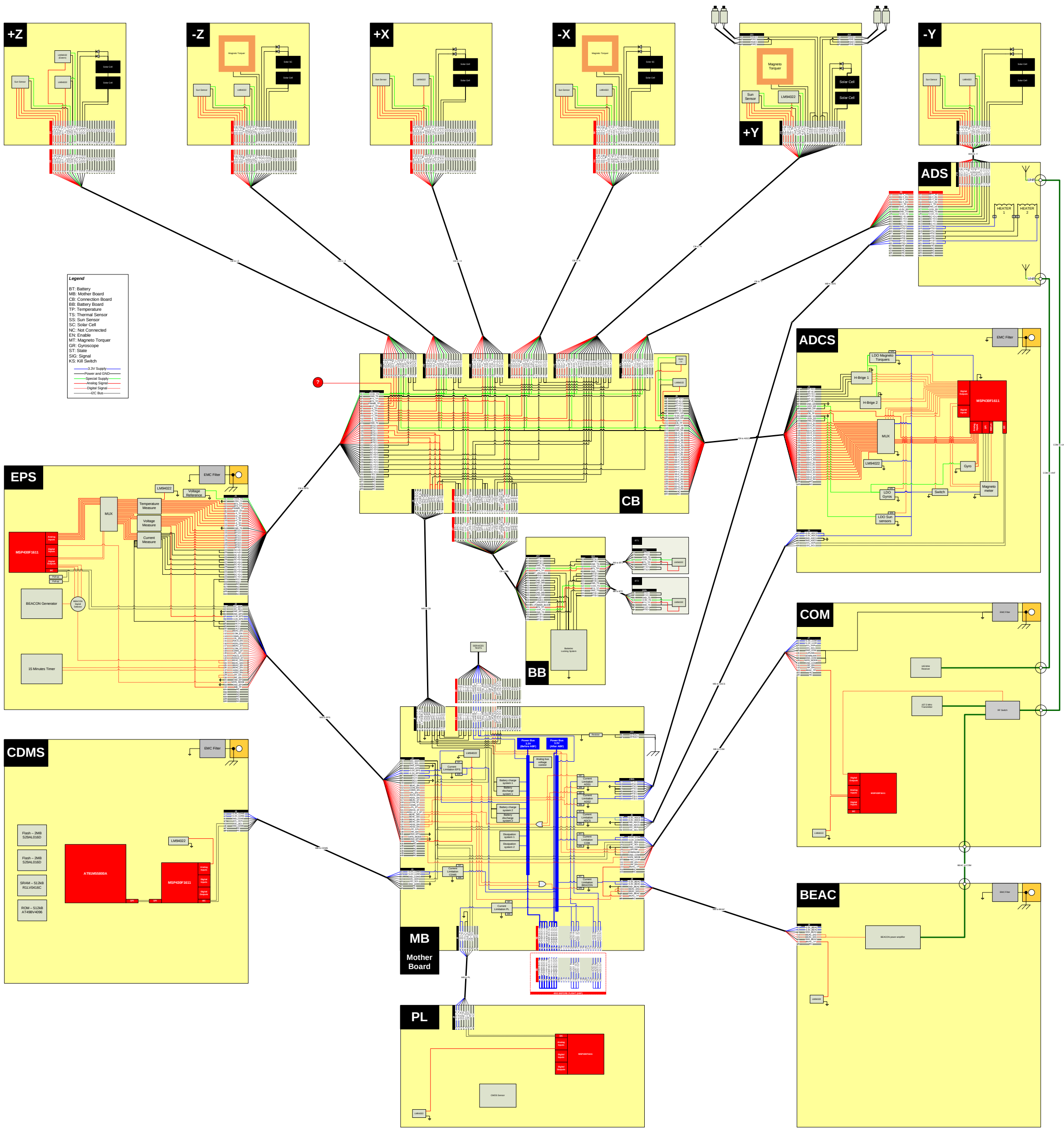


Figure IV-8: Electrical Block Diagram.

3 System Budgets

3.1 Mass Budget

The CubeSat specifications state a 1kg maximum mass allocation. The mass budget is based on the work done during Phases A and B. It was updated during the whole Phase B. After Phase A, the overall mass of the SwissCube was 913 grams. It means a margin of about 10%. During Phase B it was possible to refine the estimation made before, because structural models of the hardware were available. Table IV-5 shows the mass estimates of each subsystem. The total is about 886 grams, which means a margin of about 13%.

Subsystem	Mass [g]
Structure & Configuration	274
EPS	198
ADCS	115
CDMS	48
Payload	67
COM	58
Mechanism & antenna	21
Thermal	7
Cabling	80
Total	868

Table IV-5: Subsystem mass budget.

3.2 Power Budget

The establishment of the power budget is done using many assumptions. The worst case in term of orbit duration is considered and a 30 % margin is book kept.

3.2.1 Power production profile

The mean power production over the worst case orbit (400 km) was calculated using the STK simulator and Simulink. The incidence angle on each face is calculated with Simulink/MatLab taking different vectors from STK. With a basic model of the SwissCube, the incidence angle of the solar rays on each satellite face can be computed. The results of each face are summed and the power production profile over the orbit can be determined. The average over the orbit is then calculated. Figure IV-10 shows the power production in function of time.

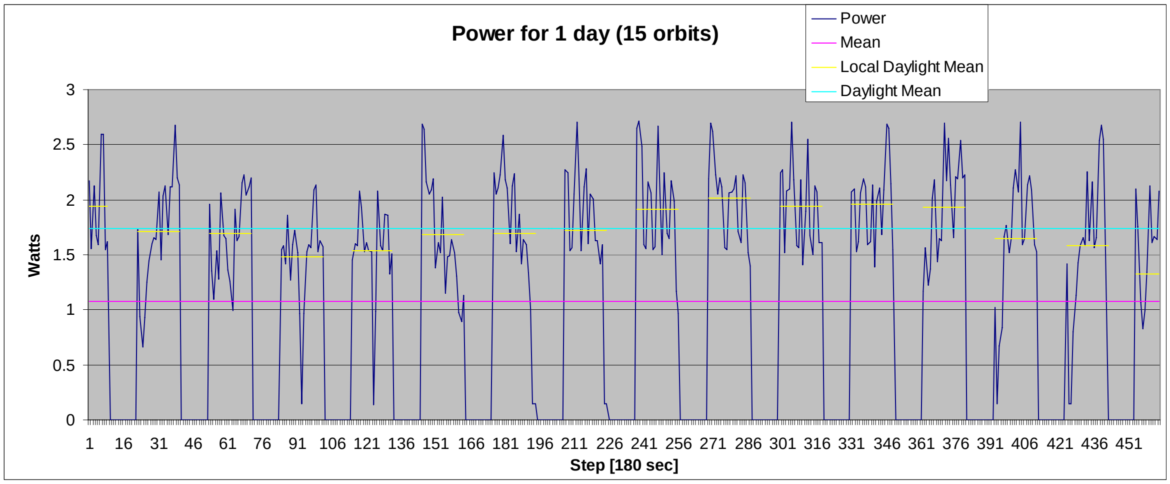


Figure IV-10: Power production as a function of time with mean value.

During daylight, the mean produced power is 1740 [mW].

3.2.2 Power and energy balance

To calculate the total amount of energy (mWh) needed for one orbit around the Earth, assumptions were made. Some of them come from the projects requirements and others will define new system requirements. These assumptions are listed below.

- One orbit is 92.6 minutes long with 36 minutes in eclipse.
- RF reception is always on and needs **80** mW.
- RF data transmission needs **3000** mW. In one day, for one ground station we have on average **2** opportunities to communicate with an average time of **6** minutes each.
- ADCS Magnetotorquers are always in use and need **160** mW (total for the three).
- The ADCS controller consumption calculated with its real "on time" is **10** mW. The sensors need **50** mW.
- The beacon sends a message every 30 seconds and is stopped during the main RF transmission.

  The peak power [**300**] mW is considered but during the real time emitting. The software message has 3 parts and we consider the worst case that is the word with the most bit set to 1. The emitting time of the worst message is ~5 seconds. So we can consider the beacon is globally in function the 20% of the time.
- The payload takes picture during the eclipse and when possible also during the day time.
- Two loss sources have to be considered. The first one is the chemical yield of the battery and the second is the yield of the converters. These two types of losses affect the charge and the discharge but in the simple model used to establish this power budget the battery charge is not considered. So all the losses have been reported on the battery discharge and **20% losses on the discharge** were considered.

The power, in watts, needed by each subsystem is presented in Table IV-6.

Loads			
Subsystem	Value	Unit	Remark
EPS	50	mW	
Payload	450	mW	
CDMS	150	mW	
Beacon	300	mW	
ADCS control	10	mW	
ADCS sensor	50	mW	
ADCS Magnetotorquers	160	mW	
Main RF control & receiver	80	mW	
Main RF transmitter	3000	mW	

Table IV-6: Power required per subsystem.

As the main RF transmitter power is large, transmission will only occur during the day. Also, as attitude determination errors grow as a function of time after a communication with the ground station (at which point, the position is updated), images will be taken shortly after a pass.

In the nominal mode of operation, there are 6 different possible power consumption states: 4 during daylight and 2 during the eclipse. In each part of the orbit the satellite can be transmitting or not and taking pictures or not. The different states are listed in Table IV-7 with their corresponding energy consumption.

Abr.	Power consumption state	Consumed energy
DTS	Daylight WITH transmission and WITH science	1177 mWh
DTnS	Daylight WITH transmission but WITHOUT science	1172 mWh
DnTS	Daylight WITHOUT transmission but WITH science	757 mWh
DnTnS	Daylight WITHOUT transmission and WITHOUT science	752 mWh
EnTS	Eclipse WITHOUT transmission but WITH science	607 mWh
EnTnS	Eclipse WITHOUT transmission and WITHOUT science	600 mWh

Table IV-7: Power consumption states in the nominal mode of operation.

In order to have to have power modes for complete orbits, the different cases above have to be combined. The combination gives 8 (4x2) different power states. Table IV-8 shows the sum of the different states (unit mWh).

	EnTS	EnTnS
DTS	1'784	1'777
DTnS	1'779	1'772
DnTS	1'364	1'357
DnTnS	1'359	1'352

Table IV-8: Power consumption combinations in mWh per orbit (includes 30% margin).

The 8 different combinations are list in Table IV-9. Once the satellite is in nominal mode of operation, it will be in one of these power consumption combinations.

The assumed scenario is that the satellite will take pictures once a day during the eclipse and that during the remaining twelve orbits, no science and no transmission will occur. The energy consumed for each orbit is summed, so the total consumption per day is known.

As can be seen in Table IV-9 the total power consumed in one day is less than the total amount of produced energy during that day.

Phase	Number of orbits	Energy/phase	Energy
DTS EnTS		1'784	0
DTS EnTnS		1'777	0
DTnS EnTS		1'779	0
DTnS EnTnS	2	1'772	3'544
DnTS EnTS		1'364	0
DnTS EnTnS		1'357	0
DnTnS EnTS	1	1'359	1'359
DnTnS EnTnS	12	1'352	16'224
Total			21'127
Power production	15	1'636	24'534

Table IV-9: Power [mWh] consumption scenario over a day (includes 30% margin).

A detailed model of the instantaneous power consumption and battery charge/discharge is being elaborated.

3.3 Data Budget

Space to ground data budget

The main bandwidth limitation is the amount of the data that can be transmitted to the ground. A space to ground data budget was done to control if the SwissCube has enough bandwidth left to manage and download payload images.

With the assumptions stated in Table IV-10, the downlink data budget is summarized in Table IV-11.

Bandwidth :				
Raw bandwidth	1200	bits/sec		
Frame AX.25	30	%	29	octets/frame
Error rate	7	%		
Net bandwidth	781.2			
Ground station :				
Min angle of elevation	15	deg		
Mean pass duration at 400 km	4.7	min		
Mean pass duration at 600 km	10	min		
SwissCube data :				
HK RT Size	200	octets	1600	bits
HK AR Size	200	octets	1600	bits
Payload image size	28200	octets	225600	bits
Payload image size + CCSDS	32296	octets	258368	bits

Table IV-10: Assumptions for the space to ground data budget.

This budget shows that bandwidth is a problem if the spacecraft is launched at 400 km. In that case, it might be difficult to fulfil the payload requirement (one image each 4.5 days). With a spacecraft at 600 km there are no problems to fulfil the requirements.

I²C Bus Data Budget

The space data bus is an I²C bus at 100 kilobits per second to exchange data between subsystems. The transferred data includes the housekeeping of all subsystems, time synchronization, the internal functions calls (contain telecommands received and telemetry to be sent to the ground) and payload data.

During transmission with the ground, the speed of the RF link represents only about 5% of this bandwidth in a very worst case scenario: RF full-duplex and two hops transfers to destination slave subsystem only (COM to master then master to slave or inversely).

The rest of the data, excluding payload, accounts only for 10% of the I²C bandwidth; once more with a worst case scenario taken very large: housekeeping of 200 octets of data at 5 Hz.

This leaves more than enough room for the payload's data. With the bandwidth left (85%), an image could still be transferred in about 3 seconds from the payload to the CDMS. And that without taking into account that the payload will never operate at the same time as the RF link due to power limitation.

The bandwidth of the I²C is thus sufficient in the worst case scenarios and can even easily handle big burst of data such as retrieving images from the payload.

	400 km		**600 km**	
Pass per day	1	2	1	2
Pass duration [min]	4.7	2.3	10.0	5.0
Pass duration [sec]	280	140	600	300
RF initialization [sec]	30	30	30	30
Load Keplerian elements [sec]	5	5	5	5
Load scheduler [sec]	30	30	30	30
Data time available [sec]	215	150	535	470
Total bits available [bits]	167'958	117'180	417'942	367'164
HK RT Frequency [sec]	30	30	30	30
HK RT bandwidth [bits]	14'933	14'933	32'000	32'000
HK RT bandwidth [%]	8.9%	12.7%	7.7%	8.7%
HK AR Frequency [sec]	3600	3600	900	900
HK AR bandwidth [bits]	38400	38400	153600	153600
HK AR bandwidth [%]	22.9%	32.8%	36.8%	41.8%
Acknowledgment RT [bits]	9600	9600	9600	9600
Acknowledgment RT [%]	5.7%	8.2%	2.3%	2.6%
Acknowledgment Store [bits]	9600	9600	9600	9600
Acknowledgment Store [%]	5.7%	8.2%	2.3%	2.6%
Other [bits]	10240	10240	10240	10240
Other [%]	6.1%	8.7%	2.5%	2.8%
Free Bandwidth [bits]	85185	34407	202902	152124
Free Bandwidth [%]	**50.7%**	**29.4%**	**48.5%**	**41.4%**
Fraction of downloaded payload image	**0.33**	**0.13**	**0.79**	**0.59**
Number of days to download a full image	**3.03**	**7.51**	**1.27**	**1.70**

Table IV-11: Space to ground data budget.

4 Subsystems Description

4.1 Electrical Power Subsystem

4.1.1 Design drivers and functional overview

The satellite's Electrical Power System's task is to provide power to the various subsystems as required by the mission timeline. It will ensure power generation by means of solar cells, energy storage in batteries and regulate the power distribution to the various subsystems.

The main SwissCube electrical load is the Communication System. The RF transmitter consumes approximately 3 Watts.

The payload will take measurements during the eclipse with a power demand of 450 mW but its operation does not consume a lot of energy.

4.1.2 Architectures

Two main architectures have been studied for EPS, the first one including a Maximum Power Point Tracking (MPPT) system the second one without. The best layout for the architecture using MPPT is shown in Figure IV-9. This design contains a maximum redundancy in case of battery or solar cell breakdown. In this case the microcontroller represents a single point of failure and its failure would cause the loss of the mission. The only solution for this would therefore be to create a redundant system with two microcontrollers where each one controls a battery and 5 solar cells.

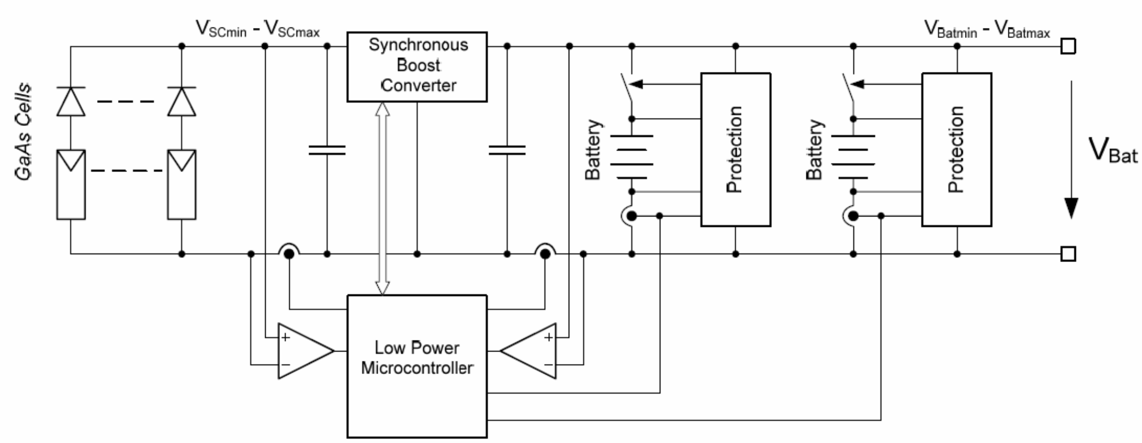


Figure IV-9: EPS architecture with MPPT, the batteries and solar cells are connected in parallel.

A second approach is to use a fixed working point of the solar cell characteristics. Major advantage is that the control of the bus can be done by an analogue circuit, as shown in Figure IV-10, further charge and discharge circuits are redundant, i.e. the loss of one battery will not cause a mission critical failure. This approach has been chosen for the baseline design.

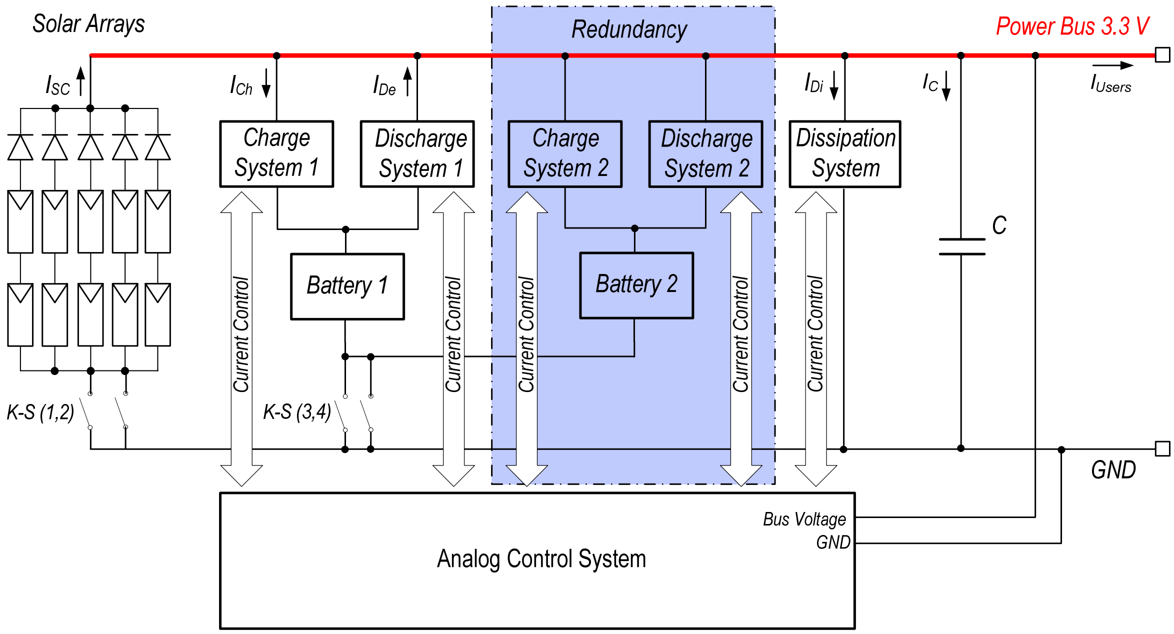


Figure IV-10: EPS architecture without MPPT.

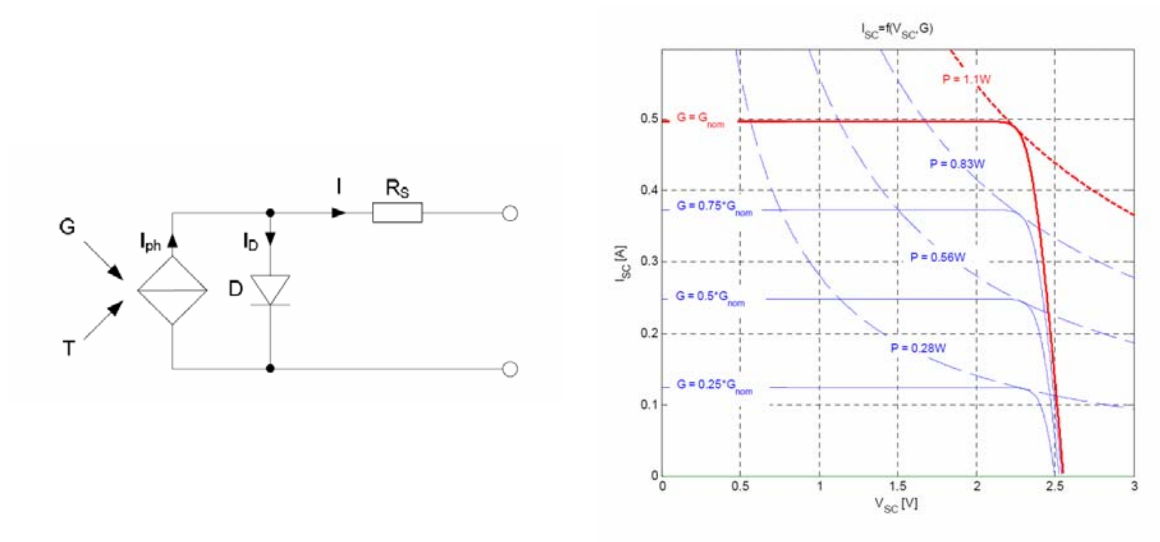
Furthermore, this concept has the advantage that 100% of the energy can be directly transmitted to the users and does not need any microcontroller to operate. To validate this concept and to show that the power losses due to a missing MPPT can be kept low, an analytical model of the solar cells was created which will be discussed hereafter.

4.1.3 Analyses

The satellites small size and power requirements require the use of the most efficient solar cells on the market. GaAs based solar cells have the potential to reach efficiencies greater than 30%. Currently cells with up to 28% efficiency are commonly available. To carry out the electrical power calculation we have initially considered the following cell type: RWE3G-ID2/150-8040[1] which has an average efficiency of 26.6 % (BOL@28°) (see Figure IV-11).

In addition, to analyze the behavior of the solar cell with respect to insolation and temperature an analytical model of the cell behavior has been created.

[1] The company RWE has been renamed to AZUR SPACE Solar Power GmbH and the cell type used for our considerations has been replaced by the type: GAGET2-ID2/160-8040.



(a) (b)

Figure IV-11: Solar cell model. (a) simplified electrical model of the solar cell (b) I-V characteristics as a function of the insolation (G_{norm} = 1350W/m², T=25°)

From the model it can be concluded that that for any insolation and temperature range the MPP does not fall below 2V. Measurements conducted with a xenon "solar simulator" gave a confirmation of the model.

4.1.4 Batteries

Currently Lithium-Ion Polymer batteries are considered for the mission. Their major advantage over Lithium-Ion cells being a higher energy density. The baseline design foresees the use of two PoLiFlex batteries from VARTA[2]. So far bulging has been a major issue with Li-Poly, recent tests at ESA [18] have shown that the PoLiFlex series, in comparison to other Li-Poly batteries, do not suffer from this problem. They further are radiation tolerant and conserve their charge under vacuum conditions.

The chosen DOD for our system is 15% if both batteries are operational and 30% in the case of one operational unit. The low DOD has been chosen to increase life expectancy of the battery.

4.1.5 Baseline design

Based on the considerations presented above, a baseline design without MPPT was chosen (Figure IV-12). The current limiters also serve as switches for the supply of the different subsystems.

[2] Investigated models include VARTA PoLiFlex PLF503759 and PLF423566

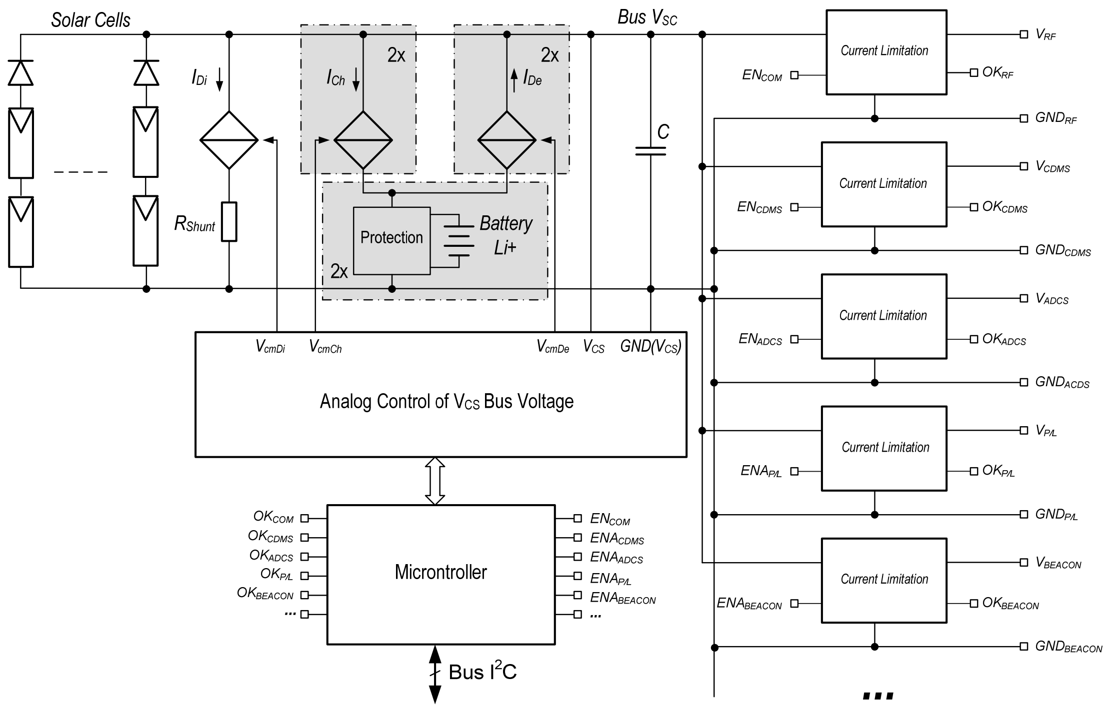


Figure IV-12 : EPS baseline design.

4.1.6 Power Management

The Power is managed by switching on/off the current limiters. This management is done by the EPS microcontroller. This microcontroller is also used for the following functions:

- Making several measures, such as battery voltage, solar cells current and temperature.
- Creating a software signal (beacon) which will replace the hardware beacon generator.
- Switching between the hardware and the software beacon signal.
- Turning ON/OFF the RF amplifier.
- Turning ON/OFF current limiters of subsystems.

In case of microcontroller breakdown, the current limiters of the critical subsystems (EPS, COM, BEACON) are naturally turning on and the beacon signal is generated by a hardware system. The Power Management Board has also a hardware 15 minutes timer in order to switch on the antennas deployment system and the beacon amplifier.

This board has been designed, built and tested (functional tests). Some vibration and shock tests are currently performed.

For the following topics please refer to the corresponding report:

- Satellite Electrical Architecture and Power System: [19]
- Latch up mitigation circuitry: [20]
- Power Management [21].

4.2 Telecommunication Subsystem

4.2.1 Design drivers

The main design drivers for the communication system are the low available power on-board the satellite. A second driver is the capability of satellite debugging and commanding at any rate or attitude.

4.2.2 Baseline design

The block diagram of the telecommunication system is shown in Figure IV-13. The design foresees to operate the beacon constantly and switch to the data transceiver once above a partner ground station.

The data transmitter sends the scientific and engineering telemetry at 1200 bits/sec and the beacon sends only simple housekeeping data at very low speed in Morse code.

The data transmitter uses FSK and the data receiver AFSK. The transmitter power consumption is in the order of 2-3 W mainly due to the power of the power amplifier.

The beacon uses amplitude modulation (OOK) and its power has to be as low as possible. The main RF transmitter and the beacon are using the same antenna; it is why there is a switch to select the source of the antenna. The chosen frequencies are both in the amateur band.

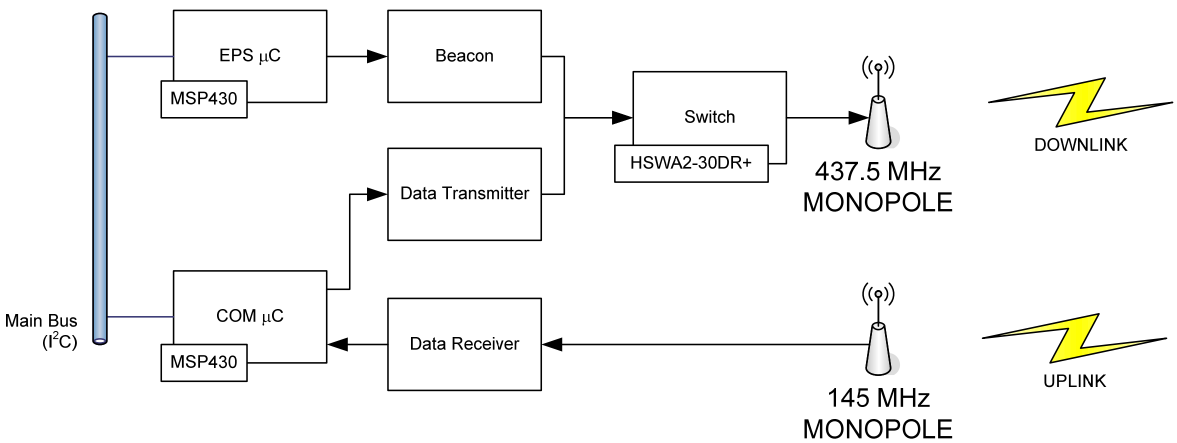


Figure IV-13: Block diagram of the RF system.

<u>Data Transmitter</u>

The architecture used for transmission is shown in Figure IV-14. The modulation is FSK. The data is directly filtered and used to drive the FM modulator. The generated FM signal is then passed through a power amplifier. The power amplifier is capable of transmitting 28dBm (>0.5 W). This is required to satisfy the link budget requirements for BER<10^{-4}. The power amplifier used is RF5110G manufactured by RF micro devices.

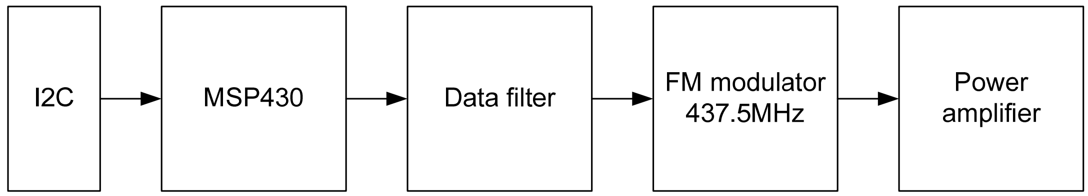


Figure IV-14: Transmitter block diagram.

Data Receiver

The receiver design is based on a dual-conversion receiver architecture, which in a nutshell means the received frequency is down-converted twice before demodulating the message signal from the carrier.

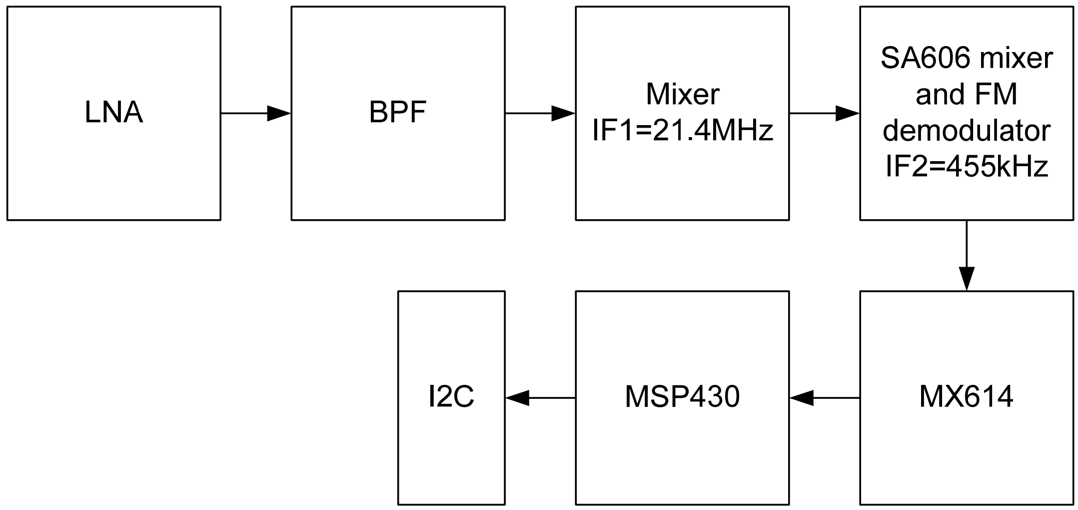


Figure IV-15: Receiver Architecture.

Figure IV-15 shows the block diagram representation of the receiver. The major building blocks of receiver are a low noise amplifier, band pass filter, 1st mixer and final 2nd mixer /demodulator (SA606). The incoming carrier frequency is at 145 MHz, it is passed through a LNA to boost the signal power while removing noise from the incoming signal. The amplified signal is then passed through a passive band pass filter. After which it is down converted to the 1st intermediate frequency (IF1) of 21.4 MHz using 1st mixer and local oscillator. Finally, the message is passed through a SA606 chip, which is a single IC that includes the 2nd Mixer, IF amplifier and the quadrature FM demodulator. The mixer in SA606 converts the incoming signal to 455 KHz (IF2) before being demodulated by the quadrature detector. The CMX469 modem then proceeds to the demodulation of the AFSK signal.

In order to be compatible with the ground station, the modem has also been changed from CMX469 to MX614 which now uses 1200 and 2200 mark and space frequencies, respectively.

Packet Format

The data for the up- and downlink are transmitted in packets. For compatibility with existing and widely used radio amateur equipment, the data are sent using the AX.25 protocol. To reduce the overhead, only the connectionless mode using unnumbered information (UI) frames is supported.

Information and Unnumbered information (UI) Frames:

Start Flag	Address Bits	Control Bits	Information no more than 2048 Bits	CRC Bits	Stop Flag
01111110	112-560 Bits*	8 Bits	8 PID Bits	16 Bits	01111110

Figure IV-16: AX.25 Protocol Packet Format.

To have a maximum flexibility of the information content, the KISS format is used for the information field.

Beacon

The architecture of the beacon subsystem is shown in Figure IV-17. When the beacon is turned on, the oscillator provides continuously the carrier frequency, in our case 437.5 MHz. Then the signal is modulated in amplitude (OOK) and finally amplified. The desired bit rate is 14 bits/s.

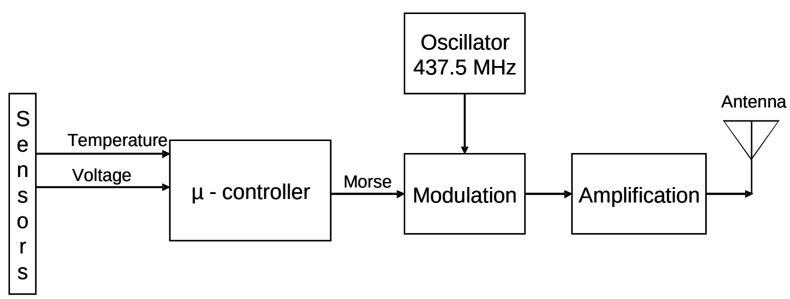


Figure IV-17: Block diagram of the beacon.

The component that controls the carrier generation and modulation is the ATA2745. The ATA2745 is a UHF ASK (OOK) Transmitter.

For the power amplifier, three different A class amplifiers were selected and tested to find the best solution. The three amplifiers are the RF2155, the RF2172 and the MGA-68563.

The ATA2745 was thus tested on the three different prototype boards (see Figure IV-18), and similar results were observed with these three systems. The main purpose of this circuit part was to generate the carrier frequency and to allow modulating this carrier using an ON/OFF input signal. This function has been successfully demonstrated. Further optimizations, such as an increasing of the output power are of course still possible.

The Morse code message was generated using a PicoKeyer and was fed into the ATA2745. The expected ON/OFF modulated at the output of the power amplifier was observed, as shown in Figure IV-19.

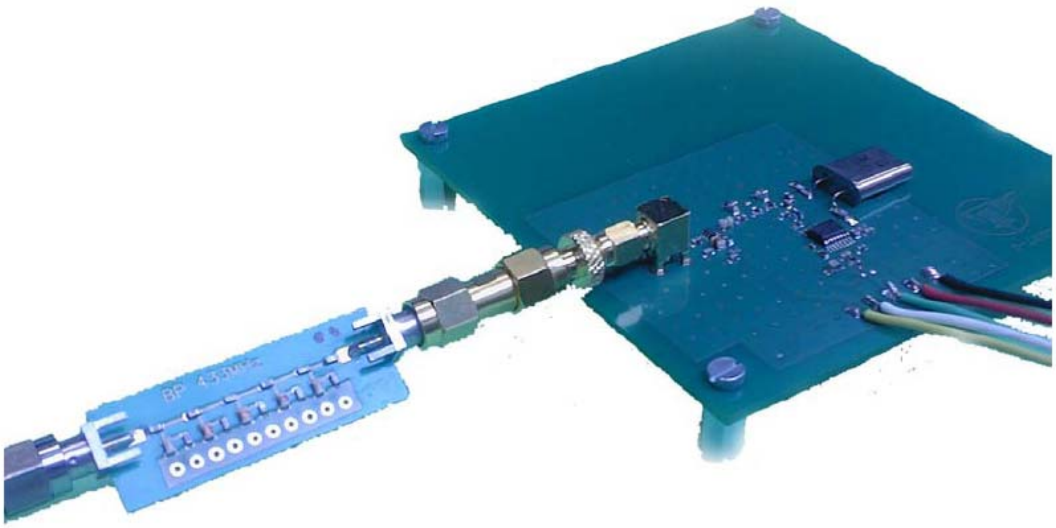


Figure IV-18: Beacon board and test set-up.

Measurements performed on the full system showed good results. The output signal showed the same shape as observed in measurements performed with an external modulator. The output power level was somewhat smaller, due to the input power level provided by the ATA2745, which was smaller than 3 dBm.

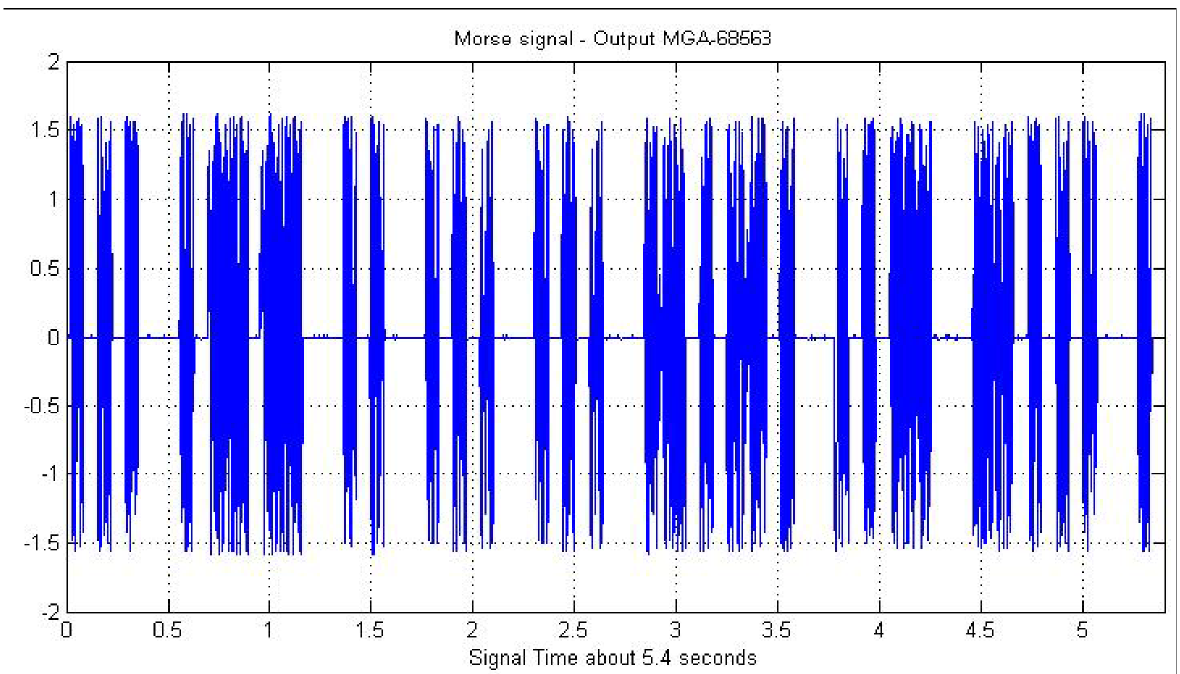


Figure IV-19: SwissCube message output.

Characteristics of the two most promising power amplifiers are given in Table IV-12. In the next phase one of both options will be selected.

	RF2172	MGA-68563
Vcc	3.3V	5V
PowerDown	yes	No
Pout at 3dBm	94.5mW	89.9mw
Consumption ON + Morse	250mW	253mW
Consumption ON	78.5mW	112mW
Consumption OFF	0W	100mW
Efficiency	37.80%	35.50%
Matching with ATA2745	bad	OK

Table IV-12: Beacon power amplifier options.

A well adapted filter placed at the output of the power amplifier should be used for the realization of the final beacon system.

Antennas

Modeling of the antennas length, satellite backplane material and position on the satellite panel was performed and several solutions were analyzed.

In convergence with the Antenna Deployment System design, the chosen antenna configurations include a quarter-wavelength monopole antenna for 145.8 MHz and another one for 437.5 MHz. Figure IV-20 shows the antenna layout and Figure IV-21 the radiation patterns for SwissCube. Both antennas are made of beryllium copper.

Several tests were performed on the antenna deployment system and on the effect of the bending of the antennas on the RF pattern. Results are discussed in section IV -4.8.3.

The VHF antenna is 610 mm long when the antenna is in straight ideal position. The maximal gain is about 2.25 dBi and the return loss (S11) goes from -15.3 dB in the first case to -14 dB in the second (3 % of power is reflected and therefore 97 % is transferred to the antenna).

The UHF antenna of 176 mm when the antenna is in straight ideal position and 181 mm when in bent position. The gain is 3.15 dBi for the first case and 3.65 dBi for the second case whereas the S11 parameter is -16.44 dB and, respectively, of -14.5 dB (3 – 5 % of power is reflected and 95 – 97% is transferred to the antenna). The final design features a length of 176 mm.

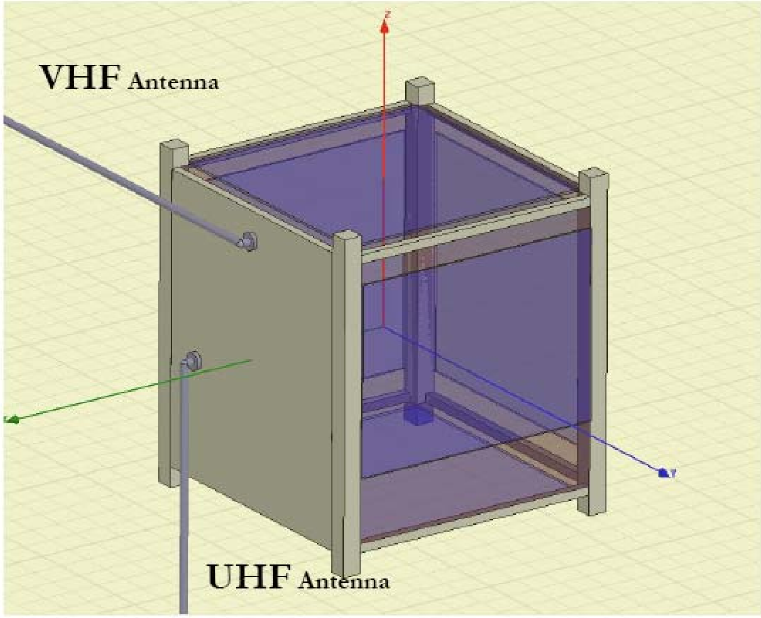
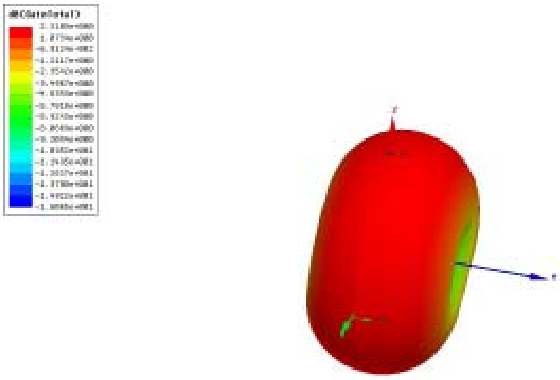
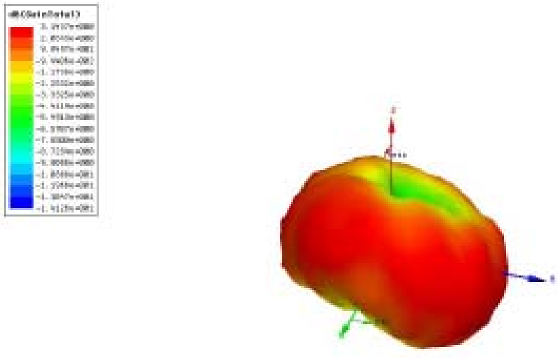


Figure IV-20: SwissCube antenna layout.



VHF 3D Polar Antenna Gain



UHF Polar Antenna Gain

Figure IV-21: Radiation pattern for the antenna baseline design.

For further information please refer to:

- SwissCube RF Beacon Design: [22]
- Telecom System Engineering [32]

4.3 Science Instrument

4.3.1 Design Drivers

4.3.1.1 Functional requirements

The payload of the SwissCube satellite is a telescope which takes images of the airglow and demonstrates a novel technology that may be used for a new earth sensor (ES). It shall satisfy the following conditions:

- The payload observes the airglow band at 762 nm, with a resolution between [10] and [40] nm.
- The payload has a spatial resolution of [0.3]° and a field of view (FOV) of [25]°.
- The payload survives a sun pointing attitude, with a permanent damage or performance degradation of less than [20]% for an exposure time of at least 10h.
- The payload is able to perform the science mission with the sun no closer than [30] ° from the sensor boresight.

4.3.1.2 Operational requirements

In a first phase airglow emissions shall be observed at different regions and under different angles of observation. These measurements will provide a first idea of expected minimum, maximum and mean intensities of airglow emissions during both day and night. Furthermore, it will allow analyzing background radiation due to scattered sun- or moonlight. The first observation phase shall last 3 months. During this period, 20 images of the airglow shall be taken.

In a second phase only observations of airglow emissions at limb between 50 and 120 km shall be carried out. Since they constitute the basis for a new low-cost earth sensor, their variation in intensity has to be studied more carefully. Hence, the variation of emission intensity depending on latitude can be observed over a longer period. The duration of this second phase will be determined by the lifetime of the satellite.

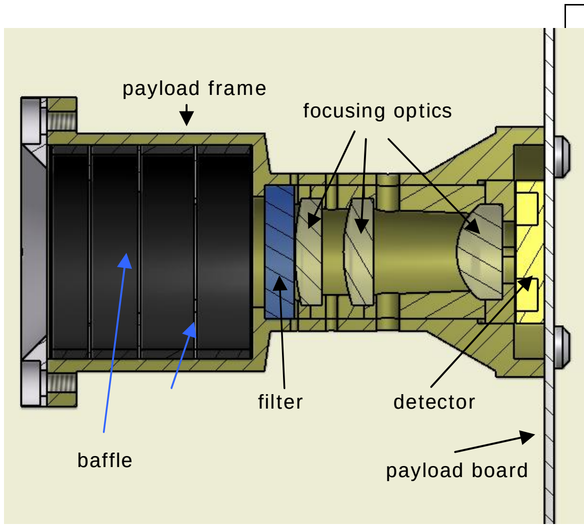
4.3.1.3 Physical constraints and power consumption

The space which has been attributed to the payload has a volume of [30 (height) x 30 (width) x 65 (depth)] mm^3 for the optical system and a volume of [80 (length) x 35 (width) x 15 (depth)] mm^3 for the payload board. The total mass of the payload shall be less than [60] g.

The payload shall consume at maximum [450] mW (peak power) during [30]s for each image capturing.

4.3.2 Technical description of the payload

Figure IV-22 shows the overall design of the payload. It consists of a PCB, including a detector and its control electronics, and an optical system, comprising a baffle, the bandpass filter which selects the desired wavelength of the oxygen emissions and focusing optics. The mass budget and a detailed description of the material used for the payload are given in Table IV-13.



Element(s)	Material	Mass
Closing cap	Aluminum	2 g
Payload frame	Titanium	20 g
Vanes	Stainless steel	2 g
Spacers for the baffle	Aluminum	2 g
Spacers for the optics	Titanium	5 g
Lenses	Glas (BK 7)	1 g
Filter	TBD	1 g
Payload board	PCB	15 g
Screws	Stainless steel	3 g
Margin (5 %)		5 g
Total mass:		**54 g**

Figure IV-22: Mechanical design of the payload.

Table IV-13: Mass budget and material used for the payload.

4.3.3 Optical system

Focusing Optics

The optical system provides a FOV of 18.8° x 29.4° and a resolution of 0.16 °/pixel for the chosen CMOS detector. As shown in Figure IV-23 a triplet design consisting of standard lenses only has been chosen due to reduced cost and complexity of the optics. Its RMS spot diagram satisfies the targeted resolution for a pixel pitch of 24 µm for 95% of the rays. A detailed analysis on the optical system including a verification of the focalization, the aberrations, and the required tolerances for assembly is planned.

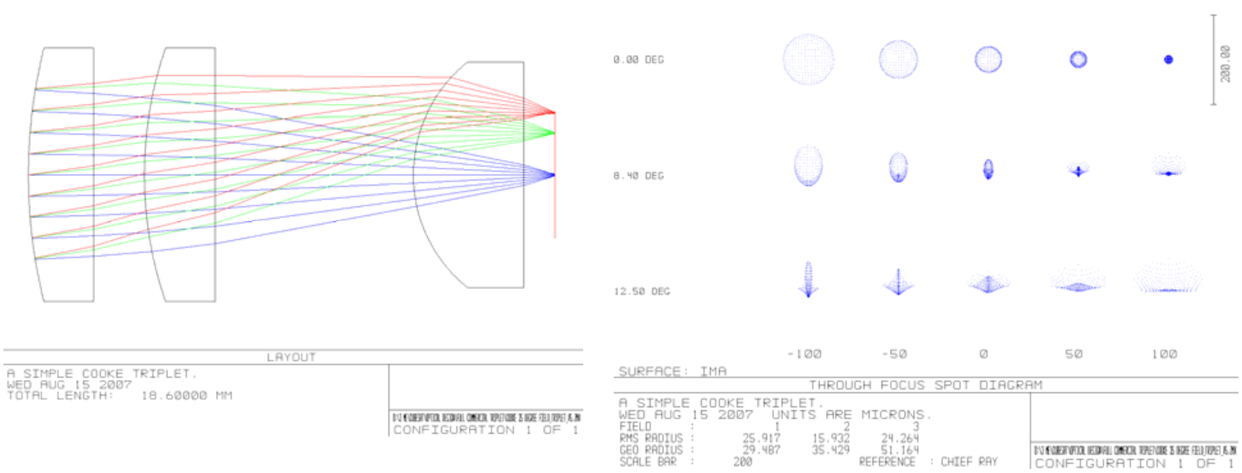


Figure IV-23: Layout and spot diagram of the payload triplet lens design.

Baffle Design

There are some basic rules for the baffle design. First of all, no optical components are permitted to view any sunlit wall or vane edge directly. This means that at least two reflections from darkened surfaces of the baffle tube or vanes are required between the straylight sources and the optical elements. Second, a minimum number of vanes edges should be directly exposed to the bright straylight. And finally, surfaces of non-optical components in the FOV have to be covered with a black material.

The targeted baffle attenuation factor of 10^{-4} guarantees that the signal perturbation due to the straylight noise which reaches the detector does not exceed the dark signal. If a reflectivity of 1% is provided by the surfaces of the baffle, two reflections would therefore be sufficient to get the targeted attenuation since it is given by $\tau_b = 0.01^n$ (with n = number of reflections).

The internal visible surfaces of the baffle will be roughened and blackened, and the reflections are expected to be rather diffuse. The baffle has therefore been designed as shown in Figure IV-24, for a solar exclusion angle (SEA) of 30°. The height of the vanes is determined by the intersection of the instrument's FOV and the most direct light path for incident angles outside of the instrument's FOV.

For further details on the optical system of the payload please refer to the following report:

- S3 Phase B Payload's Optics [24]

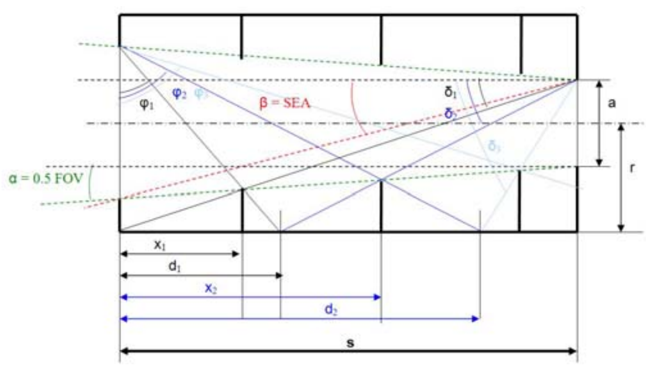


Figure IV-24: Ray traces for the determination of the position of the vanes in case of diffusely reflecting surface.

4.3.4 Detector and control electronics

The payload of the SwissCube satellite is the prototype of the ES currently developed at LMTS. Since this ES instrument will be based on a SPAD-array[3], it would be best to use this same detector for the prototype in order to provide a more realistic characterization of the ES concept. However, it might be difficult to adapt such a novel solution to the low-power and low-mass specifications of the SwissCube project; and the detector and its control electronics will not be ready in time to be

[3] SPAD – Single Photon Avalanche Diode

launched with SwissCube. Therefore, a solution, like the use of commercial CMOS-detectors or CCDs, has been studied and designed.

Detector

Analysis shows clearly that the performances of a CMOS detector are closer to those of a SPAD-array. Thus, it is this type of sensor which will be used. The MICRON MT9V032 is a highly sensitive monochrome CMOS detector, with similar power requirements and a similar size as a SPAD-array, if a Binning of 4 x 4 pixels is applied. However, the dark signal, the fill-factor and the photon detection probability are significantly higher for the CMOS sensor. Nevertheless, it will be able to detect similar photon fluxes as a SPAD-array and can therefore be used as detector for the SwissCube application.

Electronic circuit

The electronic circuit of the payload is attached to the optical system and bears the detector and the components required to successfully operate the detector and communicate with the CDMS subsystem. With exception of the connection used to operate and read the detector, its schematic is very similar to those of other subsystems using a MSP430 microcontroller. The task of the payload microcontroller consists of operating, reading and formatting the science data before transmission to the CDMS for storage until transfer to the ground station. No data compression is done aboard the satellite.

The payload program is based on interrupts coming from CDMS via the I2C interface. Such interrupts allow configuring the detector, triggering an image capture and triggering the read-out of the image. The science data will be transferred to the CDMS in packets of 235 bytes (corresponding to one image line) via the I2C interface where it is stored until transfer to the ground station. Since the CMOS detector has of 752 x 480 pixels with 10 bits/pixel, the size of one image (with a Binning of 4 x 4 pixels) is 226 kbits.

4.4 On-board Command & Data Processing Hardware Architecture

4.4.1 Design Drivers

The CDMS board realizes the processing and scheduling functions of the SwissCube satellite. The main activities are:

- to perform scheduling, execution and verification of telecommands;
- to perform data storage of housekeeping data and telemetry;
- to execute the ADCS algorithms;
- to provide a time reference aboard the satellite;
- to provide the interface for the payload detector;
- to perform TC scripts;

A summary of the controller needs of the various subsystems is given in Table IV-14.

Subsystem	Controller architecture	Memory	Remarks
EPS/Beacon	8 bit		Read EPS vital parameters, such as battery voltage, temperature etc. Generate the beacon signal
RF Transmitter	8 bit		Manage the AX.25 protocol for data transmission
CDMS	32 bit	4MB	TM/TC scheduling, storage, verification & execution, ADCS algorithms, on-board timebase. I/F with payload sensor
ADCS	8 bit		Read data from the ADCS sensors.
Payload	8 bit		Control parameters of the payload sensor.

Table IV-14: Subsystem controller needs.

A study has identified two of the following microcontrollers as best candidates for our application.

- 16-bit Microcontroller: MSP460F1611
- 32-bit Microcontroller: ATMEL AT91M558800A

Radiation tests will be done this autumn in order to confirm this choice.

The subsystem controllers will be treated in their respective paragraphs and shall not be detailed here any further.

4.4.2 CDMS functional architecture

The functional architecture as shown in Figure IV-25 features a watchdog timer for system resets. The system has been built to provide radiation robustness. The most common failure scenarios are listed in Table IV-15. Depending on the criticality of the subsystem and the complexity of the needed microprocessor/microcontroller, parts or all functional elements will be adopted. Three types of memories have been selected. ROM program storage, RAM as temporary memory and Flash memory that will be used as non-volatile memory space.

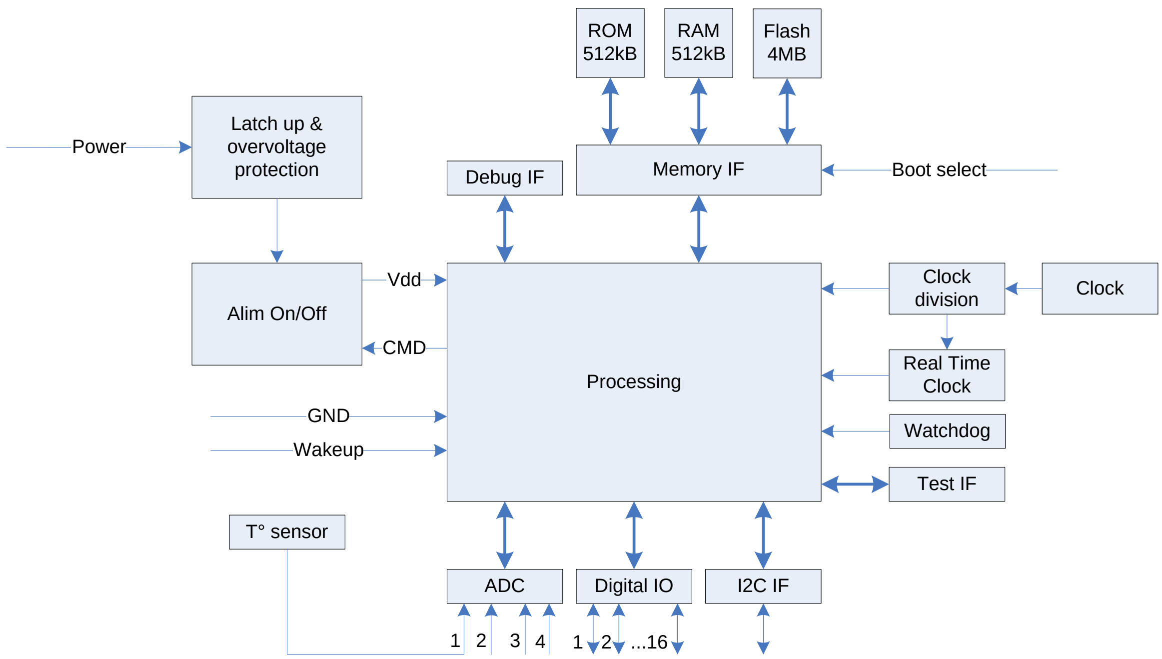


Figure IV-25: CDMS functional architecture.

Failure	**Consequence**	**Mitigation method**
Software lock-up	Wrong operations are executed, no controller not controllable.	Hardware watch-dog timer
Single event latch-up (radiation)	Short-circuit within semiconductor chip. Burn-out of component.	Latch-up protection circuit. (verification of the correct operation of the latch-up protection circuit still needs to be done)
Single event upset (radiation)	Bitflips in memory	• ROM for program code • Error detection and correction for RAM/FLASH

Table IV-15: Failure scenarios for on-board microcontrollers/microprocessors.

4.4.3 I2C Interface

I^2C has been chosen for the main bus due to its low power consumption and availability on most small microcontrollers. The 32-bit Microcontroller ATMEL AT91M558800A used on CDMS is able to communicate with a SPI bus, but not directly with the I2C. Therefore it has been decided to use an MSP430F1611 to do the SPI ⇔ I2C conversion. Moreover, with this configuration all the I2C will be managed by the same microcontroller for all subsystems. It will be easier for the tests and debug.

4.4.4 Baseline design

The CDMS board has been built and preliminary functional tests have shown a hardware mistake about the oscillator choice. This mistake is now corrected.

Various electrical tests were performed on the CDMS prototype board in order to ensure the correct operation of all components. The board is now ready to receive the software in order to continue the functional tests.

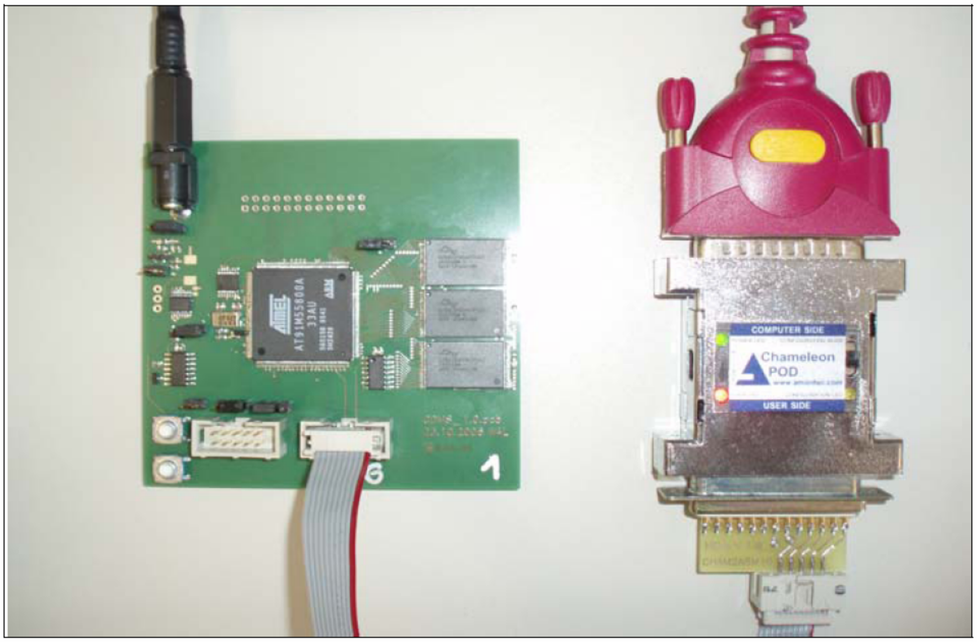


Figure IV-26: CDMS board with JTAG interface.

4.5 Attitude Determination and Control

4.5.1 Design drivers

ADCS design is driven by the payload pointing requirements which have to be satisfied, as discussed in IV -1.3.

4.5.2 Disturbances

In order to ensure the attitude control of the satellite, different environmental perturbation torques that disturb the satellite once in orbit have to be modeled. These perturbation torques include aerodynamic, gravity gradient, magnetic disturbances and solar radiation torques. The equations are summarized in Appendix A**: Satellite Perturbation Models**.

Figure IV-27 shows the magnitude of the disturbance torques as a function of altitude. The aerodynamic torque is dominant up to 550 km. Note that this graph is only plotting one upper bound of the different torques.

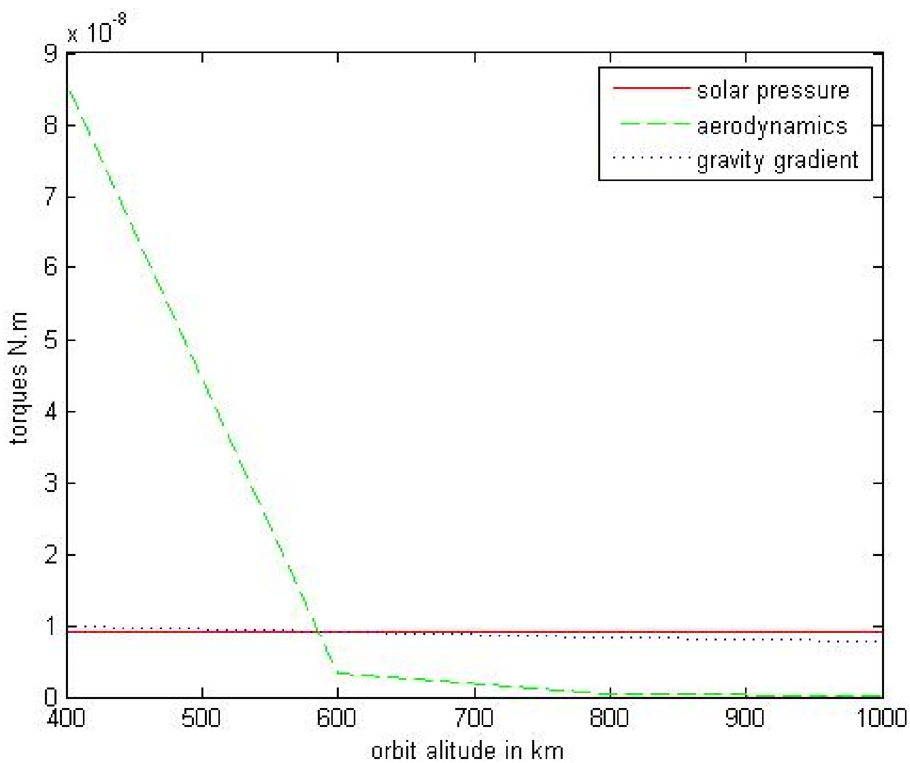


Figure IV-27: Upperbound of the disturbances torques versus altitude

4.5.3 Attitude modes

The Attitude Determination and Control System (ADCS) will be responsible to acquire the satellites current orientation and to influence its attitude.

Three distinctive modes of operation might be defined for the satellite's attitude. They are summarized in Table IV-16.

Mode	Timeline
Unstabilized	Separation from the P-Pod, rotation at 0.1 $rad\ s^{-1}$
	Start ADCS system, antenna deployment
Detumbling	Begin detumbling control algorithm
	Receive data from ground station with orbital and position parameters
Transitional	Begin attitude determination
Nominal (Science)	When de-tumbling controller has reached equilibrium begin nominal control

Table IV-16: ADCS Mode Summary.

4.5.4 Attitude Control Algorithm

4.5.4.1 Reference Frames

For the derivation of the dynamic model of the satellite three different reference frames were defined an inertial one (IRF), fixed to the Earth, an orbital one (ORF), fixed to the orbit with the positive x-direction pointing in the direction of displacement and a positive z-direction pointing toward the center of the Earth. The last one is the body-fixed referential (SRF) which coincides with the ORF when the satellite has the desired nominal orientation (cf. Figure IV-28).

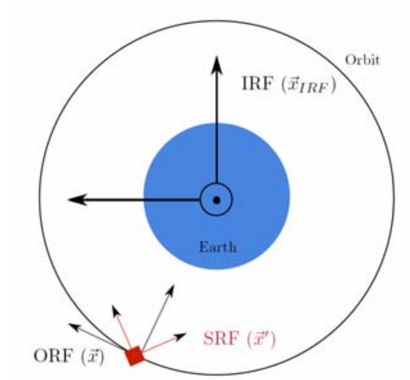


Figure IV-28: ADCS Reference Frames.

4.5.4.2 Control Algorithm Design

The appropriate control algorithms for science operations are currently being investigated. During the Phase B study three models of the satellite were created and validated these models are:

- Non-inertial quaternion based;
- Inertial quaternion based;
- Non-inertial Euler angles based.

The first one is the one of interest; the latter two were derived in order to be able to perform cross-validation by mean of simulation comparisons.

A design option at the beginning of Phase B was to fly an inertia wheel. The conclusion of the analysis is that one single reaction wheel is of limited interest for the project. Therefore the study was refocused to magnetotorquers only. The controller takes the form of a simple proportional-derivative (PD) controller (i.e. the control torque generated by the magnetotorquers is the sum of a term proportional to the quaternion imaginary part vector and of another term proportional to the rotational speed vector expressed in the body fixed referenced frame). Of course, this control torque is always constrained to the plane perpendicular to the local Earth B-field. This in turns implies that the actual torque rarely really match the desired one. It could be shown theoretically that this still works for relatively slow dynamics and therefore, for relatively low disturbance torques. The conclusion of this study has shown that full magnetic actuation is theoretically feasible, but probably not capable of rejecting the amount of environmental predicted perturbations.

4.5.4.3 Attitude Determination

In order to work properly, the controller needs to know the current satellite attitude and rotational speed. Based on measurements from the magnetometers, sun sensors and gyroscopes, the attitude determination algorithm tries to estimate the system state variables (i.e. attitude and rotational speed). The main criterions are computational greediness, “convergence reliability” and precision.

First, a complete continuous Kalman filter was implemented to serve as a comparison basis to the other simpler and easier to implement estimators.

Various more or less classical estimator implementations were written with SIMULINK and run together with the dynamics, sensors and actuator models.

Beside a discretised Kalman filter, various methods tested were

- TRIAD algorithm
- Davenport’s q-method
- Optimal Two Observation Quaternion Estimation

for the so-called “deterministic” algorithms and

- REQUEST algorithm
- Optimal REQUEST

for the so-called “recursive” algorithms.

The method of choice was found out to be the REQUEST or the optimal REQUEST. Overall performances were satisfying. However, the tests where done without perturbation torques and additional extensive simulations are needed.

4.5.5 Attitude control and determination hardware

4.5.5.1 Baseline design

The current design was based on the ADCS results obtained during phase A. Figure IV-29 shows the possible determination and control systems, the selected ones are grayed out.

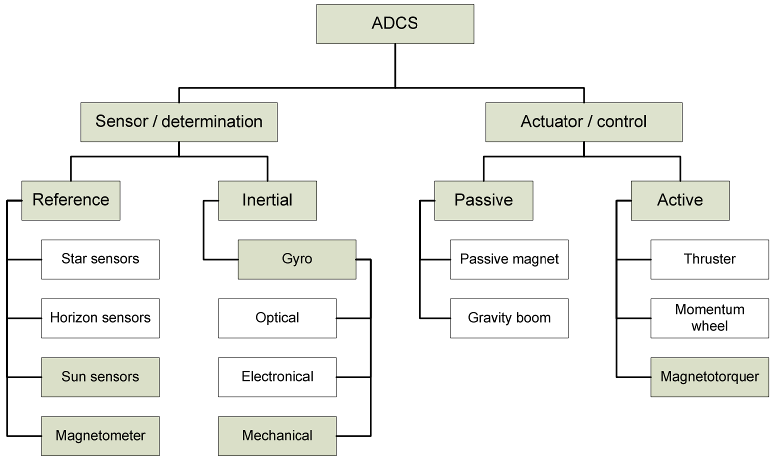


Figure IV-29: ADCS hardware trade-off.

During daylight, three kinds of sensors are used, gyroscopes, sun sensors and magnetometers. According to preliminary calculations, the solar panels do not offer enough accuracy to be used as sensors therefore dedicated sun-sensors will be used.

During the eclipse, the sun sensors will become useless. The magnetometers and the gyroscopes will still be used. Due to drift, the gyroscopes will probably need to be recalibrated. It will be done by using the sun sensors during daylight.

Figure IV-30 shows the block diagram for the ADCS board. Figure IV-31 shows the board.

At this date, almost all functionalities of the ADCS board have been integrated and tested (magnetometers, magnetotorquers, sun sensors, gyroscopes and temperature sensors, but not I2C). The first version of the ADCS microcontroller software has been written and a second iteration of the design of the board has been done. Recommendations are proposed for a third revision of the board. Although there is still a lot of work and tests to carry out, especially about temperature tests and compensation software, the present ADCS systems is able to provide basic sensors readings and actuators control.

An efficient calibration methods remains to be found for the gyroscopes. A detailed and complete model of the sun sensors (taking into account the 6 sensors) must be created; for that a model providing the direction of the sun as a function of the date, satellite orbit and position, therefore linked to the propagator, should be built.

During Phase B, additional options for passive (using a permanent magnet) and active (inertia wheel) control were investigated. Although the use of an inertia wheel could improve the control of the satellite, the development time was judged too risky to pursue this option. Developments in this area are still going on, but will not be integrated in this first version of the satellite. In addition, and according to simulations of the control algorithms, the presence of a permanent magnet will not help to control the satellite, but on the contrary it will reduce its controllability; the magnetotorquers will always stay nearly perpendicular to the Earth's magnetic field lines and will have few effects. Therefore we do not have an efficient method for controlling the SwissCube for the moment. This is a major issue for the ADCS.

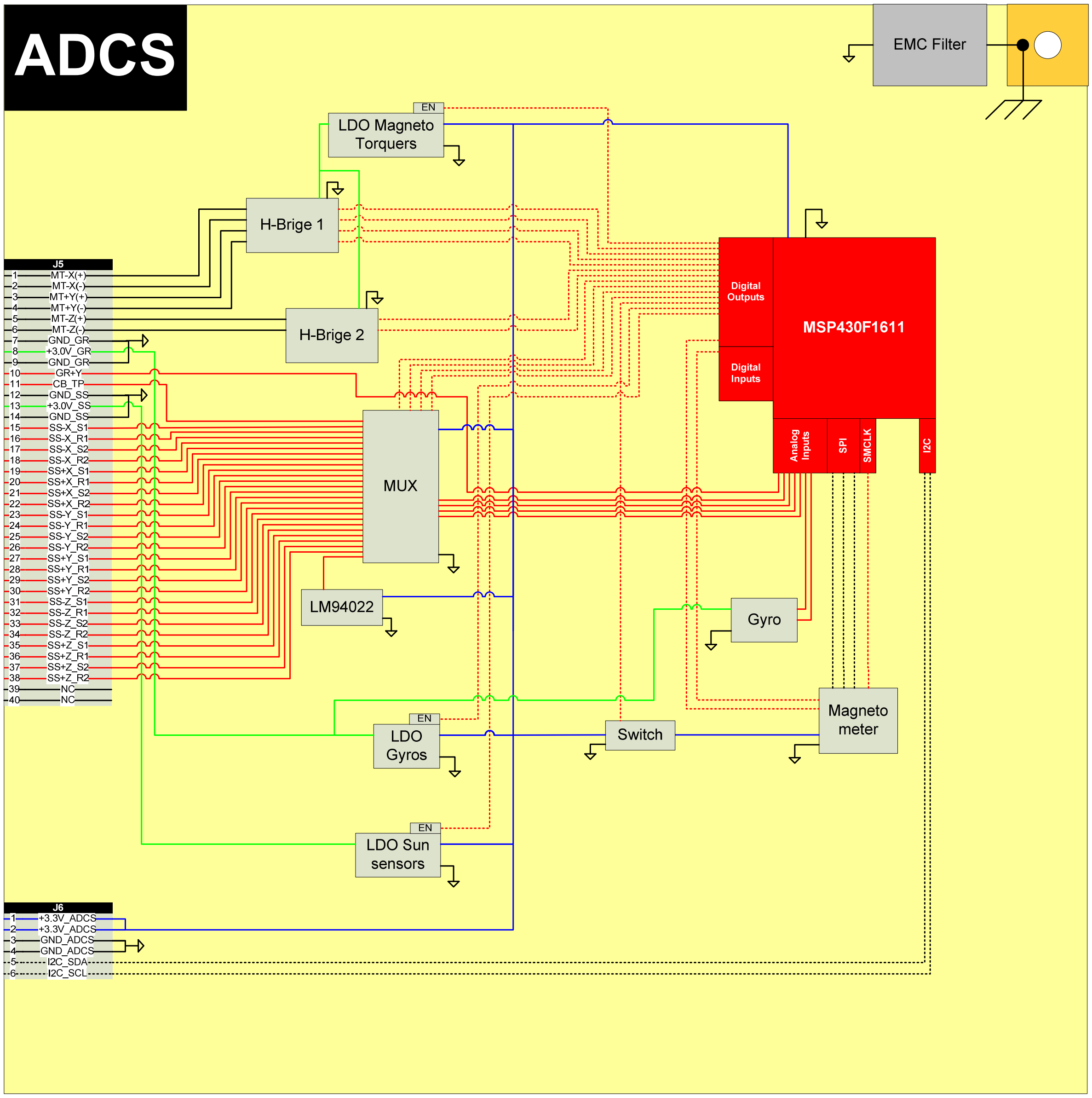


Figure IV-30: ADCS electrical and functional block diagram.

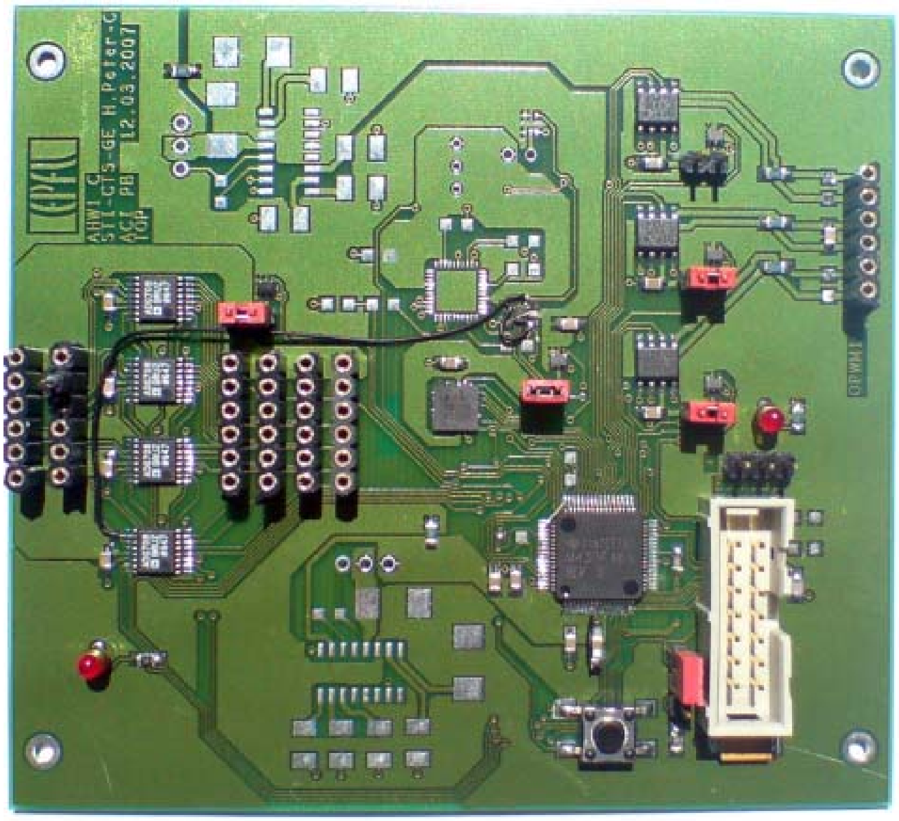


Figure IV-31: ADCS EM board.

For further details on the algorithms, sensors, actuators, board and tests please refer to the following report:

- ADCS Hardware and System Engineering report [25]
- Determination Algorithm Design: [26]
- Characterization and tests of the Sun Sensors [27]
- Characterization and tests of the Gyroscopes [28]

4.6 Structure and Configuration

4.6.1 Overview

The purpose of the structural subsystem is to provide a simple study structure that will survive launch loads and provide a suitable environment for the operation of all subsystems throughout all phases of the mission life. The structure shall also provide easy access for integration and satellite check-out. Moreover the structural subsystem shall carry, support, and mechanically align the satellite equipment.

Structural design shall aim for simple load paths, a maximization in the use of conventional materials, simplified interfaces and easy integration. Due to the size of the satellite and small expense budget, this was done with the philosophy of maximizing usable interior space, while minimizing the complexity and cost of the design.

The structural subsystem has been given a mass allocation which is a target for the overall design.

4.6.2 Principal design drivers

The proposed SwissCube structure has to be compatible with the CubeSat standard, including access ports and deployment switches. The configuration has to accommodate all platform elements, the optics payload and the antenna mechanism.

4.6.3 Baseline design

Different frame options were studied during the early phases of the design. A "monoblock" approach was selected based on weight constraints and structural strength considerations.

The payload and platform elements are inserted into the primary structure, shown in Figure IV-32. The structure will be made out of either AL-7075-T73 or Al-6061-T6 as required in [7].

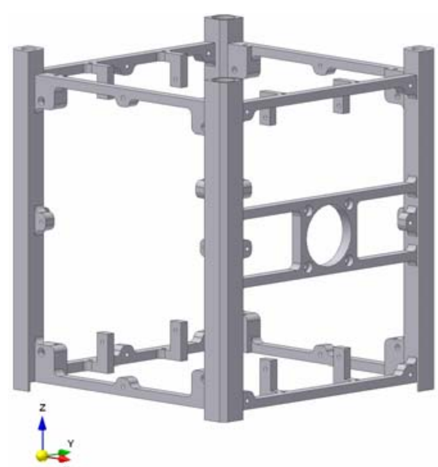


Figure IV-32: SwissCube primary structure.

Figure IV-33 shows the layout of all components forming the satellite. The satellite configuration combines three major assemblies:

- Primary structure with attached side and solar panels
- Payload assembly
- Printed circuit board stack connected to a common motherboard

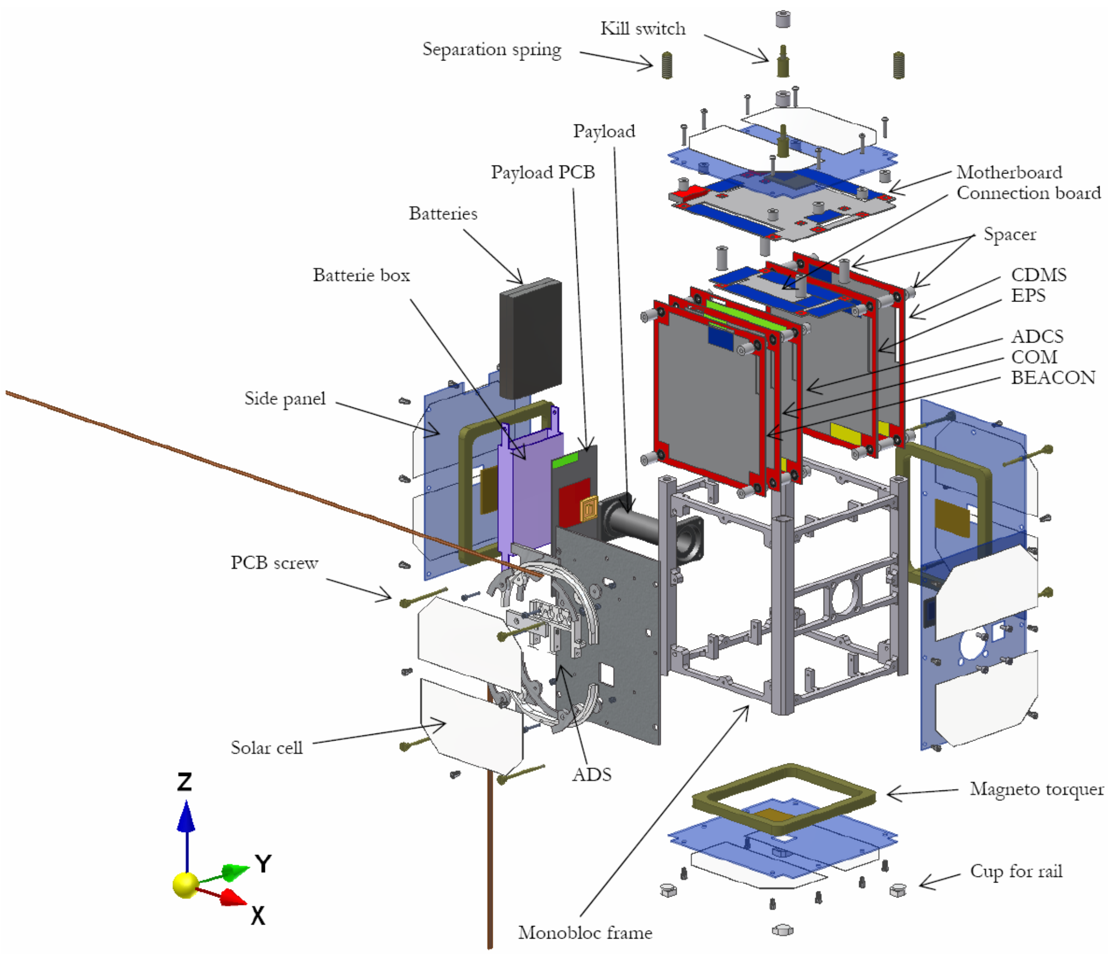


Figure IV-33: SwissCube configuration.

4.6.4 Satellite assembly

An overview of the satellite assembly procedure is given in Figure IV-34. Elements will either be threaded or glued onto the structure. The preferred fixation method is by threaded fasteners, which should allow disassembly. Some components such as kill-switches or separation springs will be glued.

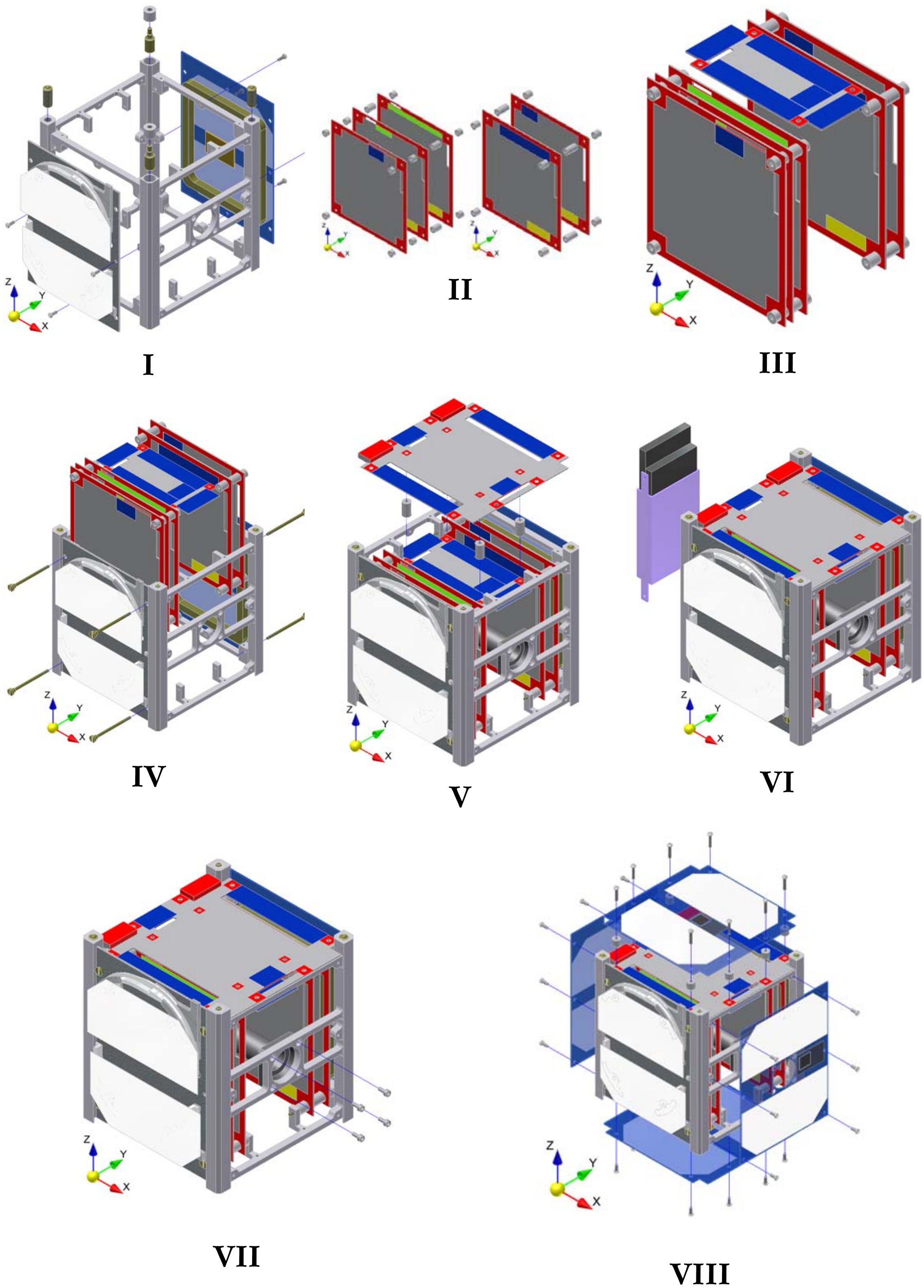


Figure IV-34: SwissCube assembly procedure.

4.6.5 Structural Analysis

To validate the selected structure and configuration baseline static and dynamic analyses were performed. Furthermore, the physical and inertial properties of the satellite were determined through the CAD model.

4.6.5.1 Physical and inertial properties of the satellite

Table IV-17 and Table IV-18 give the center of gravity and the inertial properties of the SwissCube with respect to the satellites reference point and frame.

Center of Gravity	
Axis	Value [mm]
X_c	-5
Y_c	0.5
Z_c	-0.1

Table IV-17: Center of mass properties.

Physical moments of inertia	
	Value [kg mm²]
I_{xx}	1576
I_{yy}	1691
I_{zz}	15465

Principal moments of inertia	
	Value [kg mm²]
I_1	1579
I_2	1676
I_3	1543

Rotation XYZ/principal	
	Value [deg]
R_x	-5.4
R_y	20.5
R_z	-8

Table IV-18: Inertial properties.

4.6.5.2 Static analysis

The worst case static load of 7.5g was identified for SwissCube on the Dnepr launcher. Including a factor of safety of 1.25, a worst case acceleration of 10g has been considered for the analysis. The determined margin of safety for the whole satellite is a factor of 21, the weakest points being the z-rails ends.

4.6.5.3 Modal analysis

Modal analysis has been performed with a simplified model of the satellite using the Abaqus software.

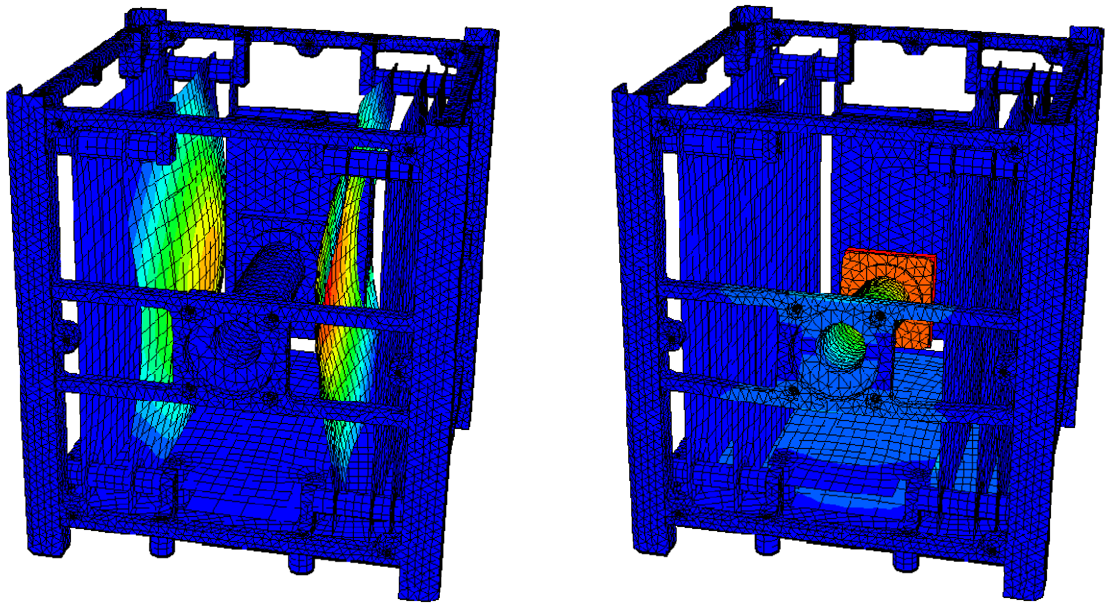
Mode	Frequency [Hz]	Zone of interest
1	174	PCBs stacks
2	175	PCBs stacks
3	175	PCBs stacks
4	175	PCBs stacks
5	200	PCBs stacks
6	317	Monobloc frame (Payload attachment)
7	392	PCBs stacks
8	392	PCBs stacks
9	393	PCBs stacks
10	393	PCBs stacks
11	399	Motherboard
12	400	PCBs stacks
13	425	Motherboard

Table IV-19: Natural frequencies of the satellite

The first five modes concern the PCBs stacks as summarized in Table IV-19. The first vibration mode of the primary structure (monobloc) happens around 317Hz. These two modes of vibration are illustrated in Figure IV-35.

Due to these results, other options for the attachment of the PCBs stacks are actually investigated. On solution will be to add spacers in the center of the PCBs stacks and to connect them to the primary structure.

For the payload attachment, possibilities to increase the eigenfrequency of this area can be to enlarge the crossbars as well as to add orthogonal crossbars. Both options are under investigations.



(a) (b)

Figure IV-35: Modes of vibration. (a) 1st mode at 174Hz (b) 5th mode 317Hz.

4.6.6 Structural Tests

Sinusoidal, random vibration and shock tests are scheduled at the end of August 2007.

For further details please refer to the following report:

- Structure and Configuration [20]
- FEA Analysis [30].

4.7 Thermal Management

4.7.1 Mathematical model and analysis

For SwissCube a passive thermal management approach has been selected, i.e. no active thermal components, such as heaters will be implemented.

The current FEM study has been done around two models:

- *Structural-thermal model* – includes frame, external panels, thermal glue, solar cells and internal PCBs.
- *Battery-thermal model* – focused on battery study and internal radiation effects. This model includes a model of thermal properties of stand-by, charging and discharging modes of the battery.

The following environmental conditions were taken into account:

- Sun emissions
- Albedo effect
- Earth emissions
- Sunlight and eclipse times
- Sun vector-function

4.7.2 Structural-thermal model

Three orbital altitudes were considered for the simulations, i.e. 400km, 700km and 1000km. The results are presented in Figure IV-36. A decrease in maximum and minimum temperature can be observed for increasing orbital height. The model does not take into consideration the variation of Earths radiation intensities for the different altitudes.

4.7.3 Battery model

For the battery model it was assumed that the battery is thermally isolated in the center of the satellite. The goal of this simulation was to determine the radiative behavior of the battery depending on its operational state and the mean temperature of the walls of the satellite. The results show that with the assumed electrical efficiency during charge and discharge, the heat generation of the battery keeps it within the desired range, provided that the battery be thermally isolated (by conduction). The design should tend to minimize thermal conduction leaks from the battery package to the structural frame.

Table IV-20 summarizes the thermal design parameters for satellite components. In addition a **maximum thermal gradient of 30°C** should be taken into account.

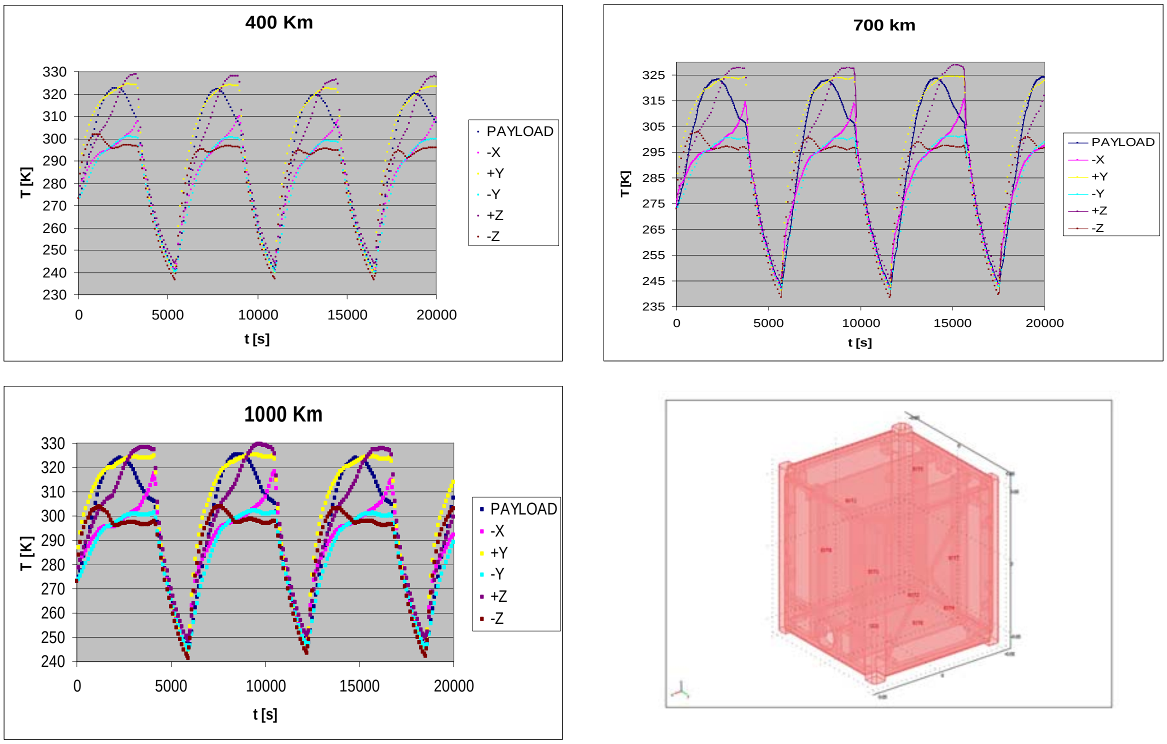


Figure IV-36: Simulation results for the thermal-structural model.

Parameter	Analysis Results [°C]	
	Internal	External
Minimum Temperature	- 20	- 30
Maximum Temperature	+ 24	+ 37

Table IV-20: Satellite thermal environment design values.

For further information please consult the following report:

- Thermal Management [31]

4.8 Mechanisms

4.8.1 Design drivers

The antennas will have a length of up to 1 meter. During the take off, the satellite shall not exceed the dimensions specified by Cal Poly. Therefore a system to deploy the antennas is needed.

The first requirement of the deployment system is high reliability. This has to be achieved with several other constraints like mass, volume and power consumption. The mass for the antenna system is budgeted to 25g. The volume used in the interior of the satellite has to be kept as small as possible and it shall not protrude out of the face of the cube more than 6.5mm. In addition to that, the system has to fulfill all the compatibility criteria for space applications.

4.8.2 Baseline design

Figure IV-37 shows a 3D view of the deployed antenna system. Figure IV-38 shows changes made during the semester for the mechanisms subsystem. A non magnetic Beryllium copper antenna is used to avoid magnetic perturbations of ADCS. A melting wire will be used as release mechanism.

The UHF antenna is bent around the lower guide. The VHF antenna goes over the UHF and around the guides for two loops and a quarter. The end of it, is retained by dyneema wire attached to the melting device going through two nichrome wires. Current passing through nichrome wire heats up and melts the dyneema. Once released, the antennas deploy with their own spring force.

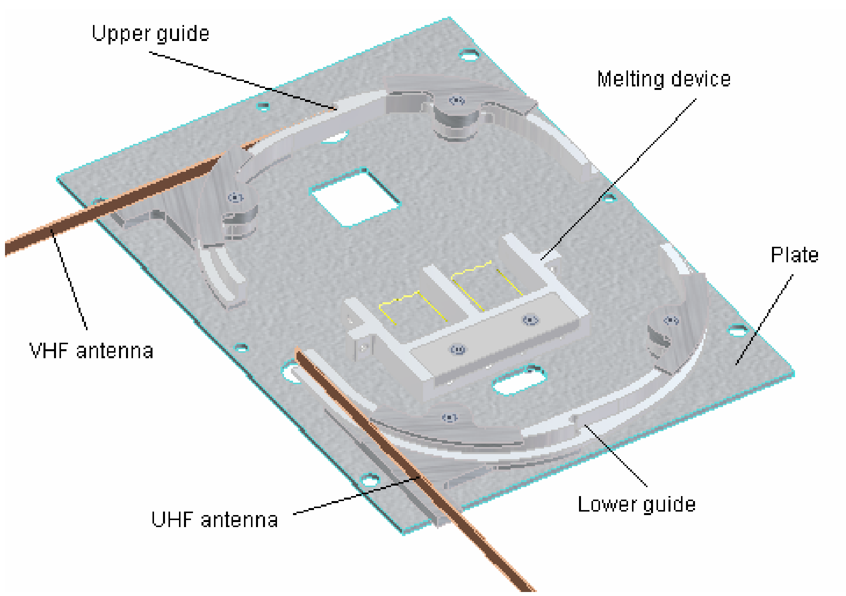


Figure IV-37: General view of the baseline design.

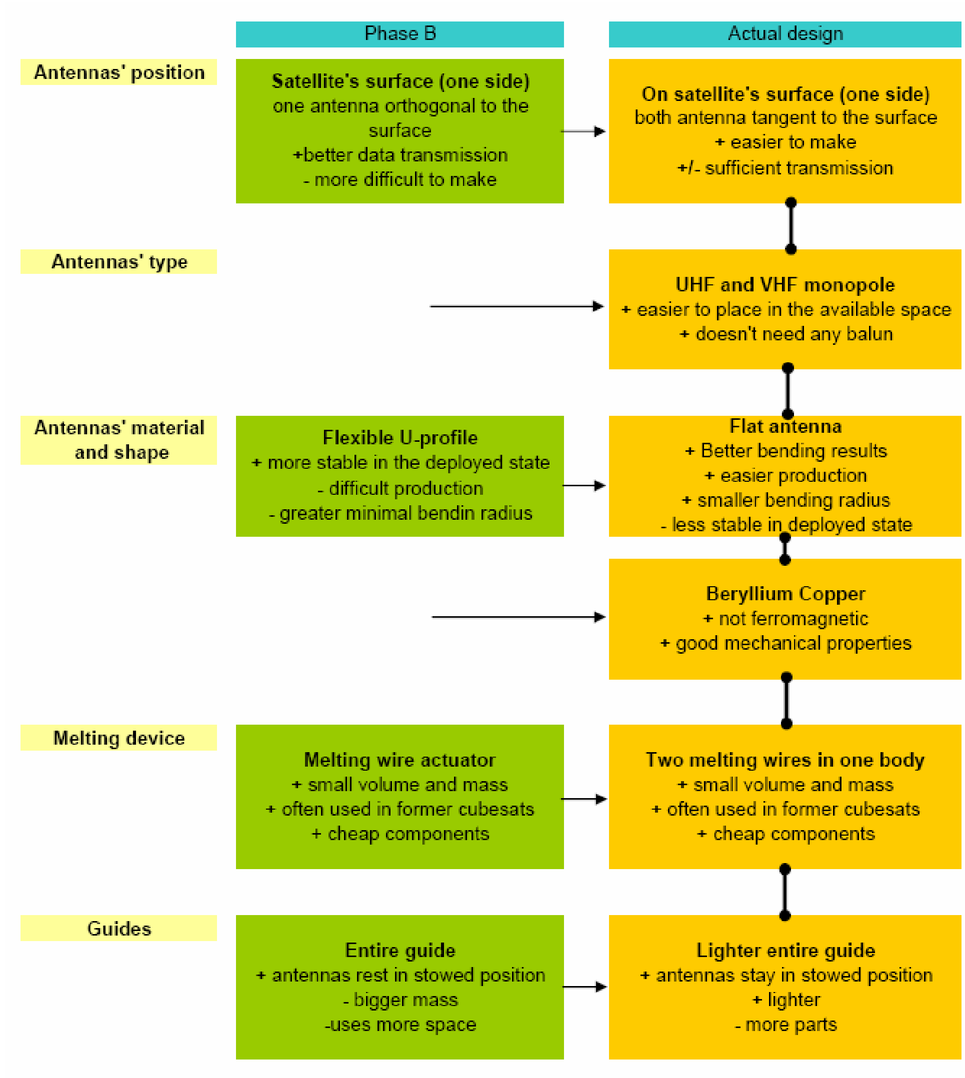


Figure IV-38: Solution summary for the antenna deployment mechanism

4.8.3 Tests

During the semester, many tests have been performed by the ADS team. The most significant are the bending tests, melting behavior under vacuum and RF tests in anechoic chamber.

4.8.3.1 Bending tests

The resulting bending of the both antennas (180mm and 610 mm) after deployment has been measured. The short antenna has a bending of 2° whereas the long one has a bending of 10°.

4.8.3.2 Melting tests

The whole melting device has been tested in vacuum at low and high temperature conditions, -56 and +73°C respectively. The scope of these tests was to determine the necessary time to melt the nylon and the required current for heating the nichrome under space conditions.

The results are a melting time of 2 seconds in both conditions and a necessary current of 1.15 and 1.09 A in cold and hot conditions respectively.

4.8.3.3 RF transmission tests

To see the effect of the resulting bending of the VHF antenna, RF transmission tests in an anechoic chamber have been performed. Transmissions of straight antenna have been compared with the case of bending antenna. Figure IV-39 shows that the influence of a 10° bent antenna is acceptable.

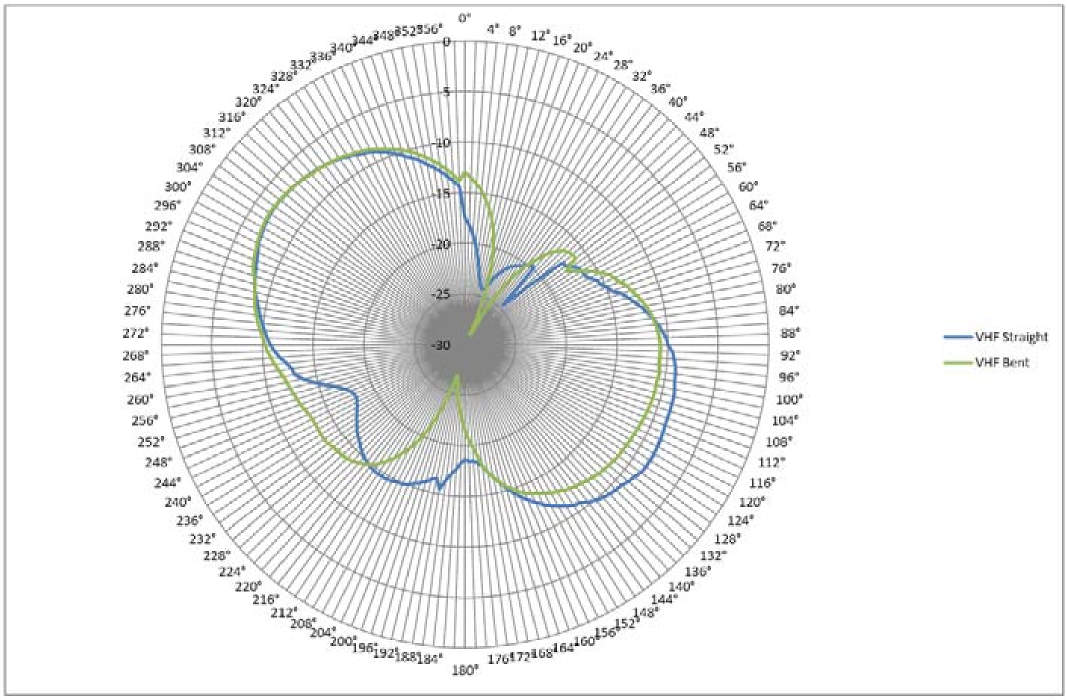


Figure IV-39: RF test results of VHF antennas.

The emission pattern of the UHF antenna has been measured and Figure IV-40 shows that the monopoles have a symmetric pattern just as in the simulations.

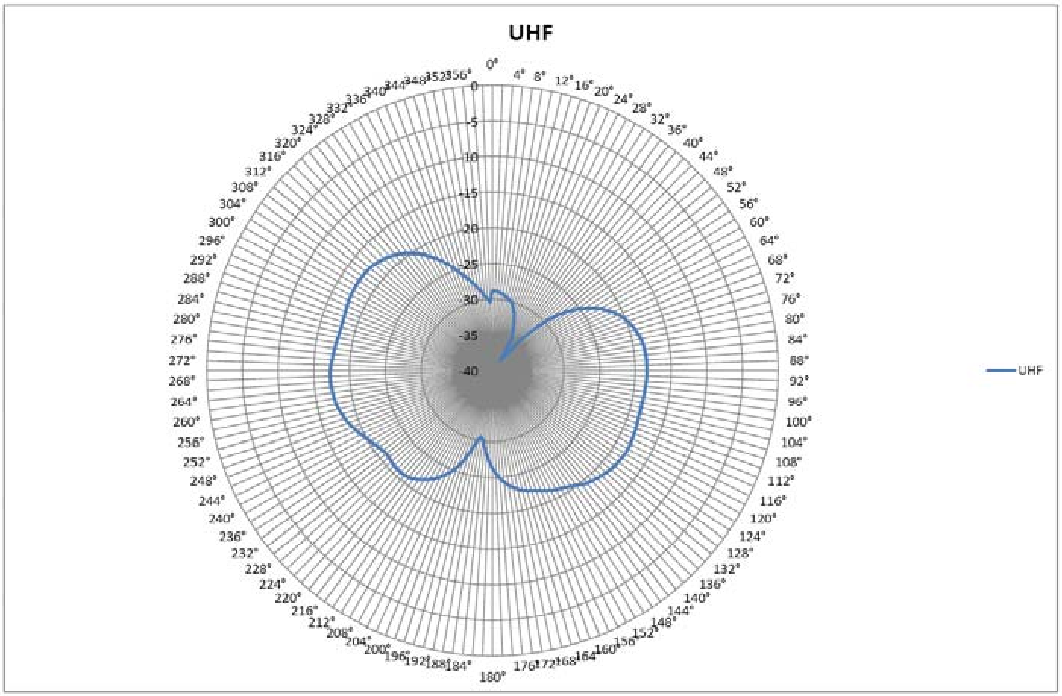


Figure IV-40: RF test results of UHF antenna.

For the further information please consult the following report:

- Antenna Deployment Mechanism [32]

4.9 Software

4.9.1 Design drivers

TM/TC standardization concepts have been elaborated by the *Consultative Committee for Space Data Systems*, CCSDS, and are being applied to a certain extent by many space agencies including ESA and NASA. Although SwissCube is not required to follow these specifications they serve as starting point for the satellites software layout. The concept used is based on the *Open Systems Interconnection*, OSI, model, a 7 layer structure that has been defined by the International Standardization Organization, ISO, as network architecture. This concept allows a standardized data exchange between the different layers and therefore simplifies the development of communication systems.

Flight software will be included in the following subsystems:

- EPS
- COM
- CDMS
- ADCS
- Payload.

The overall software architecture should therefore consider this distributed topology.

4.9.2 Baseline design

The chosen baseline features a Command and Data Management System that will provide certain services on-board the satellite, namely:

- Housekeeping
- Function management
- Scheduler
- Data storage.

These services will be accessible from ground. This will allow keeping the necessary hardware resources of the subsystems as small as possible as each other subsystems will only be responsible of providing its own housekeeping to the CDMS and exposing its own specific functions.

For further information please refer to:

- SwissCube Flight Software Architecture [33]

V GROUND SYSTEM DESIGN

1 System Overview

The proposed ground system architecture for SwissCube combines the ground station topology used by the AMSAT community with additional tools required for satellite operations. The chosen architecture tries to maximize the amount of hard- and software jointly used during satellite testing and operations. Figure V-1shows the hardware lay-out of the ground segment. Distinction will be made between the following three segments:

1. Ground Station System
2. Mission Control System
3. Satellite Check-out System

The three segments will be capable of operating in distributed manner. The OPSLAN could therefore be formed by a WAN connection using secured channels. This topology has the major advantage of allowing remote control of the satellite from mission control during testing. It further allows database population during subsystem development.

The Swisscube Ground Stations (GS) will be located on the EPFL campus and at the HES-Fribourg campus. The antennas will be installed on a roof, and a room, next to the roof, will be dedicated to the electronic devices. The Mission Control will be located at EPFL and at HES-ARC St Imier.

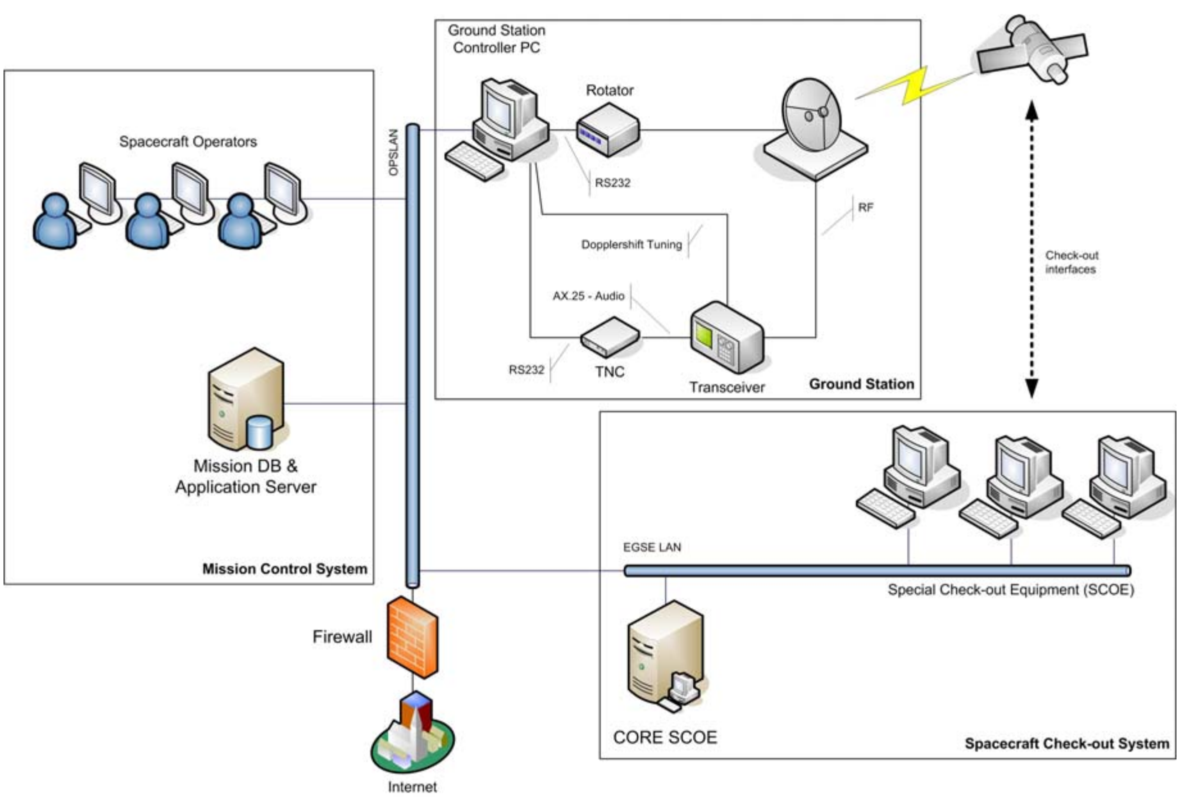


Figure V-1: Ground segment architecture.

The main drivers for the development of the ground systems are summarized here:

2_PR_11_03 **Space to ground frequencies and protocols**

The ground to space communication link shall comply with the Amateur Radio Satellites services.

For student satellite this is the easiest solution to implement.

2_PR_15_06 **Compatibility of ground system with other satellites**

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.

This requirement allows an independent way to test the ground station and train personnel before launch of the CubeSat. The same ground system might be used for future tracking network, which implies that the design shall have the possibility to conform/adapt to a standard network.

3_GS_11_xx: **Ground system operating lifetime**

The project shall design the ground system to have an operating lifetime greater or equal to [48] months.

2 Ground Station System

2.1 Design Requirements

The ground station system establishes the physical RF link between the space and ground segment. It controls the antenna rotors, the terminal node controller (TNC) and the transceiver. The ground station will be based on commercial off-the-shelf components.

The ground station design has to guarantee compatibility with the Radio Amateur systems and to reduce the development time by using COTS elements. To increase the downlink/uplink time the design should allow the compatibility and collaboration with other ground station systems.

To accommodate the requirements described above, it was determined that the SwissCube Ground Station should have the following features:

- Cross beam or circularly polarized Yagi
- Computer tracking system
- Computer controlled AZ-EL rotators
- Full-duplex dual band radio computer controlled tuning
- TNC and Soundcard interface for TLM and packet
- Transceiver control software
- Mast mounted receiver preampifiers

2.2 Design Overview

2.2.1 EPFL Ground Station

The ground-station will be built on the roof of the EL building of the EPFL. One part, the antenna system, will be installed outside on a mast. It will reuse parts of the existing infrastructure. The other part, the control electronics, will be located in a storage room about two floors below the roof.

Figure V-2 shows the system Block Diagram for the Ground Station. It shows all connections and devices. Table V-1 also shows the planned manufacturer and model of the devices.

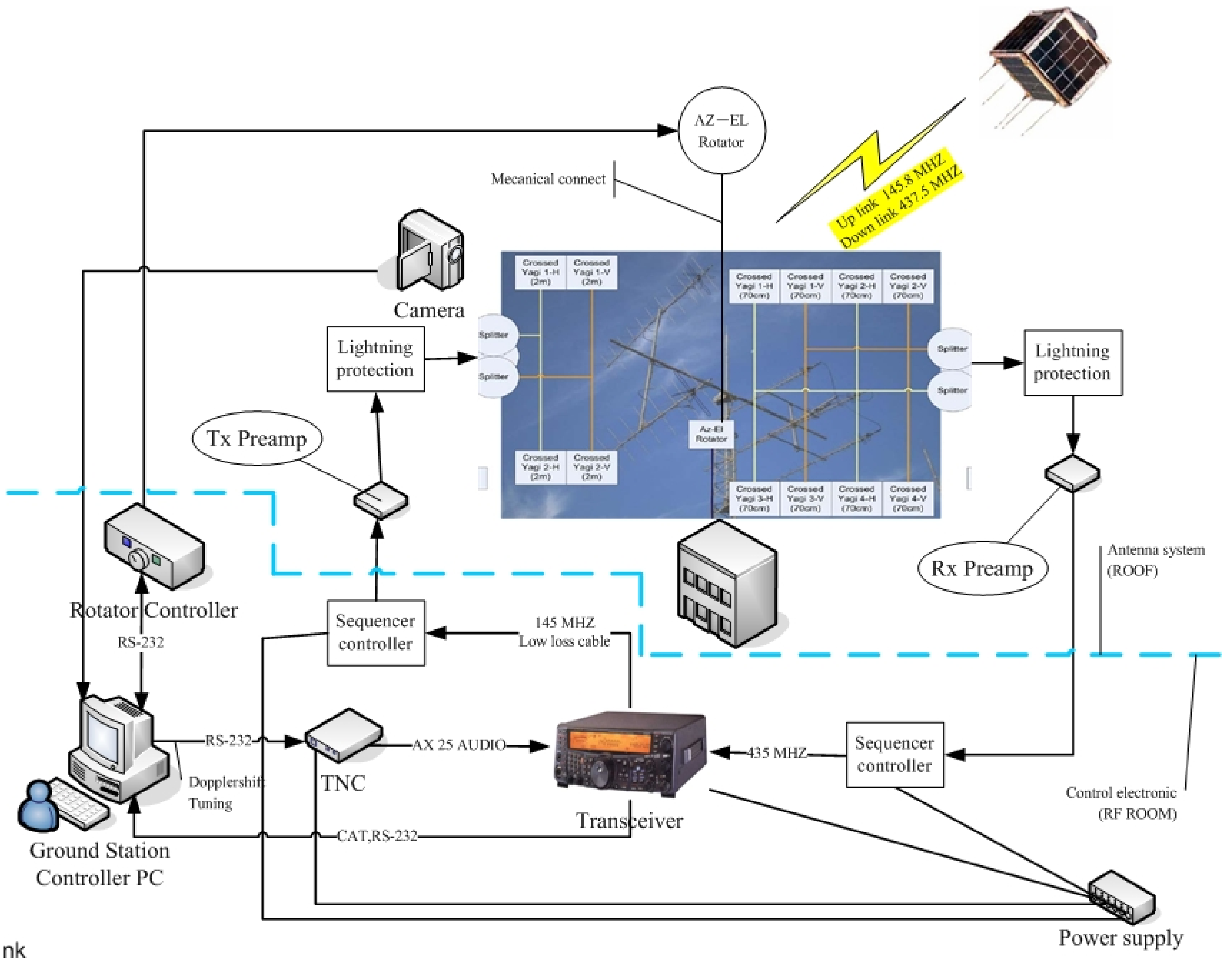


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Figure V-2: EPFL Ground Station block diagram.

The telecom data protocol between the ground and the space systems is the AX.25 and was chosen for its wide-spread use in the Amateur Radio community.

Element	Model	Function	Choice Rationale	Purchased
Control electronics				
Transceiver	Kenwood TS-2000	combination transmitter /receiver	See Note 1.	Yes
TNC	TNC2H-DK9SJ SYMEK	AX.25 packet modem	1) capable to decode the Pacsat protocol. 2) adapted to the speed of transmission going from 1200 to 38400 Bauds	Yes
Controller PC	486 IBM PC	1) control the antenna positioning motors for tracking of the satellite 2) control the transceiver, including Doppler correction	Available and free	Yes
Rotator controller	RC2800 PX-EL Controller	Command the rotator's position		Yes
SWR meter	CN-103LN or CN-801VN	Check the quality of the match between the antenna and the transmission line		No
Sequence controller	DCW-2004 for SP-2000 and SP-7000	Ensures the proper sequencing of both power amplifier and preamplifier switching		No
Amplifier (Optional)	HLV 300	Amplifies uplink signal		No
Power supply	GSV-3000			Yes
Antenna System				
Tx Preamp	SSB-Elektronik SP-7000	Low noise amplifier	Recommended by radio amateurs	Not needed
Rx Preamp	SSB-Elektronik SP-2000	Low noise amplifier	Recommended by radio amateurs	No
Lightning protection	Lynics 20310-3	Protect from lightning damage		No
Power Splitter				No
AZ-EL rotator	EL: M2 MT1000 AZ: M2 OR2800	Antenna rotators		No
Uplink Antennas 2-m	2 CP: 2MXP20 Yagis		Good G/T Optimized for stacking	No
Downlink Ant. 70-cm	4 CP: 436CP42 Yagis		Gain and F/B are excellent	No
Mast	Check if needed			
Additional clamping, beams and mounting HW	See detailed documentation			No

Table V-1: EPFL Ground Station hardware.

Note 1: The criteria for the choice of the transceiver were:

- Band of frequencies adapted to the frequencies of the CubeSat radio amateurs (145.8MHz for upload and 437.5 MHz for download).

- The transceiver must be able to recognize all the modes used for satellite radio amateur operations: FM, USB, LSB, CW, AM, AFSK, 9600 bauds packet, 1200 bauds packet.
- Possibility of controlling the transceiver by PC.
- Good compensation of the Doppler effect: the step of the synthesizer must be to the maximum of 1 kHz.
- Full Duplex: broadcast on a band and reception on the other one (VHF > UHF or UHF> VHF). The full duplex mode is currently not a requirement for the SwissCube but it is or might be for other satellites.
- Software support.

Figure V-3 shows the existing Antenna mast on the EL Building. The whole RF part will be replaced and new rotators will be installed.

Figure V-3: Current installation on the roof of the EL building.

Figure V-4 shows the baseline layout of the ground-station with two circularly polarized 2m Crossed-Yagi antennas for the uplink and four 70cm antennas for the downlink. The Yagi-Uda antenna is the standard antenna for television. It's a directive antenna that has one active dipole, reflectors and directors. They offer a good choice in terms weight/gain ratio. Figure V-5 shows the radiation patterns of available Yagi antennas for 2m and 70 cm.

For further details please consult:

- Ground Station Telecom Infrastructure Description and Specifications [34]

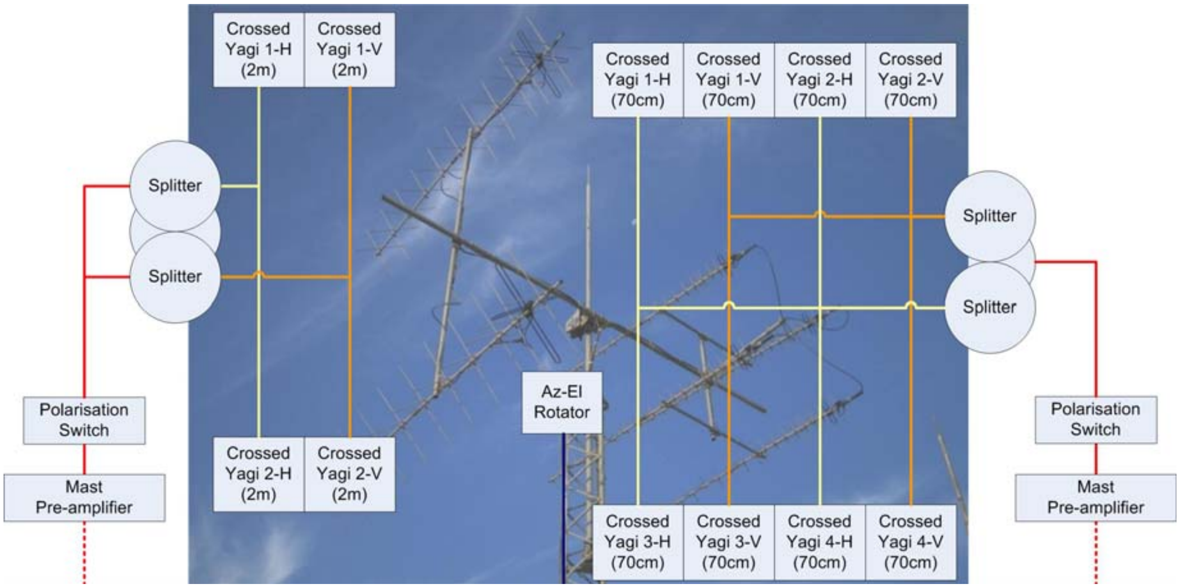


Figure V-4: Antennas layout.

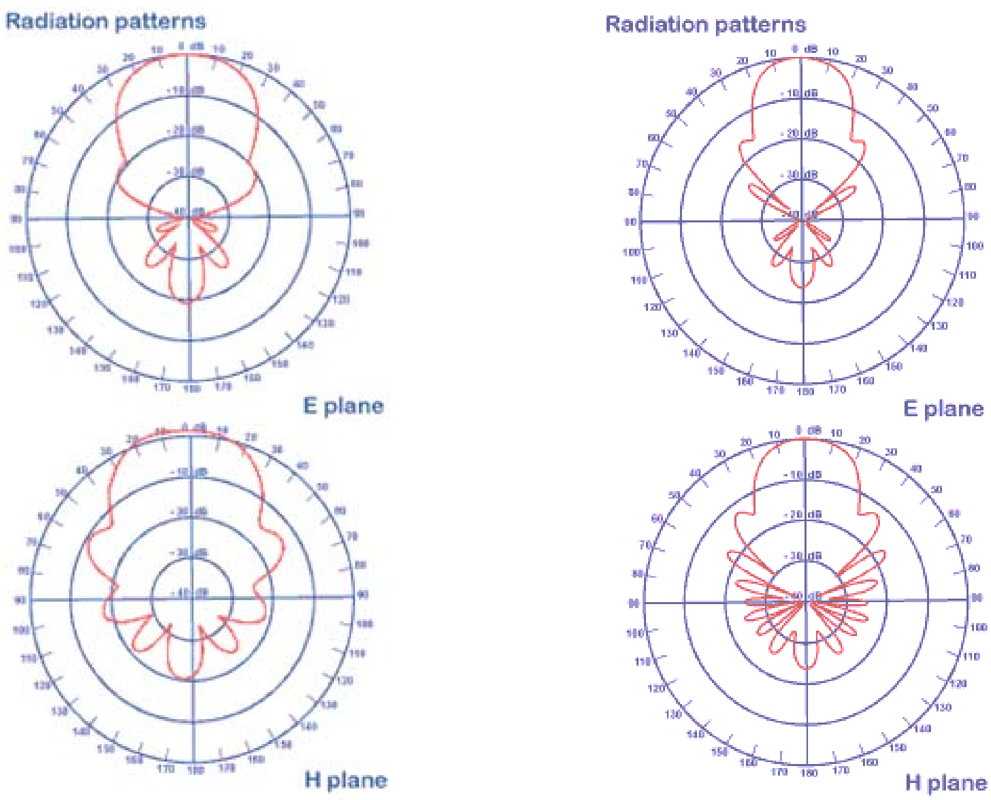


Figure V-5: Radiation pattern of a 2m and 70cm Yagi Antenna

2.2.2 HES-Fribourg Ground Station

The ground station in Fribourg was used a few years ago for Radio Amateur and educational purposes. The data collected from reports of past semester and diploma projects written by students and from spec-sheets is summarized below. These specifications will have to be confirmed and completed (G/T, EIRP etc.) during an upcoming semester project.

1) Uplink

11.4 dB of antenna-gain (crossed 9-element yagi)

17 dBW transmitter power-level (Yaesu FT-847, no external power-amplifier so far)

0.066 dB/m of attenuation for 30 m of coaxial-cable (Huber–Suhner S_07212BD)

2) Downlink

14.5 dB of antenna gain (crossed 17-element yagi)

0.125 μV of receiver sensitivity for 10 dB S/N at SSB/CW (2.2 kHz of bandwidth) (Yaesu FT-847, no preamplifier so far)

0.098 dB/m of attenuation for 30 m of coaxial-cable (Huber–Suhner S_07212BD)

2.2.3 Ground Stations Performance Summary

Table V-2 summarizes the estimated ground station performances.

	2m Antenna 144-146 MHz Uplink		***70cm Antenna 430-438 MHz Downlink***	
	EPFL	*HES-FB*	*EPFL*	*HES-FB*
Transmitter power	17 dBW	17 dBW		
Antenna type	Stack of 2 2MXP20 Yagis	2 CP Yagis	Stack of 4 CP: 436CP42 Yagis	2 CP Yagis
Antenna Gain	15.4 dBi	15.1 dBi	19-25 dBi	14.5 dBi
Beamwith			21°	21°
Elements	2*18	2*9	4*19	2*17
Feed impedance/Conn	50 Ohm / N		50 Ohm / N	
Transmission line losses	6 dB	2 dB	1.85 dB	2.9 dB
Ground station EIRP	26.4 dBW	TBC		

Table V-2: EPFL and HES-Fribourg performance estimates.

2.2.4 Ground Station Network (GSN)

It is possible that the SwissCube project will benefit from parallel developments in ground station networking (GENSO) or from the established radio-amateur community of AMSAT. Table V-3 shows the various options for SwissCube using radio amateur frequencies.

The Global Educational Network for Satellite Operations (GENSO) is a project carried out under the auspices of the International Space Education Board and coordinated by the Education Department of the European Space Agency (LEX-E).

The project aims at providing a mutually beneficial service to all educational satellite projects by combining their ground stations into a collaborative, internet-based network, vastly improving mission return by increasing the amount of available communication time.

Three different types of ground-stations might be considered:

- **Radio Amateur Shack:** This ground station is operated by a radio amateur somewhere on Earth. All types of hardware should be expected, including the manual pointing of an antenna. The data integrity for this ground station types is typically low since its material is not known a priori. Nevertheless this could be interesting choice for beacon data recovery.
- **GENSO:** Allows the user to remotely operate a Ground Station, the big difference compared to the radio amateurs is that the ground station can be operated by SwissCube staff or integrated into an automated operations scheme. In order to operate within GENSO the SwissCube stations have to use GENSO software and operation time has to be given to other satellite operators.
- **SwissCube:** Major drawback of one ground station is its small coverage.

	Radio Amateurs	GENSO	SwissCube
Number of stations	> 100	3-10	2
Coverage	Global (++)	Global (+)	Local (--)
Operator	Radio amateur	SwissCube staff (Remote operation)	SwissCube staff
Beacon data integrity	+	+	++
TC integrity	--	+	++
TM integrity	-	+	++
Constraints	Need to create a web-interface to recover/send data AMSAT requires to provide some radio amateur I/F when using their frequencies	Need to make GS compatible with GSN	No constraints

Table V-3: Comparison of potential SwissCube Ground Stations

In conclusion, it would be very advantageous to participate in the GENSO for satellite operation which would significantly increase coverage time. Further Radio Amateurs should be included to obtain extra housekeeping data. This is particularly true during LEOP where the satellite should be monitored as often as possible to determine possible errors.

3 Mission Control System

3.1 Design requirements

The mission control system design shall provide the following basic functions:

- Telemetry reception & processing
- Telecommanding (manual, automatic)
- Data displays and prints
- Real time updates
- Data storage (archiving) & retrieval

The system shall have a client – server topology where the various subsystems/flight operators shall monitor and command the satellite from terminals connected to the operations server. More advanced features such as the display of the subsystem state in graphical form (synoptic pictures) can be added once the core elements have been designed and implemented.

For mission control the approach shall be consistent with the various operational phases of the mission. Two principal modes have been identified, namely;

- In-orbit check-out of the satellite (LEOP & Safe-Mode) during which only basic commands are sent one-by-one and the results monitored in real-time.
- Autonomous operation. In this mode the ground segment is "pre-programmed" by satellite operators in advance of a pass. During the pass the ground segment performs autonomous operations by downloading telemetry data and uploading new telecommands into the satellite's scheduling system.

3.2 Design overview

The architecture of the space ground segment used is similar to the ESA ground segment. It uses a modular architecture because all components are connected to a software router (EGSE Router) developed by ESA.

The Mission Control Software (MCS), also called **GSS** for **Ground System Software**, is a monitoring and control system. Its role is to send telecommands to the spacecraft (or SCOE), and to manage the telemetries received. For this task, ESA uses the system SCOS 2000 developed by ESOC/ESA and industrial partners. SCOS 2000 is a generic system and is functionally complete, but this makes it complex to configure and not quite adapted to small spacecraft. The next section describes the GSS developed for the SwissCube project.

The user interface is vital to the GSS. SCOS 2000 has an interface to view the housekeeping, to send telecommands, etc. and a programmable one used to develop the mission specific parts. In the SwissCube GSS, we have a core and three system interfaces used to create clients around it. Four clients have been developed so far (see [35]). Figure V-6 shows the interfaces to the router.

The ground station has an antenna and uses a small software application to communicate with the spacecraft. The SCOE is a test system used to test a subsystem. The simulator is used to simulate the behavior of the spacecraft when testing the GSS or the user interfaces.

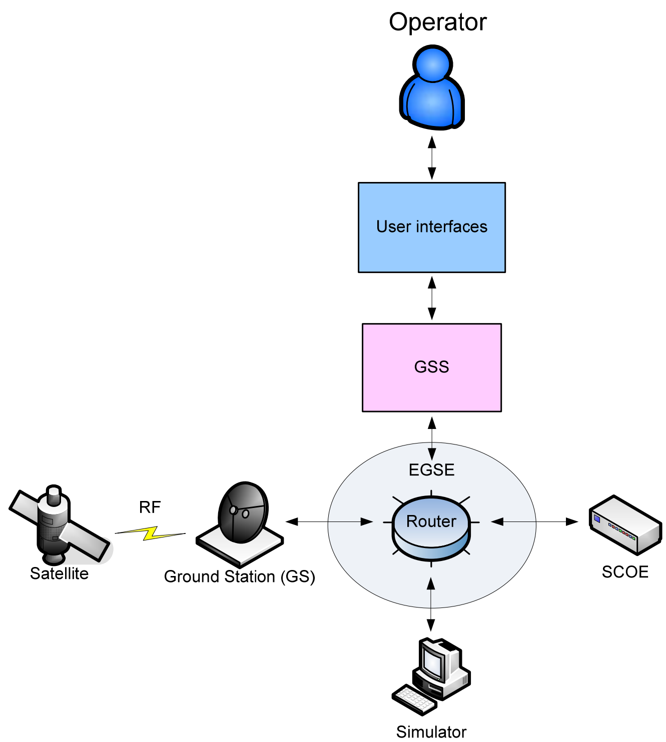


Figure V-6: Ground System router interfaces.

3.2.1 Ground System / Mission Control Software Architecture

Figure V-7 shows the architecture that have beend designed and implemented. All the components are detailed in the subsequent chapters.

The architecture of the GSS contains three layers. The first one at the bottom is composed of the Core and the TC/TM Catalog Access where all the logic and processing are done. Above it are the system interfaces that expose the system to the outside with the use of Web Services and provide things like security, data validation, etc. The system is interoperable so that clients or other project developers, the third layer, could develop their interfaces with various technologies via the use of Web Services.

The main part of this architecture is the Core. This is the central part where of all the processing of the data, to sent or receive is done. The core is designed so that there is no mission specific processing done in it (except the monitoring modules) and thus can be kept the smallest possible to maximize its reliability and scalability.

At the client level, the monitoring (rich monitoring client) and the control (TC manager client) are well separated. This was done after much discussion and with the input from experts from ESA. With the separation, the decisions are more thoroughly thought of or are even taken by two people;

one that monitors and the other that controls. This avoids controlling errors when it's too easy to interact with the spacecraft.

At the other side of the system, all the communication is done via the EGSE Router, which is a routing software developed by ESA. It was chosen as it makes the system independent of the actual communication with the spacecraft (not well defined at this point). Furthermore it allows the use of tools developed by ESA such as a "Spy" that capture all the traffic for debugging purpose.

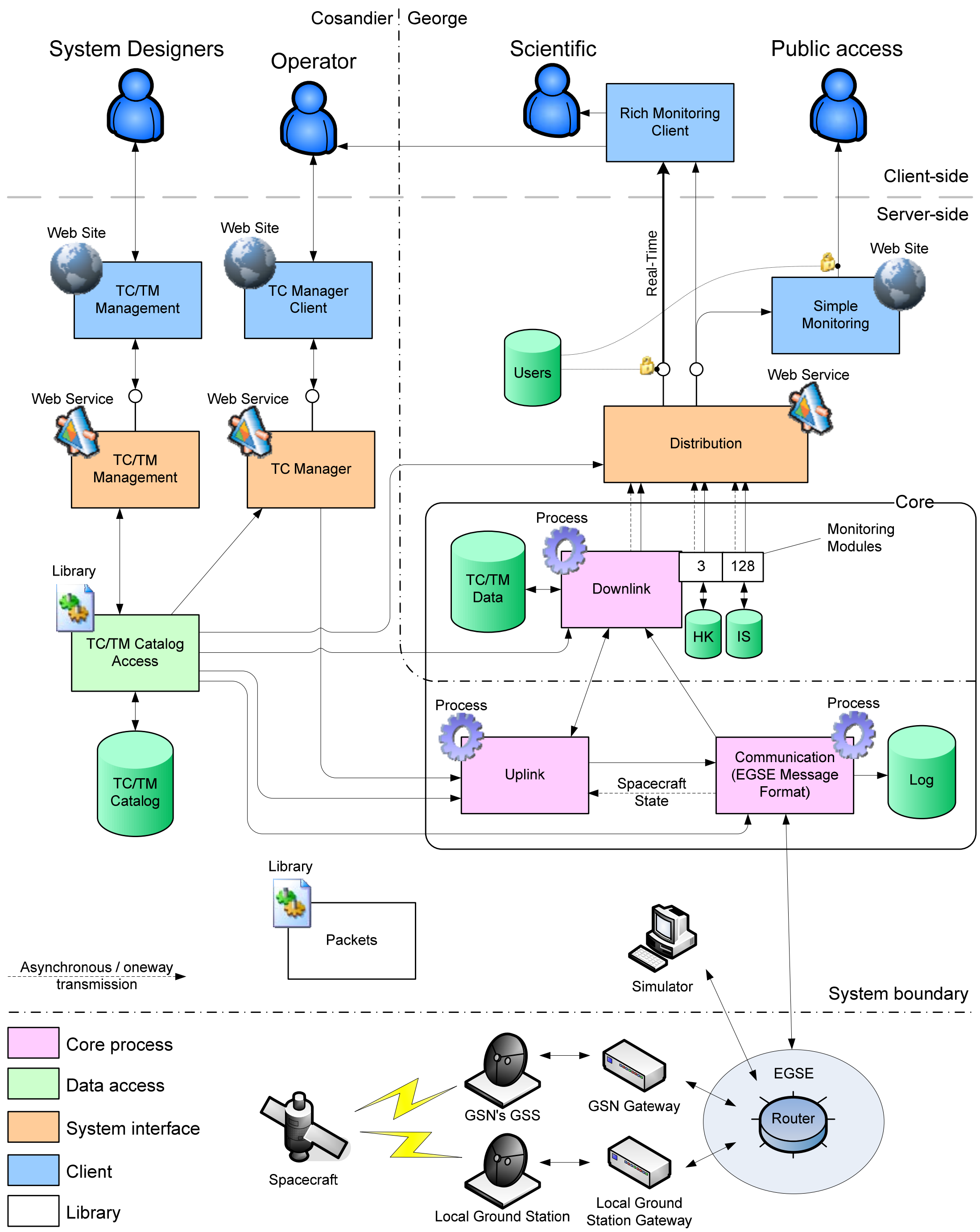


Figure V-7: GSS and user interface architecture.

3.2.2 Programming environment

Based on experience and knowledge, the OS chosen to develop the software was Microsoft Windows.

There were four possible styles of programming at our disposal:

- Native (C, C++): offers great performances, but for a program that must be able to run 24/7 for weeks (for example no memory leaks allowed), this was not a reasonable choice as we don't have the necessary experience and time to attains such quality with this kind of programming.
- LabView: Used by ESA for the new version of their EGSE Router, the choice to use it was taken seriously. But due to a lack knowledge of it and the completely different approach to programming, it would have required too much time to learn it.
- Scripting (Python, Perl, etc.): Dismissed directly due to the lack of type-safety, scalability and reliability when working with them.
- Managed (.NET, Java): These technologies bring great RAD (Rapid Application Development) and come with great framework and libraries for almost everything. The greater memory usage is outweighed by the automatic memory management especially needed for long running applications that must not suffer from memory leaks.

As a result, a managed programming environment was chosen. Based on internal experience, the selection of .NET was straightforward. Consequently, the programming language (the .NET platform support numerous programming languages) chosen is C# 2.0 as it is the most used and best supported on the .NET platform (it was developed in conjunction of it).

Having chosen the .NET Framework running on Microsoft Windows, the most natural choice of database software was Microsoft SQL Server for its great integration with the two. So no external library is needed to access the database and it adds performance counters to Windows that enable us enhanced debugging and scalability tests.

For further details about each of the elements, please consult:

- Ground System Software Description and User Manual [35, 36]

3.2.3 User Interfaces

This section briefly describes the user interfaces. More details can be found in [36].

To manage and configure the GSS, a "TC/TM Management" website was created (see Figure V-8). The telecommands can be sent by the "TC Manager Client". The monitoring can be performed via two clients, a website and rich client. The rich client is real-time and can display all the information relating the monitoring to the spacecraft operator. Simple monitoring is a simpler web-based interface available to display the monitoring for all those interested without the need to install a client. It is also intended to provide access to the telemetry for the public.

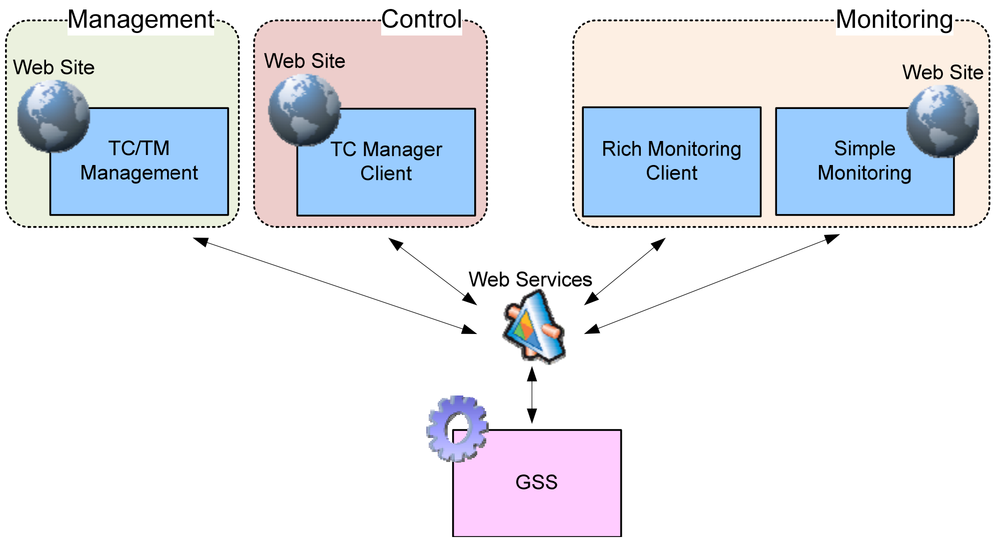


Figure V-8: User Interface to the GSS.

TC/TM Management

The TC/TM Management is a website to configure the GSS, and to parameterize the mission telecommands and spacecraft parameters (see Figure V-9). This website is used before the launch.

The GSS Configuration page is used to configure the "spacecraft ID" and mission description.

The Mission Configuration page is used to configure mission constants. In this version of GSS only the "Telecommand Checksum Type" and the "Telemetry Checksum Type" field is used by the system. The GSS supports the insertion and the modification for all parameters. It is only the interface which currently limits the configuration of certain configurations.

The APID (Application Process ID) uniquely corresponds to an on-board application process. This is used in the CCSDS packet header and the choice of APID values is mission-specific.

The service type defines which PUS service is used. Service types 0 to 127 shall be reserved for the standard ECSS-E-70-41A, service types 128 to 255 are mission-specific.

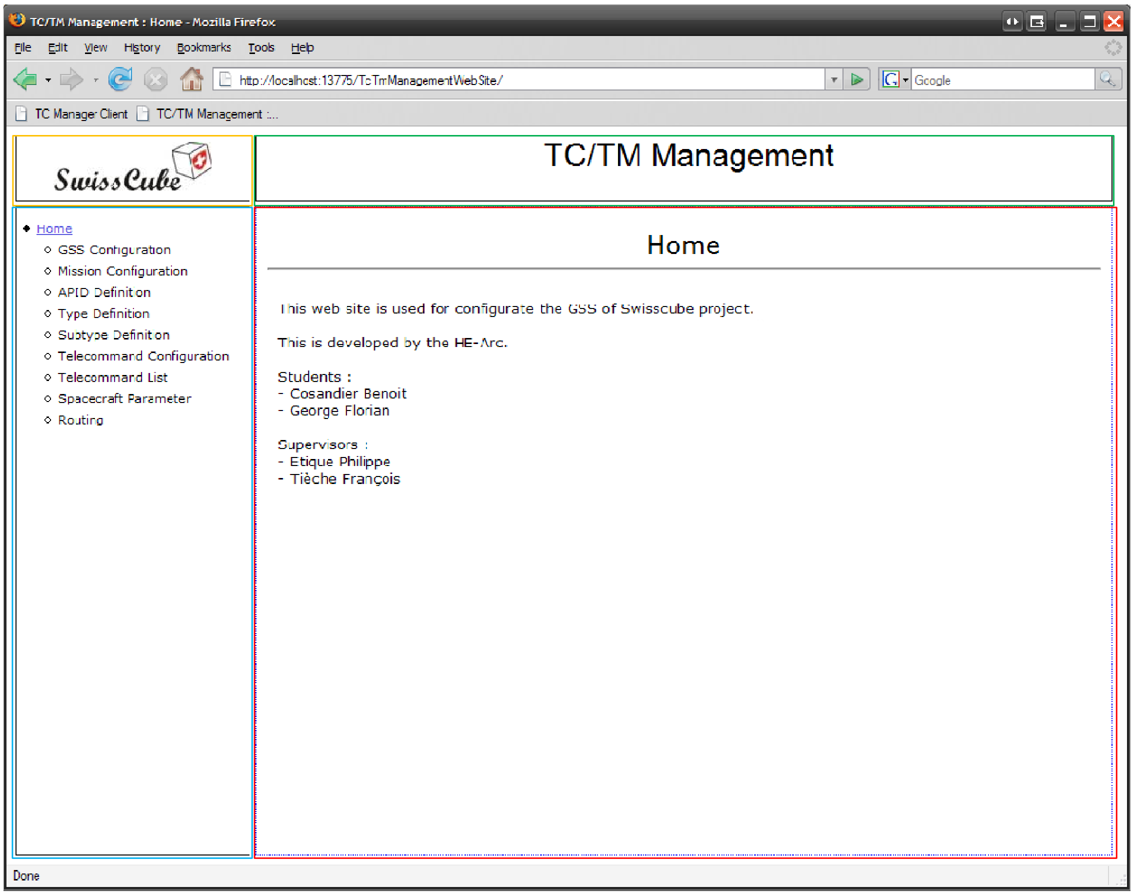
With a service type, the subtype uniquely identifies the nature of the service. Within standard services, subtypes 0 to 127 shall be reserved in the standard ECSS-E-70-41A, subtypes 128 to 255 are mission specific. Within mission-specific services, all subtypes (0 to 255) are available for mission-specific use.

The Telecommand Configuration page is used to define the telecommands and associated parameters.

The Telecommand List page lists the telecommands. This page it is used to modify and delete a telecommand. It is in relation with the page "Telecommand Configuration".

A spacecraft parameter is a housekeeping parameter which can be reported from the spacecraft to the ground station and and which specifies how the GSS should treat the parameter.

The GSS uses a system of routing. Each APID correspond to a route (the ID in EGSE Router). When a telecommand is sent by the GSS, this table is used to check where the telecommand must be sent. If the route for a APID is not defined, the GSS must use the route defined for the APID = 0, hence the APID = 0 is the default route.



Logo area

Title area

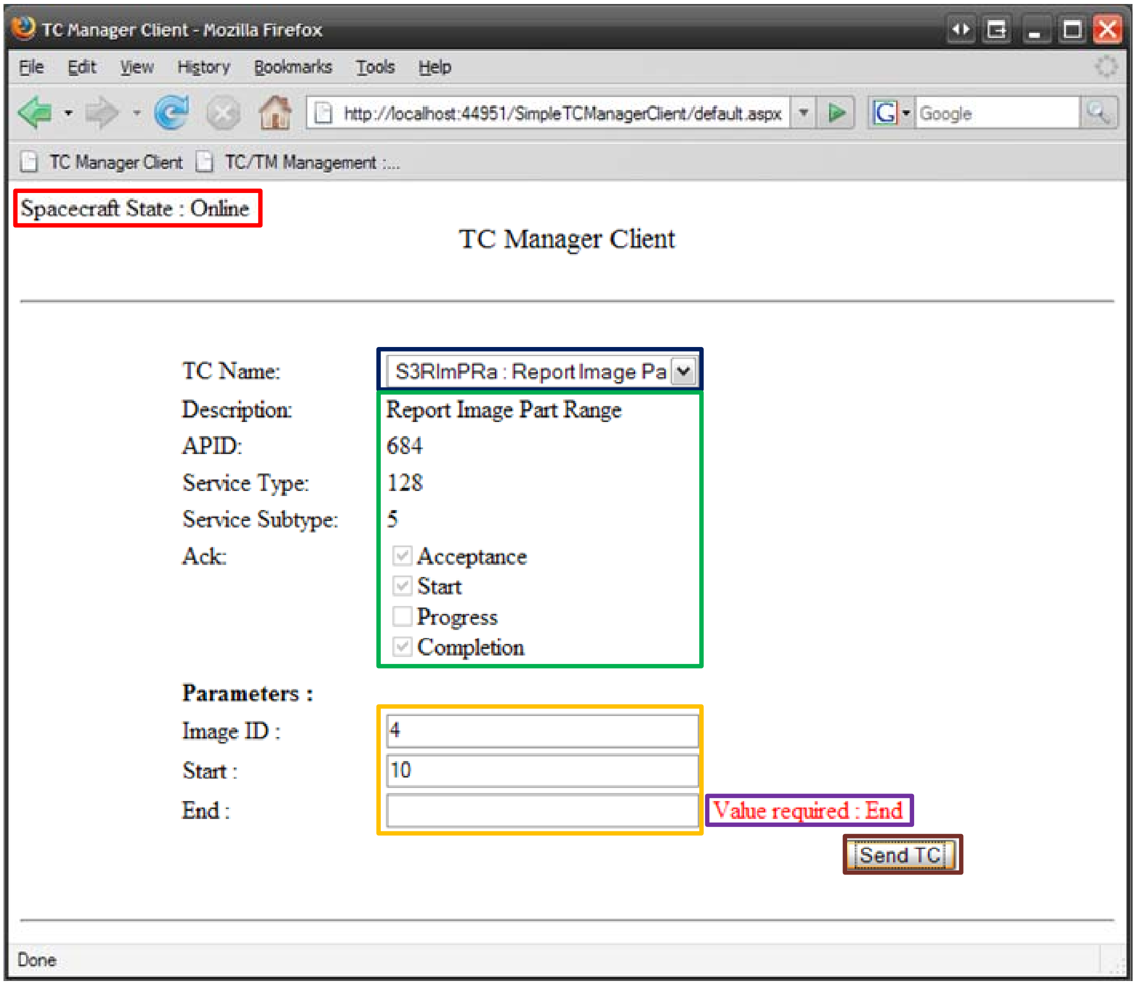
Menu area : Used the menu for selected which component you want to configure.

Content area : it's this area who display the configurations for TC/TM Management.

Figure V-9: TC/TM Management Structure.

TC Manager

The TC Manager is used to send a telecommand. This client permits sending the telecommands defined in the TC/TM Management site section. Figure V-10 shows the interface.



- Display the satellite's state (Online or Offline).
- Select the Telecommand by the name.
- Display the description of the telecommand, who is selected in the Dropdown List.
- Enter in these textboxes the parameters of the telecommand.
- It's a validator, which is display only if the textbox on the left are empty.
- Click on this button when you want send the telecommand.

Figure V-10: TC Manager Client interface.

Simple Monitoring

This web application provides a simply mean to visualize all the monitoring data. Only viewing the telemetry is publicly available, the other sections require to be authenticated.

The TC page shows all the telecommands sent or waiting to be sent. They are displayed in reverse chronological order (most recent first). See Figure V-11 for an example.

1 2 3 4 5 6 7 8 9 10 ...

Token	Date	Time	APID	SF	SC	ST	SST	R	G	T	A	S	0	1	2	3	C	Data
1362	23/12/2006	22:09:47	684	0	973	128	1	O	O	O	O	O	O				O	-
1361	23/12/2006	22:02:24	684	0	972	128	1	O	O	O	O	O	O				O	-
1360	23/12/2006	21:50:34	684	0	971	128	1	O	O	O	O	O	O				O	-
1359	23/12/2006	21:33:16	684	0	970	128	4	O	O	O	O	O	O				O	00-04-01

Figure V-11: Example of TC queue.

A similar page shows all the telemetries received. They are displayed in reverse chronological order (most recent first). See Figure V-12 for an example.

1 2 3 4 5 6 7 8 9 10 ...

Id	Token	Date	Time	APID	SF	SC	ST	SST	Data
3920	1362	23/12/2006	22:09:48	684	3	973	128	2	00-03-45-8D-90-17-00-04-...
3919	1361	23/12/2006	22:02:25	684	3	972	128	2	00-03-45-8D-90-17-00-04-...
3918	1360	23/12/2006	21:50:34	684	3	971	128	2	00-03-45-8D-90-17-00-04-...
3917	1359	23/12/2006	21:33:16	684	3	970	128	6	00-04-01-08-19-29-21-49-...
3916	1358	23/12/2006	21:28:57	684	3	969	128	2	00-03-45-8D-90-17-00-04-...

Figure V-12: Example of TM queue.

Another page displays all received values (with their reception date and time) for the selected housekeeping parameter. See Figure V-13 for an example.

Parameter: EPSV0003

EPSV0003

Date	Time	Value
11/12/2006	14:07:39	90
11/12/2006	14:07:34	160
11/12/2006	14:07:30	105
11/12/2006	14:07:24	190
11/12/2006	14:07:14	190

Figure V-13: Housekeeping parameter display.

A fourth page provides the means to retrieve the payload images received. See Figure V-14.

Figure V-14: Example image retrieval.

Rich Monitoring Client

The Monitoring Client provides both a rich and interactive interface in order to visualize all available monitoring related information regarding the spacecraft. The main window of the application that is displayed when launching the client is shown in Figure V-15. It is from this window that the different modules can be launched.

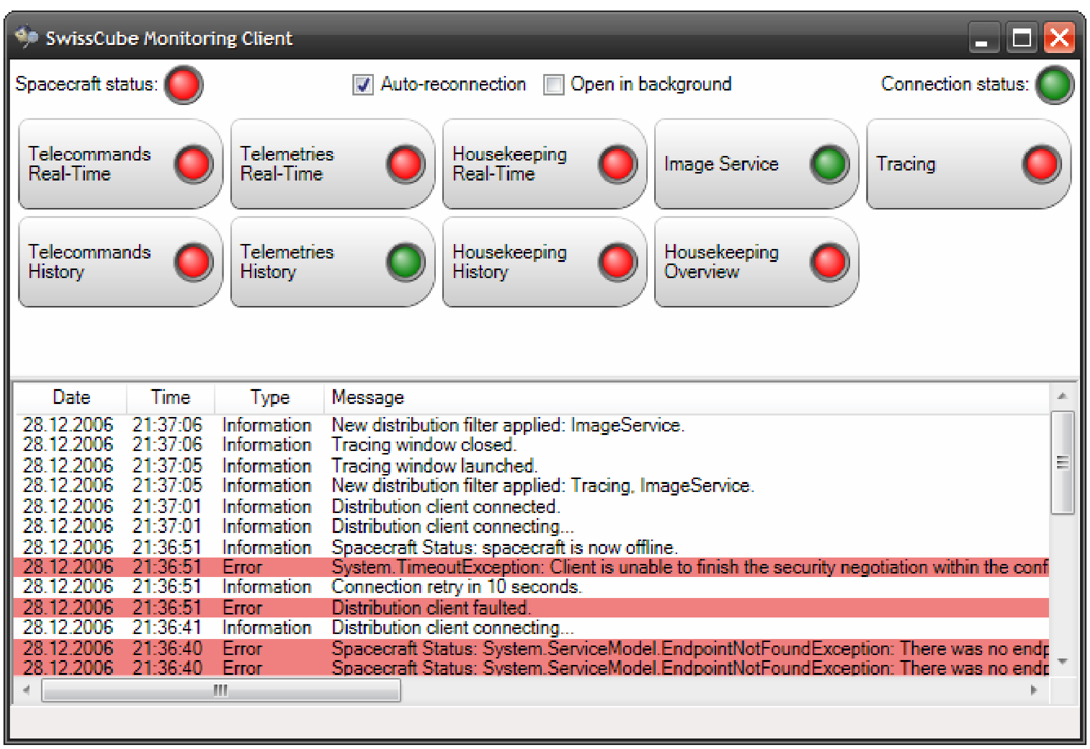


Figure V-15: Rich monitoring client display.

1. Spacecraft Status: indicates the current status of the spacecraft and is periodically updated (Green=online, Red=offline).

2. Auto-reconnection: if checked, the client will automatically try to reconnect to the distribution server if the connection is lost or can not be established.
3. Open in background: if checked, the modules windows will be opened behind the main window.
4. Connection Status: indicates the current status of the connection (Green=connected, Red=disconnected, Orange=fault)
5. This zone contains a list of all loaded modules. The application comes with bundled modules that are described in [36]. The light indicates whether the module is open (green) or closed (red). When a real-time module is closed, real-time data is not sent by the server and thus does not consume network bandwidth.
6. Message window: contains information about the client itself: such as important messages, warnings and error messages.

3.2.4 Further developments

Further developments include:

Monitoring data export

Add a functionality enabling the exportation of the monitoring data. For example the housekeeping values could be use in a program like Microsoft Office Excel to perform complex analysis.

Housekeeping graphs

Allow the user to visualise the housekeeping values as graphs. An early prototype has already been developed and is available in the source directory under the name "MC Prototype" (WPF Application).

Definitions import

Add the possibility to import mission definitions in the TC/TM Management. This would enable the transfer of a configuration from a test GSS to another one.

Time correlation implementation

Due to the lack of specification on information on time management aboard the spacecraft, this functionality is not yet implemented.

Log replay

Allow the replay of a communication log. This would provide better debugging support and equipment testing.

Error handling and display

Better handling of the malfunctions or occurring errors and improved quality of the messages displayed in this case so they can be more easily corrected.

Performances

There are places in the software where the performance can be greatly improved like in the Packet library conversion algorithms or by regrouping the Core in only one process.

4 Ground Support Facilities and Test Benches

The SwissCube verification will be done using test benches as early as possible. Functional system test benches shall replace whenever possible software models with hardware in-the-loop.

The check-out system will allow AIT operations to be performed while operating as early as possible using the mission control software (MCS). It will be composed of all the necessary special check-out equipment (SCOE) to validate the satellite. The CORE SCOE will coordinate the work between the various SCOE's. For simplification purposes, CORE SCOE could be integrated with MCS.

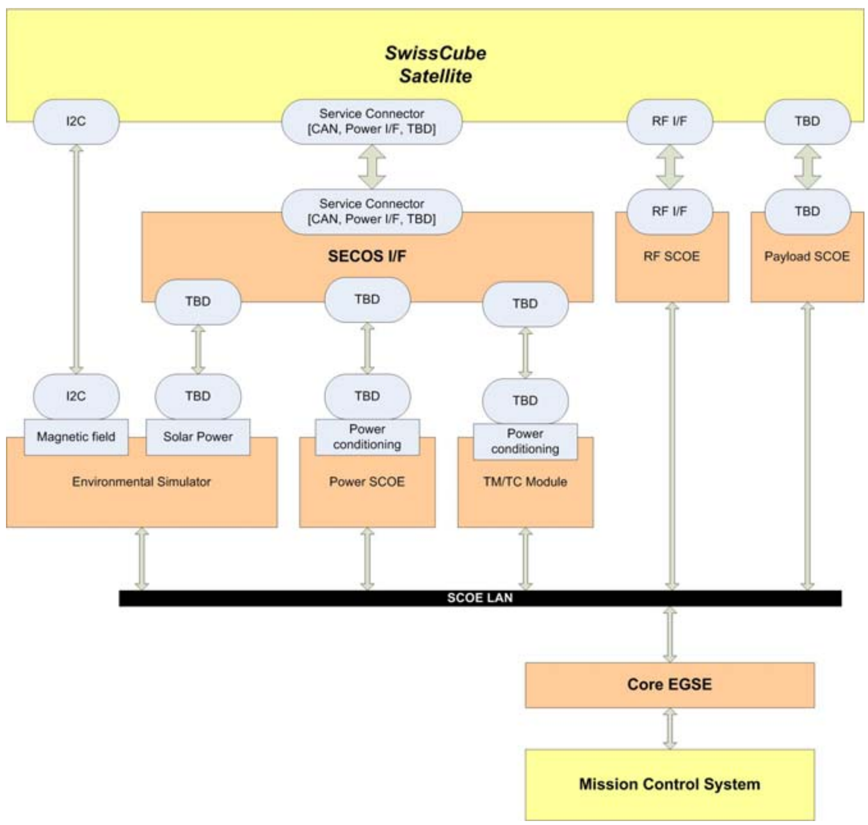


Figure V-16: SwissCube real-time test bench environment

4.1.1 Integration model test bench

The integration model will be functionally representative of the end items in electrical and software terms.

The test bench objectives are:

- to test the functional behavior of the electrical and software parts;
- to test electrical and data interfaces;

- to investigate failure modes.

4.1.2 Engineering qualification model test bench

The electronic equipment of the SwissCube subsystems and the SwissCube Payload will be installed on the EQM test bench.

It will be flight representative in electrical I/F and function, but as baseline only one functional branch will be realized.

The Test Bench configuration will be a 3-D representative and it will be flight representative w.r.t harness length and orientation as well as the unit position and orientation.

The build standard of the EQM Test Bench will not be identical to flight units: there may be deviations regarding internal redundancies, mechanical interfaces, parts quality, electrical characteristics (e.g. transient behavior).

The main objectives of the EQM Test Bench are:

- Check the electrical and functional interfaces between the units.
- Verify the functionality of the avionics subsystems and on-board software.
- S/C autonomy functional verification
- Validation of communication and power interfaces between the payload and the platform.
- EGSE validation including the EGSE software and verifying the EGSE capability to perform the planned test.
- Validation of the test sequences to be re-used for PFM test campaign.

The Test Bench will be kept operational all along the AIT sequence to be usable for potential failure analysis or for validation of software modification.

After the integration of all units at platform level (e.g. ADCS, CDMS…) the instrument will be integrated as PFM into the Test Bench for system level tests.

VI CONCLUSION

1 Conclusion

This report summarizes the work performed by the students during their semester and master projects from September/October 2006 to August 2007 (Phase B).

The project now has a good system engineering team in place and has made great progress since the last review (PDR, March 07). In addition, all subsystems have functional models that have been tested and work. Critical components have been identified and design solution found. After tests, some re-design needs to be implemented and the workforce is in place to do this next semester (starting September 07). The critical areas identified during PDR have been addressed, i.e the beacon has been designed and tested, the science payload has now a design for each sub-elements, and the flight software architecture and main function have been identified.

However, there is still a lot of work to be done in many areas. Critical design questions (that could compromise the overall planning) remain in the areas of:

- ADCS control loop: testing of the overall loop with perturbations, implementation on the CDMS computer;
- RF communication link between the satellite and the ground stations;
- Thermal design, especially for the batteries;
- Mission Control (ground software): time correlation design.

Each of these areas will be tested in the upcoming semester. The project is now ready to move forward with the detailed design and tests phase.

VII REFERENCES, ABBREVIATIONS AND DEFINITIONS

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2 Abbreviated terms

ADCS	Attitude Control and Determination System
BOL	Beginning Of Life
CCD	Charged Coupled Device
CDMS	Control and Data Management System
COM	Communication System
DOD	Depth of Discharge
EOL	End Of Life
EPS	Electrical Power System
GaAs	Gallium Arsenide
HK	House keeping data
LDO	Low-Dropout Regulator
Li-Ion	Lithium-Ion
Li-Po	Lithium-Polymer
LV	Launch Vehicle
MPPT	Maximum Power Point Tracking
I^2C	Inter-Integrated Circuit
PCB	Printed Circuit Board
P/L	Payload
PV	Photovoltaic
SEL	Single Event Latch-up
SPAD	Single Photon Avalanche Diode
TBD	To be defined
TC	Telecommand
TM	Telemetry

3 Definitions

Albedo	Albedo is a measure of reflectivity of a surface or body. It is the ratio of total electromagnetic radiation reflected to the total amount incident upon it. The average albedo of Earth is about 30%.
Dropout voltage	The dropout voltage is the minimum difference between input and output voltage that a LDO needs to do the output voltage regulation.

VIII APPENDICES

1 **Appendix A**: Satellite Perturbation Models

1.1 Aerodynamic torque

The interaction of the upper atmosphere with the satellite's surface produces a torque around the center of mass. For spacecraft at below approximately 600km, the aerodynamic torque is the dominant environmental disturbance torque.

The infinitesimal force $d\vec{f}_{aero}$, on a surface element dA with an outward normal $\vec{u}_n$ is given by

$$d\vec{f}_{aero} = -\frac{1}{2} C_d \rho (\vec{u}_n . \vec{V}) \vec{V} dA$$

Where C_d is the drag coefficient (2), ρ the atmospheric density and $\vec{V}$ the relative speed of the spacecraft. This formulation is only acceptable if $\vec{u}_n . \vec{V} \geq 0$ otherwise the force is zero.

1.2 Gravity gradient torque

Any nonsymmetrical object of finite dimensions in orbit is subject to a gravitational torque because of the variation in the earth's gravitational force over the object. This gravity-gradient torque results from the inverse square gravitational force field; there would be no gravitational torque in a uniform gravitational field. For our application, it is sufficient to assume a spherical mass distribution of the earth.

The gravitational force acting on a spacecraft mass element, dm_i, located at a position $\vec{R}_i$ relative to geocenter is

$$d\vec{F}_i = -\frac{\mu \vec{R}_i dm_i}{R_i^3}$$ where μ is the earth's gravitational constant.

Let $\vec{R}_s$ be the position of the geometrical center of the spacecraft expressed in the BRF and $\underline{\underline{I}}$ the moment-of-inertia tensor in the BRF (diagonal matrix).

Then the gravity-gradient torque can be expressed as

$$\vec{N}_{gravity} = 3\frac{\mu}{R_s^3}\left[\vec{R}_s \times (\underline{\underline{I}} \bullet \vec{R}_s)\right]$$

1.3 Magnetic disturbance torque

Magnetic disturbance torques results from the interaction between the spacecraft's residual magnetic field and the geomagnetic field. The primary sources of magnetic disturbances:

- Spacecraft magnetic moments (magnetotorquers), since magnetotorquers are already taken into account in the satellite dynamics;
- Eddy current loops: the eddy current produce a torque which precesses the spin axis and also causes an exponential decay of the spin rate. This torque is given by

$$\vec{N}_{eddy} = \kappa_e(\vec{\omega}'\times\vec{B})\times\vec{B}$$

Where κ_e is a constant coefficient which depends on the spacecraft geometry and conductivity and $\vec{B}$ the geocentric magnetic flux density.

Eddy currents are appreciable only in structural material that has a permeability nearly equal to that of free space (as copper or aluminum).

- Hysteresis

In a permeable material rotating in a magnetic field, energy is dissipated in the form of heat due to the frictional motion of the magnetic domains. In our case, no high permeability material where used.

The disturbance generates by the magnetotorquers were already computed in the attitude control and determination algorithms.

A rapid estimation of the Eddy current loops effect gives an order of magnitude of 10^{-15} N.m which can be neglected.

Further measurements have to be performed on the residual magnetic dipole of the SwissCube.

1.4 Solar radiation torque

Radiation incident on a spacecraft's surface produces a force which results in a torque about the spacecraft's center of mass. The surface is subjected to radiation pressure or force per unit area equal to the vector difference between the incident and reflected momentum flux. Because the solar radiation varies as the inverse square of the distance from the Sun, the solar radiation pressure is essentially altitude independent for the spacecraft.

Assuming that absorption, specular and diffuse reflections all play a part (without any transmission), then the total solar radiation force is

$$d\vec{f}_{solar} = -\frac{F_e}{c}\left[(1-C_s)\vec{u}_s + 2(C_s(\vec{u}_s.\vec{n}) + \frac{1}{3}C_d)\vec{n}\right](\vec{u}_s.\vec{n})dA$$

where F_e is the solar constant, c the light speed in vacuum, C_s the specular reflection coefficient of the elementary surface, C_d the diffuse reflection coefficient of the elementary surface, $\vec{n}$ the outward normal and $\vec{u}_s$ the unit direction of the sun.