COMPASS-1

Phase A Study

2003-Nov-06

Hochschule Aachen

Preface

This paper reflects the results of a four-weeks effort of a group of eight students elaborating the phase A study of a pico satellite project, Compass-1. The contribution from all of them has led to an extensive documentation about the feasibility and to the possibility of the project realization.

The purpose of this paper is twofold. Primarily of course it shall path the way for the later research and facilitate the design process. The given results provide an essential point of departure for the following works. By reading through its content, everyone will be able to get an complete understanding of the matter of the Compass-1 satellite project.

In addition, this little book might also serve as a guideline for next generations of space technology students. Perhaps in a possible line of CubeSat developments at the FH Aachen, this work can help to get other projects started.

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Contents

Abbreviations

ADCS	Attitude Determination and Control System
BOL	Beginning of Life
C&DHS	Command and Data Handling System
CAM	Camera
CCD	Charged Couple Device
CMOS	Complementary Metal Oxide Structure
COM	Communications System
COTS	Commercial-of-the-shelf
EM	Engineering Model
EOL	End of Live
FM	Flight Model
FOV	Field of View
LEO	Low Earth Orbit
MCU	Micro Controller Unit
OBC	On Board Computer
PCB	Printed Circuit Board
P-POD	Poly Picosatellite Orbital Deployer
PSU	Power Supply Unit
SNR	Signal-to-Noise ratio
STM	Structural/thermal Model
TBD	To be determined
TCS	Thermal Control System
WBS	Work Breakdown Structure

Introduction and Background

When we look at the present activities on the space market we can identify a trend that goes to the development of small to very small satellites. The reasons for that are obviously the reduced costs and shorter development time. When we take the launch costs in proportion to the payload mass, we will quickly understand that it is worth to save every kilogram. This calculation is of course not applicable for all cases, especially not for primary payloads. In case however, we consider launching a satellite as a secondary payload or piggybacking it, we definitively can relate to the previous statement. This is true for many smaller companies and for universities as well.

It should be mentioned that small satellites in this context still are quite large and have masses of even more than 100 kilogram. But there is a new borderline for satellites that attracts increasing interest by the space community, especially by universities. These are pico satellites with a mass below 1 kilogram.

In the end of the last decade, professor Robert Twiggs from Stanford University in the U.S. proposed the concept of a cubic pico satellite bus, which was named CubeSat. It defines a 10x10x10cm cube with a mass of maximal 1kg. A group of students at CalPoly University developed a deployer for those satellites, which goes under the name P-POD. With it, three CubeSats can be deployed into orbit at a time. This deployer is also the main advantage of the concept, because it facilitates the interface definition of satellite and launcher. The CubeSat groups only have to meet the requirements of the P-POD and do not have to worry about the launcher itself. The other point is that the launching costs can be further reduced since it will be split up among the participating groups.

A lot of universities worldwide have started to work on their own CubeSat projects. Six of them are already in orbit, launched in end of July 2003. The contact with this groups and their published documentation makes it easier for prospective developers to step on ground in this new area.

The initial idea to build a student satellite at our university was encouraged by Prof. Dr. Wittmann, who is the Director of GSOC (German Space Operation Center). To accomplish such a complex enterprise we were in need of some guidelines. By chance we discovered the CubeSat concept. The very low costs, rapid timescale and manageable proportions made this concept very attractive for us. It will help to provide us with excellent expertise in space technology through an affordable small satellite project.

Since we plan to use an amateur frequency band for the communication architecture we feel committed to emphasize on a common interest in terms of mission operations. Therefore we have worked on the mission design in such a way, that radio amateurs worldwide can benefit from this satellite as well. We aspire a strong collaboration with the radio amateur community and look forward to receive vital feedback from them.

Mission Analysis

Artur Scholz

1. Introduction

The mission analysis is the first thing to do when we talk about a project. This is so, because we want to know what the project is aiming at. What are the goals and why do we want to put such a lot of effort into this? Is it feasible? Is it worth doing it? In the following chapter we will exam the basic idea behind Compass-1 comprehensively. After all, we want to answer the last two questions with a 'yes'!

2. Mission Objectives

Likewise other student satellite projects, our main driver is to actually gain experience, which we could not by just reading books and listening to lectures. The bottom line here is that even so we have profound lectures, dealing with all aspects of space mission design; the learning factor is much greater when putting our own hands on it. But still, we want to achieve a goal as well. Our satellite will be a success when we have finally finished it, but we also want it to operate it in orbit of course. The initial idea that was brought up and which is also a very common one among the other CubeSat groups is to implement a camera on board the satellite and take pictures of the earth. This is a challenging task since it influences virtually all subsystems and therefore their design. Principally, we can distinguish our objectives into two categories, listed below:

Educational objectives:
- collaboration and contacts with industry, universities and other CubeSat groups;
- insight into the system engineering process and team dynamics;
- deeper understanding of subjects.

Scientific / research objectives:
- technology demonstration and validation for space application of COTS products;
- communication link for pico satellite;
- attitude control system for pico satellite.

The following table describes the distinct levels of success for the Compass-1 project.

Table 1.2.1: Mission Success Levels

Full Mission	A full mission comprises a reduced mission plus: - a reliable communication and data transfer between spacecraft and ground in both directions; - all subsystems fully operational as described later on.
Reduced Mission	The reduced mission would be a minimum mission plus: - the reception of signals from the satellite.
Minimum Mission	The minimum would be: - the complete development of spacecraft and ground segment; - the launch of the satellite.

3. Mission Design

Now that we have seen the mission objectives we will next formulate a mission statement that helps to underpin the basic idea, respectively the vision and which will be the fix point in the iterative process of mission analysis.

Compass-1 Mission Statement Because of the importance of future-oriented training and motivation of prospective space engineers, we, students of the University of Applied Sciences in Aachen, will develop a pico satellite in accordance with the CubeSat concept. We will conduct a mission, which is to take pictures of the earth and transmit them to the ground.

3.1 Mission Requirements

Taking into account the previous information we will now be able to identify the mission requirements as the next step. The analysis of the mission requirements needs the inputs from the mission statement, the mission objectives and other parameters that predominantly evolve from the project environment. We use the table from Larson & Wertz[1].

Table 1.3.1: Top-Level Mission Requirements

Functional Requirements	
Performance	On request, the satellite takes a pictures of the earth and transmits them. The image area has to be large enough to enable identification of costal lines or regions. The images have to be in color. The camera needs to be aligned with the nadir axis with a pointing accuracy of better than 8°.
Coverage	The precise coverage area is of minor interest but it is strongly encouraged to cover a large piece of populated areas in order to have a broad group of interested users.
Responsiveness	It needs to be ensured that an image can be downlinked during an access frame, which is at least 5 minutes.

Operational Requirements	
Duration	The satellite payload shall operate for half a year from the time of launch.
Availability	Due to security reasons, the gap between access frames shall be below 12 hours.
Survivability	To secure its operational time the spacecraft and its components have to withstand the environmental conditions.
Data distribution	The data will be distributed in two ways. Images can be downloaded by direct access from ground with adequate equipment or via the web site of our group.
Data content, form and format	All data from the satellite will be in clear language, meaning no encryption mechanism is used. The images are in a clear format.

Constraints	
Cost	The cost for the launch is about 30.000 Euro. The total cost budget for the satellite development (including facilities, material and other expenses) is 50.000 Euro.
Schedule	The tight time schedule for the phase B and phase C/D is below one year.
Regulations	For management, product assurance and system engineering proceedings we will act according to ECSS standards. For amateur radio frequency use international regulations apply. The operator needs a radio amateur license. For the development and testing of the CubeSat, Stanford and CalPoly University have published regulations.
Environment	The satellite will be exposed to space conditions in LEO.
Interfaces	The operator interface depends on the chosen communication architecture. The user interface will be through Internet or direct access, respectively. User shall have access to the transmitted images freely by visiting the web page of the Compass-1 project or by using own ground station equipment.
Development Constraints	The spacecraft development has to be done according to the CubeSat specifications and is restricted in mass, size and power consumption.

3.2 Mission Modes

This section deals with the various mission modes that are implemented in the spacecraft mission operation. The control mode is made up of functions to guarantee a regular operation mode, in terms of attitude stabilization and other crucial adjustments. The regular operation modes consist of the functions the spacecraft shall supply to the end-user. The emergency mode though will only be active if something on the spacecraft goes wrong, e.g. the power subsystems doesn't supply sufficient energy.

3.2.1 Control Mode

The control mode is necessary for the spacecraft to fulfill the mission requirements on the spacecrafts attitude, i.e. the pointing accuracy. Therefore it has to secure a stabilization of the satellite shortly after its deployment from the P-POD. This phase is called detumbling and will be described in more detail in the ADCS subsystem section in chapter 3.

Based on a periodical time basis the system switches into control mode to acquire the spacecrafts attitude and measure its discrepancy from the nominal value. If the discrepancy exceeds the tolerance level, it uses actuators to adjust the attitude. When finished, it returns to regular operation mode.

3.2.2 Regular Mode

In this paragraph we will get an understanding of how the spacecraft will operate when it is in orbit and functions as it is supposed to. Here we need to separate two main operational aspects. There is one method that is intended for any user and another one that is reserved for the operator only. Let us first define what we understand under the terms user and operator:

- A user is any person that holds a ham license and has the necessary equipment to communicate with the satellite.
- The operator either consists of students involved in the project or persons authorized by them. Of course the operator status also comprises the user status.

User Interaction:

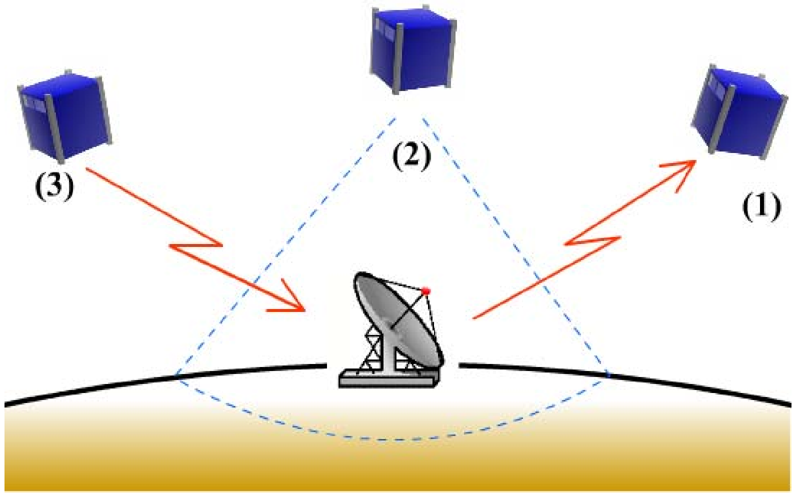


Fig. 1.3.1: User Mode Operation

When the satellite is in view of a user, the user can send a command to the satellite to take a picture (1). In the next step the OBC checks if the available power is above a minimum margin and if so, it commands the payload to capture an image (2). Just after that, the communication system downloads the taken picture to the user (3).

Operator Interaction:

For the operator the above mode is valid also. In addition the operator is authorized to send commands concerning the housekeeping and control of the spacecraft and furthermore advanced imaging commands.

A request from the operator to the spacecraft causes it to downlink the housekeeping data log. The log consists of essential information about the satellite's vital status from various sensors.

The operator can send a command to the satellite that holds information about when to take a picture and where to store it in memory. In this way different places can be photographed. The operator can then download the various memory slots separately and publish it on the web page.

3.2.3 Emergency Mode

If for any reason, crucial components of the spacecraft do not work in the expected way, and will therefore hinder the spacecraft to go into the regular operation mode the satellite will react with sending emergency signals. The system switches to the emergency mode and sends beacons periodically. In case that the problem resolves, the system switches back from emergency mode to regular mode.

3.3 System Operation

The discussed modes are presented in a block diagram in figure 1.3.4. A sensor of the ADCS periodically measures the discrepancy of the satellites attitude to the given tolerance and inputs this data to the computer. In case the satellite has drifted too much from the required position, the system switches to the control mode and corrects this drift. When finished, the system goes back to regular mode. The emergency mode is expected to occur seldom or hopefully never during mission life time and has its main purpose to accelerate the recharge process of the batteries by running the system with very reduced power consumption. All regular functions are off in this mode. When the problem is solved, e.g. the battery is recharged, it goes back to regular mode.

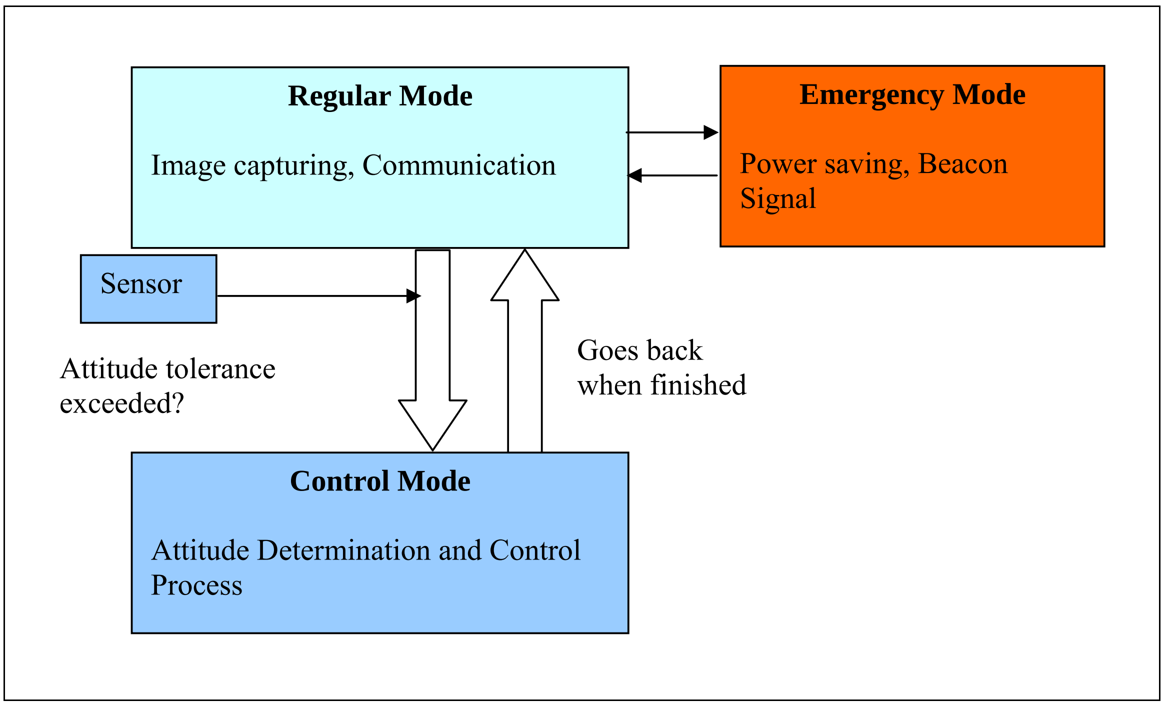


Fig. 1.3.4: System Operation Block Diagram

4. Launch and Deployer

From the very start of a spacecrafts development process the final launch needs to be taken into account. The main reason for that is because the launch creates the strongest mechanical loads on the spacecraft. Moreover we have to design the satellite in a way, that it does not interfere the launch rocket or other satellites and CubeSats. This section deals with the possible launch rockets and describes the carrier of the CubeSats, the P-POD. In addition, we will examine the initial boot sequence, which begins right after the deployment from the P-POD.

4.1 Launcher

The first set of CubeSats was launched on June 30, 2003 with the Rockot by Eurockot. On board were CubeSats from Japan (XI-IV and Cute-I), Denmark (AAU CubeSat and DTUSat), Canada (CanX-1) and U.S. (QuakeSat). It should be mentioned that the Japanese did not make use of the P-POD but developed their own deployer. The orbit is a near sun-synchronous (inclination of 98.73°) with mean altitude above the geoid of 820km. For the near future yet there is no launch opportunity with Eurockot obtainable. The next launch of CubeSats is scheduled Sep. - Nov. 2004 with a Dnepr rocket as shown in figure 1.4.1. This launcher goes in a sun-synchronous orbit with an altitude of 500-700 km and an inclination of 98 degrees. 4-5 P-PODs will be considered for this launch. Although there are many applicants, there are still places offered. The launch costs are expected to be below 30k$.

Even so we are not yet in the position to fix a launch contract for Compass-1, the launch parameters provide us with numbers we can stick to. We assume an orbit altitude of 600km and a circular orbit with a high inclination of about 98° for future launches. With this orbit we cover a wide range of other orbits as well, as lower inclination would just simplify the system requirements, e.g. the ADCS and the power subsystem. The orbit parameters however shall not vary too much from the assumed ones, i.e. the inclination needs to be high enough to ensure a reliable access with our ground station. Also the altitude should by no means exceed 700km because the communication would suffer critically.

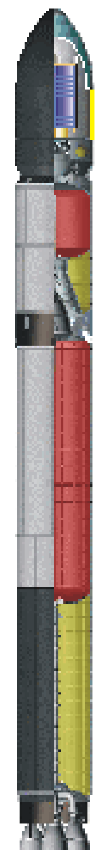


Fig 1.4.1: Dnepr

Figure 1.4.2 displays the separation sequence of the primary payload, the P-POD and eventually the CubeSats.

Although as mentioned above we do not hold a launch contract yet, we want to have our CubeSat to be developed and ready according to the time schedule. It is better to have a finished satellite on the desk than to launch a half finished one. Thus we will be in the position to realize any launch opportunity that might emerge quickly and unexpected.

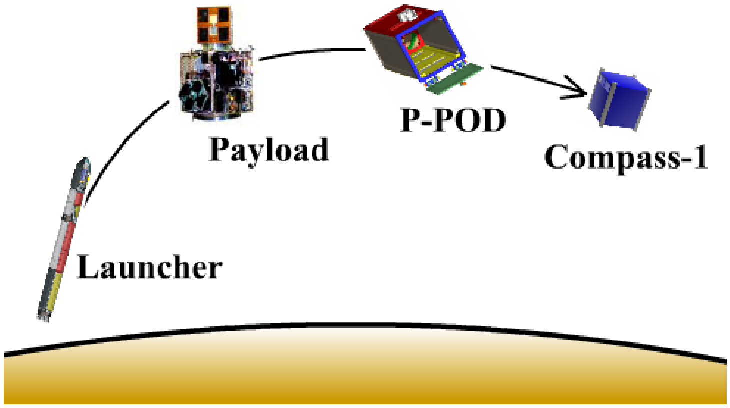


Figure 1.4.2: Launch Sequence

4.2 P-POD

The P-POD is the carrier of the CubeSats. It hosts up to three of them at a time. For the design of the satellite we have to concentrate on the requirements from the P-POD supplier. The general guidelines for a CubeSat are:

- CubeSats must not present any danger to neighboring CubeSats in the P-POD or to primary payloads, meaning that all parts must remain attached to the CubeSats during launch, ejection and operation. No additional space debris may be created. Furthermore CubeSats must be designed so as to not jam on ejection.
- All satellites must be powered off during integration and launch to prevent any electrical or RF interference with the launch vehicle and primary payloads.
- CubeSats must use designated space materials approved by NASA (http://epimsogsfc.nasa.gov/og/) to prevent contamination of other CubeSats and primary payloads during integration, testing, and launch.
- Cal Poly and Stanford hold final approval of all CubeSat designs. Any deviations from this document must be discussed with Cal Poly/Stanford launch personnel before the final CubeSat design is approved for launch.
- Absolutely no pyrotechnics are allowed inside the CubeSat.
- The P-POD ejects the CubeSats with an exit velocity of no greater than 0.3 m/s.
- A final check of specifications will be conducted prior to launch.

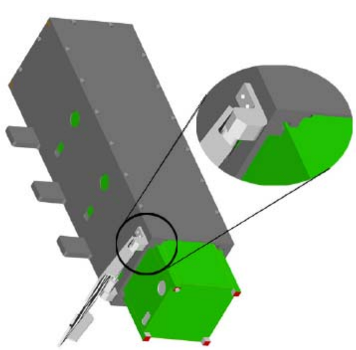


Fig. 1.4.3: P-POD ejecting CubeSat

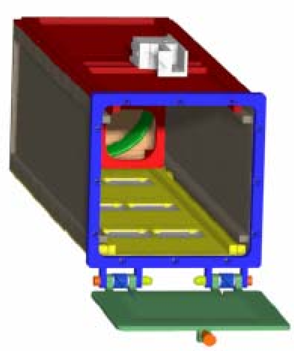


Fig. 1.4.4: P-POD Model

Table 1.4.1: Specific Requirements for CubeSats

Mass
- Each satellite may not exceed 1 kg of mass. - The CubeSat center of mass must be within a sphere with a diameter of 2cm of the geometric center.
Structure
- All edges that contact the rails must be rounded. CubeSats must have at least 75% (85.125 mm of a possible 113.5mm) of flat rail contact with the deployer. - To prevent cold-welding, raw metal is not allowed as the contact surface of the bottom standoff. Derlin inserts, or a hard anodize are examples of acceptable contact surfaces. - The outer surfaces of the CubeSats are required to be hard anodized in order to prevent wear between the sliding rails and the CubeSats. - Separation springs must be included at designated contact points. Recommended springs are manufactured by M.J. Vail part number SSMD-51P (http://www.mjvail.com), can also be located at McMaster-Carr part number 84985A76 (http://www.mcmaster.com). A custom separation system may be used upon approval by Cal Poly/Stanford launch personnel. - One deployment switch (also called kill switch) is required (two are recommended) for each CubeSat. The deployment switches should be attached to the top surface of at least one of the four feet of the CubeSat.
Material
- The use of Aluminum 7075 or 6061-T6 is suggested for the main structure. If other materials are used, the thermal expansion must be similar to that of Aluminum 7075-T73 (the P-POD material) and approved by Cal Poly/Stanford launch personnel.
Deployables
- A time delay, on the order of several minutes, must be present between release from the P-POD and any satellite hardware deployment, to allow for satellite separation. - P-POD rails and walls cannot be used to constrain deployables.
Communication
- There must be a time delay, on the order of several minutes to an hour, before all primary transmitters are activated. Low power beacon transmitters may be activated after deployment. - Operators must provide proof of the appropriate license for frequency use.
Power
- CubeSats with rechargeable batteries must have the capability to receive a transmitter shutdown command, compliant with FCC regulations. - Satellites that require testing and battery charging must provide an external hardware interface to access the power/data port. Developers can use any kind of connector in their CubeSat, but a proper interface must be provided between standard Cal Poly equipment and the satellite. This could include interface boxes, software, a laptop, etc. Contact Cal Poly with the desired design requirements. - A 'remove before flight pin' is required to deactivate the CubeSats during integration outside the P-POD. The pin will be removed once the CubeSats are placed inside the P-POD.

Furthermore there are some test procedures that are obligatory for each CubeSat. Those tests strongly depend on the chosen launch rocket and will be examined in later phases of the project.

4.3 Separation Sequence

Just after the ejection of Compass-1 from the P-POD the kill switches release and cause the procedure depictured in figure 1.4.5.

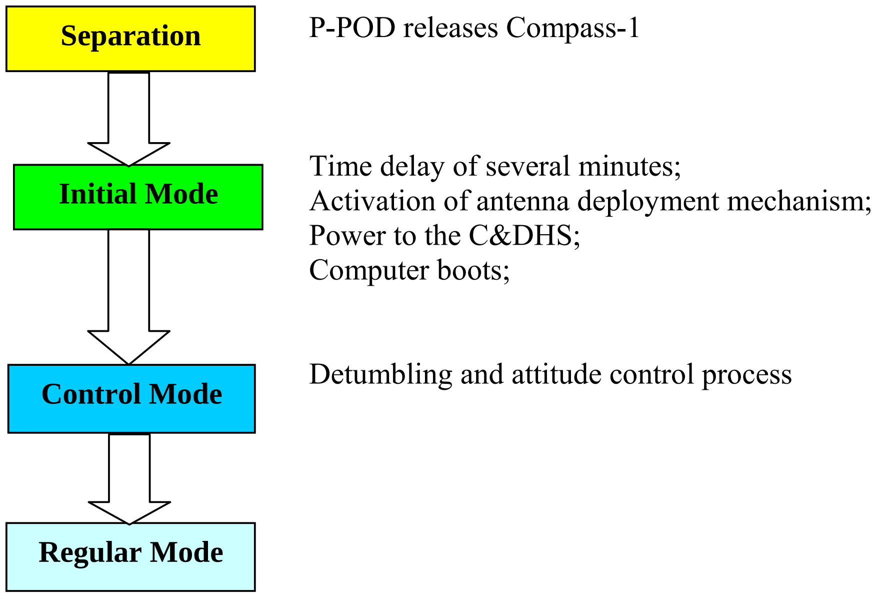


Figure 1.4.5: Initial Sequence of Compass-1

The initial mode is a combination of mechanical and electrical activities that will occur only once in the satellite's lifetime. First there will be a time delay compliant with the CubeSat specifications. During that time the systems are still disconnected from the power bus. Then the antenna deployment mechanism is activated which will set the antennas free from their stored position. Shortly after that the power bus is supplied with current, causing the OBC to boot.

The OBC will initially load its operating system from a ROM and handle over the process to the ADCS subsystem, which in turn has to care for an initial stabilization of the spacecraft. The torques caused by the separation from the P-POD are expected to be very high compared to the natural occurring torques in this orbit, which result in extreme spin rates that have to be compensated.

When this initial stabilization, which is called detumbling is finished, the ADCS hands over the treat to the OBC and the system is operational.

5. Orbit Analysis

We do not have precise information about the launch since we do not have any contract fixed yet. And even when we eventually have one, it is not unlikely that the orbit parameters might change, because the launcher is designated to serve its primary payload. The CubeSats as secondary payloads have to accept the changes and plan in advance to be prepared for such cases. The calculations throughout the entire document are all referring to a reference orbit that we will characterize first. Although we can make very good approximations concerning this orbit, we have to remember the mentioned uncertainty in order not to forget about possible changes concerning the altitude and the inclination.

Table 1.5.1: Reference Orbit Parameters

Reference Orbit	
Type	sun-synchronous
Altitude	600 km
Inclination	98°
Eccentricity	0
LTAN	00:00:00
Period	~97min

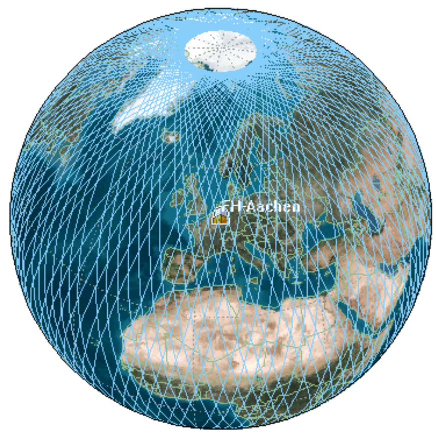


Fig. 1.5.1: Ground Tracks after one Week

5.1 Ground Tracks

We enter the data from the reference orbit in a propagator, a J4 Perturbation model. This propagator accounts for secular variations in the orbit elements due to earth oblateness. However, it does not model atmospheric drag or solar or lunar gravitational forces. J2 and J4 are zonal harmonic coefficients in an infinite series representation of the earth's gravity field. J2 represents the dominant effects of earth oblateness. The even zonal harmonic coefficients of the gravity field are the only coefficients that result in secular changes in satellite orbital elements. The J4 propagator includes the first- and second-order effects of J2 and the first-order effects of J4. The J3 coefficient, which produces long period periodic effects, is not included in either propagator. Figure 1.5.1 illustrates the ground tracks of Compass-1 after one week.

As it can bee seen virtually the whole globe is covered evenly by the satellite's ground tracks. Yet we should take notice of the fact that certain combinations of altitude, eccentricity and inclination can lead to a repeated ground track, which would limit the coverage area significantly.

5.2 Sunlight and Eclipse Times

In space the definition of day and night is completely different to what we are used to on earth. Times of direct sunlight and times of total darkness follow each other in very short periods. Given the orbit we can calculate the duration the satellite will be sunlit and the time it will spend in darkness, because of the earth blocking the sun. The given results are all referring to a time period of 24 hours in order to obtain representative mean values.

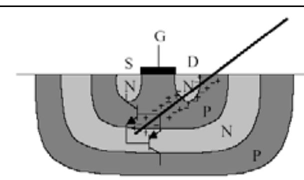
Table 1.5.2: Sunlight and Eclipse Times

Sunlight (during 24h)	
Mean time	61min 30sec
Total time	922min 38sec (15h 22min 38sec) ≈ 64%
Penumbra & Umbra (during 24h)	
Mean Time	32min 13sec
Total Time	517min 22sec (8h 37min 22sec) ≈ 36%

The table shows that the bigger portion of time is spent in sunlight. It is a rule of thumb that a spacecraft in LEO orbit spends about 2/3 in sun and 1/3 in shadow.

5.3 Space Environment

Table 1.5.3: Environmental Conditions

Plasmas and Spacecraft Charging	**Communications, Structure**
Magnetic substorms may charge the surfaces of satellites with high negative voltages.	
Trapped Radiation	**Power, Comm., C&DH, ADCS, Structure**
Electrons and ions (mostly protons) trapped in the Van Allen radiation Belts degrade material and electronic components. The total radiation dose over a period of time consists of three components: proton dose, electron dose and bremsstrahlung X-ray dose produced by the interaction of electrons with the shielding material. Because the centroid of the magnetic dipole pattern of the earth is offset from the earth's center, a portion of the inner radiation belts is closer to the earth than elsewhere. This is the South Atlantic Anomaly (SAA). It is a region of enhanced radiation.	
Galactic Cosmic Rays	**Power, Comm., C&DH, ADCS**
Galactic cosmic rays are particles, which reach the vicinity of the earth from outside the solar system. They pose a serious hazard because a single particle can cause a malfunction in common electronic components, e.g. microprocessor. Those so-called single-event phenomena are:	
Single-event upset (SEU)	It Causes bitflips in RAM circuits. It neither damages the component nor interferes with its subsequent operation. Protection methods for electronics need to be considered, e.g. watchdog timer, redundancy, lockstep, voting and repetition.
Single-event latchup (SEL)	In this case the part hangs up, draws excessive current and does not operate proper until power is turned off and then back on. The excessive current drawn in the latched condition can destroy the device if the power supply cannot handle the current. 

Single-event burnout (SEB)	This causes the device to fail permanently. The failing component may cause the entire subsystem to fail.
Solar Particle Events	**Power, Comm., C&DH, ADCS**
Protons from solar flares degrade materials and electronic components. Likewise galactic cosmic rays cause single-event effects in semiconductor components.	
Upper Atmosphere	**Structure, Power**
Atmospheric drag	The very small atmospheric density causes friction and reduces the velocity of the satellite. The orbit decrease is for high altitudes >600 km very weak and usually leads to a lifetime of more than ten years.
Atomic oxygen (AO)	It reacts with advanced composites and metallized surfaces, resulting in lost or degraded sensor performance.
Temperature	**All Subsystems**
Surfaces	The outer surfaces, e.g. solar cells may experience temperatures from –80°C to +80°C.
Inner parts	The inner parts of the CubeSat, namely the electronic components, have to withstand temperatures ranging from -20°C to +40°C.
Out-gassing	**Mechanical, ADCS**
Out-gassing deposits on cold surfaces, e.g. the camera lens or sun sensors. It can be avoided or minimized by proper selection of materials. Particles deposited on optical apertures can be removed with heat from heating elements or by turning towards the sun.	

6. Conclusion

It was described the mission analysis of the Compass-1 CubeSat project at the University of Applied Sciences Aachen, Germany. Starting with a mission statement and the mission objectives, the system requirements were analyzed. Those are valid at all stages of the spacecraft development process.

The mission modes specify the functionality of the spacecraft in orbit. We defined what the spacecraft has to be capable of and how we access those functions. The correct implementation of the mission modes is a combination of hardware and software, which will be subject to the spacecraft engineering. The subsequent systems have to provide the necessary functions to comply with the mission requirements.

Next the elements that make up a space mission were examined. The (always critical, because high costly) issue of launching was discussed and the envisaged solution was explained. A reference orbit was introduced and its features and consequences for the spacecrafts operation were shown.

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The Payload

Georg Kinzy
Artur Scholz

1. Introduction

From the very beginning of the payload discussion it was clear to implement a camera to our first satellite. Although there will be no deep scientific interest behind it, it accompanies several aspects that underpins this choice. Firstly, a camera supplies data that is easy to understand. Therefore it facilitates the detection of failures or flaws in the data acquisition process or transmission, respectively. Secondly, a camera as payload has a very strong influence on virtually all subsystem requirements and therefore their design. We will have to adjust the CubeSat design, in particular the ADCS to fulfill the payload requirements, which will be a great educational experience.

Contrary to the other subsystems the payload is going to be defined already in detail in this chapter and will be purchased quite soon. This is so because alike in usual system engineering cases we start with an already existing payload and the task of the engineering team will be to build a spacecraft to host it. We want to use a COTS product, which brings the advantage of low cost and a short delivery time.

2. Requirements Analysis

Since we are not experts in the physics of optics we do not intend to construct a camera by our own. The lens system needs high precision assembly procedures, which we cannot supply. Also we do not want to depend on an external manufacturer who could develop a solution for us. As mentioned above, our aim is to select the most suitable module out of the growing market of COTS miniaturized camera modules for our mission. We have chosen to use a CMOS camera chip because of its very low power consumption compared to CCD chips. Both have in common that they are flat area image chips. Shortly explained the CCD- and CMOS- chips consist of a two dimensional array of very small photosensitive detectors also known as sensor elements. The amount of light falling on each of these elements creates a small charge, which is proportional to the energy of the light. The difference between these two sensors types is the way they handle this charge from now on. The CCD-chip transfers each pixel's charge packet sequentially to a common output structure, which converts the charge to a voltage, buffers it and sends it off-chip. In the CMOS-chip the charge-to-voltage conversion takes place in each pixel, which makes it more compact and easier to handle. The output is a digital signal usually made up by 8 bit per pixel. This signal can then be stored in memory or transmitted to ground. The main disadvantage of the CMOS compared to CCD is its poorer resolution. This might improve in future as the technology is moving towards reaching the same capabilities as the CCD sensors. Luckily we do not have a stringent requirement on the spatial solution for the camera, thus we can go for the solution that seems easier to handle, the CMOS chip.

We want to obtain color images from the camera. Since as single pixel would only delivery us with an 8-bit information on its illumination, which is a gray scale ranging from black to white, we would need to implement some methods to filter out the color information. This could be done be using multiple chips for different wavelengths or by adding a filter to each single pixel sensor separately. The second method obviously is far from our capabilities. Fortunately camera modules exists that provide this essential feature already. The single chip option also allows to keep the payload extremely small compared to the use of multiple (at least 4) chips.

The chip shall operate in the visible light band of the electromagnetic spectrum, that reaches from about 0.45 µm wavelength to 0.7 µm. The size of an image shall be VGA standard size, i.e. 640x480 pixels.

The following table shows the requirements that are addressed to the camera payload from system level.

Table 2.2.1: Camera Module Requirements

Mass	< 50g
Power consumption active	< 500mWatt
Power consumption standby	< 10mW
Size	< 40x40x20
Pixel area size	Color VGA (640x480)
Mechanical interface	No moving parts (will be fixed to satellite)

3. Design Analysis

As mentioned above we were looking for a complete solution for the camera that consists of a CMOS sensor chip, complementary processor and a lens system. With the given requirements in table 2.2.1 we begun a market research to find out about products matching our needs. The research was somewhat disappointing as it brought up far fewer products than we had anticipated.

Table 2.2.2: Commercial Camera Modules

Model number	C3188A	LZ0P3916	VS6502V015	MK00-D190
Supplier (Manufacturer)	Quasar	Unitronic (Sharp)	ST Microelectronics	Pictos
Voltage supply	5V ± 5%	2.8V	~2.8V	3.3V
Power active	<120mA	90mW	<20mA	<100mW
Power standby	<10µA		<10µA	
Temperature range [C]	0°, +40°	-20°, +60°	-25°, +70°	
Bus interface	I²C	I²C	I²C	8 bit parallel
Lens system	f=6mm F1.6 FOV=30°	f=3.3mm F2.8 FOV=58°	F2.8 FOV=44.5°	f=3.1mm F2.8 FOV=55°
Costs	~105€			
Remarks	available; same as CubeSat “XI” used	Only OEM	still in development; only OEM supply	only OEM supply

Much more disappointing was the fact that the best products in terms of small mass and minimum size are either still under development or will only by supplied to OEM customers, which are companies that order a huge quantity of those products. Even so we tried to convince them of our research intentions they refused to make an exception for us.

Because of this there were two more options discussed during the meetings. The first was to disassemble used mobile phones with proper camera modules. The other would be to take a module out of a web cam. Both options were put aside, because we chose to use the one available module.

We agreed that the C3188A module looks promising and decided to go for that choice. This CMOS camera is also used by the Japanese CubeSat "XI" and supplied with reliable images, indicating that it is 'space-proof'. However we will have to do some testing with this module to ensure its functionality.

Fig. 2.3.1: Camera Images, © ISSL, University of Tokyo, JAPAN

3.1 Sensor Chip

The C3188A is a 1/3" color camera module with digital output. It uses OmniVision's CMOS image sensor OV7620. Combining CMOS technology together with an easy to use digital interface makes C3188A a low cost solution for higher quality video image application. The digital video port supplies a continuous 8/16 bit-wide image data stream. All camera functions, such as exposure, gamma, gain, white balance, color matrix, windowing, are programmable through I²C interface. The size of the board is 40mm x 28 mm.

Fig. 2.3.2: C3188A Module

Pin Description

1~8	Y0~Y7	Digital output Y Bus.
9	PWDN	Power down mode
10	RST	Reset
11	SDA	I²C Serial data
12	FODD	Odd Field flag
13	SCL	I²C Serial clock input
14	HREF	Horizontal window reference output
15	AGND	Analog Ground
16	VSYN	Vertical Sync output
17	AGND	Analog Ground
18	PCLK	Pixel clock output
19	EXCLK	On chip video oscillator clock output
20	VCC	Power Supply 5VDC
21	AGND	Analog Ground
22	VCC	Power Supply 5VDC
23~30	UV0-UV7	Digital output UV bus.
31	GND	Common ground
32	VTO	Video Analog Output (75Ω monochrome)

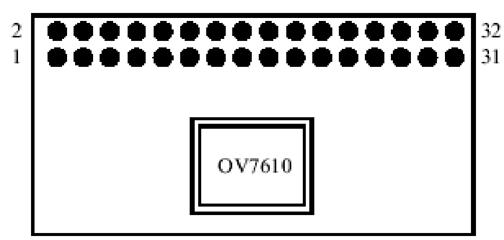

Fig. 2.3.3: PCB Layout of Camera

Features:

- 307,200 pixels, VGA / CIF format
- Small size : 40 x 28 mm
- Lens: f=6mm (Optional)
- 8/16 bit video data : CCIR601, CCIR656, ZV port
- Read out - progressive / interlace
- Data format -YCrCb 4:2:2, GRB 4:2:2, RGB
- I^2C interface
- Electronic exposure / Gain / White balance control
- Image enhancement - brightness, contrast, gamma, saturation, sharpness, window, etc
- Internal / external synchronization scheme
- Frame exposure / line exposure option
- Single 5V operation
- Low power consumption (<300mW)
- Monochrome composite video signal output

Specification

Imager	OV7610, CMOS image sensor
Array Size	644 x 484 pixels
Pixel size	8.4 x 8.4 µm
Scanning	Progressive / interlace
Effective image area	5.4mm x 4mm
Electronic Exposure	500:1
Gamma Correction	0.45/1.0
S/N Ratio	42dB
Min Illumination	20lux @F1.4
Operation Voltage	5 VDC
Operation Current	200mW Active 100 µW Standby
Lens (Optional)	f6mm, F1.6

An important parameter of digital cameras is how convenient the interface is. Essentially, it is a whole image capture system in a single chip. Since the internal AEC has a range of 1:260, and AGC have 24dB, for the most of applications, the camera can adjust itself to meet the lighting condition without user intervention.

3.2 Lens System

There is a strong connection between the choice of sensor chip and choice of lens. The diameter of the lens determines the amount of light getting through to the chosen chip. However, with the chosen sensor it is not critical to get the lens diameter entirely correct because the sensitivity of the chip can be adjusted to suit the right environments. The Japanese team of “XI” recommended us to insert a ND filter because they experienced an three times higher brightness in orbit than on earth, thus they had to set the exposure time to the minimum to obtain good images. Using this additional filter would allow us to still have some variance for the exposure time adjustment.

In addition they told us that the FOV is about 30°, giving an image area of about 350kmx350km for the reference orbit altitude when the pointing is in nadir axis direction. Obviously the covered area increases enormously when the satellite turns away from that axis, as it can be seen in the figures 2.3.1.

The exact dimensions of the lens system are not known, but referring to figure 2.3.4 it is roughly about 20mm in height.

Fig. 2.3.4 EV-Board for Camera Module

The covered area for a 24h period is illustrated below for an FOV of 30°.

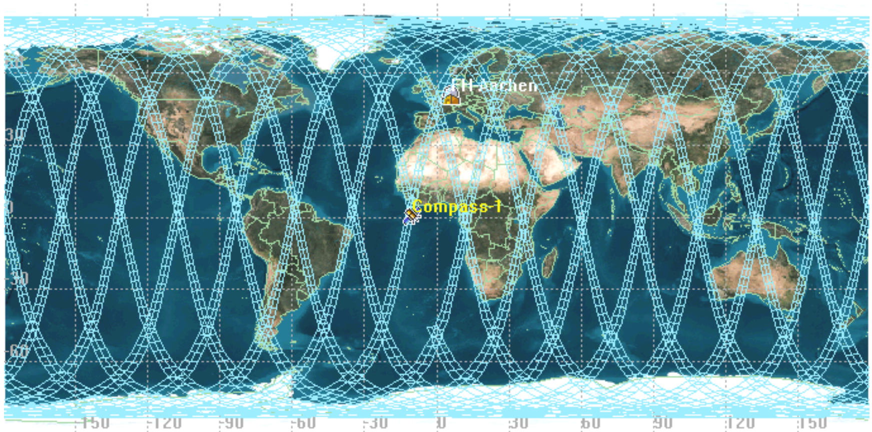

Fig. 2.3.5: 24h Ground Coverage of Camera

Fig. 2.3.6: Footprint of Camera

3.3 Data Format

The data from the camera is an array of bytes, in total 640x480 bytes for an VGA image. To decrease data transfer rate the camera chip provides a solution, that is it can output QVGA resolution image. This mode decreases pixel rate one half. The resolution default value is 320x240 and can be programmable. Every line only outputs one half of the data. For Interlaced Mode, all field line output (320), for Progressive Scan Mode, only one half line data output. The digital video port also offer RGB Raw Data 16 Bit/8 Bit format.

4. Conclusion

The chosen module would fit perfectly in the scope of the Compass-1 mission goals. We have examined its functions and demonstrated its capabilities in the previous sections. There is an evaluation board available for the chosen camera module, shown in figure 2.3.4. It can be purchased separately. This board can help to validate the camera when undertaking environmental tests, e.g. the exposure to vacuum conditions. The board allows to display the camera data on a TV-screen and a computer. Furthermore several parameters can be adjusted with the auxiliary software.

The Spacecraft Bus

1. Introduction

The spacecraft bus is the housing for the payload. It provides the mechanical, electrical and communication interface. For designing the spacecraft we have to take two key aspects into account. The first is to follow the specifications of a CubeSat design. The other is to accommodate the payload and make sure its requirements are met. For that reason we have done a major research in the various options for designing a satellite. To facilitate this extensive work we took a look through the solutions other CubeSat groups had found and compared its usefulness with our suggestions. When needed we modified our design but in some aspects we had to go different ways. This chapter describes the requirements that are exposed to each subsystem resulting from the mission requirements analysis. In addition, our solutions to meet those requirements are given.

2. System Overview

2.1 System Specification

Most of the requirements for a CubeSat are addressed in table 1.4.1 because the P-POD (the deployer of the CubeSats) drives most of the constrains. Additional information is provided by Stanford University which depictures the basic layout for designing the spacecraft.

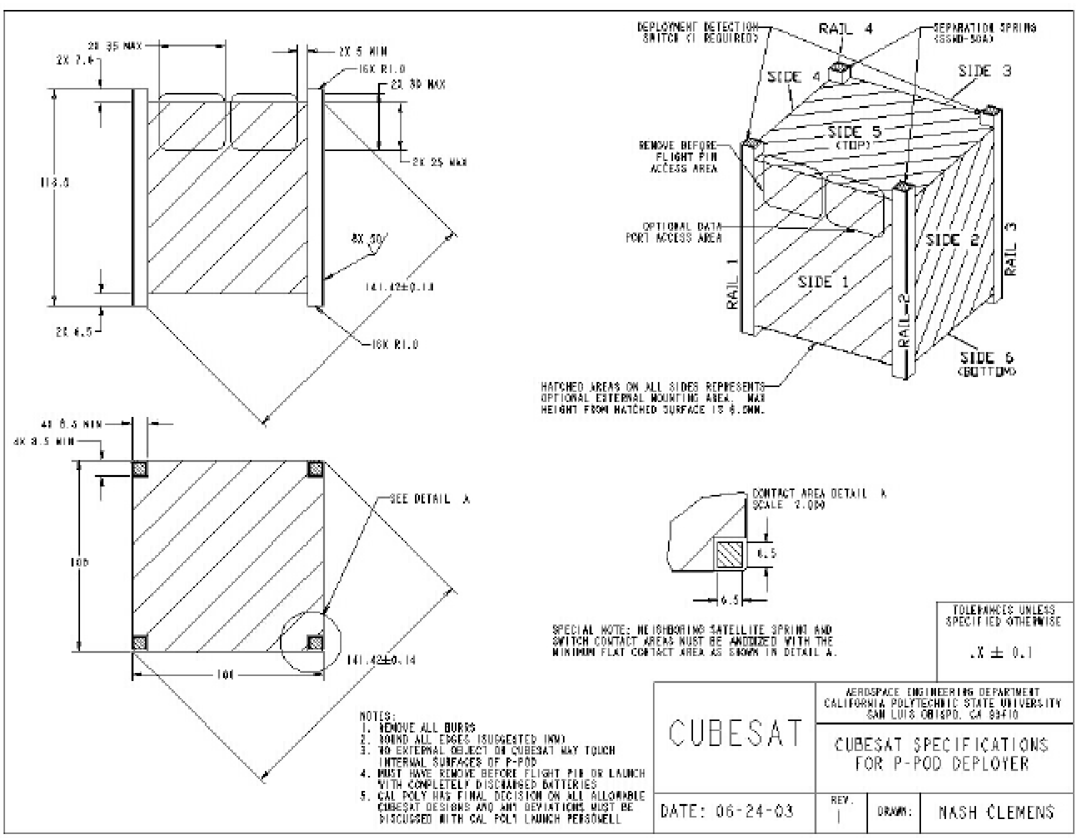

__Figure 3.2.1: CubeSat Drawing__

2.2 Subsystem Budgets

With the overall specification for a CubeSat given by Stanford University and summarized in table 3.2.1 we can draw up the system budgets, necessary to allow an accurate distribution and allocation of the valuable recourses mass and power to the subsystems.

Table 3.2.1: CubeSat Specifications

	Consideration	Remarks
Mass	<1 kg	Including Payload
Size	10x10x10 cm	Including Payload
Power	1 W	Alike other CubeSats
Bus Voltage	5 V / 3.3V	We will work with components that have different voltage level demands.
Mission Duration	6 months	Specific requirements for Compass-1

Table 3.2.2: Mass Budget

	Percent of total	Mass (g)
Payload	5%	50
ADCS	15%	150
Communications	13%	130
C&DH	7%	70
Power	15%	150
Thermal	3%	30
Structure & Mechanism	28%	280
Total allocated	86%	860
Margin	14%	140
Total mass	100%	1000

Table 3.2.3: Power Budget

	Percent of total	Standby (mW)	Peak (mW)	Peak time (%)	Average (mW)
Payload	1%	0,1	500	1	5,1
ADCS	24%	200	1000	5	240
Communications	34%	160	2000	10	344
C&DH	6%	60	60	100	60
Power	0%	0	0	0	0
Thermal	17%	80	1000	10	172
Structure & Mechanism	0%	0	0	0	0
Total allocated	82%				821
Margin	18%				179
Total power	100%				1000

Note that the values for power consumption in standby are given for the module when it is not actively working but is connected to the power supply. In emergency mode or cases where the modules are disconnected from the supply, the power consumption is zero, obviously. The peak consumption is when the subsystem is fully busy and draws the maximum current. The peak times indicate the estimated proportion of total time for those events.

2.3 Subsystem Interfaces

The layout of the power and data transfer among the subsystems is shown in the block diagram in figure 3.2.2. Note that the subsystems are shown in blocks, which do not directly refer to their structural layout. For example, the camera will sit on the backside of the main bus board as it is illustrated in section 8 - Structures and Mechanism. This block diagram shall serve as a help to understand the internal communication and the power distribution. Both issues are addressed in more detail in section 5 and 6, that are C&DHS and Power.

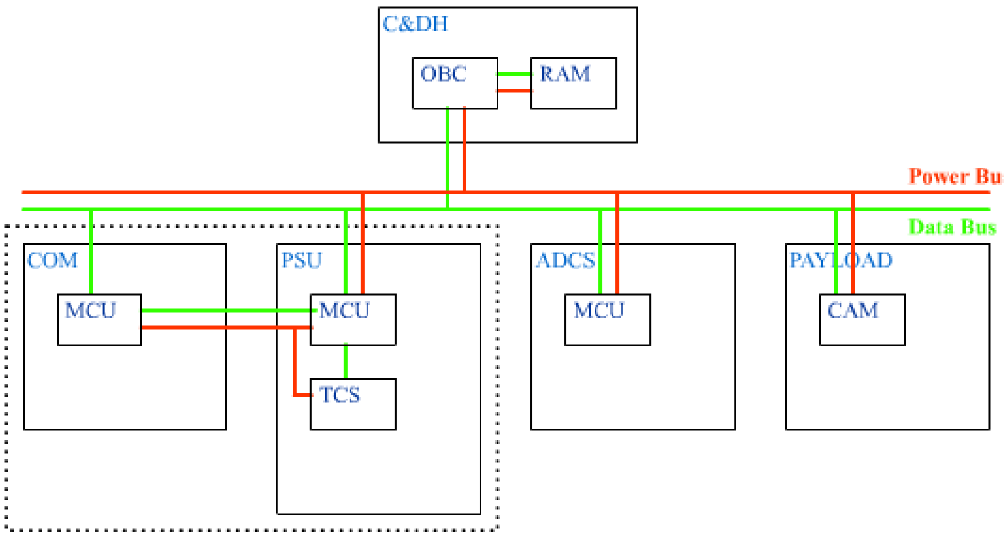

Fig. 3.2.2: System Block Diagram

The micro controller units act as the brain of the respective subsystem and coordinate the data and power transfer to and from its components. A significant feature is that the power system together with the communications system form a minimum operational system for the emergency mode. Even in case the MCU of the power supply unit disconnects all other systems from the power supply bus, the COM is still operational because it has an external line to the PSU. Also there is an external data line, thus enabling the spacecraft to send beacon signals when in emergency mode.

As mentioned the block diagram only reflects the power and data bus layout. Section 8 - Structures deals with the mechanical interfaces.

3. Attitude Determination and Control

Jens Giesselmann

The goal of the Attitude Determination and Control System (ADCS) is to stabilize the spacecraft against all attitude disturbing influences resulting from the environment in the earth orbit in order to point the payload towards a predetermined point on the earth's surface within a specified margin of error. Furthermore the ADCS should be capable of reorienting the spacecraft into a predetermined attitude, thus providing controllability to high extend. This goal must be achieved regarding a stringently limited mass, power and size.

3.1 Requirements Analysis

Table 3.3.1: ADCS Subsystem Requirements from System Level

Functional Requirements
- Stabilize spacecraft against external disturbances
- Realize pointing along nadir with an accuracy of 8°
- Reorient the satellite into alternative predetermined attitude
Constraints
- Power consumption for sensing < 200mW
- Power consumption for actuating < 1W
- Mass below 150g

Another very stringent constraint that is usually not of a major concern in regular satellites is the size and fitting of the subsystem. It is difficult to express specific values for these issues in the early stage of the design since it all depends on the structural integration of the subsystems. This said it should be kept in mind that all required components for the ADCS subsystem should be either extremely small and/or independent of mounting positions within the cube. Thus all systems required to coincide with the center of mass should be examined carefully for options of substitution with a less constrained solution.

In order to be able to point the camera payload to a specific location on the Earth's surface, say a small country, a minimum pointing accuracy of 8° is aimed at. If manageable this accuracy should be maintained at every point of the polar orbit while the satellite is in normal operation mode. This is a somewhat lenient requirement which should be met without incorporating highly complex systems. More uncommon will be the aim for slew angles up to 80° in each direction from nadir. This will be a secondary non-crucial requirement in order to be able to, as an example, take interesting pictures of the terrestrial horizon for general interest. For comparison, a Danish group of students involved in the AAU CubeSat set a minimum pointing accuracy of 8.13° and a maximum slew angle of 56.84° as design goal. These parameters were set to meet the specific requirement of taking photographs of Denmark only.

A logistic requirement of the ADCS will be the capability to run autonomously for the longest part of the mission. Intervention from the ground should be minimized.

3.2 Design Analysis

3.2.1 Disturbance Torques

During the total life time the spacecraft is subjected to variable disturbances resulting from the various properties of the low earth orbit (LEO) environment. Approximated simple models are used for the preliminary analysis. Recall, the targeted circular orbit has an altitude of 600km and an inclination of 98°. The disturbances encountered on this orbit are computed conservatively, i.e. the worst case conditions serve as basis for the design of the ADCS. Parameters taken from the documentation of previous CubeSat missions will serve as parameter estimates for the described computation.

The disturbances acting on the spacecraft emerge from

- The geomagnetic field and its interaction with residual spacecraft dipoles
- The gravity gradient
- The solar pressure
- The aerodynamical interaction with the rarefied gas environment in earth orbit

Magnetic Interaction

At low altitude, the earth magnetic field is approximately that of a magnetic dipole while at high altitude it is strongly distorted by the interactions with the solar wind. The dipole which approximates the near-earth field is both tilted and offset with respect to the earth's rotation axis, so that the geomagnetic poles do not coincide with the geographic poles. Additionally, the field strength is not independent of longitude. This configuration is called an eccentric dipole. An eccentric dipole has axial poles but also dip poles where the field lines are normal to the earth's surface. The 1985 axial northern pole (geomagnetic south) was at 82.05º N, 270.2º E.

The total field strength assuming the earth being an ideal dipole is given by

$$B = MR^{-3}\sqrt{1+3\sin^2\lambda} \tag{1}$$

where

λ is the magnetic latitude [rad]
R is the radial coordinate [km] (R, λ constituting a polar coordinate system)
M is the earth magnetic dipole moment ($M = 7.9\times10^{24}$ nTm^3 or 30400 nTR_e^3)

The spacecraft's motion across the geomagnetic field induces a motional electromagnetic field in the spacecraft which in return interacts with the geomagnetic field, generating a disturbance torque. The magnetic torque imposed on the satellite can be expressed by

$$\vec{T}_m = \vec{D}\times\vec{B} \tag{2}$$

with

$\vec{D}$ residual magnetic dipole vector
$\vec{B}$ local geomagnetic field vector

In the worst possible case the vectors are perpendicular to each other and the cross product turns into a product of scalar values. Furthermore, B becomes a periodic maximum for $\lambda = n\pi/2$ for n=1,2,3…, which is located above the magnetic poles. At these positions the worst case moment can be described as

$$T_m = D \cdot \frac{2M}{R^3} \tag{3}$$

Equation 3 is illustrated in Figure 3.3.1 plot showing the magnetic disturbance torques as a function of orbit altitude for various values of the residual spacecraft dipole. Estimating the residual dipole is difficult. In a system that is not controlled by a magnetic torquer it should be very small. For magnetically actuated satellites like the Korean Hausat-1 or the Danish AAUSat the values with activated control are 0.022 Am² and 0.075 Am² respectively. The following plot shows the magnetic disturbance torques for an uncontrolled satellite with residual dipole D estimates of 0.0001, 0.001 and 0.01 Am². Solutions for the targeted 600 km orbit are highlighted.

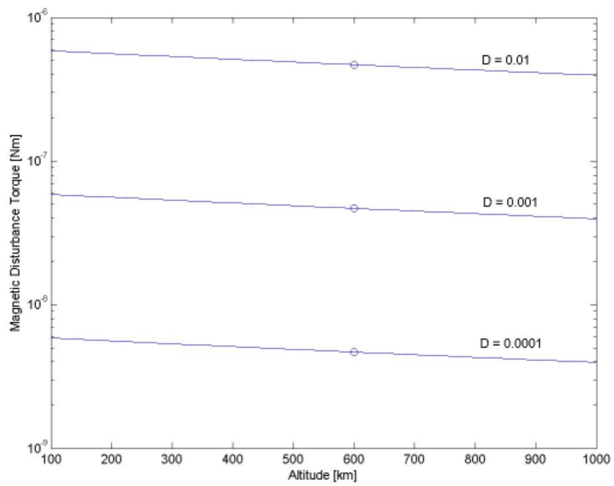

Fig. 3.3.1: Magnetic Interaction Torques

Gravity Gradient

Gravity gradient torques are resulting from the fact that two opposing points of the spacecraft have a finite distance in a declining potential field. Only the center of mass of the satellite experiences a static equilibrium. Beginning with the Newtonian model of gravitation

$$F = -\frac{GMm}{r^2} \tag{4}$$

one can derive a model of the gravity gradient torques acting on the spacecraft

$$T_{GG} = \frac{3\mu}{2R^3} \cdot |I_{zz} - X| \cdot \sin(2\Theta) \left| \begin{matrix} I_{yy} \le Ixx : X = I_{yy} \\ I_{yy} \ge I_{xx} : X = I_{xx} \end{matrix} \right. \tag{5}$$

with

T	is the resulting torque [Nm]
μ	is the gravitational parameter of the earth [m³/s²] ($\mu = 3.896 \cdot 10^{14}$ m³/s²)
R	is the radius coordinate [km]
I	moments of inertia about respective body axis [kgm²]
Θ	is the maximum deviation angle from local vertical [rad]

Note the conditional assignment for the difference of moments of inertia. A typical set of values for the moments of inertia taken from the Korean CubeSat design is

$$I_{xx} = 1.14 \cdot 10^{-4} \, kgm^2$$
$$I_{yy} = 7.64 \cdot 10^{-5} \, kgm^2$$
$$I_{zz} = 6.35 \cdot 10^{-5} \, kgm^2$$

In this example the inertial moment about z is smaller than about y. The gravity gradient torque will behave as depicted in the next plot. Solutions for the targeted 600 km orbit are highlighted. The worst case torque will be experienced at Θ = 45°.

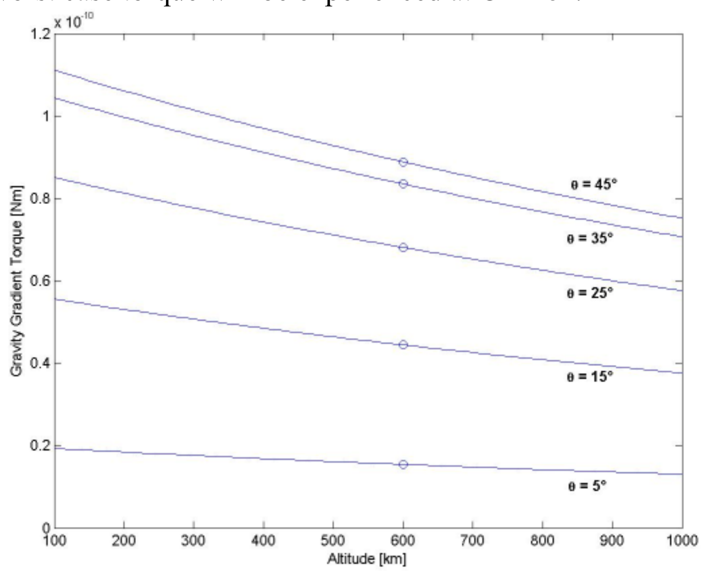

Fig. 3.3.2: Gravity Gradient Torques for Various Deviation Angles

Aerodynamic Interaction

In order to calculate the aerodynamic drag acting on the satellite one must first model the density as a function of altitude. The following equation depicts the atmospheric density in an adequate manner for the preliminary investigation

$$\rho = \rho_0 e^{\frac{h_0 - h}{H}} \qquad (6)$$

where

ρ is the density at a specific altitude [kg/m³]
h is the specific altitude [km]
ρ_0 is the reference density [kg/m³]
h_0 is the reference altitude [km]
H is the scale altitude [km]

Table 3.3.2: Atmospheric Densities [1]

Altitude [km]	Atmospheric scale height [km]	Atmosperic density	
		Mean [kg/m³]	Maximum [kg/m³]
500	64.5	4.76×10^{-13}	2.82×10^{-12}
550	68.7	2.14×10^{-13}	1.53×10^{-12}
600	74.8	9.89×10^{-14}	8.46×10^{-13}
650	84.4	4.73×10^{-14}	4.77×10^{-13}
700	99.3	2.36×10^{-14}	2.73×10^{-13}

For the calculation a scale height of 20km has been selected. This value is inconsistent with the value given in table 3.3.2 but a scale height of 74.8km for a 600km kilometer orbit showed too high density results. For the relatively wide range of investigated altitudes (100km to 1000km) the scale height changes significantly. However, selecting sea level conditions (density: 1.225kg/m³; scale height at sea level: 8km) as reference condition and using a mean scale height of 20km seemed most reasonable for the preliminary model.

The model for the torque due to aerodynamic effects is

$$T_a = \frac{1}{2} \cdot \rho \cdot c_D \cdot A \cdot v_c{}^2 \cdot \left(c_{pa} - c_g\right) \tag{7}$$

where

ρ	is the density [kg/m³]
cd	is the coefficient of drag = 2.2
A	is the projected Area [m²] = 0.0141 m²
v_c	is the orbital velocity [m/s]
c_{pa}-c_g	is the distance between center of pressure and center of gravity [m] = 0.02 m

This model is as simple as can be. In the high altitude of 600 km there is no continuous flow regime anymore. For more accurate calculation of flow in a rarefied gas regime a statistical Monte-Carlo method should be applied. The Newtonian slipstream theory of rarefied gas dynamic predicts a coefficient of drag of 2.0 for a sphere. The cube should experience slightly higher coefficients. However, during the calculation it became quite obvious that the torque is not very much dependent on the drag coefficient. The following plot shows the aerodynamic torque for a coefficient of 2.2. The solution for the targeted orbit altitude of 600 km is highlighted.

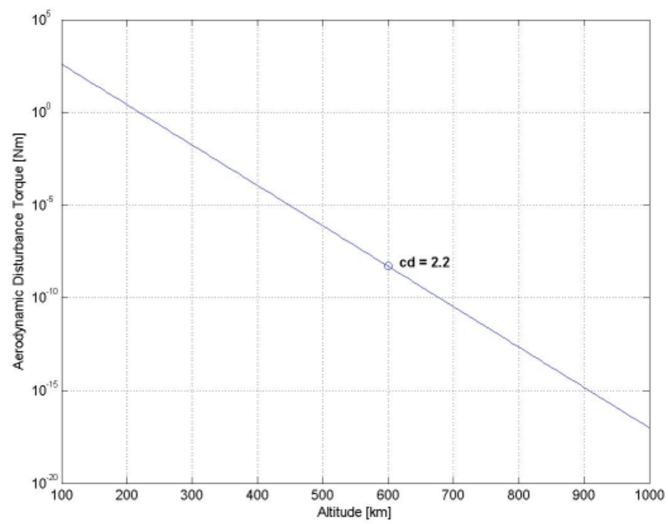

Fig.3.3.3: Aerodynamic Disturbance Torque for coefficient of drag of 2.2

Solar Pressure

The pressure and the torque generated by the radiation emitted by the sun are governed by the solar constant. It is defined as the normal energy flux onto a unit area per unit time, outside of the atmosphere, at one astronomical unit (1 AU = average earth-sun distance). The solar constant has an uncertainty of about ±10 W/m^2. The standard values for the electromagnetic radiation of the sun are as follows:

Solar constant at 1 AU	1371 W/m^2
Max. solar energy flux (European winter)	1428 W/m^2
Min. solar energy flux (European summer)	1316 W/m^2

The resulting torque follows the equation

$$T_{sp} = \frac{S_0}{c} \cdot A \cdot (1+q) \cdot \cos i \cdot (c_{ps} - c_g) \tag{8}$$

with

S_0	solar constant [W/m²] = 1428 W/m² (max)
c	speed of light [m/s] = 3*10^8 m/s
A	projected Area [m²] = 0.0141 m²
q	reflectance factor (0: perfectly absorbing, 1: perfectly reflecting)
i	angle of incidence [rad]
c_{ps}-c_g	is the distance between center of pressure and center of gravity [m] = 0.02 m

The solar pressure disturbance torque is the only one that is not dependent of the orbit altitude. However, it is dependent of the sun incidence angle i. The worst case torque arises at i = 0°. The following plot shows the solar disturbance torques as a function of the sun incidence angle for various reflectance parameters. For the calculation the maximum solar constant was assumed.

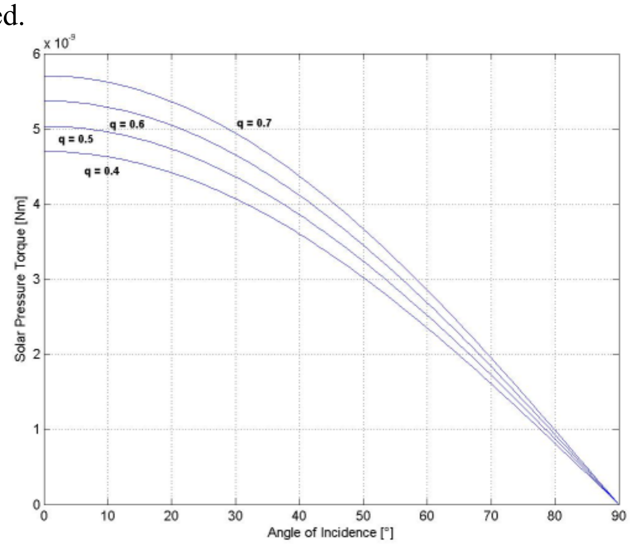

Fig. 3.3.4: Solar Pressure Disturbance Torques for Different Reflectance Parameters

Total Disturbance Torque

The individual disturbance torques have been summed up to calculate the total disturbance torque. The following list summarizes the assumptions made for the calculation:

Solar Constant	S_0 = 1428 W/m²
Residual Spacecraft Dipole	D = 0.001 Am²
Reflectance Parameter	q = 0.6
Projected Area	A = 0.0141 m²
Coefficient of Drag	c_d = 2.2
Deviation angle	Θ = 45°
Sun Incidence Angle	i = 0°

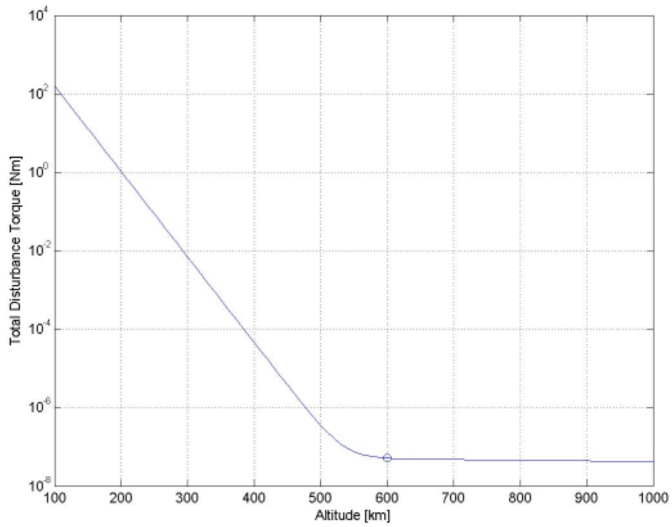

Fig. 3.3.5: Total Disturbance Torque

According to the graph the total disturbance torque for an orbit altitude of 600 km is $5.112*10^{-8}$ Nm. The shape of the graph is mainly resulting from the dominant influence of the aerodynamic drag at lower altitudes. Shortly before 600 km the magnetic torque takes over and slows down the decrease of torque. After that the constant solar pressure torque but mainly the magnetic torque dominate the further shape of the graph for altitudes up to 1000 km. The weak influence of the gravity gradient with in the order of 10 to minus 11 only has a negligible impact on the overall disturbance torque. Since the aerodynamic influence is dominant the analysis of the total torque is very much dependent on the model selection for the density as a function of altitude. As described earlier the applied model cannot be deemed to be highly accurate. Figure 3.3.5 can only serve as a rough estimate of the order of magnitude of the torques that will be imposed onto the spacecraft.
However, judging from the graph the orbit altitude selection seems to be a fortunate one. Up to an altitude of 600 km the torque decreases rapidly. Higher orbits benefit only slightly from the greater distance to the earth.
Again, the shown results depict the worst case situation. This means that the individual torques are simply summed up as it would be the case if all torques would act on the same axis in the same direction. In reality this case is highly unlikely to happen. Torque cancellation is much more probable resulting in a much lower total torque.

3.2.2 Sensors and Actuators

The ADCS can, obviously, be divided in an attitude determination branch and an attitude control branch. The determination branch, which is the sensor part of the system, can again be subdivided in reference sensors and inertial sensors. The control branch, which is the actuator part of the system, can be subdivided in passive and active control mechanisms, although it has to be noted that only active mechanisms truly allow control.

The following section will discuss the advantages and disadvantages of the different determination and actuation mechanisms. It will become clear at this point already that most of the options are not suitable for the stringent budgets of the CubeSat concept. Mostly this is due to the lack of miniaturization of the off-the-shelf components.

Table 3.3.3: Sensors

Reference Sensors
Star Sensor
The Star Sensor is a reference sensor which provides the control algorithm with information about the orientation of the sensor with respect to one or more stars in a known constellation of stars. In other words it provides the control system with one of two orientation vectors necessary to determine the attitude in an unambiguous fashion. So far, available systems would clearly exceed at least one budget (mass) assigned to the attitude subsystem. A group of Canadian students involved in the CubeSat project Can-X developed a star sensor based on CMOS imager technology. However, this system was considered to be a primary payload for technology demonstration and as such had higher budgets available. The Canadian team was not able to establish contact with the satellite since its launch.
Sun Sensor
A sun sensor acquires the spacecrafts orientation with respect to the sun. In order to measure a vector to the sun it is necessary to measure the suns position in two planes. A group of Danish students involved in the CubeSat project DTUSat managed to build a linear two-axis sun sensor in a Micro Opto Electro Mechanical System (MOEMS). This sensor is extremely lightweight but its reliability could not yet be verified since no contact could be established with DTUSat either.
Horizon sensor
The horizon sensor as a reference sensor is similar to the star sensor. But instead of providing a vector to a field of stars it images the earths horizon in an infrared optical band and determines the satellites position with respect to the horizon image. A more simple version of the horizon sensor is an infrared sensor which tells the control algorithm if the earth (or any other infrared source for that matter) is in the field of view of the sensor. A disadvantage of this simple method is the very limited accuracy for attitude determination.
Magnetometer
A magnetometer measures the orientation and strength of the magnetic field of the Earth inside the spacecraft. The controller requires an additional reference vector to unambiguously acquire the satellites attitude. It also requires information about the current position of the satellite due to the variation of the geomagnetic field along one orbit.
Inertial Sensors
Gyros
Gyros are inertial sensors, i.e. they measure angular rates or angular acceleration opposed to angles to a reference. They exist in different forms: - Optical: a laser beam sent into opposite directions into a light conducting ring creates measurable interference as a function of angular rates. - Mechanical: High spin wheels alter their measurable orientation about one axis in a cardianic frame when a torque about another perpendicular axis is applied. - Electronical: Piezo sensors react with a measurable internal voltage when an acceleration is imposed that results from angular velocity. Inertial sensors have in common that they have to be updated by a reference typically from one of the above reference sensors, since they all show a drift in the angle measurement which is the necessary information for the attitude algorithm. This drift is due to the integration of an errous spin rate measurement. Gyros, in which form ever, are typically used for high accuracy attitude determination on larger scientific platforms. While electronical gyros can be manufactured very lightweight and reliable, mechanical gyros are somewhat bulky and heavy and will exceed the mass budget. It should be considered though for the piezo-gyros that accessories are necessary for conditioning of the weak output signal. Piezo-electric gyros can have drifts up 20% resulting in large errors if not updated frequently.

It has been concluded that the desired accuracy will be met by means of reference sensors only.

Table 3.3.4: Actuators

Active Actuators
Thrusters
Offset thrusters actuate the satellites attitude by imposing a torque on the spacecraft. Typically thrusters are used for attitude control on bigger and heavier satellites. The tight mass budget and the CubeSat regulations will not allow to carry any propellant on-board for attitude control. The tight power budget prohibits the use of ion thrusters. Hence, this option will not be feasible for Compass-1.
Momentum Wheel
The mechanism based on momentum wheels is sometimes referred to as dual spin stabilization. Momentum wheels generate torque by acceleration of a spinning mass. This actuator requires momentum dumping when saturation is reached, i.e. when the maximum momentum storage capacity is reached. In this case another control mechanism must be able to compensate the reverse torque generated by the dumping process. So far, momentum wheels have only been manufactured for Mini- and Micro-Satellites. The smallest momentum wheel that has been found during the initial research was Dynacon's MicroWheel200. It has the following specifications[2]: Size: 102 x 94 x 89 mm Mass: 0.77kg (minimum) Power: 3,2W + 1W (for optional rate sensor), 14/28V It becomes obvious at this point that this option is not feasible for Compass-1.
Passive Actuators
Passive Magnet
A permanent magnet installed on a satellite simply aligns the spacecraft with the geomagnetic field vector. Limitations of this system arise from the fact that the magnetic field undergoes dramatic changes during one polar orbit. Continuous nadir-pointing is thus not possible, which makes the passive magnet an uninteresting option for Compass-1 (although one has to admit that this actuator would perfectly fit into the scope of the satellite's name).
Magnetorquers
Magnetorquers is principally a set of controllable electromagnets, which are capable of aligning an arbitrary vector inside the satellite with the geomagnetic field. A disadvantage is the requirement of implementing a model of the earth's magnetic field on the on-board computer. Additionally, this system requires information about the current position. However, this requirement appears to be manageable.
Gradient Boom
A gravity gradient boom aligns the satellite with the earth's center of mass ideally. The stabilizing mechanism is initiated by a single in-orbit manipulation of the moments of inertia of the spacecraft. A boom, which constitutes a mass, has to be deployed after orbit insertion. After this deployment sequence, if the tensor of inertia is looked after carefully, the spacecraft will 'fall' into a stable position with the boom pointing towards the earth. However, the generated torques are extremely small. A small damping of the satellites motion might make a detumbling process based on this technique very time consuming. Also, a deployment of a gravity gradient boom always constitutes a single point of failure (SPOF) and is thus often avoided in the mission design. But since it is considered to establish radio communication via deployable dipole antennas constituting a gravity gradient boom it might be possible to support the ADCS as long as the antennas are remotely nadir-pointing. However, the effect of the gravity gradient on the satellite's attitude must be taken into consideration when developing the simulation tools.

Spin Stabilization
Spin stabilization is the passive form of a momentum wheel. The complete satellite is spun about the principal axis of inertia with the highest momentum of inertia. However efficient this mechanism may be at spin rates of about 20 up to 50 RPM, since the primary payload is a camera the maximum spin rate will be limited to no more than 3 RPM to prevent image blurring. At these low spin rates no significant stabilization will not be achieved. Hence, this option will not be feasible for Compass-1.

There are more passive attitude control systems available than the listed options. These additional systems include nutation dampers and hysteresis rods. However, it has been concluded that the design goal of high controllability can not be met by incorporating passive systems.

After ruling out all system options not suited for Compass-1 for reasons discussed above only one option for the complete ADCS persists. Adequate and unfeasible options are also listed below:

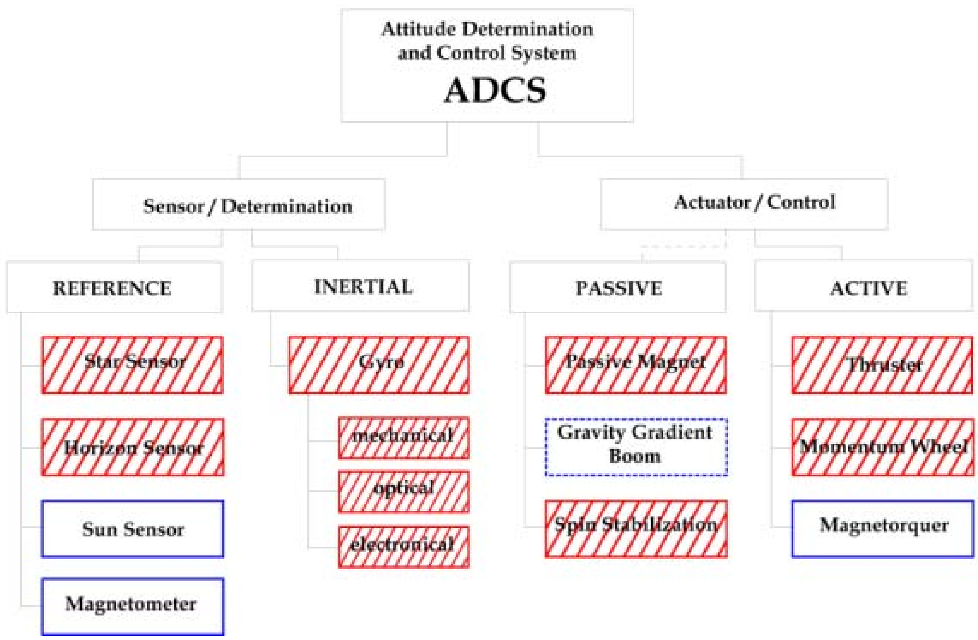

Fig. 3.3.6: Feasible ADCS System Options

The ADCS will consist of three systems being supported by a fourth. The attitude acquisition will be performed by a combination of sun sensors and magnetometer. The control of the attitude will be done using a magnetorquer which will be supported by the gravity gradient generated by radio-wave dipole antennas pointed towards the earth. In order to comply with the design goal of high autonomy it will be necessary to feed the controller with real time data about the current position with respect to the earth. This information is crucial for the control algorithm to calculate the satellite's attitude based on the information gained by the magnetometer. The acquisition of the current position will be done by an on-board GPS receiver. A different solution involving the NORAD service has been denied. A periodical update of position data and on-board trajectory projection would not only significantly increase on-board computation requirements but also the complexity of operations. By using a simple lightweight GPS receiver the satellite will autonomously be informed about position and attitude without any further intervention from the ground station. A GPS antenna will be mounted into one of the side faces of the cube. A possible hardware choice for the antenna could be the GPS Dielectic Patch Antenna DAX1575MS63T manufactured by Toko America. This consideration will be evaluated in later design studies. However, the antenna should not exceed the dimensions 20 by 20 mm and should have a low geometric profile.

The sun sensors should be built in linear two axis lightweight MOEMS technology demonstrated by DTU (Danish Technical University). It is envisioned to establish a collaboration with DTU to reduce development time and cost. Hardware testing can be done at the facilities of FH Aachen. For reasons of redundancy and continuous attitude acquisition 5 sensor chips should be applied to all cube faces except the payload face. This will be done in order to cancel occultation of the sun by the satellite's body. The flat sensor itself has a dimension of 7mm x 8mm and a weight of 116mg each. It will be mounted onto a PCB, adding some more weight. The magnetometer should be 4-axes, 3 for nominal operation and 1 backup axis for redundancy. Magnetometers are very lightweight microchips based on the 'Hall-Effect'. The magnetorquer will consume the major part of the mass budget (as well as power budget). It should be three simple rectangular coils manufactured from inexpensive materials like copper or aluminum.

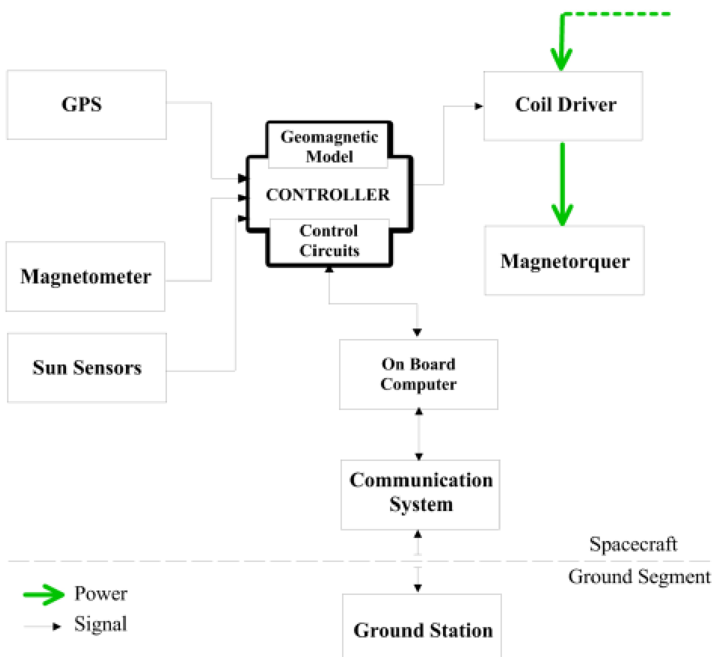

Fig. 3.3.7: ADCS Layout

3.2.3 Operational Modes

The ADCS will perform differently at different stages of the satellite's life time. The operational modes will be

- Initial attitude determination
- Detumbling
- Normal
- Slew
- Safe

Table 3.3.5: ADCS Operational Modes

Initial Attitude Determination
Upon orbit insertion and release from the P-POD platform it is possible that the satellite will be tumbling about one or more axis. Tumbling rates specified by the CubeSat documentation predict maximum tumbling rates of up to 0.1rad/s on all three axes. Additionally, during impulsively deploying the inflatable dipole antenna a torque will be acting on the spacecraft which is difficult to predict. However, before starting normal operations the tumbling rates must be compensated. As a first step the initial attitude determination must acquire the actual tumbling rates in a sequence of several determination loops and provide this information to the controller. Information about the dynamic state of the satellite after separation and antenna deployment should be downlinked to the ground station for mission analysis purpose as soon as a communication link can be established.
Detumbling
After the initial attitude determination the controller calculates the necessary means to counteract the tumbling motion. The on-board determined procedure will be conducted in an open loop fashion, i.e. the controller will make the actuator counteract the motion while not receiving any more attitude information other than the initial conditions what so ever until the precomputed procedure is completed. In this mode power can be rerouted from the sensors to the actuator. It is probable that the satellite will experience the detumbling mode only once, right at the beginning of its life.
Normal
After detumbling the controller is going to be switched into the normal mode. Here the complete ADCS will try to keep the satellite fixed onto one specific pointing (usually nadir pointing unless otherwise specified by the operator). In this mode only a limited power supply of about 200mW will be accessible for the ADCS. The sensors and the actuators will share this power. Since the magnetometer cannot operate while the magnetorquers are active the current to the magnetorquers will be cut off in regular intervals. The total time for reduction of the induced collapsing magnetic field and magnetometer sampling will not exceed 1msec enabling a quasi-continuous closed control loop. The generated torques will be small compared to the detumbling torques since the motion/attitude of the spacecraft will not be altered significantly. The ADCS will only actively compensate disturbance torques.
Slew
The slew mode has to be activated by a command that is uplinked to the satellite during a communication window. The slew command will include a vector and a time information. After confirming the command the controller will set its aiming value to the new vector specified by the operator. Consequently the satellite will actively reorient itself to the new attitude and keep it until told otherwise. The slew procedure is comparable with the normal procedure, possibly with higher power demands for a limited time duration. Keeping the spacecraft's new attitude will be achieved with the modified normal mode, updated with the current control variable. The time information in the command function will allow initiating a slew maneuver at a predetermined time even when the satellite is not accessible from the ground station.
Safe
In the safe mode the actuator is deactivated and possibly the controller set into a low power consumption mode. The sensors will continue to acquire the attitude in a discrete manner. Time intervals might be fixed-step 5sec or variable step dependent on the rate of change of attitude. As soon as a predetermined deviation from the nominal attitude is achieved the system will automatically switch into the normal mode to restore the nominal attitude. Switching into the normal mode from safe mode can only be done if the OBC flags permission. Permission may happen to be delayed when a high power system like the communication prohibits the activation of an additional power intensive subsystem.

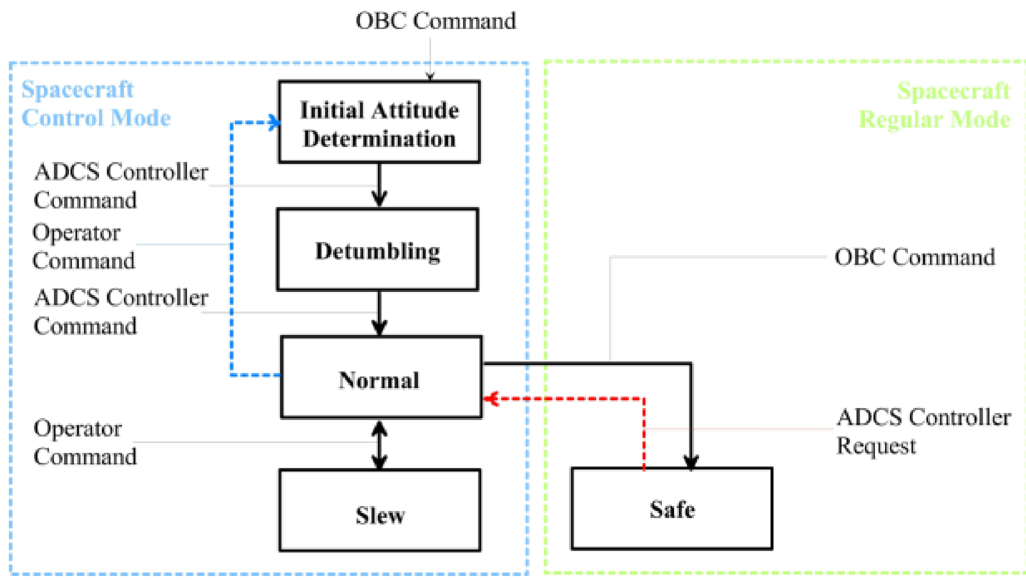

Fig. 3.3.8: ADCS Operational Modes

3.3 Conclusion

Rational decisions on the coil material have to be made. Weight considerations make aluminum the more favorable candidate as coil lead material. But limitations are set to the number of turns due to the lack of thin aluminum wire available. It has to be investigated whether suppliers of required wire are existent.
An option to incorporate 2 x 3 opposing coils = 6 coils making up 3 pairs of Helmholtz coils instead of 3 single coils with equal total mass will be investigated in terms of better homogeneity of the magnetic field. Testing is envisioned to accomplish a more effective magnetic field.
A simulation software has to be developed, preferably using the programming environment MatLab and Simulink, to simulate the dynamics of the spacecraft, its reaction to external influences during the different operational modes and to verify if the stated requirements have been met. The software should incorporate realistic models of the external influences. The simulation will help to determine the necessary properties of the controller and the magnetorquers. The controller will be selected and the magnetorquer designed according to the results of the simulation. The software will furthermore support the power subsystem in determining the total power generation during different sections of the spacecrafts operation.

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4. Communications

Artur Scholz
Oscar Moreno

The Communication subsystem is the gateway for the user/operator to the functionality of the satellite. A communication with the spacecraft takes place over two types of links. The *uplink* carries commands from a ground station to the spacecraft. These commands are redirected to the OBC and then either processed immediately or stored and executed at a specified time. The *downlink* carries data, which consists of two different types of information. One are the data generated by the payload and those are in our case images. The other data are information about the spacecraft's vital characteristics, so-called housekeeping data. This is the total of all information for a specific time gathered by various sensors, e.g. temperature, voltage and current.

This subsystem is an inevitably complex area and its design has to be done carefully to avoid failures, which would lead to a void mission. Whenever possible we will make use of conventional components that are in use by other small satellites and have proven their reliability. We also envision a strong collaboration with the radio amateur community to profit from their profound knowledge.

4.1 Requirements Analysis

The requirements for the Communication subsystems derived from the system level are described in the table below.

Table 3.4.1: Communications System Requirements from System Level

Functional Requirements
- Receive commands from ground - Transfer the command to OBC - Transmit data to ground - Modulate and demodulate the basis signal on/from a carrier frequency - Prepare data packets in protocol format for the RF link - Send beacon in emergency mode
Constraints
- Power consumption for transmission < 2W - Power consumption in standby < 160mW - Mass below 130g

As for the other subsystems the COM subsystem shall be as autonomous as can be. The overall intention is, that the other systems do not directly have contact with the components of the communications subsystem but are served by a processor on the PCB of the COM. It handles the whole process of transmitting and receiving signals. Data, which has to be sent is directed to the COM processor via the C&DH main bus. The same way it works for reception of commands. The uplinked commands are made available for the system via the bus.

4.2 Design Analysis

An overview of the communications subsystem is depictured in figure 3.4.1. As noted in the subsystems interface section at the beginning of this chapter, the power subsystem and the communications subsystem will make up a minimal operational configuration, permitting the transmission of beacons in the emergency mode.

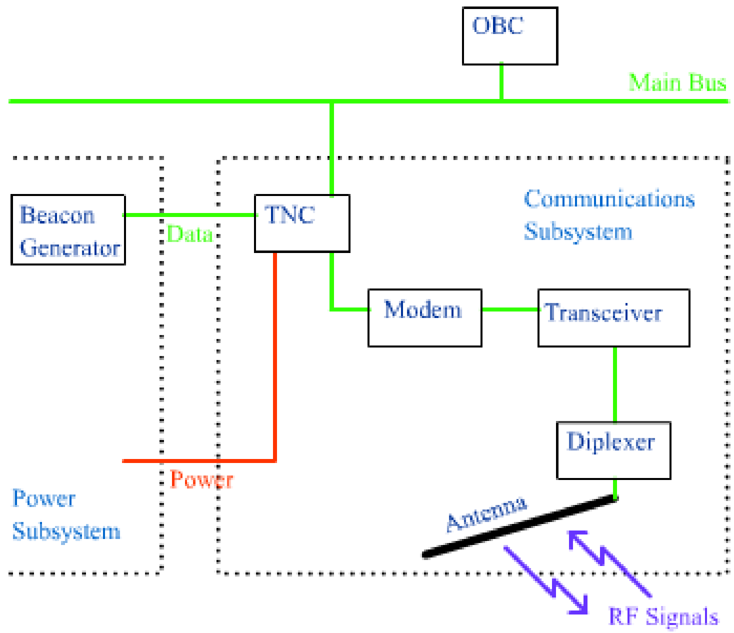

Fig. 3.4.1: Overview of Communications Subsystem

The data the TNC receives from the bus and from the Power subsystem is of digital nature. The same is true for the commands that the TNC redirects from the modem onto the bus.

The next paragraphs examine the specified main components of the COM system in enough detail to get an understanding of their scope of work.

4.2.1 TNC

A TNC (Terminal Node Control) is a microprocessor, previously also introduced as micro control unit (MCU), which has the following functions for downlink:

- read the data to be transmitted and store them if necessary;
- pack the data into protocol format;
- send data to modem.

The other way around it works in the following manner:

- receive commands from modem;
- unpack commands;
- redirect commands to OBC.

4.2.2 Modem

Modulation is a process by which a signal (in our case digital information) is used to vary a characteristic of the carrier wave. Therefore a subcarrier is modulated to represent the information about the binary ones and zeros. Then the carrier wave is modulated by the subcarrier. The carrier wave has a much higher frequency than the subcarrier frequency, permitting a practical antenna length.

4.2.3 Transceiver

The transceiver is a combination of a transmitter and a receiver as the name indicates. The transmitter part is needed because the signal from the carrier phase modulator is too weak for transmission. Therefore power amplification is required. Basically there are to types of amplifier: solid-state and traveling wave tube. Although the output of the latter ones is higher, we will use solid-state amplifiers, because they are small and light. They are called HPA (high power amplifier). The output from the power amplifier is then filtered and passed to the antenna for transmission.

With the receiver it works vice-versa. The signal from the antenna is amplified and filtered to provide a stronger input signal to the demodulator process. The abbreviation for this component is LNA (low noise amplifier).

4.2.4 Diplexer

Because we want to use one only antenna for both uplink and downlink we will need to put a diplexer before it. This is a switch to connect the antenna either with the transmitter or with the receiver.

4.2.5 Antenna

Antennas are essential components in a satellite. They are the interface between free-space and electronic devices. Their purpose is to provide a transition from a guided wave on a transmission line to a free-space wave and vice versa in the receiving case. If the antennas fail to work the satellite can be considered dead.

To establish a communication link between satellite and ground station even in conditions where the position and attitude of the spacecraft is not exactly known we will need an antenna that has a broad beamwidth. Furthermore we cannot provide any pointing mechanism on board the satellite for the antenna. Thus parabolic dishes and other narrow beam devices are not applicable. Antennas that theoretically radiate equally in all directions are called omni directional. The disadvantage here is that those antennas require much more power than directional antennas. A compromise between large beamwidth and available power is a dipole antenna or a monopole antenna. The radiation pattern of a dipole antenna is demonstrated in figure 3.4.3.

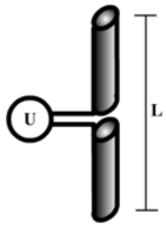

Fig. 3.4.2: Dipole Antenna

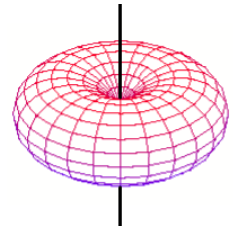

Fig. 3.4.3: Radiation Pattern of Dipole Antenna

The dipole antenna has its best RF link when the transmitting and the receiving antennas are parallel to each other. In any other cases the performance decreases. In addition, along the axis of the dipole there is no signal radiation. In this case the received signal is zero.

A constructive method for dipole antennas to avoid directions of zero electromagnetic radiation is the application of turnstile antennas. Turnstile antennas are nothing else but crossed dipoles.

A monopole antenna however works similar to the dipole antenna. It is a dipole that is divided in half at the center feed point and fed against the ground plain. Thus a monopole antenna obviously has only half of the length of a proportionally dipole, which is important for a pico satellite. The current and charges are the same as the upper half of the dipole counterpart, but the terminal voltage is only half that of the dipole. The radiation pattern of a monopole above a perfect ground plane is the same as that of the dipole similarly positioned in free space. The antennas need to be insulated against the structure.

For the Compass-1 satellite it seems to be more useful however to modify the crossed dipole or monopole antenna assembly in the way illustrated in figure 3.4.4. The reason for this configuration is that we take for granted the spacecraft's orientation along the nadir axis in regular mode due to the payload requirements. With the proposed layout the directivity of the antenna is increased and therefore the performance of the link budget will be better, compared to an single dipole or monopole antenna as used by many other CubeSats.

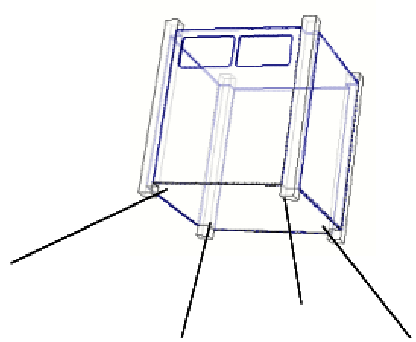

Fig. 3.4.4: Proposed Antenna Layout

As said, the four monopole antennas need to be insulated from the main structure. The length of the monopole antennas is a quarter of the wave length, in our case about 175mm for the 435Mhz band. They are fed in quadrature, i.e. with a phase shift of 90° each to provide circular polarization. The material of the antennas needs to be determined by the structural group. In general copper would be perfect, it is the most used material for antennas. But we have to take into account the elasticity. In fact, this might be much more important than to use the best conductive material. Since the antennas will be stored packed, they have to have good spring like capabilities. Steal might serve the purpose better.

4.3 Conclusion

For most of the communications subsystem there exist integrated products, which can be disassembled and reconfigured for our purpose. This seems to be the best solution because the products are reliable and easy to purchase. The selected products have to be tested intensively to find out the best configuration.

As the communications subsystem, the communications architecture and the ground station are interdependently connected to each other, changes in one will have an impact on the design of the others.

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5. Command and Data Handling

Georg Kinzy
Artur Scholz

The command and data handling subsystem provides the main bus for the data exchange between all other subsystems. All data exchanged through the bus is in binary format made up by zeros and ones. If any component on a subsystem generates an analog signal, it needs to be converted to a digital one before being supplied to the bus.

The system manages three digital data streams, each critical to the spacecraft and each with distinctive characteristics. Those are:

- data from the payload
- housekeeping data
- commands

Moreover the C&DHS has to be capable to store those data streams, because of two reasons. The first is that data can only be transmitted from time to time, i.e. when there is a communication link with a ground station. Furthermore some commands from ground have to be processed at later times, making it necessary to store them as well.

The controller of the main data flow is the OBC. It interacts with all the subsystem controllers and has to secure a stable operational status of the spacecraft.

5.1 Requirements Analysis

Combining the above-mentioned general tasks of a C&DH subsystem together with the specific requirements for the Compass-1 satellite we get the requirements listed in table 3.5.1

Table 3.5.1: C&DH Subsystem Requirements from System Level

Functional Requirements
- Assure and control the data exchange among subsystems - Provide storage for data and commands - Create an operating system that controls the spacecraft
Constraints
- Power consumption <60mW - Mass below 70g

5.2 Design Analysis

The development of the C&DHS is made up of two major subjects, which are the hardware part and the software programming. Both are addressed in the next sections.

5.2.1 Hardware

Hardware is everything physical, e.g. the PCB for the main bus and the electrical ICs. This section deals with the main components, which are necessary for the command and data handling subsystem.

Main Bus

The main bus assures the dataflow within the spacecraft. Every subsystem is connected to the bus.

We decided to use the I^{2}C bus because of its suitable data rate and high flexibility. The controllers of all subsystems have an inbuilt I^{2}C bus hardware module, making it easier to write the software. The temperature sensors are also connected through this bus, so every subsystem can independently read the temperature value if needed.
The I^{2}C bus was developed for connecting different integrated currents to save space and costs for the PCB. This two-wire bus has a data and a clock wire. Every component has its own address like a phone number and is connected to this data lines.

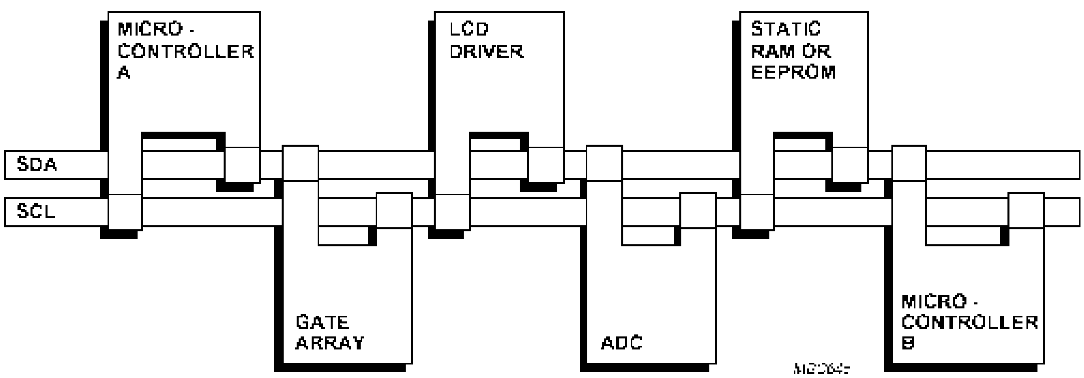

Fig. 3.5.1: I^{2}C Bus Concept

OBC

It controls the spacecrafts operations in normal mode. The OBC is the main data handling system. It handles the exchange of data between payload, memory and communication. Every command received from ground will be decoded by the OBC and stored in the memory. Because of its relatively high energy consumption, the OBC will be shut down in case of entering the power saving emergency mode.

To simplify the programming and improve computing power for future projects, we decided to use a 32-Bit controller. Even more complex or precise calculations could be implemented without the need for much code and long calculation times.

The major disadvantage of a larger processor is the higher energy consumption, but there are enough of them available on the market, which satisfy our needs. Our preliminary candidate is the Atmel AT91M40800, with only 4mW/MHz, giving us enough I/O´s and hardware modules like I^{2}C bus.

To prevent the OBC to hang up, an external WDT, watchdog timer, will also be mounted on the PCB. It forces the MCU to restart and reboot, so it can go on doing its work.

Memory

The memory on the OBC PCB has to provide space for data from the payload, the housekeeping data and the commands received trough the communication subsystem.

Most of the capacity will be needed for the pictures from the payload, every picture has an average size of about 300kbytes. With 4Mb memory it is possible to store 10 pictures and the needed data for the satellite's operation.

We decided to use a DataFlash AT45DB321 from Atmel with 32 megabit. This uses a serial interface to connect the flash with the OBC, referring to Atmel documentation this not only reduces PCB space, it even reduces EMI compared to parallel flash.

5.2.2 Software

The software does not only include the operational system of the C&DH system but is part of all other systems as well. The language used for programming will be C/C++ whereas some routines, which need to be optimized for speed, might be written in Assembler code. The table 3.5.2 list the functions the software needs to carry out while in operation.

Table 3.5.2: Software Functions[1]

System	Software Function
C&DHS	Command verification Command distribution Data collection Data formatting Data encoding
ADCS	Sensor data processing Data filtering Attitude determination
Power	Battery charge control Load control
Fault protection	Redundancy management Anomaly response
Operating system	Executive task Device drivers Run-time kernel

5.3 Conclusion

The C&DH system provides the capability for the subsystems to exchange data and this in a controlled way. The software part needs to be implemented to each system. A modular concept shall be applied in order to be able to test the systems independently and to allow a transfer to other projects.

References

[1] Brown, C.D. (2002) *Elements of Spacecraft Design*. AIAA Publishers, Virginia, USA

6. Power

Georg Kinzy
Artur Scholz

The spacecraft can only perform as long as it has power. As we could see with other CubeSats, a failure in the power supply system results in decreased functionality and omitted mission goals. Next to the C&DHS and communications subsystem the power subsystem substantially determines the spacecrafts operational status.

6.1 Requirements Analysis

The requirements for the power subsystems derived from the system level are given below.

Table 3.6.1: Power Subsystem Requirements from System Level

Functional Requirements
- Supply a continuous source of electrical power during mission life - Control and distribute electrical power - Measure and control vital status of elements
Constraints
- Mass below 150g

6.2 Design Analysis

An electrical power system provides, stores, distributes and controls the spacecrafts electrical power as illustrated in figure 3.6.1.

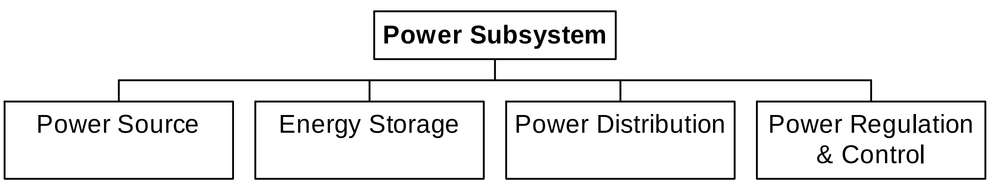

Fig. 3.6.1: Functional Breakdown of Power Subsystem

We will now analyze those four main functions.

6.2.1 Power Source

Here there is no need for a trade-off between the common types of power sources, which are photovoltaic, static (nuclear) and dynamic, respectively. Due to the constraints of a CubeSat the only realistic option is the use of solar cells. So we need to select the most appropriative type of cells for our spacecraft. Beforehand, we should recall an important fact. We intend to cover nearly the whole area of the spacecraft's outer surfaces with cells, except the bottom area where the payload is placed. Also there are no deployable areas for additional cells. Hence the efficiency of the solar cells will drive the available power.

The satellite's power budget is 1W in total. This means the solar cells have to provide a value greater than that to cover eclipse time and losses. The extra energy will be used to recharge the batteries, which will be the only power source during eclipse periods.

To estimate the required power during daylight period for one orbit we use a very simplified equation for sizing the solar array, neglecting all losses:

$$P_{sa} = \frac{(P_E T_E + P_D T_D)}{T_D}$$

where

Factor	Description	Consideration
P_E	is the required power during eclipse;	1 W
T_E	is the time of eclipse;	32 min
P_D	is the required power during daylight;	1 W
T_D	is the time of daylight;	61 min
P_{sa}	is the required power from the solar array;	**1.52 W**

For a comparison of photovoltaic cells we determine the worst case, that is only one surface of the cube is sunlit. The available power is given by:

$$P_{sa_{\min}} = \eta \cdot A_{\min} \cdot G_S$$

where

Factor	Description	Consideration
η	is the solar cell efficiency;	
A_{min}	is the minimum area facing the sun;	0.01 m²
G_s	is the solar constant;	1371 W/m²

The next table gives the results from this term for common solar cells.

Table 3.6.2: Comparison of Solar Cells

Type	Efficiency	Result
Silicon	η = 14%	1.92 W
Gallium Arsenid	η = 18%	2.47 W
Indium Phosphid	η = 19%	2.60 W

As we can see the Gallium Arsenid cells together with the Indium Phosphid cells meet and even exceed the required power budget by far. Additionally it should be noted that the Gallium Arsenid cells are better resistant to radiation than the silicon cells and provide greater EOL (end-of-life) power for a given area. Moreover they are more matured than the Indium Phosphid cells.

Methods that could further increase the generated electrical power are known as Multi Band Gap and Multi Junction Cells. The idea is to place two or more layers of cells on top of each other.

6.2.2 Energy Storage

This section deals with the selection and layout of the secondary power system. This system has to store the required power for eclipse times and needs to be functionally over the entire mission lifetime.

The batteries determine the bus voltage. Batteries can be connected in series to increase voltage or in parallel to increase current output. LEO spacecraft will encounter one eclipse period each orbit (about 15 per day) of approximately 32min. Therefore the batteries must charge and discharge about 2700 times for half a year, while the average depth-of-discharge (DOD) is only 15-25%. Depth-of-discharge is simply the percentage of total battery capacity removed during an eclipse.

Before we determine the needed capacity of the battery we will have a look on the various types of battery cells available and used for space-application. In brackets the nominal voltage per cell is given.

Table 3.6.3: Batteries for Space-application

Type	Description	Remarks
NiCd (1.2V)	The most matured battery cell for space-use. Must be charged with a constant current. Nominal Voltage is 1.2V.	Heavy but reliable, memory effect
NiMH (1.25V)	Comparable to NiCd, but higher energy density	Heavy, big size
NiH_2 (1.2V)	Higher DOD compared to NiCd.	Heavy and large, not suitable
Li-Ion (3.6V)	Has a dry electrolyte.	Light, various shapes possible
Li-Ion Polymer (3.4V)	Gelled electrolyte is added to decrease the internal resistance. Packed in foil package, which leads to swelling. Careful charge and discharge must be applied otherwise it can lead to destruction of the cell.	Light, flexible, good temperature range

We have chosen to use Li-Ion type batteries as they represent the best alternative for a CubeSat satellite. Next we will estimate the number and connection of the batteries.

The total power capacity required during eclipse time is:

$$C_r = P_E T_E = 1W \cdot 0.5hrs = 0.5Whrs \quad \text{or} \quad C_r = 0.5Whrs/5V = 100mAhrs$$

The total capacity of the battery pack yields from this equation:

$$C_{total} = C_{pack}(DOD)N$$

where

Factor	Description
C_{pack}	Is the capacity of one battery pack;
DOD	is the depth-of-discharge;
N	is the number of batteries;

Again we have not considered losses in the power lines. It can be seen very easily that the number of battery packs required is inverse proportional to the DOD and decreases linearly with the capacity of a single pack.

6.2.3 Power Distribution, Regulation & Control

An overview of the basic layout of the power subsystem is illustrated in figure 3.6.2.

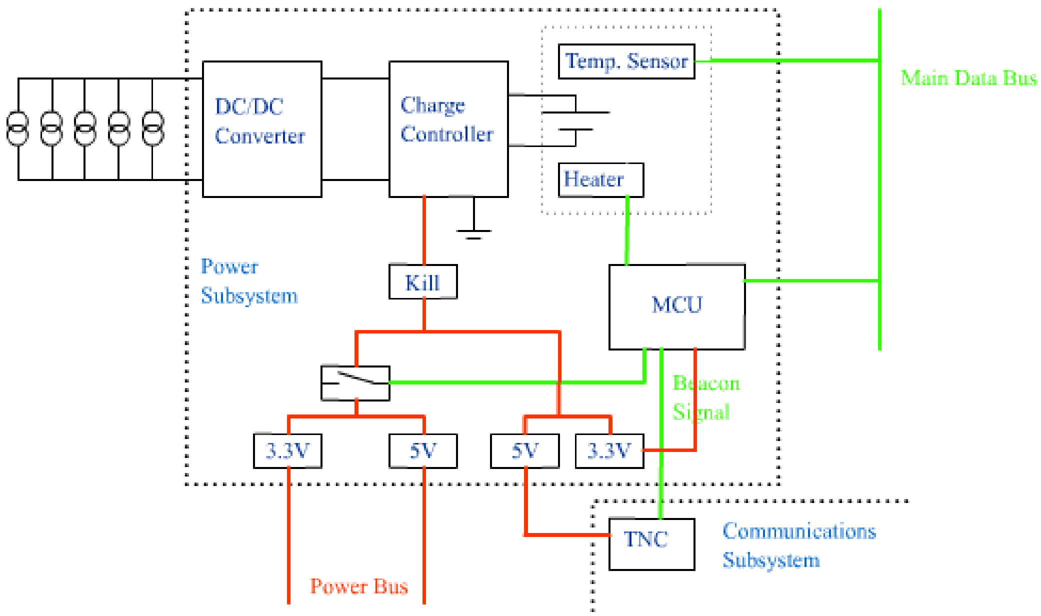

Fig. 3.6.2: Overview of Power System

The input from the five with solar cells covered sides needs to be converted to a stable voltage level. The charge controller manages the charging of the batteries and supplies the power. When the kill switch is open, current flows through the power lines to the power board controller and the communications system controller always. The other systems can be disconnected from the supply by the micro controller unit of the power system. This will be done in emergency mode. In addition the MCU can send beacon data to the TNC, containing information on the satellite's vital status.

6.3 Conclusion

The given values for the solar cells are theoretical only. In reality they must be decreased by an inherent degradation factor. Also path losses were neglected, which need to be taken into account for ongoing calculations. The several components of the system need to be defined in more detail, in particular the choice of the batteries.

References

[1] Larson, W.J. and Wertz, J.R. (1996) *Space Mission Analysis and Design*. Kluwer Academic Publishers, Dordrecht, The Netherlands
[2] Ley, W. (2002) *Energieversorgungssysteme*. FH Aachen, Aachen, Germany
[3] Brown, C.D. (2002) *Elements of Spacecraft Design*. AIAA Publishers, Virginia, USA
[4] www.dtusat.dtu.dk
[5] www.hau.ac.kr/ssrl/

7. Thermal

Artur Scholz

Contrary to most subsystems the thermal subsystem is not independent of the others. All elements in the spacecraft have an influence on the thermal housekeeping by either emitting or absorbing energy or both respectively. Since the CubeSat is very limited in space and mass we cannot expect to have the components insulated from each other. Here the solution lies in the useful configuration of the elements in order to protect vulnerable components with stringent temperature boundaries.

7.1 Requirements Analysis

Table 374.1: Thermal Subsystem Requirements from System Level

Functional Requirements
- Keep temperature of components between their boundaries - Measure temperatures at designated points
Constraints
- Peak power < 1W - Average power < 80mW - Mass below 30g

Typical temperature ranges for components are listed below.

Table 3.7.2: Common Temperature Limits [1]

Components	Typical Temperature Range [°C]
Electronics	0 .. +40
CMOS Camera	0 .. +40
Batteries	+5 .. +20
Solar cells	-100 .. +100
Structures	-45 .. +65

As we can the batteries and also the electrical components have the most narrow temperature ranges and therefore we must take special care of them. For the solar cells it should be mentioned that they work better at lower temperatures.

7.2 Design Analysis

The design of the thermal system shall be based on passive methods. This is so because we want to avoid power consumption in cases were it would not be necessary. On the other hand we must be aware of the fact that we will have to implement active methods as well.

7.2.1 Heat Sources

In figure 3.7.1 we will get an idea about the satellites thermal environment.

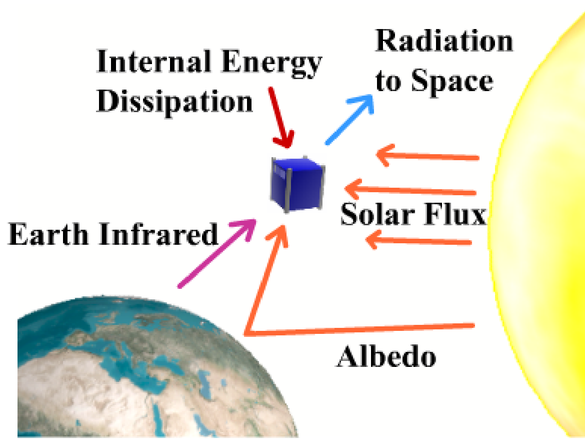

Fig. 3.7.1: Thermal Environment of Satellite

The largest source is the direct solar flux. The mean value of this solar flux at the mean sun-earth distance is called the “solar constant” and is 1371 W/m^2. It is not really a constant but varies from 1316 W/m^2 (Europe summer) to 1428 W/m^2 (Europe winter) because of the slightly elliptical orbit of the earth about the sun. The fraction of incident sunlight that is reflected by the earth is termed albedo. For an orbiting spacecraft the albedo value depends mainly on the sunlit part of the earth, which it can see. Albedo radiation has approximately the same spectral distribution as the sun. Albedo is highly variable across the globe and depends on surface properties and cloud cover. It also depends on the solar zenith angle. We need to consider that during eclipse there will be no influence of solar flux and albedo.

Table 3.7.3: Summarizing the Various Heat Sources

Heat source	Magnitude
Direct solar flux	1371 W/m^2 (1316 .. 1428)
Albedo	30% of direct solar flux
Earth infrared radiation	230 W/m^2
Energy dissipation inside spacecraft	1 W

7.2.2 Thermal Control Components

The following components and devices are used extensively in thermal control subsystems of common satellites.

Table 3.7.4: Common Thermal Control Methods

Component	Description	Remarks
Coatings	Surfaces with special radiation properties that provide the desired thermal performance. Examples: paints, mirrors, silverized plastics	Very useful for our purpose
Insulation	Most common is MLI	Could be useful for the batteries, but expensive
Heaters and thermostats	Controls temperature of delicate components and heats them to the desired temperature	Possible need for batteries etc.
Radiators	Heat exchanger on the outer surface of the spacecraft that radiates waste heat to deep space	Could be a black painted surface
Heat pipes	Transfers heat away from one component towards e.g. a radiator	Not useful for us; to complex

A typical heater element weights about 4 gram and consumes less than 300mW. We could afford up to three heater elements and still would stay in the budget. The sensors masses can be neglected and they consume only around 8mW.

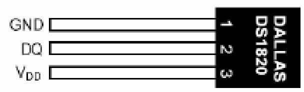

Fig. 3.7.2: Thermostat

Fig. 3.7.3: Heater

7.2.3 First-order Thermal Analysis

In order to get an initial idea about the temperatures at the spacecraft we use the energy balance equation, which is:

$$\alpha_{sun} A_{sun} G_s + \alpha_{earth} A_{earth} q_I + a \cdot \alpha_{sun} A_{earth} G_s + Q_{internal} = \sigma \varepsilon A T^4$$

where

Factor	Description	Consideration
α_{sun}	is the solar absorptivity;	0.38 for Aluminum 6061-T6, 0.98 for black paint, 0.67 for solar cell.
A_{sun}	is the area facing the sun;	
G_s	is the solar constant;	1371 W/m²
α_{earth}	is the earth infrared absorptivity;	α(T,λ) = ε(T,λ) see values for ε
A_{earth}	is the area facing the earth;	0.01 m² (bottom plate)
q_I	is the earth infrared emission;	230 W/m²
a	is albedo;	0.3
$Q_{internal}$	is the internal heat dissipation;	1 W
σ	is the Stefan-Boltzmann constant;	5.67×10^{-8} Wm^{-2}K^{-4}
ε	is the infrared emission of the satellite;	0.035 for Aluminum 6061-T6, 0.88 for black paint, 0.83 for solar cell.
A	is the total emitting surface area;	
T	is the satellites temperature.	

Furthermore we will assume the following properties for the spacecraft:

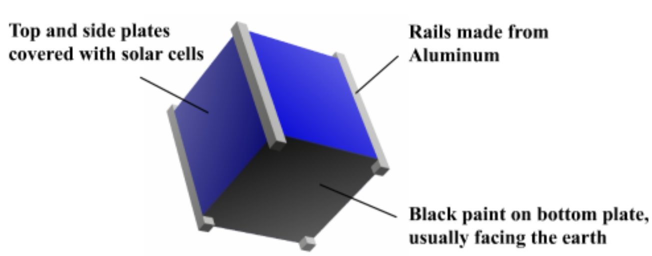

Figure 3.7.4: Thermal Properties

With that information we can now get the equilibrium temperatures assuming a steady state of the hot and the cold case. The hot case is when the spacecraft is exposed to all heat sources, whereas the cold case is when the satellite passes the shadow of the earth. Then there is no influence of albedo or sunlight.

Hot Case			
$\alpha_{sun} * A_{sun} * G_s$	0.67 * 0.01 m² * 1371 W/m²	9.1857 W	Top plate facing sun
$\alpha_{earth} * A_{earth} * q_I$	0.88 * 0.01 m² * 230 W/m²	2.0240 W	Bottom plate facing earth
$a * \alpha_{sun} * A_{earth} * G_s$	0.3 * 0.98 * 0.01 m² * 1371 W/m²	4.0307 W	Bottom plate receiving albedo
$Q_{internal}$	1 W	1 W	
$\sigma * \varepsilon * A$	5.67×10^{-8} Wm^{-2}K^{-4} * (0.83*0.05m² + 0.88*0.01 m² + 0.035*0.00768 m²)	$2.87*10^{-9}$ W/K	4 sides and one top plate with solar cells, bottom plate with paint and rails of aluminum.
$T_{equilibrium}$		**274 K = +1°C**	

Cold Case			
$\alpha_{earth} * A_{earth} * q_I$	0.88 * 0.01 m² * 230 W/m²	2.0240 W	Bottom plate facing earth
$Q_{internal}$	1 W	1 W	
$\sigma * \varepsilon * A$	5.67×10^{-8} Wm^{-2}K^{-4} * (0.83*0.05m² + 0.88*0.01 m² + 0.035*0.00768 m²)	$2.87*10^{-9}$ W/K	4 sides and one top plate with solar cells, bottom plate with paint and rails of aluminum.
$T_{equilibrium}$		**180 K = -93°C**	

7.3 Conclusion

With these figures we are somewhat in the temperature ranges but at the very lower end. This might not be too bad as we should notice that this calculation is done for the outer surfaces. Hence they could be used as a heat sink for the inner components. Also these surfaces are mostly covered by solar cells, which have a better performance at lower temperatures. On the other hand we have made some assumptions that surely need some revisions in later phases. For example, we have assumed that the body perfectly conducts heat and therefore having the same temperatures at any points at all times. By doing so we have neglected the thermal capacity of materials that helps to store some heat in the spacecraft, thus raising the temperate.

For the next phase computer based thermal calculations have to be done. Also the inner components and their properties have to be taken into account. Certainly we will need to protect certain components with heaters, namely the batteries.

References

[1] Larson, W.J. and Wertz, J.R. (1996) *Space Mission Analysis and Design*. Kluwer Academic Publishers, Dordrecht, The Netherlands
[2] Fortescue, P. and Stark, J. (1995) *Spacecraft Systems Engineering*. Wiley & Sons Ltd, West Sussex, England

8. Structure and Mechanism

Marco Hammer
Robert Klotz

The structure must provide housing for the payload and the subsystems. A satellite's structure normally appears to be subdivided in a primary structure that carries the spacecraft major loads and a secondary structure that supports wire bundles, propellant lines, non-structural doors and brackets for the components. Because the structure of a pico satellite is to be designed, which imposes much less complexity than a normal-scale satellite and parts such as doors and propellant lines would be a misunderstanding of the mission concept, it seems advantageous to only consider one general structure. Yet the most important issues when designing a pico satellite structure is to constrain the mass to a minimum. To do so, the material thickness choice is a compromise between the necessary stability on the one hand and a feasible mass reduction on the other.

Spacecraft mechanisms are always a source of critical failures. Therefore we intend to use as few as possible. But there are some cases where mechanisms become unavoidable, e.g. the antenna deployment mechanism. The antenna has to be stored packed during launch and needs to be unfolded in space. Also important are the kill switches that set the satellite in operation mode when it leaves the P-POD deployer.

8.1 Requirements Analysis

Combining the general specifications from the CubeSat concept (table 1.4.1) together with the specific requirements evolving from the Compass-1 mission goals (table 1.3.1) we can establish a system level requirements list shown below.

Table 3.8.1: Structure Requirements from System Level

Functional Requirements
- Provide housing for the payload - Provide mechanical interfaces for the subsystems - Withstand the mechanical stresses the spacecraft is exposed to - Provide a radiation shielding for inner parts - Fulfill deployer and launcher requirements - Supply harness for components - Structure will have to be made as light as possible
Constraints
- Mass below 280g - Center of mass within 20mm sphere of geometric center - Thermal expansion factor for main structure material 23.6µm/m°C - Test margin 125%

8.1.1 Environments

Since in space the environment is very different to the earth atmosphere environment we experience every day there are certain points we will have to keep in mind during the satellite design process. Environmental facts that will have an effect on the satellite's structure will be the followings:

- Radiation
- Out-gassing
- Temperature
- Upper Atmosphere

The structure must possibly be designed in such manner that the temperatures of the satellite's components stay within their functionality limits. Estimated values from -60°C to 80° Celsius for the structures surface and from -20°C to +40°C for the inner parts will be sufficient for the preliminary design.

8.1.2 Configuration of the Subsystems

In terms of testing and possible replacement of the components it is very important to use releasable attachments that can be used more than 1 time. Payload and attitude stabilization has the most stringent influence on the spacecraft configuration and packaging. Sensing devices always require aligning accuracy. The packaging designer must locate sensors to be unobstructed by antennas or solar arrays. Communication antennas also require rigidity, thermal stability, and a field of view. Components for command and data handling are often vulnerable to the environments of outer space, so we usually bury them in the center of the spacecraft to shield against radiation. Finally, batteries should be accessible for pre-launch testing or replacement and placed where they will be at they're optimum temperature. We should design special joints to connect members with different materials because their varying rates of thermal expansion and contraction can be detrimental.

8.1.3 Launch Load

As said before, the highest loads the structure will have to withstand will appear during launch phase. Important for the structure is that:

- the satellite's structure shall not respond to launch vehicles natural frequencies. Natural frequency of satellite shall be higher than the natural frequency of the launch vehicle;
- transients and steady-state accelerations originating from engine thrust changes at ignition and burn-out, wind gusts and vehicle maneuvers are to be analyzed and defined;
- random vibration from engines and other sources like acoustic noise from turbulences during flight are a critical source of load especially for thin walled structures such as skin sections and solar array panels.

The launch system will probably be a Dnepr LV (Russia) based on SS-18 ICBM from the launch agency "Internationa1 Space Company, Kosmotras". The characteristic launch loads are given by the agency:

- Maximum axial quasi-static g-load: 7.5
- Maximum lateral quasi-static g-load: 0.8
- Integral level of sound pressure: 140dB

8.1.4 Connections and Bonds

Possible are screws (M2 for example), epoxy based glue , adhesive bonding . The several connections of each part are TBD.

8.1.5 Stiffness and Strength

A CubeSat should be designed with a structural stiffness. Fundamental frequencies are not less than 20Hz in the longitudinal axis and 10 Hz in the lateral axis.

8.1.6 Stability

The load is only inertial load. The satellite structure must get over a maximal g-load during start phase of 8.3g. Required is, with a design margin of factor two a 16,6g load.

2 separation springs must be included at designated contact points.
>TBD

A Kill switch is required to power off the satellite inside the P-Pod.
>TBD

8.2 Design Analysis

The design process of the structure is as with the other subsystems an iterative process that accounts the upcoming necessary changes evolving from the interaction between the subsystems.

8.2.1 Frame Material

Stanford proposed to use either aluminum 7075-T6 or 6061-T6 for the frame. It would also be possible to use a different material, provided that its thermal expansion factor equals them of the P-POD material (7075-T73). To evaluate the suitable material for our satellite we have done a comparison between the two aluminum materials, which is shown in the table 3.8.2.

***Table 3.8.2: Frame Material Comparison* [1]**

	7075-T6	**6061-T6**
Density	2.81 g/cm³	2.7 g/cm³
CTE, linear 20°C	23.6 µm/m°C	23.6 µm/m°C
Modulus of Elasticity	72 GPa	69 GPa
Heat Capacity	0.96 J/g°C	0.896 J/g°C
Tensile Strength, yield	505 GPa	275 GPa
Costs	Expensive	Cheap

As the table shows there are not many discrepancies between those two aluminum types. The main advantage of the 7075 is its very high strength, optimized for highly stressed structural parts. A look at the other CubeSat group's documentations and the characteristics of the two launch rockets demonstrate that this material would be oversized for our purpose.

In general 6061-T6 provides excellent joining characteristics, good acceptance of applied coatings and combines relatively high strength, good workability, and high resistance to corrosion. Moreover it is widely available and much cheaper than the 7075-T6.

8.2.2 Reference Model

Based on the structural drawing provided by Stanford University, which is illustrated in figure 3.2.1, we have designed a reference model that will help to establish a nomenclature for the parts the frame consists of. Also this model declares the axis origin in the geometrical center of the spacecraft.

The reference model will also support the assessment of the preliminary physical properties, such as the mass, the moments of inertia and the center of gravity.

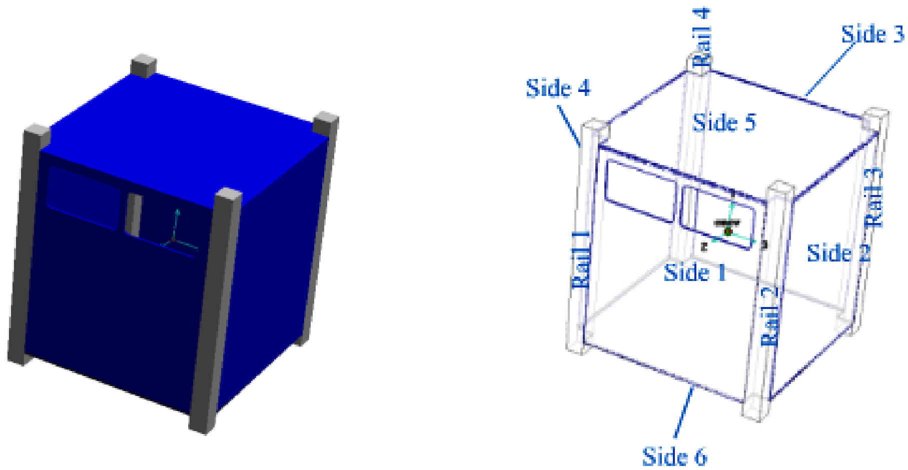

Fig. 3.8.1: Reference Model

The model is based on the following simplifications:

- the thickness of each plate is 1mm
- the rails are made solid

With those properties we will get a total mass of about 240g. This value is accordingly high because of the simplifications we made and needs to be reduced by constructive approaches as described in the next sections.

8.2.3 Structure Elements and Interfaces

The main frame consists of the following elements:

- rails
- beams
- plates

It accommodates the following parts:

- main bus board
- subsystem boards
- antenna
- other elements (kill switches, etc.)

In the next sections each element's preliminary design is illustrated and explained. The further detailed design process will be oriented on those proposals but obviously we will need to carry out necessary modifications and adjustments.

Rails

The rails are the only parts of the satellite that will have contact to the P-POD. All edges that contact the rails must be rounded. CubeSats must have at least 75% of flat rail contact with the deployer to prevent cold-welding. All contact surfaces with the P-Pod and other CubeSats have are not to be made of raw material.(Hard Anodize or Derlin Inserts for example).

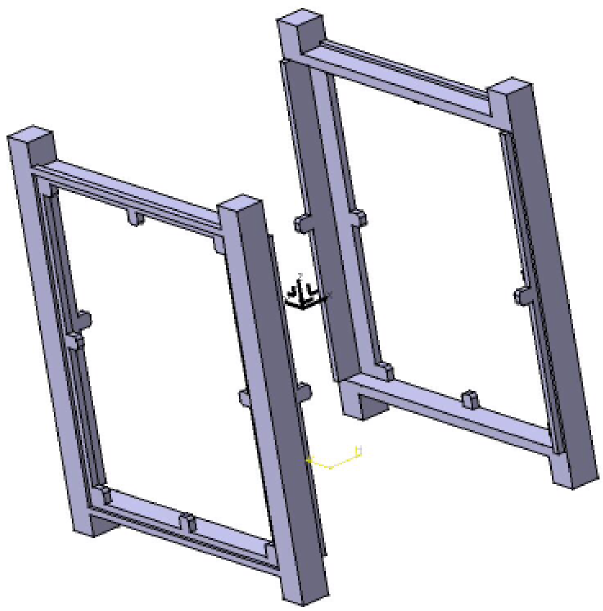

Fig. 3.8.2: Rails

Beams

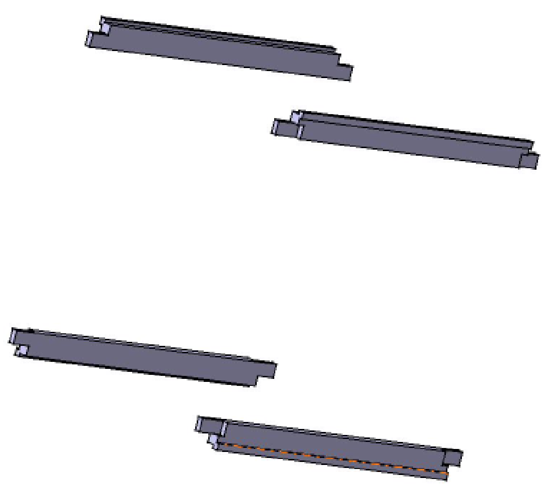

Fig. 3.8.3: Beams

Plates

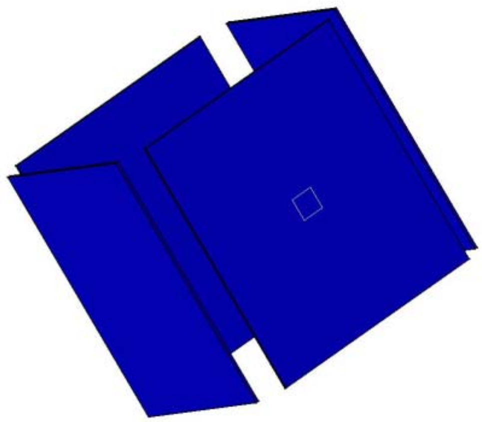

Fig. 3.8.4: Plates

Main Bus Board

The primary payload is a CMOS camera chip and lens system ,which is directly mounted on the main bus board. The camera will be located at the bottom of the satellite (side6).

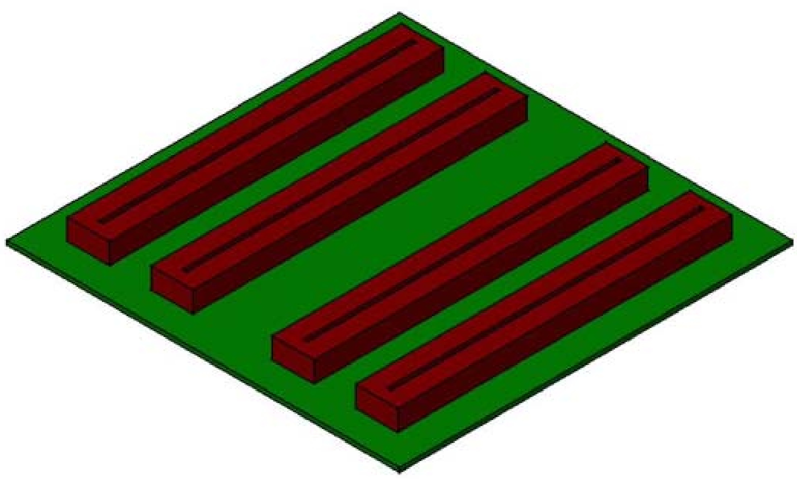

Fig. 3.8.5: Main Bus Board

Fig. 3.8.6: Bus Holders

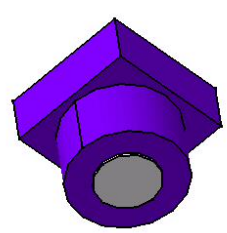

Fig. 3.8.7: Camera Module

Subsystem Boards

Attitude Determination and Control System

The ADCS system will consist of a main bus compatible board with external actuating torquer coils.

Communication

The communication system will consist of a main bus compatible board with external deployable UHF monopole antennas.

Command and Data handling

The command and data handling subsystem is constituted of electronic parts and will be located on a main bus compatible board.

Power System

The power subsystem contains several solar arrays which provide the main power and a minimum of two batteries, which are used as back up and buffer batteries for providing enough power during shadow phases and by current peaks. The best efficiency is obtained when the solar cells point directly towards the sun. Since the satellite always shows a different side to the sun during it's mission and mechanisms that would orientate the solar cells as the position is changing are much to complicated for a spacecraft of this size, these will be mounted directly on each available surface of the satellite. Because of the difference of thermal expansion of the solar cells and the sides of the CubeSat the bonding will be made with glue.

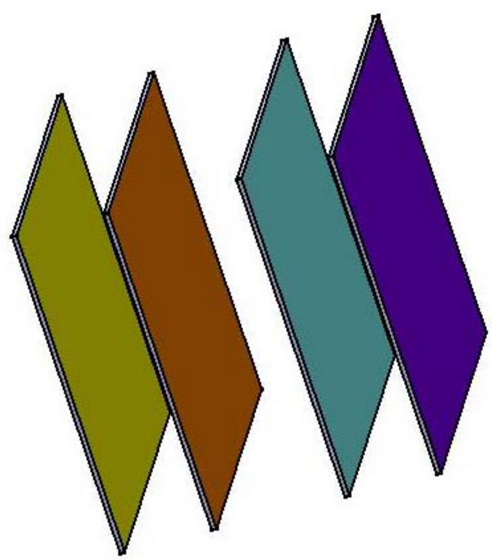

Fig. 3.8.8: The Boards Layout

All boards are additionally fixed on a board holder with aluminum-angles.

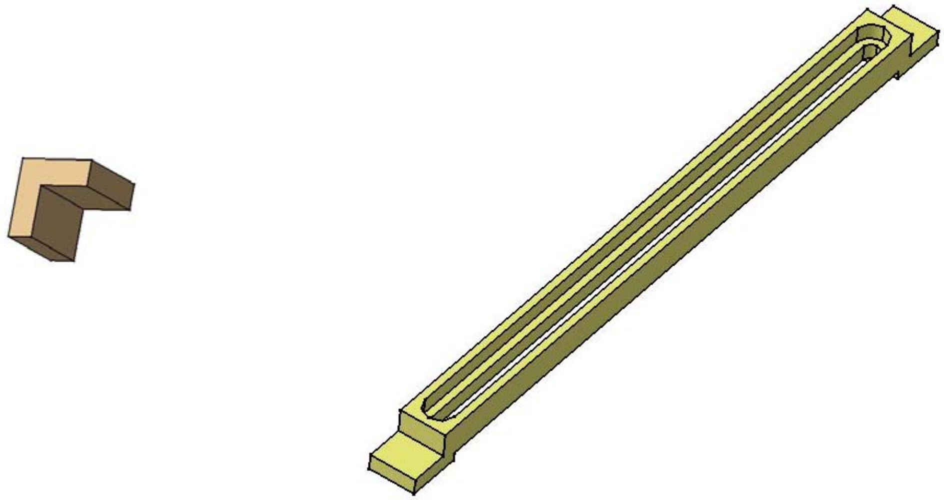

Fig. 3.8.9: Card Holder System

Antenna

For maximum coverage the antenna has to be perpendicular to the vector of emission. The first thought of mounting the antenna on the side 6 and lying in the same plane was replaced by a configuration of four 17cm antennas pointing in different directions as depictured.

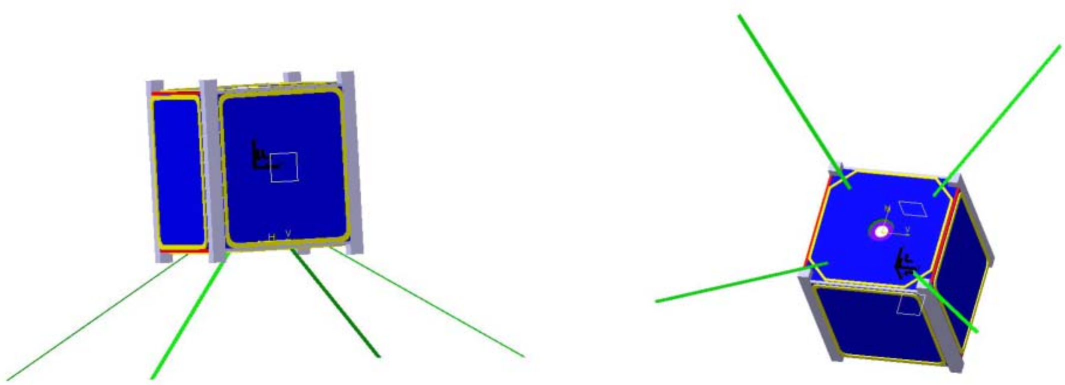

Fig. 3.8.10: Antennas

Other Elements

Kill switch
>TBD

Remove before flight Pin
>TBD

ADCS Coils
The bigger the included surface of the coils is, the better is their efficiency. We intend therefore to make square coils that follow the edges of the side plates and which will be mounted on them. This will avoid extra coil supporting parts. A minimum of 3 coils, 1 for each axis, is required. The possibility of using 6 coils, 2 for each axis (Helmholtz coils) to obtain a better uniformity of the electromagnetic field seems to be a better solution.

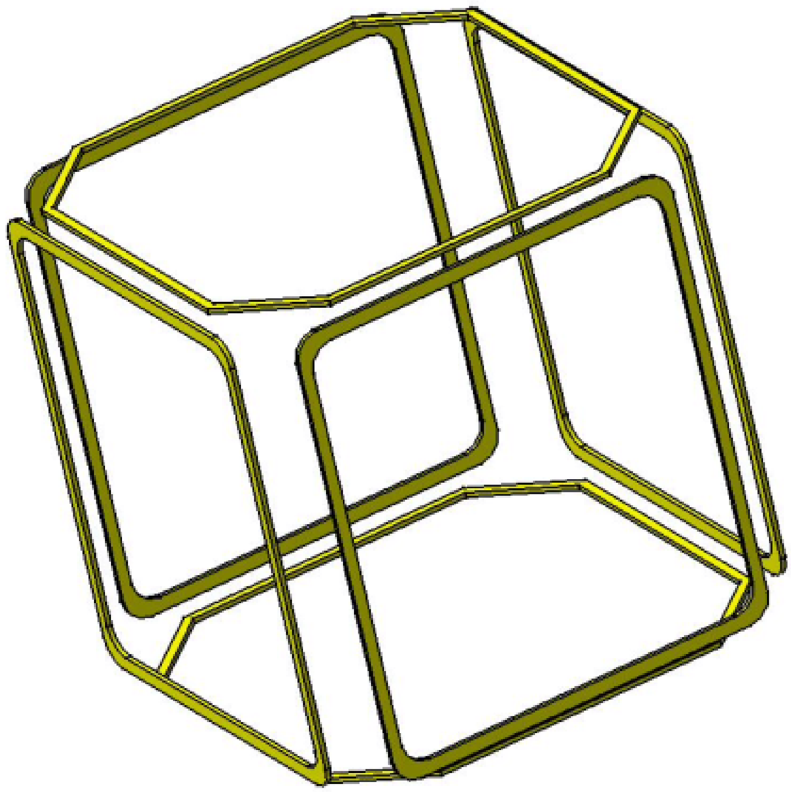

Fig. 3.8.11: ADCS Coils

8.4.2 System Integration

This section gives an overview of how the mentioned elements will be fitted into the main frame. The mechanical interfaces between the elements have to be discussed in the later design phases. The electrical interfaces are subject of the C&DHS and the Power subsystem sections. An important detail that should not be forgotten is that the whole structure is connected to electrical ground.

The figures below show the completely assembled Compass-1, using transparent plates to give insight in the inner configuration as well.

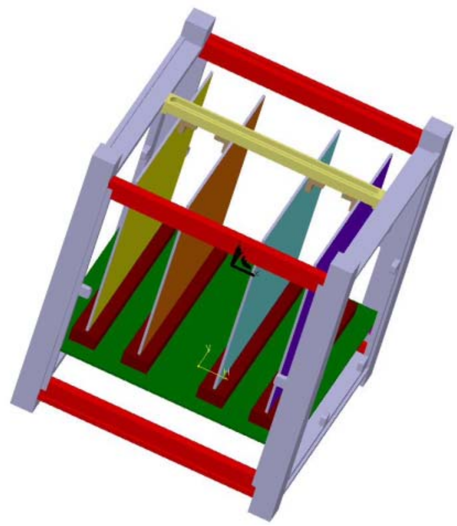

Fig. 3.8.12: Board Integration

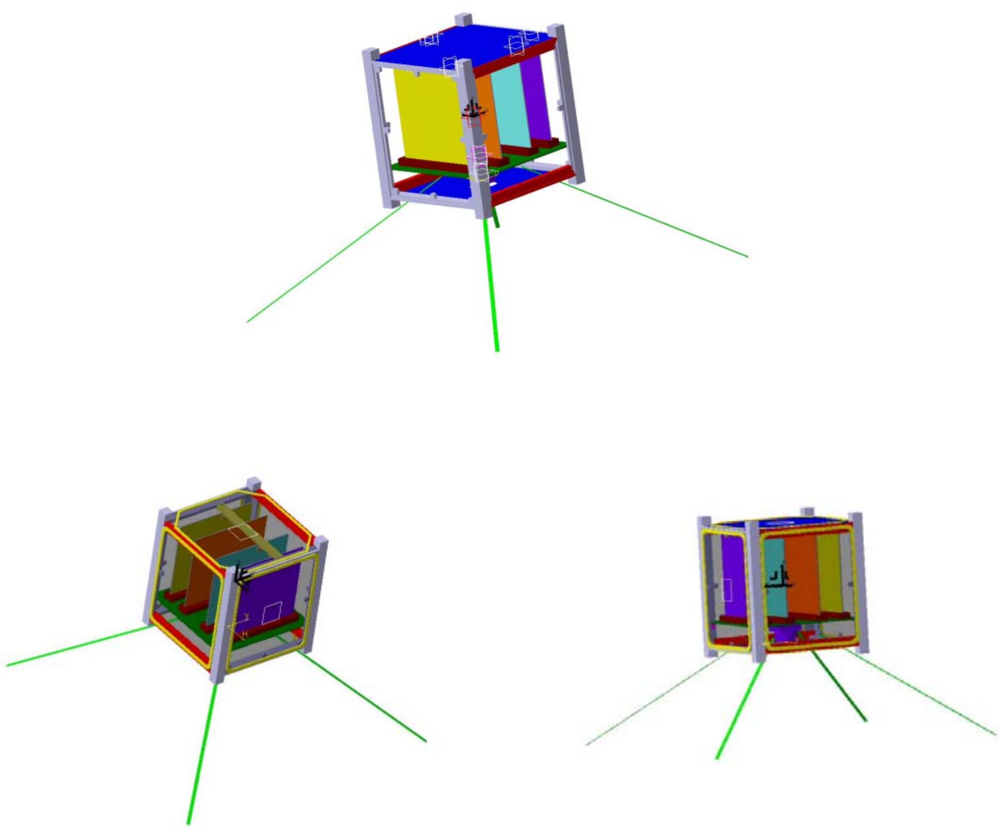

Fig. 3.8.13: Assembled Layout

8.3 Conclusion

It was discussed the preliminary layout of the Compass-1 structure. We selected the 6061-T6 aluminum as the material we will use for the manufacturing. A reference model was established to provide an orientation for other subsystem groups as well. Then the several elements that make up the spacecraft were examined and solutions for their design were presented. In the end we have shown the complete system in an assembled configuration.

References

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Mission Operations

Oscar Moreno
Artur Scholz

1. Introduction

The mission operations really start when the spacecraft is in orbit, but its properties have to be defined from the very start of the design process. That is why we have to concentrate on this topic with the same precision as with the other systems.

This chapter deals with three main issues concerning the mission operations aspect. Those are the communication architecture, the ground station and the user interface. Simply speaking the communication architecture expresses the attributes of the link between spacecraft and ground station. As the satellites communications system is more or less fixed, in terms of available power and antenna size the ground station layout has to be carried out in such a manner that it builds a commensurate counterpart. Finally the user interface is the software solution for the publishing of downlinked images and information.

2. Communications Architecture

The most basic point of departure for the link design is the frequency band we are going to use for the signal transmissions. The next would be to define the desired bit error rate. Also the choice of the polarization of the electromagnetic radiation is somewhat arbitrary. Then there are much more characteristics attached to the RF link design which will be discussed in the following sections.

2.1 Link Attributes

Frequency Band

The frequency band provides the carrier wave for the signal to be transmitted. RF bands are only a small part of the electromagnetic spectrum. The common denominations for distinguish RF bands are given in Table 4.2.1.

Table 4.2.1: Frequency Bands [1]

Band	Frequency	Service
Ka	17.3 .. 30 [GHz]	Sat.com (fixed), High-speed communication
Ku	10.3 .. 14.7 [GHz]	Sat.com (fixed), Exploration
X	7 .. 8.5 [GHz]	Sat.com (fixed), Military Sat
C	3.4 .. 7.0 [GHz]	Sat.com (fixed), Aviation radio navigation
S	1.7 .. 2.7 [GHz]	Sat.com (fixed), Aviation radio navigation
L	1.2 .. 1.7 [GHz]	Sat.com (mobile)
UHF	300 .. 3000[MHz]	Amateur HAM, data
VHF	30 .. 300 [MHz]	FM Broadcast, TV Broadcast, Mobile radio, Amateur HAM

According to its definition, the amateur service is for the purpose of self-training, intercommunication and technical investigations carried out by amateurs without pecuniary interest. Since that description can be applied to the projects principles without a doubt and the amateur band has the great advantage that it is free of charge, there was not a long hesitation to go for that opinion.

The local AMSAT group for Germany was contacted and general enquiries for frequency coordination were started. As a preliminary response, three RF frequency bands were offered for a discussion. Those are 144Mhz, 435Mhz and S-band. A look at the other CubeSat groups showed that all of them use at least two frequencies, one for uplink and another one for downlink. The reasons for this choice could not be clarified from the published documentation. Most probably this is so because when using only one frequency for up- and downlink the spacecraft can either send or receive at a time, giving no chance for a transmitter shutdown commando in case of an error loop in the transmission process.

As we want to keep the spacecrafts design simple, we intend to go for the solution of occupying only one frequency. The illustrated event of a transmitter error has to be managed on board, rejecting a faulty end-less transmission cycle.

Modulation

Modulation varies the characteristics of the carrier wave to transport a signal. The signal itself is of relative low frequency or digital. Generally there exist three ways to modulate the carrier wave, either by modifying its amplitude, its frequency or its phase. Since the use of computers in spacecrafts, digital modulation or pulse code modulation respectively are used exclusively. The major advantage of digital modulation is that coding techniques can substantially reduce the bit error rate. The academic field of data transmission is loaded with modulation strategies. Some of the more notable techniques are:

- Frequency Shift Keying FSK
- Multi-level Frequency Shift Keying MFSK
- Continuous Phase Frequency Shift Keying CPFSK
- Minimum Shift Keying MSK
- Gaussian Minimum Shift Keying GMSK
- Phase Shift Keying PSK
- Quadrature Phase Shift Keying QPSK

Each of the modulation formats listed is suitable to specific applications. There is a lot of information and literature available for all of those formats, so we won't go into detail here. But as a recommendation the use of GMSK is strongly suggested. The reasons are that this format is relatively simple, yet so powerful. It comes along with a possible data rate of 9600bps and the occupied bandwidth is 12.5KHz, but as mentioned by AMSAT it will be more likely 25KHz.

Protocol

For the protocol we are envisaging on a widely spread standard. This is so because we want to permit a lot of radio amateurs to participate in the satellites operation.

A common protocol that emerged from a computer protocol is the AX.25. It is a protocol that can accept and reliably deliver data over a variety of communications links between two signaling terminals. As defined, this protocol works equally well in either half- or full-duplex amateur radio environments, and has been improved for operation over partially impaired HF circuits.

This protocol is also used by virtually all other CubeSat groups and performs very well.

Bit Error Rate

For analog transmission we use the signal-to-noise ratio (SNR) to categorize the quality of the RF link. For digital transmissions however it is more convenient to apply the BER term to express its performance. The bit error rate states the probability of a faulty bit to occur in a transmission link. Common values are 10^{-7} for uplink commands and 10^{-5} for downlink data.

Due to the fact that we will use the same equipment for both uplink and downlink the chosen BER value is valid for both. A value of 10^{-5} for the bit error rate will satisfy our demands on the link quality. In other words, for each one hundred thousand bits transmitted, only one will be flawed.

Polarization

Antenna output can be polarized. Most space applications use circular polarization. If polarization of transmitter and receiver do not match exactly, there will be a polarization loss, which can be very large. For example, if the transmitter send right circular and the receiver is set to left circular, it will not receive any signal. The table lists the polarization losses for different combinations.

Table 4.2.2: Polarization Loss

	Wave polarization			
Antenna setting	Vertical	Horizontal	Right circular	Left circular
Vertical	0 dB	Infinitive	3 dB	3 dB
Horizontal	Infinitive	0 dB	3 dB	3 dB
Right circular	3 dB	3 dB	0 dB	Infinitive
Left circular	3 dB	3 dB	Infinitive	0 dB

As we intend to use circular polarization, the ground station antenna has to have the same one.

2.2 Link Budget Parameters

The link budget is the sum of all the power losses and gains in an RF transmission. Basically three areas can be distinguished. Those are the transmitter, the path and the receiver. Since we will have different parameters for the uplink and the downlink ways, it is needed to look at those two cases separately.

The object of a link design is to determine if the transmitted power of a given design is adequate to successfully transfer the desired data rate. The link is evaluated by systematic tabulation of gains and losses, as mentioned already. The parameters the link budget consists of are discussed in the next paragraphs.

Effective Isotopic Radiated Power

The power per unit area or power density at a given distance from a transmitting source is

$$P_0 = \frac{P_t G_t}{4\pi \cdot R^2}$$

where

P_t	is the power input to transmitting antenna [W];
P_0	is the power density, power per unit area on the surface of a sphere of radius R centered at the transmitting source [W/m²];
G_t	is the gain of the transmitting antenna;
$4\pi R^2$	is the surface are of a sphere of radius R [m²];
R	distance between transmitter and receiver [m].

The power density at a receiver is proportional to the product P_tG_t. This leads to the one of the major trades between antenna gain and transmitter power. The product is called effective isotopic radiated power:

$$EIRP = P_tG_tL_t$$

Hence, in order to be able to calculate the EIRP we will need information on the transmitter power and the antenna gain. Those values we will use for the link budget are only assumed ones, based on other CubeSat projects and similar projects.

A conservative estimation on P_t would be 0.5W. For the antenna gain we assume 0.6dBi, which is the peak gain for turnstile antenna[1]. L_t is the transmitter line loss, which is usually estimated with –1dB.

Free Space Path Loss

The power at a receiver is

$$P_r = P_0 A_r \quad \text{respectively} \quad P_r = \frac{P_t G_t G_r \lambda^2}{(4\pi)^2 \cdot R^2}$$

where

P_r	is the received power at the antenna [W];
A_r	is the effective area of the receiving antenna [m²];
G_r	is the gain of the receiving antenna.

The power loss is caused entirely by the distance between the two antennas. This is the free space path loss and can be described by

$$L_S = \frac{(4\pi)^2 R^2}{\lambda^2} \quad \text{or} \quad L_S = \frac{(4\pi)^2 R^2 f^2}{c^2}$$

Expressed in decibels this term forms to:

$$L_S = 147.55 - 20\log(R) - 20\log(f)$$

where

R	is the range [m];
f	is the frequency [Hz].

To get the maximum range between our satellite and a ground station we will assume that the minimum elevation angle for the ground antenna to see the spacecraft is 5°. If it would be zero degrees, it would mean that the satellite is in sight as soon as it appears at the horizon. We will calculate for the second one, as it will result in a higher value for the maximum range, thus providing us with some margin. The spacecrafts altitude h is defined by the reference orbit, which are 600km. From figure 4.2.1 we can calculate with some simple geometrical terms the maximum range R.

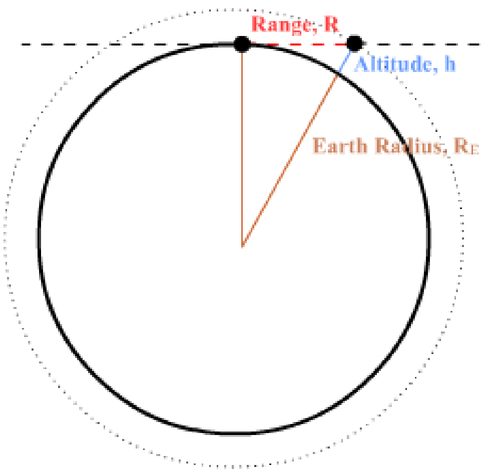

Fig. 4.2.1: Maximum Range Calculation

From Pythagoras we get a maximum range of about 2830km. Together with the frequency of the selected amateur channel, 435MHz, the path loss yields to L_S = -154.26dB.

With figure 4.2.1 we can also do estimation on the pointing error. Assuming that the satellite is pointing in nadir direction towards ground, the angle between spacecraft antennas boresight axis and the receiver antenna axis is about 66°.

Noise

Signal power is always received along with unwanted noise. This noise disturbs the correct interpretation of the signal. The four principle sources of noise are:

- cosmic noise, which comes from all bodies in the universe;
- atmospheric noise from rain, fog and lightning;
- man-made noise of all kinds from electric motors, spark plugs, and the like;
- thermal noise in all electronic devices, called Johnson noise.

To calculate the system noise temperature we have to take three temperatures into account:

$$T_S = T_{(antenna)} + T_{(cable)} + T_{(receiver)}$$

However, it will be more convenient to read the values from a table, as in [3].

2.3 Link Budgets

The link equation or link budget relates all of the parameters needed to compute the signal-to-noise ratio of the RF link. The basic equation used in sizing a digital data link is

$$E_b / N_0 = \frac{(EIRP)L_S L_a G_r}{kT_S R}$$

where

E_b/N_0 is the ratio of received energy-per-bit to noise density;
k is the Boltzmann's constant [1.381×10-23 J/K];
R is the data rate.

Data Downlink Budget

Item	Symbol	Units	Source	Value
Frequency	f	MHz	Defined	435
Transmitter Power	P_t	dB	10*log(0.5W)	-3.01
Transmitter Line Loss	L_t	dB	Estimate	-1
Peak Transmit Antenna Gain	G_t	dBi	Turnstile antenna	0.6
Effect. Isotopic Radiated Power	EIRP	dB	$P_t + G_t + L_t$	-3.41
Transmitter antenna half power beamwidth	θ_t	deg	Estimate	140
Transmitter antenna pointing error	e_t	deg	Estimate	66
Transmit antenna pointing loss	L_{pt}	dB	$-12(e_t/ \theta_t)$	-2.67
Free Space Path Loss	L_s	dB	Calculated	-154.26
Propagation & Polarization Loss	L_a	dB	Estimate	-0.5
Receiver Antenna Peak Gain	G_r	dB	To be determined	X
System Noise Temperature	T_S	K	Estimate[3]	1295
Data Rate	R	bps	Defined	9600
Bit Energy/Noise Ratio	E_B/N_0	dB	From Equation	-0.52 + X
Bit Error Rate	BER	-	Defined	10^{-5}
Required E_B/N_0	$req(E_B/N_0)$	dBHz	For MSK[3]	9.6
Implementation Loss	-	dB	Estimate	-2
Margin	-	dB		-(0.52+9.6+2)+X

As we can see the receiving antenna peak gain needs to be at least 12.5dB. But we will have to seek for a much higher value in order to have some margin, to compensate the losses and the pointing errors.

Command Uplink Budget

Item	Symbol	Units	Source	Value
Frequency	f	MHz	Defined	435
Transmitter Power	P_t	dB	$10*\log(Y_W)$	Y_{dB}
Transmitter Line Loss	L_t	dB	Estimate	-1
Peak Transmit Antenna Gain	G_t	dBi	Unknown	X
Effect. Isotopic Radiated Power	EIRP	dB	$P_t + G_t + L_t$	$Y_{dB} - 1 + X$
Transmitter antenna half power beamwidth	θ_t	deg	Neglected	-
Transmitter antenna pointing error	e_t	deg	Neglected	-
Transmit antenna pointing loss	L_{pt}	dB	$-12(e_t/\theta_t)$	-
Free Space Path Loss	L_s	dB	Calculated	-154.26
Propagation & Polarization Loss	L_a	dB	Estimate	-0.5
Receiver Antenna Peak Gain	G_r	dB	Turnstile antenna	0.6
System Noise Temperature	T_S	K	Estimate[3]	375
Data Rate	R	bps	Defined	9600
Bit Energy/Noise Ratio	E_B/N_0	dB	From Equation	$Y_{dB} + X + 7.88$dB
Bit Error Rate	BER	-	Defined	10^{-5}
Required E_B/N_0	$req(E_B/N_0)$	dBHz	For MSK[3]	9.6
Implementation Loss	-	dB	Estimate	-2
Margin	-	dB		$Y_{dB} + X - 3.72$dB

Here we can see that we can trade the ground station antennas transmission power against the antennas gain. Nonetheless the sum of both must be at least about 4dB. Again it becomes very important to exceed this value by far as we have not calculated for losses in the ground station and furthermore we will need some margin.

2.4 Conclusion

In the first section we have had a look on the parameters important for the establishment of a link budget. With the preliminary defined properties for the spacecraft's communications system we calculated the link budget, leaving the ground station parameters variable. The next step will be to set the ground stations parameters.

As the design of the communication system on board the spacecraft and the ground station matures, the link budget needs to be updated iteratively. Only by doing so, a reliable communication link can be guaranteed.

3. Ground station

The ground station is the counterpart to the satellites communications system. As we have very stringent constraints on the spacecraft's power, the ground station has to come up with the needed features to substitute the loss we have on the spacecraft. Those features are the transmitter power and the antenna gain.

3.1 Requirements Analysis

From both the communications architecture and the spacecraft's communications system preliminary design, we can establish a requirements list for the ground station.

Table 4.3.1: Requirements for Ground Station

Functional Requirements
- Uplink commands and downlink data - Provide margin for the RF links - Track the satellite - Process the received data

3.2 Design Analysis

Basically the ground station layout is the same as on the satellite. Hence, the big difference is that the components on earth can be much bigger, heavier and they have a merit of power available. A typical layout of a ground station is shown below.

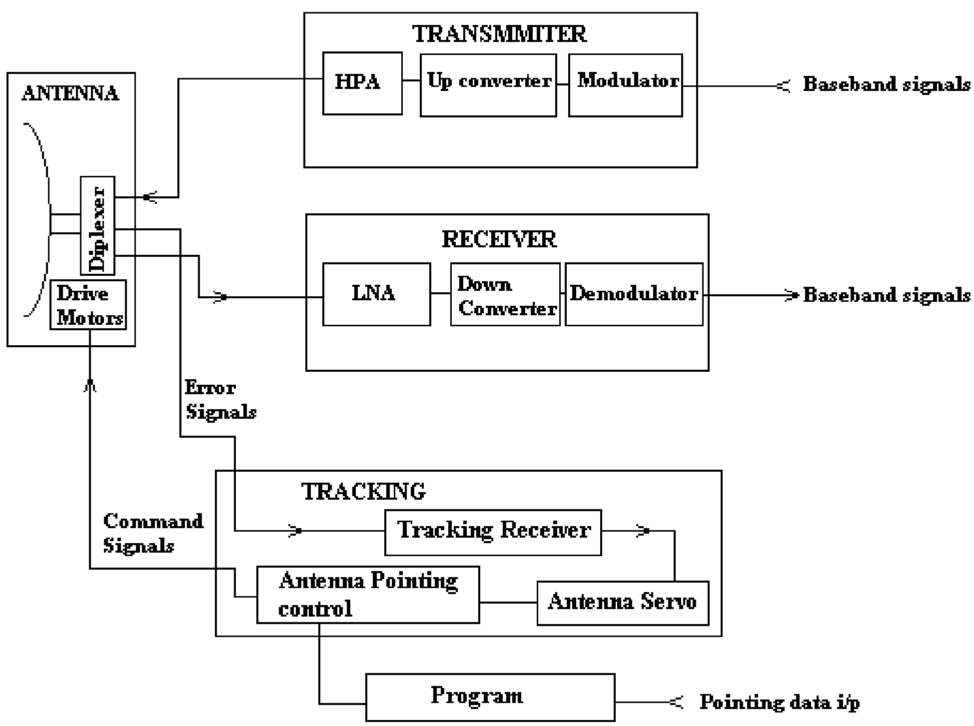

Fig. 3.5.1: Typical Ground Station Layout

3.2.1 Antenna

As we could see from the link budget, the demands on the antennas gain are critical. A gain of at least 12.5dBi would be necessary, not including margin or compensation for the losses. Therefore it will be most important to select an adequate antenna form. A short list of antennas is given in table 4.3.2 together with their usual value for gain.

Table 4.3.2: Some Usual Antennas for Ground Station

Type	Gain	Remarks
Yagi	~12dBi	Can be increased easily by joining more Yagis together.
Parabola	10-65dBi	Has a narrow beamwidth, therefore an exact tracking of satellite becomes important.
Helix	5-20dBi	Similar pattern to parabola.

It shall be noted, that this is an uncompleted list and more and deeper investigation have to be made in further stages.

3.2.2 Transceiver

The link budget for the command uplink states that the quality depends on two factors, the transmitting antenna gain and the transmitting power. One can b e used to compensate the other. Nonetheless, the gain will be defined upon the requirements for the data downlink, thus having a given value for the gain we will have to evaluate the radio power as a next step. Various devices are on the market that can perform for our purpose. A market research has to be done as soon as there are more results from the antenna design.

3.2.3 Computer

The computer might be a usual personal computer with an up to date performance characteristic. The software for the tracking needs to run on the PC as well as the program that decodes the protocol format received by the radio.

Tracking

Tracking a satellite means to determine its position in order to point the antennas boresight on it. Only by doing so the ground station will work with its antennas peak gain.

For the issue of tracking the satellite a combination of hardware and software is used. The hardware is gadgets that can turn the antenna about two axes, the azimuth and the elevation. This is enough to cover the whole sky.

But how do we know where to turn the antenna? Here the software comes into play. It uses the satellite's keplerian elements to predict its trajectory and controls the antennas turning mechanism in such a way to follow the spacecraft's path. An example for such software is WiSP. The information for the satellites keplerian elements can be accessed from the Internet. This data is supplied by NORAD, an organization that tracks everything in space that is bigger than a tennis ball. Another method would be to use the GPS on board the spacecraft for a more precise localization.

3.3 Conclusion

As we see, the ground station is a vital element in the mission operations phase. On the other hand the ground station is not fixed to the Compass-1 spacecraft only but is capable to work with similar satellites, e.g. most amateur satellites as well. Therefore the purchase and/or development of ground station equipment can boost the students' interest in amateur satellite radio and the accompanying technology.

References

[1] Brown, C.D. (2002) *Elements of Spacecraft Design*. AIAA Publishers, Virginia, USA
[2] www.hau.ac.kr/ssrl/
[3] Larson, W.J. and Wertz, J.R. (1996) *Space Mission Analysis and Design*. Kluwer Academic Publishers, Dordrecht, The Netherlands

Programmatics

Artur Scholz

1. Introduction

This chapter is the part that addresses the process of realization of the Compass-1 project, including its studies and later on its hardware and software development and test – both in terms of content, time and cost. The first activities will lead to a comprehensive documentation and a first demonstration model, followed by test results on a more sophisticated model.

It also covers the urgent questions regarding the implementation of the project into the students daily activities, the use and extension of the existing infrastructure and of course its funding and support probabilities.

2. Organizational Structure

2.1 CubeSat Program Organization

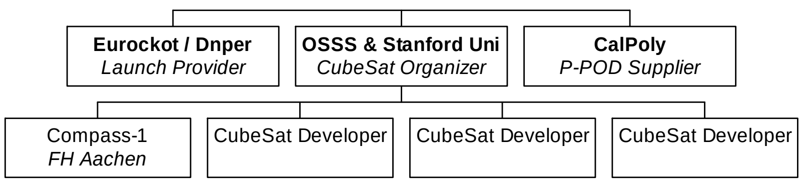

Fig. 5.2.1: CubeSat Program Structure

The persons in charge for the CubeSat program are from Stanford University, namely Prof. Bob Twiggs and assistants. They control the launcher contracts negotiations in order to provide future launches for CubeSats. Four times a year a newsletter is published on the official CubeSat homepage (http://cubesat.calpoly.edu), stating actual developments and contracts. OSSS serves as the commercial contractor for legal issues and sells also CubeSats components. The prices for a launch greatly vary with the number of CubeSats participating. A conservative value would be 30.000 US-$ nonetheless. Calpoly provides the P-POD and is responsible for the pre-launch testing.

2.2 Compass-1 Organizational Structure

Even for a student organized project the structural layout of a commercial organization can be adopted with some little modifications. The structure strongly orients on the suggestions of the ECSS. It is indispensable that participants have an equal vote on the satellites design in order to feel committed to its development and to get the most out of this project. Each student however has to decide for him/herself how much time to devote to this endeavor and is held responsible to act accordingly. As in case of a task postponement from one participant all others have to suffer, each student shall communicate perceptible obstacles as soon as possible to the team.

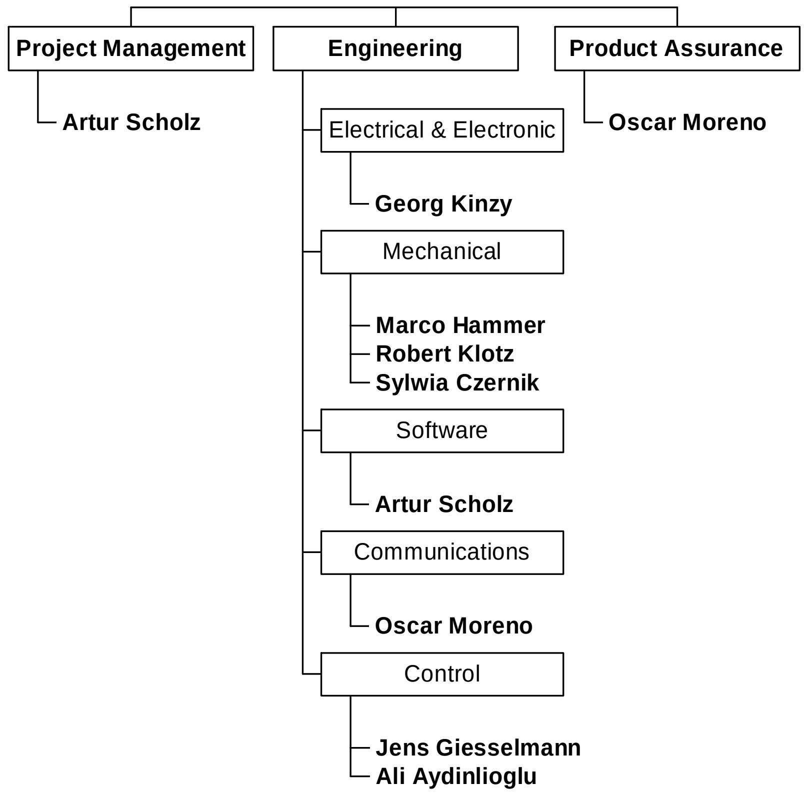

Fig. 5.2.2: Compass-1 Organizational Structure

The above figure lists the disciplines the development of Compass-1 consists of. The organizational structure must not be mixed up with the work breakdown structure. The scope of each discipline is not discussed here in further detail as there is enough information provided in the ECSS documents.

2.3 Work Breakdown Structure

The WBS gives an overview of the scope of work connected to the project. Starting with the basic elements for a space project, a tree of tasks can be drawn up. Each block is a working area that can be divided in subtasks, which again can be split up in subtask and so on. Up to now the tree is split to the second level only as the students have to draw up the appropriate subsequent levels during the definition phase.

The students need to create work packages that relate to the respective working area. Therefore they have to write a work package description, which states the inputs and outputs of the work, i.e. the necessary information for carrying out the task and the expected results. The work packages in turn shall then be processed and end up in a detailed description, enabling others to get insight and an understanding of the planned development. These work packages form the basis for the Compass-1 development and students may declare them as an accredited study work. For comprehensive work packages it might be possible to evolve a diploma thesis out of it. The detailed definition of work packages would also allow students from other universities or even other countries to contribute to this project. Moreover the packages would permit us to carry out the work on other sites as well, e.g. at international partner universities and institutes.

The most important issue here is that the interfaces between the packages have to be defined carefully and exact. Otherwise it would end up in a bunch of subsystems that serve their purposes but would stop working when they are integrated to a system.

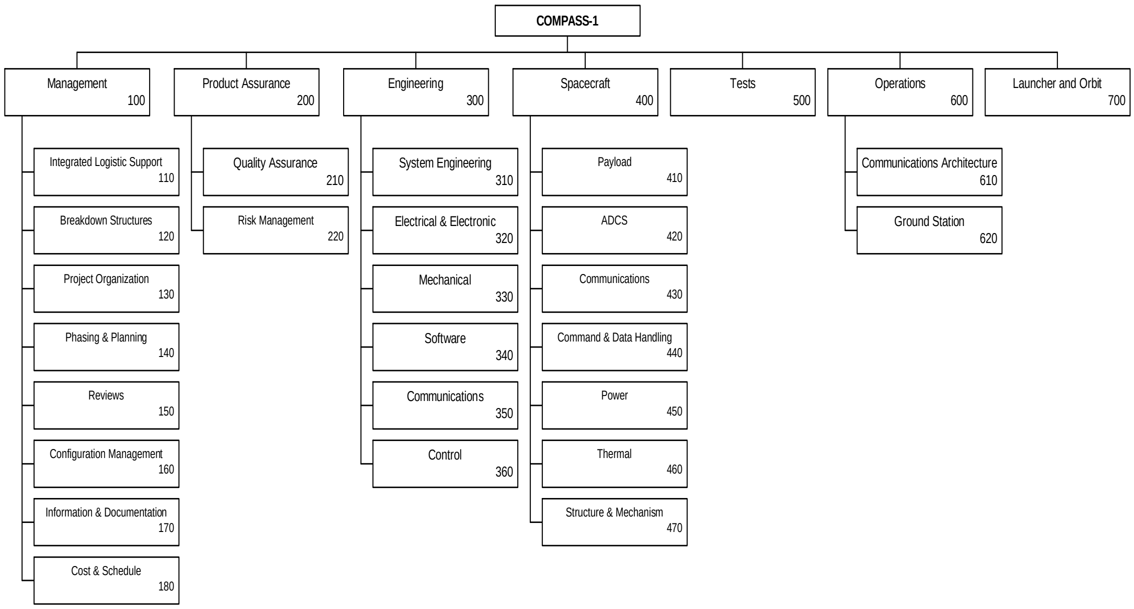

Fig. 5.2.3: Work Breakdown Structure

The WBS is a powerful tool to help organizing the work to be done. It also serves as the basis for the documentation process, by providing numbers to the documents. Those numbers directly correspond to the WBS numeration and are therefore called WBS numbers.

3. Time Schedule

The projects lifetime ranges from the commencing feasibility studies through the whole satellite development process, its launch and operation in orbit till the spacecrafts final disposal through atmospheric re-entry. The time span from detailed definition studies to a spacecraft ready for launch is envisaged with one year. This is a tight schedule considering that all participants are students and can only devote parts of their spare time to the project. But it seems to be quite important to keep the timeline so constrained in order to allow the students to follow the complete process from beginning to end. By this the gained experience will be high accordingly and the students will feel much more connected to their project.

The overall time schedule is shown in figure 5.2.4. It comprises the several phases that correspond to the definition of the ECSS documents. Milestones are indicated as well. The milestone philosophy greatly helps to evaluate the actual status against the planned status. Each milestone represents a review at which all participants present their work done and compare their outcome to expected results. Furthermore failures in interface definitions or other misunderstandings can be easily allocated and resolved. Ad-hoc activities, which are not planned but evolve and become urgent are noted down in a so-called action-item list.

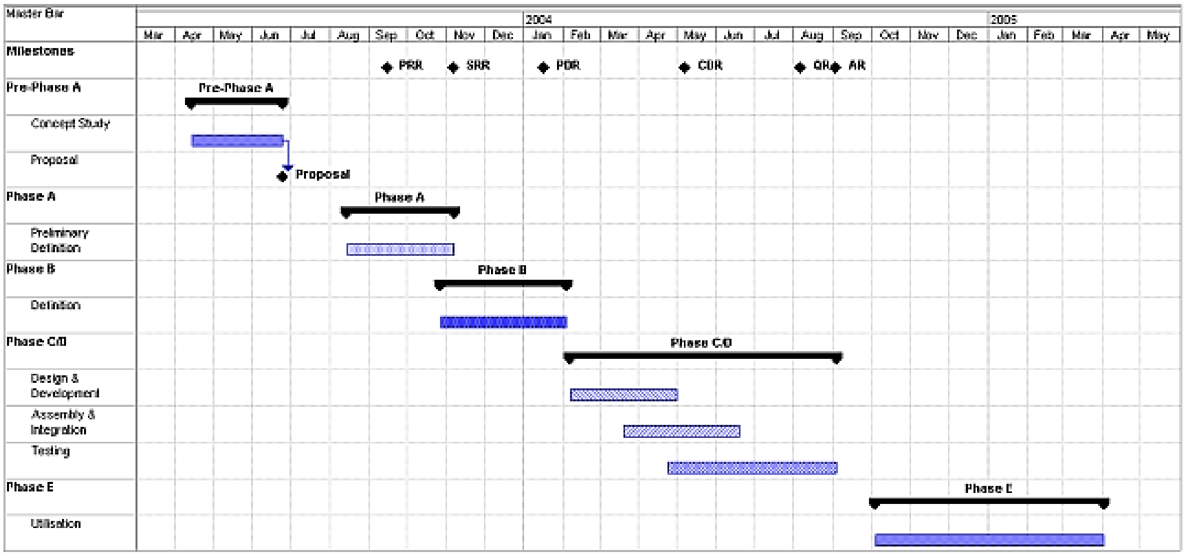

Fig. 5.2.4: Overall Time Schedule

In order to support the development process it becomes necessary to build some spacecraft models that serve as a verification basis for the subsystem developments. Usually those models are the structure/thermal model, the engineering model, a qualification model and a flight model. The costs for those would be accordingly high. This is why we will reduce the model to:

- a structural/thermal model STM;
- an engineering model EM;
- and a flight model FM.

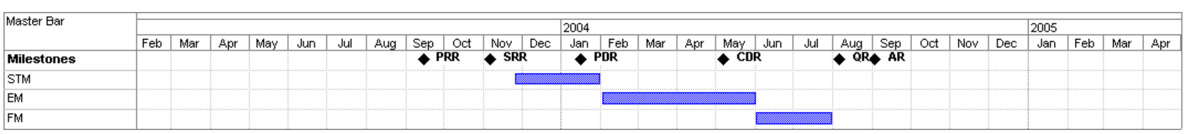

Fig. 5.2.5: Model Time Schedule

The structural/thermal model has the purpose to evaluate the physical design of the spacecraft and to enable tests based on mechanical loads and stress. The inner components might be replaced by dummy boards and components.

The more sophisticated engineering model consists of the refined structure and the inner parts, which beforehand had undergone some intensive tests on functionality under certain conditions. The qualification loads are applied to this model as well.

The flight model will be the finished spacecraft. It will be exposed to the required tests explained by the CubeSat documents. It becomes necessary to build this flight model because we cannot envisage a reuse of equipment that was exposed to qualification loads.

4. Risk Analysis

As there is no project that cannot go wrong, the detection and assessment of possible risks becomes not only important but also very essential for a realistic chance to have success. The philosophy needs to be applied by all participants at all levels and all phases of the project lifetime. The usual procedure of risk management is described in figure 5.2.5.

Whereas in the earlier phases the risk management is done somewhat automatically, the later phases will need to implement a routine for that. In the actual study results the risk management decisions are not mentioned separately but have had great influence on the design. In the commencing investigations, the most common way to reduce the possibility of risks is the application of redundancy to the subsystems. For example we decided to use two kill switches, in case one would jam. As this would cause the whole satellite to not work at all, a second switch reduces this single point failure probability to a minimum.

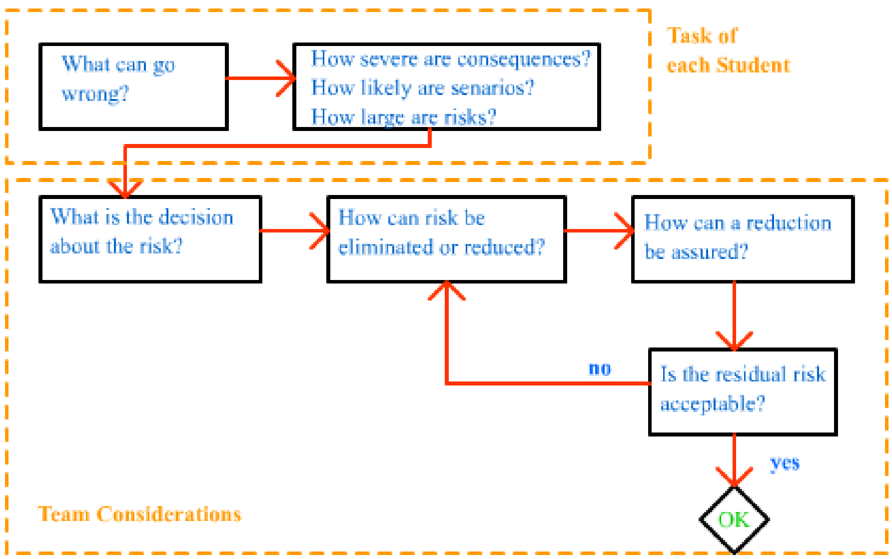

Fig. 5.2.5: Risk Management Process

As said we will have to comprehensively document decisions on risks in the further stages of the project. A template for risk analysis will be distributed to each participant. As most possible subsystem failures become apparent as soon as a deeper research is done in its area, every student has a great responsibility to properly document those threats and communicate them to the team for further reflection.

5. Cost Analysis and Support

In this last section we want to argue on the project's assumed costs for the purchase of components and material, the fabrication and refinement of parts and other expenses. Then we will look into the current and prospective sources for both, the financial and the educational support.

5.1 Costs Budget

The cost analysis is based on the expected expenses. Up to now there were no costs for the first phase since the work consisted of research and documentation only, prepared by students. From the study results described in this document we could estimate on the costs for the following stage, phase B. However, an estimation of the real costs is hard to do, because important factors are not known yet. For example, the manufacturing and test of the structural model will consume both, material costs and man-hours. The costs for the latter one strongly depend on the FH Aachen facilities and the co-operation with the personal. Therefore the cost analysis plan is not implemented at this report. It is maintained separately by the project management.

5.2 Support

As said, the support is understandably twofold. We will of course need financial support in order to compensate for the previous mentioned expenses. But another support becomes equally important, in particular for the success of the project, which is the educational or knowledge support respectively.

5.2.1 Funding Programs

The programs listed here are selected for a certain criteria. This is that those programs aim on supporting theoretical and practical research on present and new technologies in an educational non-commercial context. Surely the entire results will be offered to the supporters and can then be used for other purposes as well. But again, the Compass-1 project itself has no pecuniary interest.

Table 5.5.1: Funding Programs

5.2.2 Educational Support

As well as the inevitable financial support we will also need contacts to people that can help us out with some answers. Apart from the profound human knowledge resources we have at our university, there exists good contacts to other universities worldwide. In addition the student's contacts to person in the industry can become very vital to the project. Another source of possible knowledge support are the other students we have met at congresses and symposia. There is a list kept by the management that contains all those contacts but due to reasons of confidentially we will not list the persons her without permission.

6. Conclusion

This last chapter discussed the implementation of the necessary work to proceed with this project. This implementation encompasses the aspects of performance, schedule and cost.

To proceed with the next phase, for each subsystem a student was nominated as its supervisor, being the person in charge for internal and external enquiries. This will facilitate the communication and gives the students more independence and responsibility. In addition it will make it easier to get this work accredited by the university, when declaring it as a study work or even diploma work, dependent on its scope.

References

[1] www.ecss.nl