Delft University of Technology

Laser Swarm

Baseline Report

Design Synthesis Exercise

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Abstract

In February 2010 the ICESat mission ended after 7 years of measuring ice sheet mass balance, cloud and aerosol heights, as well as land topography and vegetation characteristics, using a space based Light Detection And Ranging (LiDAR). ICESat followed only one of the possible approaches for LiDAR, namely the use of a high energy laser and a large receiver telescope. The other approach is using a high frequency low energy laser and a single photon detector. The advantage of the latter approach is that it has a much lower mass, but it is unsure if even a single photon per pulse reaches the receiver. One possible solution could be the use a swarm of satellites around the emitter, each equipped with a single photon detector. However, the technical feasibility of this concept has not yet been proved.

This baseline report provides an overview of the initial look into this concept. This document contains the requirements analysis, functional breakdowns, risk assessments and initial design options. Preliminary business assessment is also conducted at this stage. It provides the basis for the trade-off made later in the project to find the most feasible system making use of this concept.

Contents

List of Acronyms

ADCS Attitude Determination and Control Subsystem

CMG Control Moment Gyro

DET Direct-energy-transfer

EOL End-of-life

EPS Electrical Power System

FBS Functional Breakdown Structure

FFD Functional Flow Diagram

GLAS Geoscience Laser Altimeter System

LiDAR Light Detection And Ranging

MNS Mission Need Statement

OM Operations and Management

PDS Power distribution system

PPT Peak-power tracker

RAMS Reliability, Availability, Maintainability and Safety

RDTE Research, Development, Test and Evaluation

TFU Theoretical First Unit

Chapter 1

Introduction

In February 2010 the ICESat mission ended after 7 years of for measuring ice sheet mass balance, cloud and aerosol heights, as well as land topography and vegetation characteristics. To do all this, ICESat had only one instrument on board: a space based LiDAR, allowing an unprecedented 3D view of the Earth's surface and atmosphere. However the laser lifetimes were severely limited because of a manufacturing error in one of the laser components. But ICESat followed only one of the possible approaches for LiDAR, namely the use of a high energy laser and a large receiver telescope, the other approach is using a high frequency low energy laser and a single photon detector. The advantage of the latter approach is that it has a much lower mass, but it is unsure if even a single photon per pulse reaches the receiver. One possible solution could be the use a swarm of satellites around the emitter, each equipped with a single photon detector, however the technical feasibility of this concept has not yet been proved.

This is the baseline report on the technical feasibility of this approach to achieve one or more applications of ICESat, it will mainly go into depth on the requirements, technical risks and define the initial design options. This way it will be the basis for the technical trade off to be made,which specifically requires the in depth understanding of the subjects treated in this report. Although decisions regarding the requirements can already be made, all decisions between technical concepts have to be postponed until the mid term report.

In chapter 2 the functions and requirements are described, in chapter 3 budget breakdowns in several areas are treated, in chapter 4 the technical risks are investigated, in chapter 5 the different design options are presented, in chapter 6 the sustainable strategy development is discussed, in chapter 7 the return on investment and operational profit are discussed and finally in chapter 8 the Reliability, Availability, Maintainability and Safety (RAMS) are studied.

Chapter 2

Functions and Requirements

2.1 Functional Flow Diagram

The Functional Flow Diagram (FFD) shows the functions the system needs to perform during its mission life. The schematic representation can be found in figure 2.1 on page 6.

The first thing that needs to happen, after being build, is that the satellites are put into their orbits and pointed towards Earth. After that the measurements can start; the emitter sends down laser pulses and notifies the receivers signals are sent. The receivers can adjust their attitude and then pick up reflected photons, turn them into an electric signal and inform the computer, which puts the data in a buffer. The data of the receivers can be either directly to a ground station or first send to the main satellite (the emitter) and then to the ground. The data on the ground can be split up made into data packages, which can be distributed to research institutes and other interested parties. With those datasets the terrain model can be produced. At the End-of-life (EOL) of the mission the satellites are decommissioned to make room for other satellites.

2.2 Functional Breakdown Structure

The Functional Breakdown Structure (FBS) shows the functions the system needs to perform broken up in subtasks from other functions. The schematic representation can be found in figure 2.2 on page 8.

The main function for the system is to be able to produce a terrain model, which is the reason to make the measurements. To be able to produce the terrain model the measurements need to be made, the data needs to be transferred to the ground and the data needs to be analysed. Because the mission needs to be sustainable the satellites are decommissioned at the end of their life, so they will not pose a threat to other satellites in the same orbit. The making of measurements depends on laser pulses to be send, returning photons to be detected and for the emitter and receivers to be in orbit with the instruments calibrated. For the data to be transmitted to Earth the antenna needs to be pointed to the ground station and the data package has to be transmitted. Data analysis depends on the data to be received on the ground and the distribution between the research institutes.

To have the satellite send out laser pulses the satellite needs to point down (nadir-pointing) and the emitter sends the pulses. The receiving of photons depends on the receiver being pointed at the ground target and

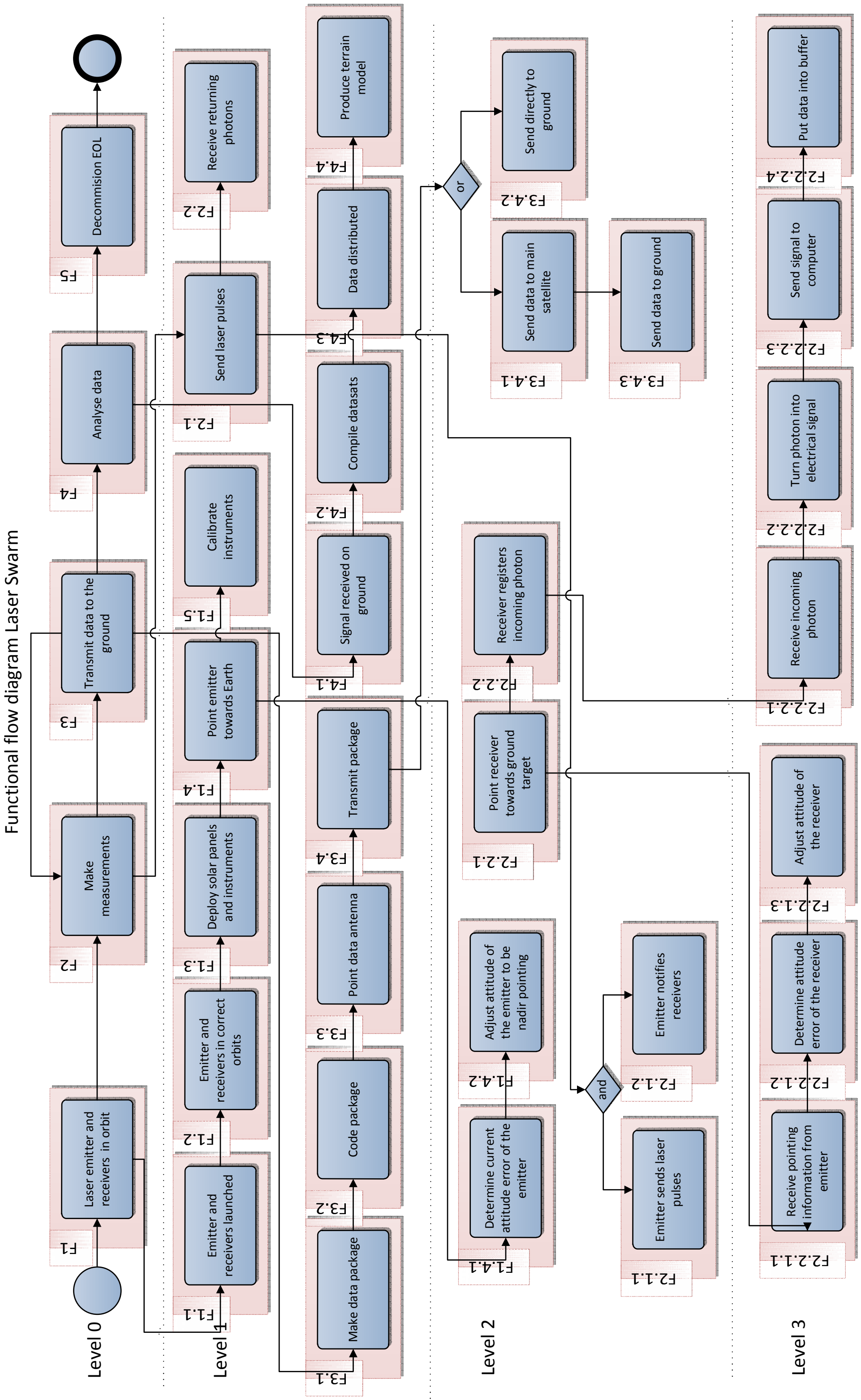

Figure 2.1: Functional flow diagram of the Laser Swarm mission

the receiver is able to register incoming photons. For the data package to be transmitted, data is put into a buffer, a data package is made, the package is code and either the receiver sends the data to the ground directly or to the main satellite, which in turn forwards it to Earth. When the emitter and receiver are in orbit, the emitter and receivers have been launched and they need to be on the correct orbits. It is important to have the solar panels and instruments to be deployed before they can be calibrated.

Determining the attitude error of the emitter and adjusting the attitude accordingly are required to point the emitter towards Earth. When an incoming photon is received, it is transformed into an electrical signal and the signal is sent to the on-board computer the photon is registered. For the receiver to be pointed at the ground target, the emitter needs to notify the receiver it has sent pulses, the receiver needs to receive the message, the attitude error of the receiver needs to be determined and the attitude should be adapted accordingly.

2.3 Requirement Discovery Tree

The requirement discovery process begins with the rephrasing of the mission need statement. From there, the top level requirements and their derivatives can be analysed.

2.3.1 Mission Need Statement

Demonstrate that a satellite constellation, consisting of a single emitter and several receivers, will perform superior (in terms of cost and lifetime) to existing spaceborne laser altimetry systems.

2.3.2 Requirement Discovery

From the Mission Need Statement (MNS) in section 2.3.1, it possible to deduce the top level requirements of this project. They are as follows:

- Cost budget below existing spaceborne laser altimetry systems.
- Lifetime above existing spaceborne laser altimetry systems.

Furthermore, several more requirements were provided by the principle tutor:

- Mass budget below or equal to existing spaceborne laser altimetry systems.
- No scanner may be used.

The last requirement is mainly considered as a pure constraint. The constellation should be designed as a collection of pointing devices.

The other three top requirements have been put in respective Requirement Discovery Trees (RDT) in appendix A. The following sections contain a brief discussion of each of these breakdowns.

Cost Budget Requirement

The cost budget requirement is mainly based on the analysis of the costs of current laser altimetry systems. As a reference point, the estimated budget of the ICESat system was taken: around $200m[3]. From hereon, the cost requirement was broken down into three main parts: payload, bus and *wraps*.

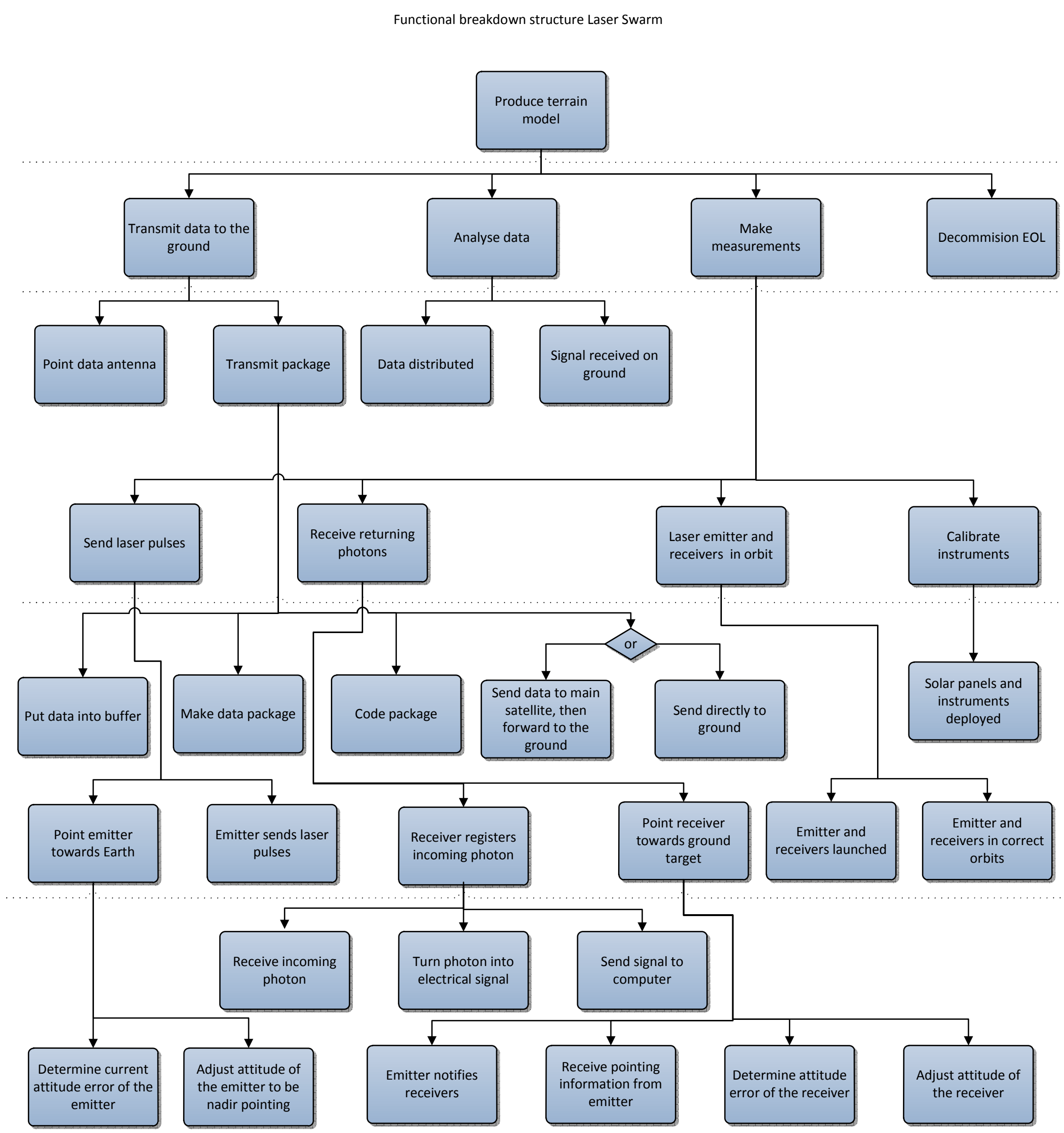

Figure 2.2: Functional breakdown structure of the Laser Swarm mission

The payload defines the design requirements for the emitter and the receivers. These are then further broken down into smaller components.

The bus requirements are those imposed on different spacecraft subsystems, excluding the payload. Only those systems that fall under the scope of the feasibility study are examined. Spacecraft structures and thermal control are taken to have a standard budget percentage and are not elaborated. Spacecraft power, data handling and Attitude Determination and Control Subsystem (ADCS) are considered to be critical design parts, thus have their requirements listed to maximum detail.

The final section - *wraps*, contains non-physical factors, such as system engineering, management and product testing. Since *wraps* typically account for close to 30% of the total budget[8], it was imperative that these systems would be accounted for, yet their design was assumed to be similar to the design of current laser altimetry systems.

Mass Budget Requirement

The mass budget is also a very important requirement. In order to keep total mass to a minimum (to ensure a cheap and unified launch), all critical subsystems and the payload have to be examined. In this sense, the requirement discovery tree for the mass budget looks very similar to that of the cost requirement. This is because all these design choices effect both factors. Some preliminary dry mass percentages (based on statistical data[8]) have been added to the tree to illustrate a primitive order of importance.

Lifetime Requirement

The lifetime requirement is quite crucial. From the experience of ICESat it is apparent that payload quality (especially that of the laser) plays a pivotal role. The ICESat mission provided the satellite with three lasers in the Geoscience Laser Altimeter System (GLAS), the first of which stopped emitting pulses on operating day 37[1]. The Anomaly Review Board has determined that it was the manufacturing flaws in the laser diode arrays that had led to unexpected behavior of the emitter[1].

It is therefore required to ensure component quality and reliability in order for the mission to succeed.

Furthermore, in terms of lifetime, consideration is given to the power generation. Power source degradation will have to be carefully looked at, as the instrument will not fulfill its requirements without sufficient power supply.

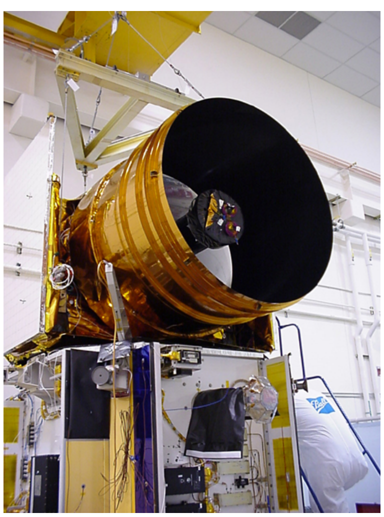

Figure 2.3: GLAS installed on the ICESat. *http://icesat.gsfc.nasa.gov/*

Chapter 3

Budget Breakdown

3.1 Cost Budget Breakdown

For a preliminary estimation of cost breakdown it is necessary to realize the different stages of the life-cycle costs. These are the costs of the entire space mission from the first stages of planning until end of life.

Figure 3.1 shows a typical life-cycle for a space mission. The Research, Development, Test and Evaluation (RDTE) stage includes the planning, development and testing of all prototypes and qualification units but does not include the technology development for different subsystems. In the case of the laser swarm this is largely dependent on the single emitter and one receiver units. This stage is also mostly consistent of non-recurring costs. The production stage consists of actual manufacture of the physical satellites. The cost estimation in this stage is based on the Theoretical First Unit (TFU). This is done because it is assumed that the first unit (in the case of the Laser Swarm, that would be the emitter and one receiver) would be the most expensive to produce. The rest of the swarm constellation satellite costs are calculated by taking a theoretical learning curve[8].

The final stage - Operations and Management (OM) is self explanatory. This phase can be very expensive for large constellations. The ground segment is the prevailing factor.

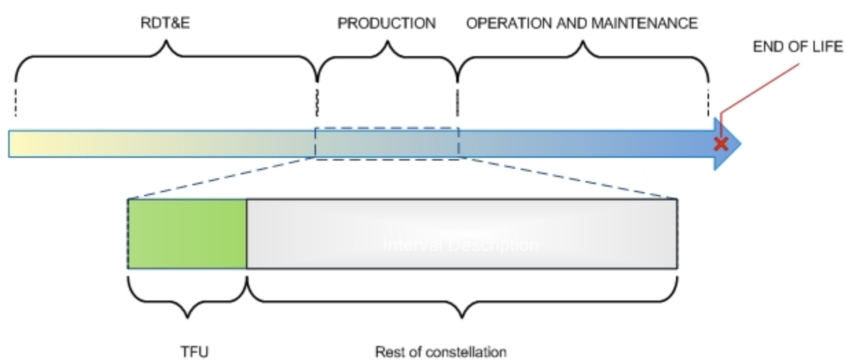

Figure 3.1: Life-Cycle

The cost breakdown can be found in figure 3.2 on page 13. In this diagram, the cost is broken down into segments. This way it is easier to follow cost distribution down to all subsystems.

Because of the nature of the system that is being designed - as a collection of satellites, it is very hard to make an accurate prediction of costs before critical decisions have been made. Most subsystem costs are mass driven, so design options create a large margin of error at early estimations. Furthermore, due to high cost of actual production of individual satellites, a swarm is hard to predict without a rough idea of the number of space platforms.

Table 3.1 on page 12 contains percentage estimations of all aspects covered in figure 3.2.

SEGMENT	% OF SEGMENT	% OF TOTAL
Space Segment		
RDTE	34	44
Payload	34	45
Bus	34	46
Structure	34	47
Thermal	34	48
EPS	34	49
Communications	34	50
ADCS	34	51
IAT	34	52
Program Level	34	53
GSE	34	54
LOOS	0	0
Software	34	56
Production	34	57
TFU	34	58
Payload	34	59
Bus	34	60
Structures	34	61
Thermal	34	62
EPS	34	63
Communications	34	64
ADCS	34	65
IAT	34	66
Program Level	34	67
GSE	34	68
LOOS	34	69
Swarm	34	70
Launch Segment	34	71
Launcher	34	72
Ground Segment	34	73
First Ground Station	34	74
Consecutive Ground Stations	34	75
Software	34	76
Operations and Maintenance	34	77
Operations and support of ground stations	34	78

Table 3.1: Cost Breakdown

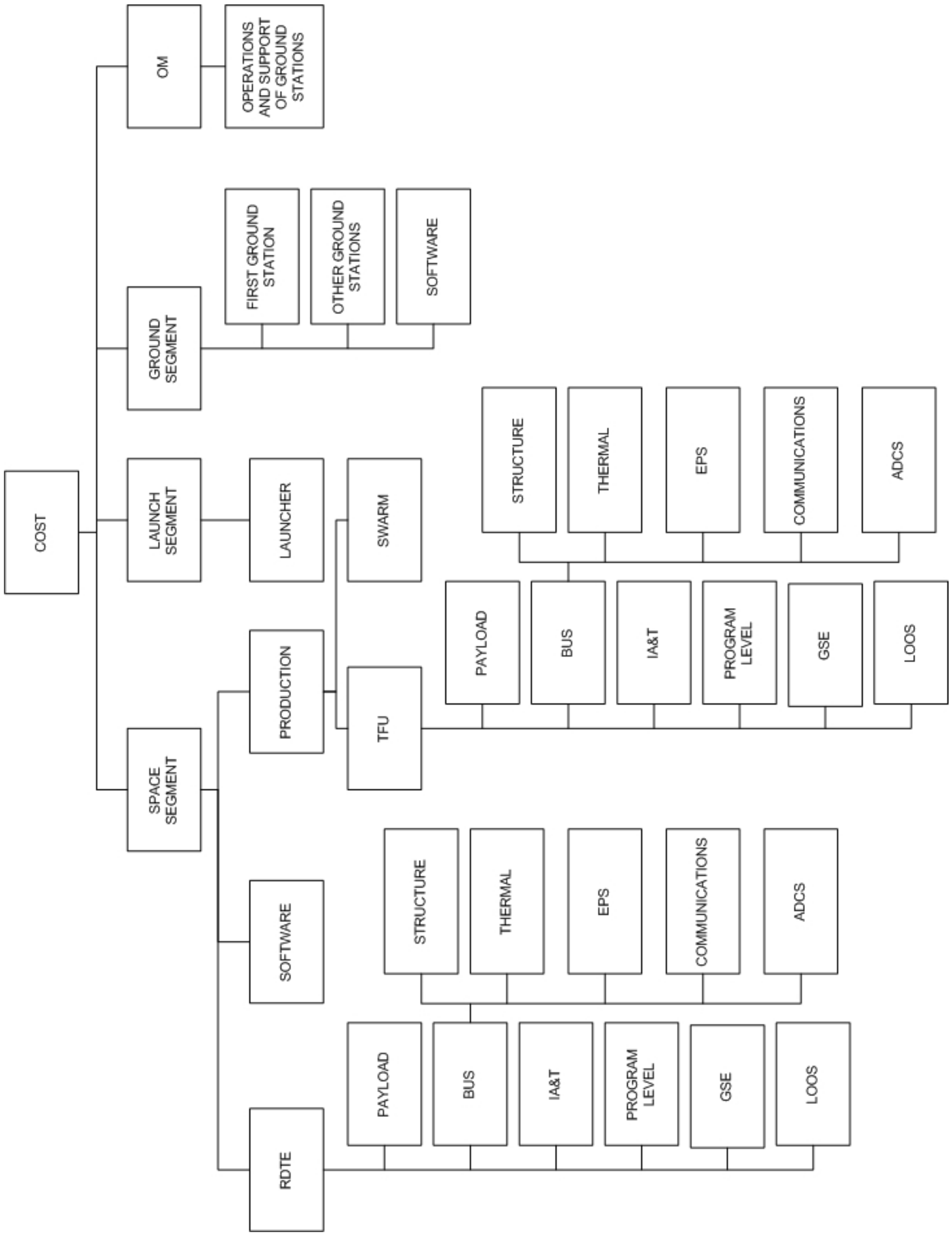

Figure 3.2: Cost breakdown

Chapter 4

Technical Risk Assessment

The main objective of the technical risk assessment is to determine the reliability compared to the possible (functional or financial) consequences per specific event. To be able to determine any of these reliabilities, a definition of reliability should be stated. In this case, reliability is formulated as: "The probability that a specific (part of a) subsystem will function without endangering the level zero requirements over the expected lifetime".

Next to formulating the definition of reliability, it should be noted that the determined reliabilities in this section are relative reliabilities, i.e. the probability that a particular subsystem outperforms another subsystem with the same core function in terms of reliability. Hence, no absolute values of reliability are determined in this section. The relative reliabilities allow for comparison material during the trade-off between multiple design options. The risk assessment analysis is divided into four main sections:

I Ground segment (before vehicle leaves Earth's atmosphere)

II During mission

III Measurement protocol

IV Post-mission

The possible events, with their respective reliability, are outlined in these sections and after that the expected consequences are shortly explained.

4.1 Ground segment

A ***Financial***

A1. Insufficient funds or low market-demand
The approximate costs are determined in the cost budget. The mission data and the final results can be very interesting for a vast number of commercial companies and research or educational facilities. Every space mission is created for at least one specific (user-demanded) requirement. This third party is responsible for covering the cost. Since the space mission is developed after this request is set, the probability that there will be insufficient funds is low (especially when more than one company can be considered as the user). However, the consequence can be severe if the funds are not enough to start or continue the development.

B ***Technological readiness***

B1. Technology for level zero requirements are not available
If the technology for measuring, detecting or processing the level zero requirements is not available at present, the requirements can't be met and alternatives should be revised (or the mission should be terminated). In our case, the technical readiness level of the payload is relatively high and hence has a high reliability. If, however, the specific payload would have a low technical readiness level, the mission should be terminated or delayed. Therefore, it has important consequences to the mission.

C ***Launch***

C1. Total launch failure
Total failure indicates complete failure of launch vehicle and all laser swarm constellation components. Needless to say, the reliability is relatively low; however, the consequences of this event are catastrophic.

C2. Partial launch failure
Partial launch failure indicates non-complete failure of laser swarm constellation components, i.e. some of the satellites (emitters or the receiver) can still perform core tasks. Considering historical launch data, the reliability that no partial failure will occur during launch is relatively high. The consequence can be very different, depending on which part (or what fraction) of the constellation can't perform its core task. If one of the receivers will be destroyed, the level zero requirements might still be achievable. However, if the emitter is (partially) destroyed, the mission will surely be endangered.

C3. Delayed vehicle launch
Delaying the vehicle launch isn't particularly a problem from the technological side of the mission; however, it will affect the financial situation. Next to the fact that the data and results are delayed, extra costs will be imposed due to an increase in launch vehicle pad costs, extra personnel costs and others. The reliability of this event is actual not that high, since it is dependent on a lot of criteria like third-party companies, the weather, and atmospheric properties. The consequences are mainly financial.

4.2 During mission

D ***Orbit accuracy***

D1. One or more satellites are in a wrong orbit.
After launch and orbit initializing, it is possible one or more of the satellites are in a wrong orbit. If this deviation from the desired orbit is relatively small, the ADCS subsystem should be able to cope with this minor error and set the orbit in its right orbit. If the altitude error is large, major altitude changes should be imposed. Assuming a low to moderate error, the consequence is not really severe if the ADCS system is working properly. The chance of actually putting a satellite in the wrong orbit is also pretty small.

In the next section, the reliability of the ADCS subsystems is compared. Assuming a non-hybrid spacecraft, i.e. a spacecraft which uses one of the ADCS subsystems considered in the design option tree, the consequences of failure are equal for all subsystems and thus shall not be inspected individually. The consequences are severe considering not only the loss of pointing accuracy, but also a decrease in vehicle stability and the total failure of controllable altitude control.

E ***Altitude and control determination***

E1.Passive systems

E1a. Gravity-gradient.
The gravity gradient technique is only dependent on gravity fluctuation in nadir direction, which makes it relatively reliable.

E1b. Passive magnetic.

The passive magnetic technique is only dependent on magnetic fluctuation near a celestial body. Since this is the only dependency, the technique is really reliable.

E1c. Zero momentum.

The zero-momentum technique uses a momentum-bias wheel, initially with no angular velocity. Like with all mechanical systems, the presence of (angular) motion will decrease the reliability (due to possible mechanical failure like static failure, fatigue etc.). The reliability however is pretty high, but lower relative to passive magnetic and gravity gradient.

E1d. Momentum-bias wheel.

The momentum-bias wheel technique uses, like the name already predicts, momentum wheels to dump and correct torques. In that sense it has the same reliability as the zero-momentum subsystem. However, since these momentum wheels are constantly spinning, the reliability is slightly lower than the previous mentioned subsystem.

E1e. Spin stabilization.

Spin stabilization can be achieved using rotation about one principal axis (single-spin) or two principal axes (dual-spin). Next to the fact that due to external torques (debris collision, aerodynamic drag) the spacecraft can become unstable, i.e. this subsystem has more dependencies, making it relatively unreliable.

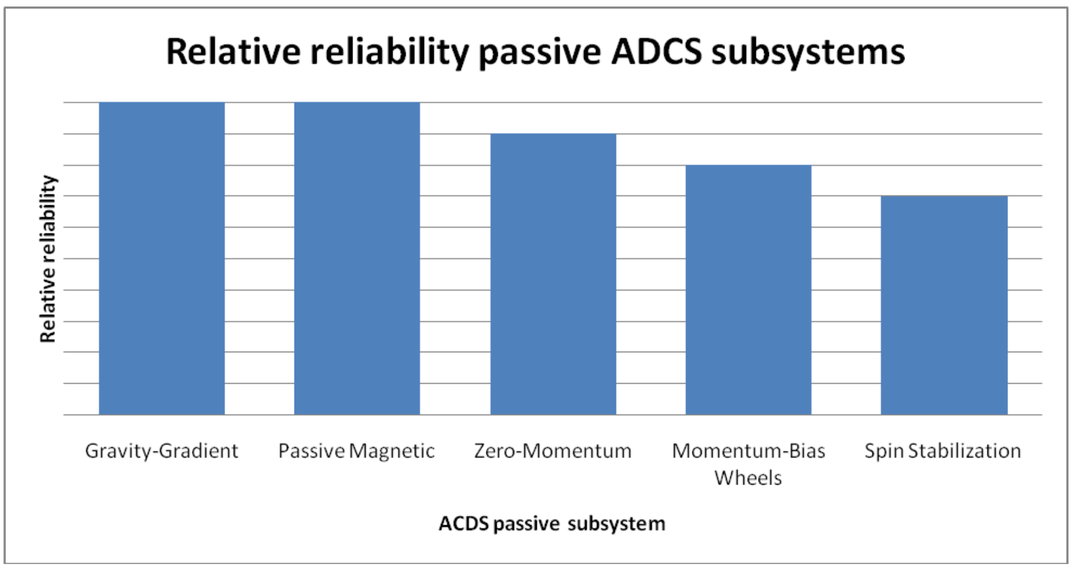

Figure 4.1: Relative reliability passive ADCS subsystems.

F ***Active systems***

F1. Actuator

F1a. Thrusters (hot and cold gas).

Multiple-axes thruster systems are very efficient ways for determination and controlling altitude and stability. The system is dependent on fuel consumption, combustion and mechanical properties. Each of these dependencies decreases the reliability.

F1b. Reaction and momentum wheels.

Mechanical reliability is an import aspect for using active reaction and momentum wheels.

F1c. Control Moment gyros.
A control-moment-gyro system consists of a spinning rotor and one or more motorized gimbals that tilt the rotor's angular momentum. Mechanical reliability is an import aspect for using this. Since it is also dependent on the motorized gimbals, the reliability is slightly lower than the reaction and momentum wheels.

F1d. Magnetic torquers
The magnetic torquers interact with the Earth's magnetic field, creating compensating torques to induce stability. Reliability is high due to the fact that the magnetic field is known and the system is dependent on a low number of parameters.

F2. Sensors
Assuming a high technical readiness level of the sensors, the reliability is considered high. Also, the consequences of failure are high as the continuation of the mission may be impaired.

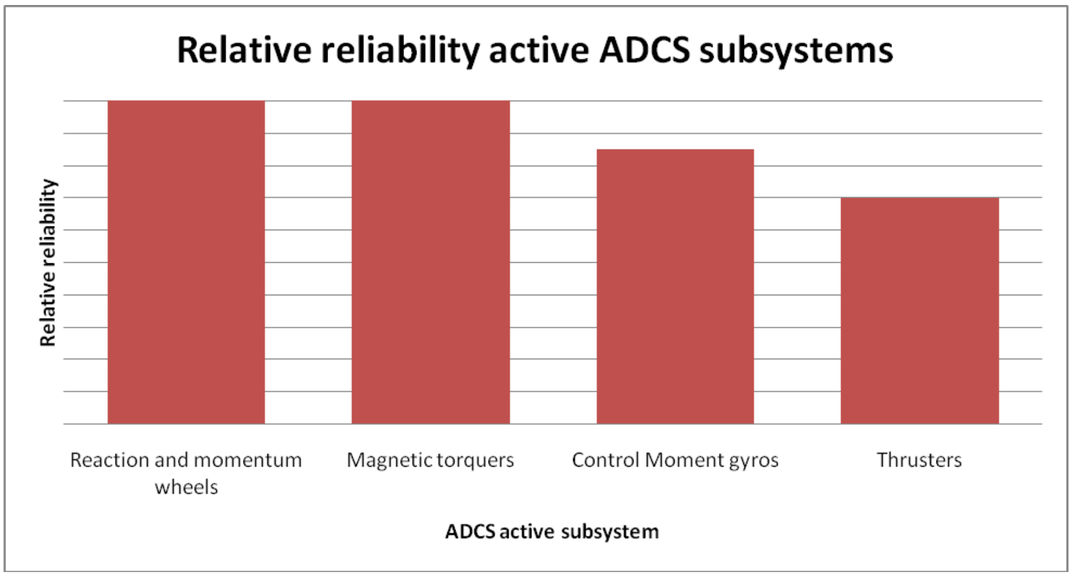

Figure 4.2: Relative reliability Active ADCS subsystems.

G ***Electric Power System (EPS)***

G1. Solar Panels

G1a. Solar panel deflection error or mechanical failure.
During launch, the solar panels are retracted to achieve the lowest volume as possible. During the initializing of the mission (assuming the spacecraft is in the right orbit), the solar panels need to be deflected. Errors can occur due to mechanical reasons or external disturbances. The probability of this is pretty low. The consequence can be however that the effective solar panel is decreased and hence a decrease in electrical power will occur. This makes the consequences pretty severe. Any other mechanical failure (broken joints, internal PN-junction failure, maybe even losing an entire solar panel) will have severe consequences as well.

G1b. Solar panel characteristics reliability (degradation).
Degradation of solar panels should always be considered during mission development. Since this (should be) known upfront, the consequences are relatively low. The probability of this actually happening is nearly 100 percentage.

G1c. Severe degradation (due to external phenomenon)
Atomic oxygen, hazardous radiation, debris collision and other external factors can influence the performance of the solar panels. Since these are not known from the start, it's difficult to cope with them. The probability of this happening is pretty small, but will have pretty severe consequences.

G2. Batteries

G2a. Initial internal failure
Considering a high level of technical readiness level, the internal reliability is high. The consequences do alter the functional capacity of the mission, since no energy can be stored if the energy capacity system would completely fail, meaning that during eclipse no energy should be used.

G2b. Decrease in capacity
Considering a high level of technical readiness level, the reliability is high. Consequences are low, because they are known and should be part of the mission analyses.

4.3 Measurement protocol

Since actual measurements are an important level zero requirement, the consequence of the items in the measurement protocol are all really severe. Unless stated otherwise, the consequences in the following section can thus be stated in this way.

Measurement

H ***Emitter***

H1. Laser pulses can't be sent/ no photon generation
Considering a high level of technical readiness level, the reliability is high.

H2. Pointing towards nadir
This is dependent on ADCS risks.

H3. Laser notifies receiver (time adjustment)
Considering a high level of technical readiness level, the reliability is high.

H4. Laser degradation
Laser degradation is dependent on multiple parameters: thermal properties, input power interval, external factors and internal mechanical errors (manufacturing or design errors). However, due to extensive research and development concerning laser technology, the probability of severe laser degradation within the lifetime is relatively low.

I ***Receiver***

I1. Point towards target This is dependent on ADCS risks. *I2. Receive and detect photons*
Considering multiple satellite receivers, the probability of total failure to receive and detect photons using advanced single photon receiving devices (like SILAT, Glass or photon-receiving modules) is negligible.
I3. Turn photon into electrical signal
Considering a high level of technical readiness level, the reliability is high.

Communication

J ***Inter satellite communication***

J1. Determine relative position receiver and emitter
Considering a high level of technical readiness level, the reliability is high.

J2. Time differences
Considering a high level of technical readiness level, the reliability is high.

K ***Data handling***

K1. Store data/ make data package
Considering a high level of technical readiness level, the reliability is high.

K2. Transmit package
Considering a high level of technical readiness level and relative low-tech technology, the reliability is high.

K3. Interpreted results
Historical data comparisons for the interpretation of altimetry missions are sufficient, but not elaborate. However, the reliability is still pretty high.

K4. Reproduce terrain model
Historical data comparisons for the interpretation of altimetry missions are sufficient, but not elaborate. However, the reliability is still pretty high.

L ***Housekeeping/ Ground communication***

L1. Housekeeping data from ground station to satellite
Considering a high level of technical readiness level, the reliability is high.

L2. Adjusting space segment characteristics
Considering a high level of technical readiness level, the reliability is high.

M ***Structural***

M1. Joints
Considering a high level of technical readiness level, the reliability is high.

M2. Connection points
Considering a high level of technical readiness level, the reliability is high.

M3. Thermal limits
Thermal limits will alter the characteristics of pretty much all subsystems. However, thermal will be excluded in this analysis.

M4. Fatigue
High-cycle loading is usually not present (except for momentum wheels) and should therefore only play a minor role. The probability is low. The consequences are medium to high if high-cycle loading will lead to fatigue and hence partial failure.

M5. Electrical overlay failure
See EPS.

N ***External***

N1. Debris collision
Unknown external phenomenon. The probability of this event causing total failure, which is a function of multiple parameters like orbit and celestial position, is low to medium.

N2. Dangerous radiation
Unknown external phenomenon. The probability of this event causing total failure, which is a function of multiple parameters like orbit and celestial position, is low to medium.

N3. Charged particles collision
Unknown external phenomenon. The probability of this event causing total failure, which is a function of multiple parameters like orbit and celestial position, is low to medium.

N5. Politics or international influence
Political decisions or international influences can alter the space mission considerably. With altimetry

missions, the probability of these external influences causing a delayed or complete stop of the mission is negligible. The consequences could be very severe though.

N6. Classified information (army)
Some parts of the measurement are classified information, for example military ground stations or governmental classified areas. The government and/ or military can pressure the vehicle engineers to keep certain information classified. However, if this is the case, most of the measurements still can be taken and analyzed. So where the probability is medium, the consequences are very low.

4.4 Post-mission

O ***Satellite decommission***

O1. Decommission LEO
At the end of life the satellites have to be decommissioned to allow new mission to take their place. To decommission satellites in low Earth orbit one could just wait a couple of years and air drag will cause the satellites orbit to degrade to the point they burn up in the atmosphere, so the consequences are low. However, it is desirable to have the satellites burn up faster, so as to remove the risks like satellite collision. The probability to no longer be able to eject the satellite from orbit depends on whether or not its propulsion is still working; as such this probability is low.

O2. Decommission GEO
Satellites in GEO can not be placed in an orbit that will cause them to burn up in the atmosphere because they will cross paths with too many other satellites. Because risk is so high these satellites are instead decommissioned by ejected them from orbit further into space. This way, new GEO satellites can take the place of the swarm. If this in the dead satellites will continue orbiting the Earth, wasting space that can be used by other satellites, as a result the consequence of failure is high. Being able to reposition a satellite depends on the ADCS systems, as such the probability of this event is low.

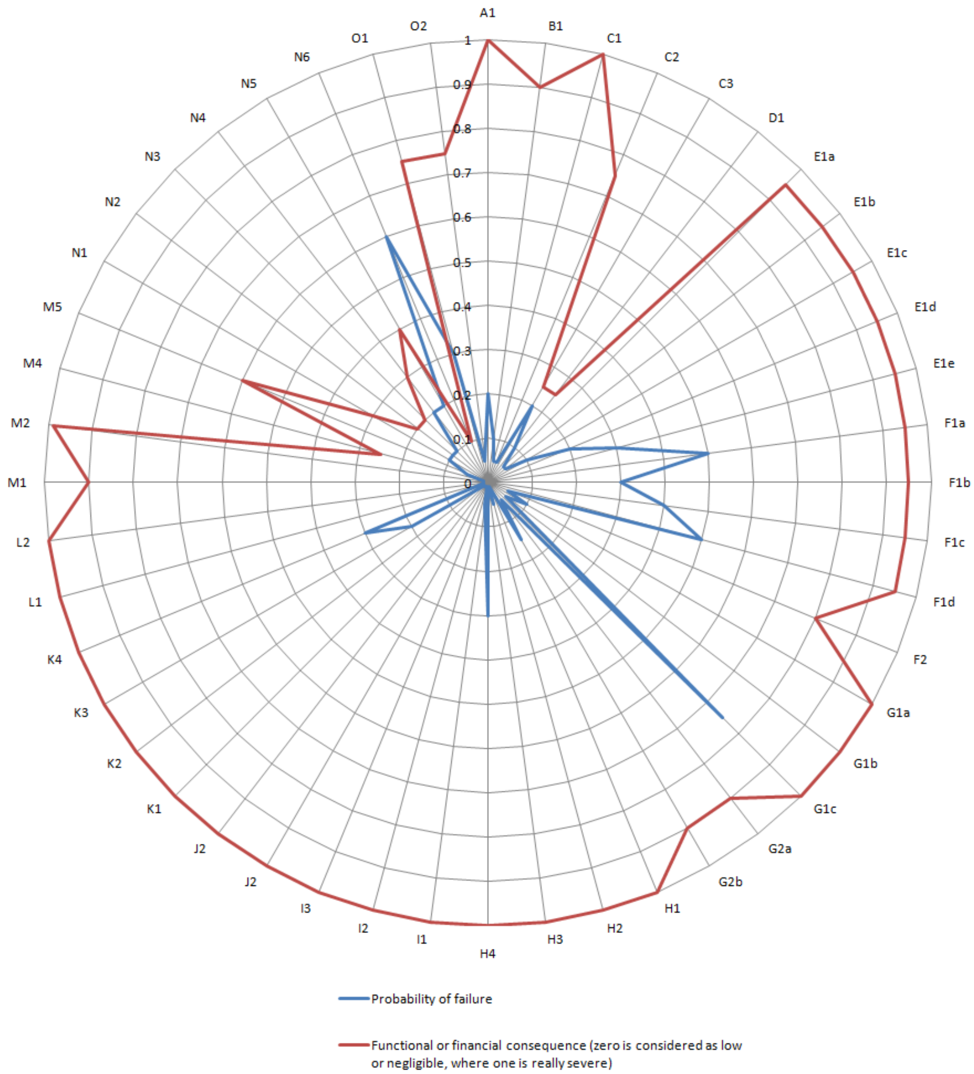

Figure 4.3: Technical Risk Map

Chapter 5

Design Options

5.1 ADCS

The design of the ADCS is twofold. On one side there is the attitude determination, on the other the attitude control. With the attitude determination the state (attitude and attitude rate) of the satellite is established, if the state of the satellite is not as it should be the attitude control part adapts the attitude. It is also possible to have a passive system by using the gravity gradient of the Earth, using the Earth's magnetic field or by having a spinning spacecraft. In all cases the satellite is only stable along two axis. A design option tree can be found in figure 5.1 on page 24.

5.1.1 Attitude determination

For attitude determination a number of different sensors could be used. First of all there are inertial measurement units like gyroscopes and accelerometers. These systems measure the attitude deviation from a set point in time. Then there are Sun sensors, which use the position and angle towards the Sun, Earth scanners scan the Earth's limb, Star trackers use the positions of known stars to determine the attitude of the satellite. A recent technique to determine the attitude is using accurate relative **GPS!** (**GPS!**) measurements with multiple antennas on the satellite body.

5.1.2 Attitude control

The attitude control is able to change the attitude of the satellite. Basically there are four kinds of active systems to do this: using thrusters, momentum wheels magnetic torques and Control Moment Gyros (CMGs). Thrusters exert gasses and momentum wheels spin up to to give a momentum to the satellite. In magnetic torquers a current through a coil produces a Lorentz force, using the magnetic field of the Earth. Because there is no friction, an equal amount of momentum has to be added in the other direction as well to stabilise the attitude again. CMGs have a constant speed, but the angle in which the force vector is directed is adapted by a single or double set of gimbals. Passive means of attitude control include gravity gradient, spin stabilisation and passive magnetic. Due to the Earth's gravity field gravity gradient satellites will always tend to point towards Earth with their smallest moment of inertia axis. Spin stabilised spacecraft are rotating around one axis to stabilise along the other two. Passive magnetic stabilising uses the magnetic field of the Earth and permanent magnets for its stabilisation. Passive means of stabilisation are often only usable for stabilisation along two axis, if stabilisation or even control along three axis is required a different system is required.

5.2 Electrical Power Subsystem

The Electrical Power System (EPS) is divided in to four parts: the power source, the energy storage, the power regulation & control and the power distribution. These are also the four main branches in the EPS design options structure. Each of these will be considered individually in this section.

5.2.1 The power source

Launch vehicles use primarily batteries as power source, because launches are quite short and thus the batteries can be kept fairly small. For missions lasting from weeks to years however, batteries required would be too large to make a mission affordable. Typically, there are four types of power sources for longer missions: static power sources, dynamic power sources, fuel cells and photovoltaic solar cells.

Static power sources use a heat source for thermal-to-electric conversion. This conversion can be done by either a thermoelectric or a thermionic concept. The thermoelectric converter uses the fact that the radioactive source (typically plutonium-238 or uranium-235) has a slow rate of decay. Because of this, there exists a temperature gradient between the p-n junction of individual cells which is used to provide the desired direct current electrical output. The efficiency of such a system is about 5 tot 8%.

Thermionic energy conversion on the other hand, uses a hot electrode facing a cooler electrode to convert thermal energy to electrical. These electrodes are sealed in a chamber containing an ionized gas. The hotter electrode can be seen as the emitter: it emits electrons that flow across the inter-electrode gap towards the receiver (the cooler electrode). Once arrived, these electrons condense and return to the emitter through an electrical load connected externally between the two electrodes. Typical system efficiency is about 10 - 20%.

Dynamic power sources function somewhat differently. They also use a heat source (typically concentrated solar radiation, radio isotopes or a nuclear-fission reaction) to produce thermal energy but the conversion method to electrical power is different. The generated heat is used to heat up a fluid to drive an energy-conversion heat engine. This is done using a Stirling cycle (efficiency of 25-30%), a Rankine cycle (efficiency of 15-20%) or a Brayton cycle (efficiency of 20-35%).

Fuel cells are self-contained generators that convert the chemical energy of an oxidation into electrical power. They consist of two half-cells, each with an electrode and an electrolyte. The two half-cells may use the same electrolyte or they may use different ones. In the fuel cell, one half-cell gets oxidized, it loses electrons, and the other is reduced, it gains electrons. As the electrons flow from one half-cell to the other a difference in charge and thus an electric current is created. Fuel cells can be regenerative or not, unfortunately regenerative types have not been space-proven yet[7]. The efficiency of fuel cells can be as high as 80%, but will drop significantly at higher currents.

Photovoltaic solar cells are most commonly used in space. They convert incident solar radiation directly in to electrical power. They consist of a semiconductor with metal plates on the top and bottom. Part of the incident solar radiation gets absorbed and is transferred to the semiconductor. The energy excites electrons who are then free to move around. The metal plates move the electrons, which creates a current, to power different subsystems. An efficiency of 29% has been achieved [2] in the lab, but production efficiencies are around 22% [8]. Several options are available for the placing of the solar cells. The biggest difference is between solar panels and body-fixed solar cells. Body-fixed solar cells require a spinning satellite to be able to make optimal use of the cells. Solar panels however, can be pointed towards the Sun to have minimum cosine loss. The panels can be rigid or flexible, flexible panels being easier to transport but less efficient compared to rigid panels.

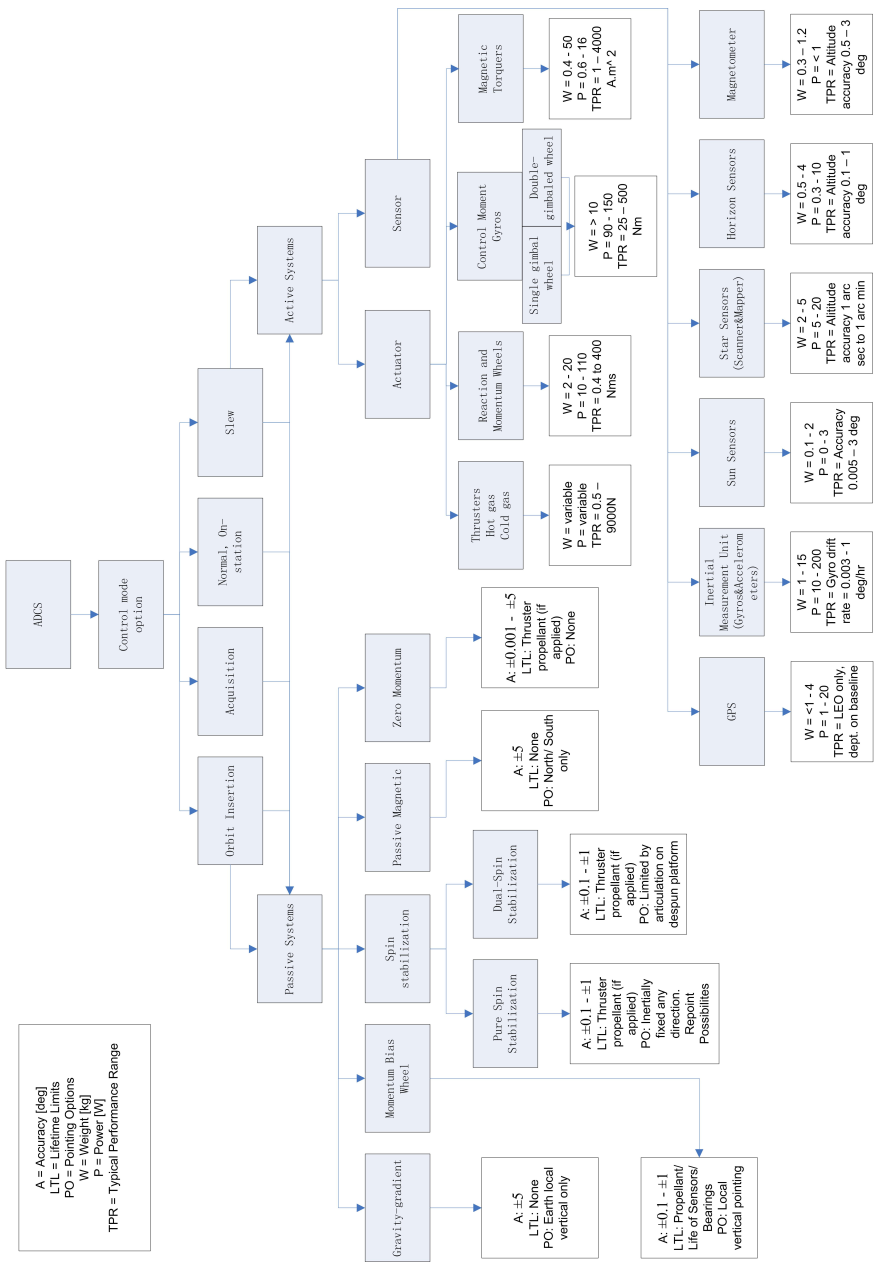

Figure 5.1: Design option tree for the ADCS

A comparative table for the different power sources can be found in table 11-35 on page 410 of [8].

5.2.2 Power storage

Power storage is the second subdivision of the EPS. They can either provide all the power for short missions (primary batteries), provide power during eclipse or they provide back-up power for longer mission(secondary batteries).

Batteries can be both a power source and a power storage system. The following design options apply for both uses.

Primary batteries usually are used for short-term missions, up to about one day. Sometimes they are also used for long-term mission for tasks that require small power usage like memory back-up for example. They have a high specific energy density, which makes them a good choice for short missions. The number of batteries and their corresponding weight and size required for longer mission makes them a bad candidate. The most typical battery types use silver zinc, lithium thionyl chloride, lithium sulfur dioxide, lithium monofluoride and thermal cells.

Secondary batteries are mostly used on missions which use photovoltaics as a power source. Here they provide power when the solar panels are eclipsed and at moments when power requirement peaks. To keep the secondary batteries from becoming too large, they are required to be rechargeable. Some common secondary batteries are: Nickel-Cadmium, Nickel-Hydrogen (both space qualified), Lithium-Ion and Sodium-Sulfur (both under development). The nickel-hydrogen batteries have three space-qualified design variants: individual, common and single pressure vessel.

The individual pressure vessel contains only one electrochemical cell within. Usually they are connected in series to obtain the desired voltage. The internal electrode stacks are connected in parallel. The only difference between the individual and common pressure vessel is that the common pressure vessel has two electrode stacks, connected in series. This means that there are only half as much pressure vessel and less pieces, resulting in a higher specific energy density. Finally, the single pressure vessel has multiple cells connected in series that share a common supply of hydrogen. The cell stacks are each contained in a container with its own electrolyte supply[8].

Lithium-Ion batteries offer a significant advantage over nickel based ones. According to [8], these types of batteries should be space qualified in the near future. At the moment, the effect of temperature on the performance is being researched. At the moment there are no components that are expected to be critically effected by temperature, resulting in a wide temperature range. Thus lithium-ion batteries may be a very interesting candidate for space applications in the near future[5].

5.2.3 Power regulation and control

The power regulation mainly concerns the bus regulation. The bus is the connection between power source and the different loads. It can be unregulated, quasi-regulated or fully regulated. The unregulated bus has its converters at the individual loads (see section 5.2.4). A quasi-regulated bus has a battery charge regulator and a fully regulated bus also has a battery discharge regulator. Batteries can be charged individually or in parallel. Batteries that are charged in parallel degrade faster. Because the current is not controlled, one battery could receive all the charge current. Eventually the batteries will balance out, but the battery life will be limited to about five years. To ensure a longer lifetime, they should be charged individually (for example by using a linear, charge-current-control design). A parallel charging system, however, is simpler and smaller than an individually charging system.

Because the optimum power source output and the bus input are different, a system has to be put in place to deal with this. An example of two possibilities will be given and explained for solar panels. There are two ways of doing this: with a Peak-power tracker (PPT) and with Direct-energy-transfer (DET). A PPT is a non-dissipative system: it exacts the exact power the satellite requires (up to the arrays's peak power). Every solar panel has a peak power point, which can be seen from the panel's IV-curve. So the panel produces the most power at a certain current and voltage. But when a battery is charging, it is charged at its own current and voltage. The result of this is that the battery will change the panel's current and voltage to its own, forcing the solar panel to under perform. A PPT uses a DC/DC converter to change the solar panel's output to the required battery input, thus letting the panel perform at its peak power point, increasing the efficiency. However, because the PPT is connected in series to the solar array, it uses about 4-7% of the total power.

A DET system uses a shunt regulator to shunt away the excess power from the subsystem, usually at the array, to avoid internal energy dissipation. Shunt regulators can keep the bus voltage at a predetermined voltage. These systems are extremely efficient and has a lot of advantages over a PPT system: lower mass, less parts and a higher total efficiency at EOL.

5.2.4 Power distribution

A Power distribution system (PDS) consists of the electrical load profile, the control options and fault protection. After the load profile has been determined, the first choice to make is the type of current of the distribution system. Mostly direct current is used because spacecraft generate direct current. An AC/DC converter would need more electronics, resulting in more parts and a higher mass. But systems working with alternating current can be single phase or multiple phases[6]. The main difference is that in a single phase system all voltages from the source vary in unison, whereas the different currents of the multi-phase system reach their peak value at different times. Each mode has its advantages and disadvantages. Single phase systems are simpler and thus cheaper but when higher loads are required multi-phase system are more useful. They can help to reduce vibrations, for example.

The PDS can either be centralized or decentralized. The decentralized option requires a converter at each individual load, resulting in an unregulated bus. The centralized option regulates power to all loads from the main bus. The advantage here is that the EPS does not have to be tailor-designed.

The fault protection mainly is about isolating failed loads. If it is not isolated, a short circuit can occur. This will draw excess power and stress the cables. The isolation is usually done with fuses. The different design options here are the fuse types and location. The short circuit can also be dealt with by using cables that have extra current carrying capabilities. These come in varying types and sizes. A last option is to foresee extra power storage capabilities to cope with a short circuit. Of course, the fault protection options are not limited to the use of just one of these. In practice, the three are used simultaneously.

Chapter 6

Approach with respect to sustainable development

6.1 Production and Logistics

Since a swarm of satellites is envisioned, most of which are likely to be identical, it is possible to produce them in series which is more efficient in terms of resources than one large satellites with a lot of unique components. This also implies that the number of different spare parts could be reduced. Smaller satellites could also use smaller facilities for production and testing.

Transportation can be split up in two pars: transportation to the launch site and the launch from the surface to its final orbit in space. On both occasions the system can again profit from its small size, if chosen to bring the satellite into orbit on different launches, they can piggyback on the rocket but also on the airplane to the launch site (assuming main payload is built on the same location).

Spreading the swarm over different launches has several advantages, first of all the amount of emissions is lower than in case of a dedicated launcher, secondly if the first satellite fails before the launch of the rest of the swarm, the others can be repaired and thus less resources are wasted.

6.2 Operations

Once in orbit, its influence on the Earth is very limited, the only real concern is the debris it leaves behind during launch and deployment, which can be dangerous to other satellites if the debris stays in orbit. However, the deployment mechanisms, which are solely responsible for the debris, are not included in this technical feasibility study. Later studies continuing on this feasibility study should keep an eye on it, since more satellites could mean more deployment mechanisms and hence more debris. One aspect that can be dealt with is the efficient use of resources, the swarm can be designed in such a way that if one of the satellites fails a replacement satellite can be sent and the other still working satellites can be reused.

6.3 End Of Life

Each satellite will be at the end of its life it can't perform its function anymore or if its onboard propellant is low. It is important that after the mission is over no satellites are removed from space and burn up in the atmosphere so that they don't pose any danger to other satellites. Final decommissioning of the swarm will be more complex than for a regular satellite, since every individual satellite has to be decommissioned separately.

Chapter 7

Return on Investment and Operational Profit

Chapter 8

R.A.M.S.

In this chapter the approach to the characteristics of the satellite system are described. In the upcoming sections all aspects are threaded separately.

8.1 Reliability

Compared to conventional laser altimetry missions, a swarm of receivers is used in this system. Spreading the receivers over a number of satellites reduces the risk of the mission failing. If one receiver satellite fails there will still be others to perform the task of detecting photons. Using a low-power laser emitter extends the lifetime of the mission compared to missions using a higher power laser, since low-power lasers generally wear out slower. Figure 8.1 on page 32 gives a overview of subsystem contributions to satellite failures after different time span from a non-parametric analysis. More detail reliability analysis can be found in chapter risk assessment.

8.2 Availability

The system is designed functional all the time to cover the entire Earth. To safe power it is also possible of switch off the laser to only make measurements in a designated area. Since the system use a swarm of receivers the system can still make measurements if some of the receivers in pointed in the wrong direction.

8.3 Maintainability

Maintainability is defined as the ability of our operating system specific item to be maintained. In our case, as with most other satellite missions, it is not possible to do regular on-board maintenance to the satellites themselves. The main focus in maintenance is on the ground segment and can be divided in two parts: preventive maintenance and corrective maintenance. They are considering separately as follows:

1. Preventive maintenance. During the regular system operation time, there are periodic maintenance and condition dependent maintenance. System software or simulator servicing maintenance of ground station is mandatory and data link rate needs to be adjusted in some cases. On the other hand,

conditional maintenance is set to do some specific inspections to prevent something going wrong in the future.

2. Corrective maintenance. This is mainly carried out after something goes wrong. For instance, if one of the photon receiver is not functional, the system can relocate the rest of the receivers to decrease the functional influence mostly. But if the laser emitter is broken, it is no way to obtain the maintenance. Corrective maintenance is also used during analyzing measurements data to obtain better resolution or accuracy.

Since the mission consists of a swarm of satellites it will be possible to add more satellites to the swarm, for example to upgrade the mission or extend the mission life.

8.4 Safety

The system safety is mainly considered during launch and decommissioning. The safety risks during launch are mainly covered by choosing a reliable launcher. For decommission it is important to choose a decent orbit in such a way that it ensures the satellite to burn up in the atmosphere entirely. Most of the time the satellite is on its orbit in space. The orbits of the satellites need to be designed in such a way that they will not intervene with the orbits of other satellites, even when the satellite fails.

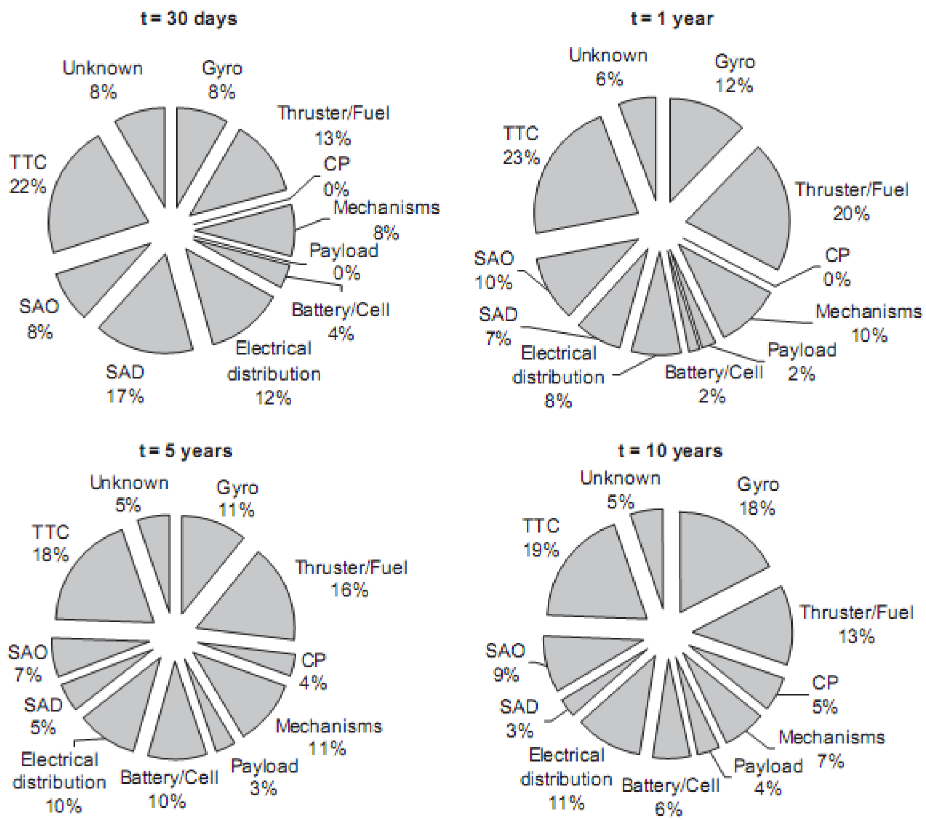

Figure 8.1: Subsystem contributions to satellite failures after 30 days, 1 year, 5 years, and 10 years in-orbit.[4]

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Appendix A

Requirement Discovery Trees