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) system. ICESat used 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 uncertain if even a single photon per pulse reaches the receiver. One possible solution could be to 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 proven.

The 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. A 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, which incorporates this concept.

Contents

List of Acronyms

ADCS Attitude Determination and Control Subsystem

CCD Charge-Coupled Device

CDH Command and Data Handling

CMG Control Moment Gyro

EOL End-of-Life

EPS Electrical Power System

FBS Functional Breakdown Structure

FFD Functional Flow Diagram

GEO Geosynchronous Earth Orbit

GLAS Geoscience Laser Altimeter System

GNC Guidance, Navigation and Control

GPS Global Positioning System

HEllO Highly Elliptical Orbit

HEO High Earth Orbit

IAT Integration, Assembly and Test

laser Light Amplification by Stimulated Emission of Radiation

LEO Low Earth Orbit

LiDAR Light Detection And Ranging

MANS Microcosm Autonomous Navigation System

MEO Medium Earth Orbit

MNS Mission Need Statement

OFOV Optical Field of View

OM Operations and Management

ORD Optical Receiving Device

PDS Power Distribution System

QDRD Quantum Dot Receiving Device

RAMS Reliability, Availability, Maintainability and Safety

RDTE Research, Development, Test and Evaluation

ROI Return On Investment

TDRS Tracking and Data Relay Satellite

TFU Theoretical First Unit

TTC Telemetry, Tracking and Command

Chapter 1

Introduction

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. To do all this, ICESat had only one instrument on board: a space based LiDAR system (Geoscience Laser Altimeter System (GLAS)), allowing for an unprecedented 3D view of the Earth's surface and atmosphere. The laser lifetimes, however, were severely limited because of manufacturing errors in one of the laser components.

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 uncertain if even a single photon per pulse reaches the receiver. One possible solution could be the use of 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 proven.

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. 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.

Chapter 2 describes the functions and requirements of the system as a whole, whereas Chapter 3 shows the multiple budget breakdowns and resource allocation. In Chapter 4 the technical risks are investigated. Chapter 5 illustrates the different design options. Since sustainable engineering is an important factor, Chapter 6 is devoted to this subject. 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 7.

The first thing that needs to happen, after having been built, 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 that the signals are sent. The receivers can adjust their attitude, 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 stored and then transmitted directly to a ground station or sent to the main satellite (the emitter) and then to the ground. The data on the ground can be split up into data packages, which can be distributed to research institutes and other interested parties. With those data sets, a terrain model can be produced. At the End-of-Life (EOL) of the mission the satellites are decommissioned to make room for other satellites.

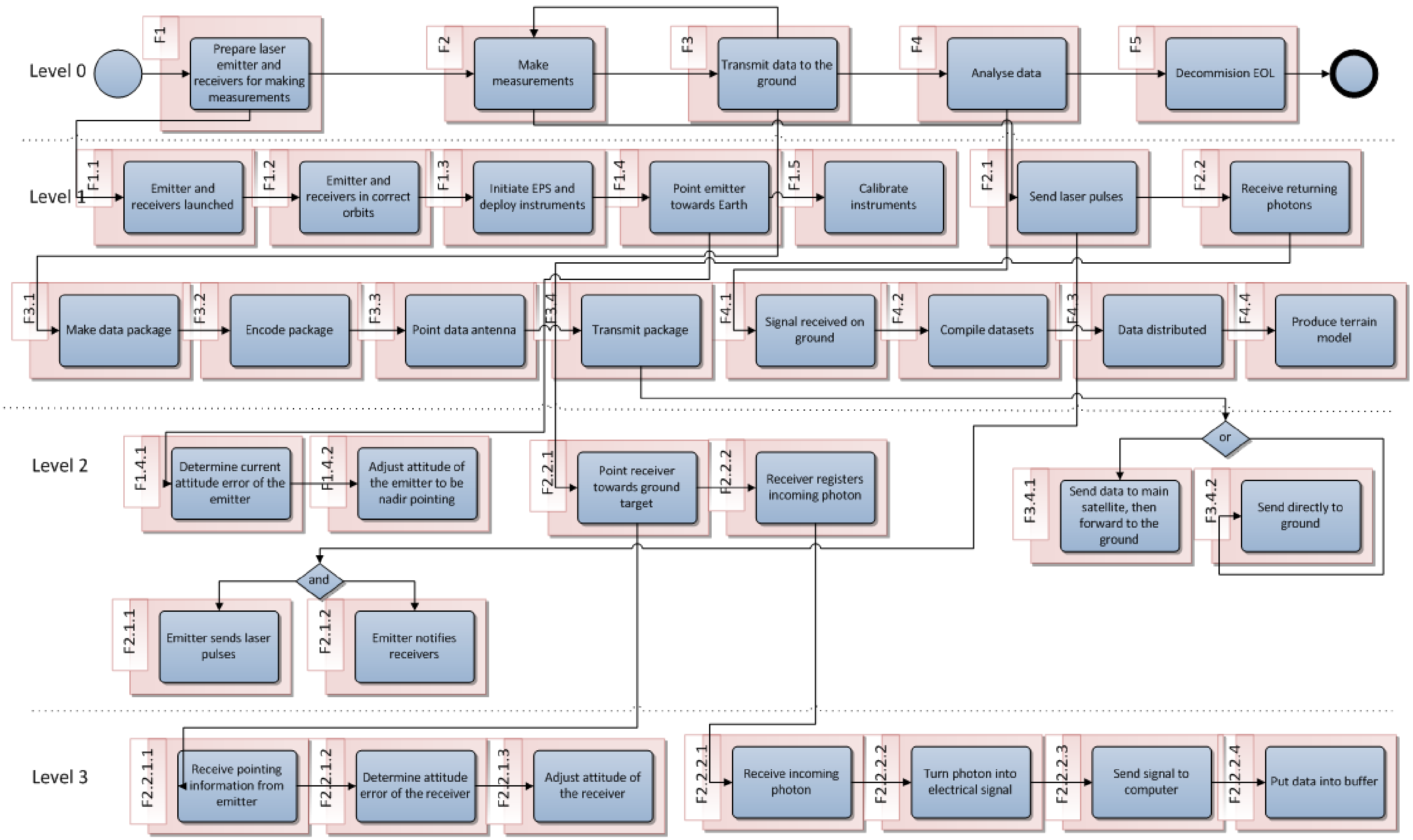

Figure 2.1: Functional Flow Diagram of the Laser Swarm mission

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.

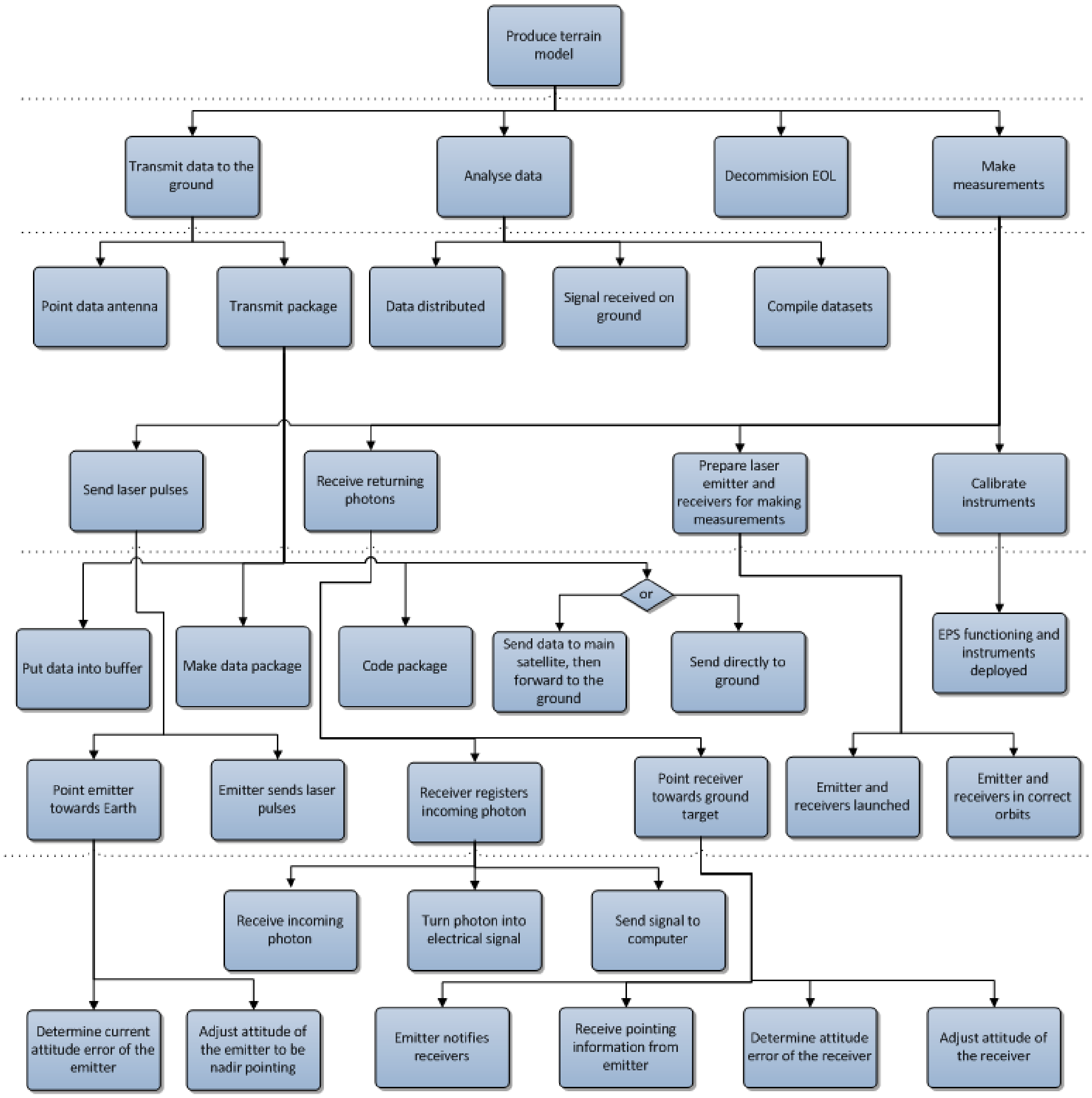

Figure 2.2: Functional breakdown structure of the Laser Swarm mission

The main function for the system is to be able to produce a terrain model, which is the reason for making 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 analyzed. Because the mission has 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 measurements depend on laser pulses to be sent, 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 needs to send the pulses. Receiving photons depends on the receiver being pointed at the ground target and 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 coded and either the receiver sends the data to the ground directly or to the main satellite, which in turn forwards it to Earth. It is important to have the Electrical Power System (EPS) properly functioning and instruments deployed before they can be calibrated.

Determining the attitude error of the emitter and adjusting the it accordingly is 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 so 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, interpret it, determine attitude error and adapt accordingly to make sure that data gathering will be carried out properly.

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 must be below existing spaceborne laser altimetry systems.
- Lifetime must be above existing spaceborne laser altimetry systems.

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

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

The last requirement is mainly considered as an arbitrary constraint. The constellation should be designed as a collection of pointing receiver devices. Furthermore, an inherent top level requirement is gathering usable data. Since this requirement depend virtually on all aspects of this design, no explicit requirement discovery tree is made.

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 [8]. From hereon, the cost requirement was broken down into three main parts: *payload*, *bus* and *wraps*.

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 [16], it is imperative that these systems would be accounted for, yet their design was assumed to be similar to the design of current laser altimetry systems.

Since this innovative design is being mostly compared with existing systems, which include just one payload, it is imperative to recognize the main aspect of this requirement: cost of bigger, more powerful instruments vs the cost of multiple weaker platforms.

A detailed look at the cost budget breakdown can be found in section 3.3.

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

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 combined 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 requirements effect both factors.

A detailed look at the cost budget breakdown can be found in section 3.2.

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 GLAS, the first of which stopped emitting pulses on operating day 37 [4]. 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 [4].

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.

Chapter 3

Budget Breakdown

3.1 Human Resource Allocation

The first part of the project will involve a team of five specialists designing the five critical subsystems and one person designing the orbits of the swarm. The other four members will concentrate on the development of the software that will be used to assist the trade-off and verify the design. At later stages, some of the the software engineering personnel will be heavily involved in designing proper algorithms for the processing of mission data, while others will be brought in to assist with detail design.

A schematic representation of the resource allocation can be found in figure 3.1 on page 11.

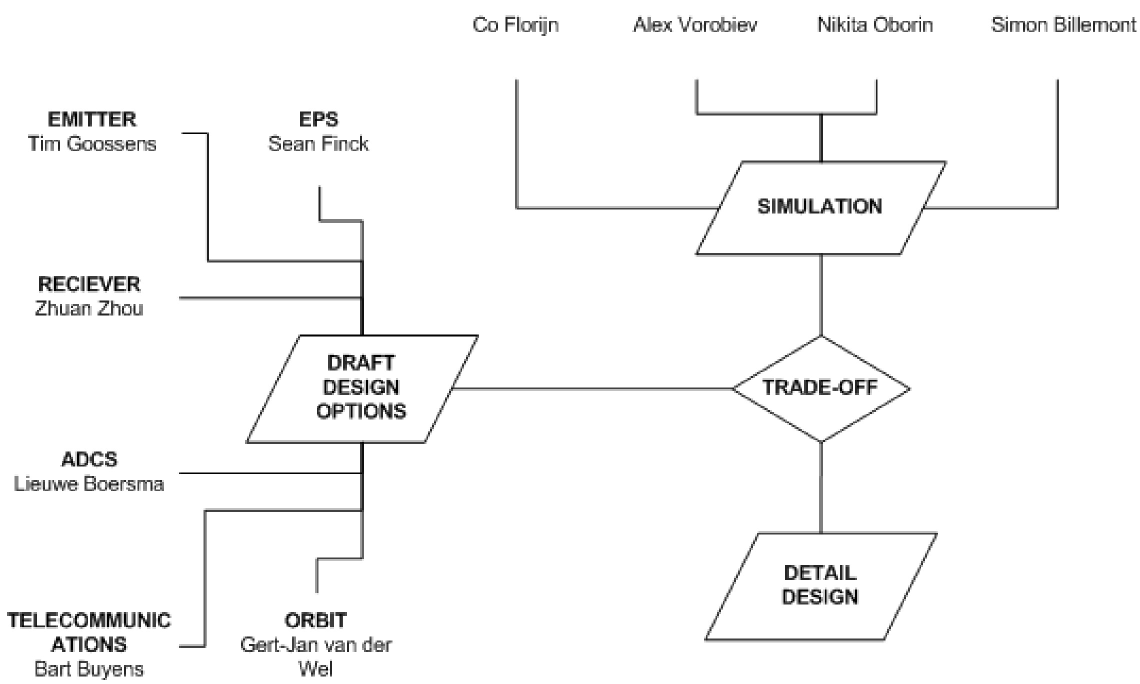

Figure 3.1: Human resource allocation chart

3.2 Mass Budget Breakdown

A satellite is built up from a number of subsystems, which all have a mass. Figure 3.2 on page 12 shows the different subsystems and some examples of components that contribute to the mass of the subsystem. In table 3.1 rules of thumb for finding the mass of the different subsystems are given in terms of the satellite's dry mass. It has to be noted that the rules mainly count for conventional satellites. When using micro- or even nano-satellites there might be another distribution of mass, since some systems like computers can be miniaturized, while other systems like antennas can not.

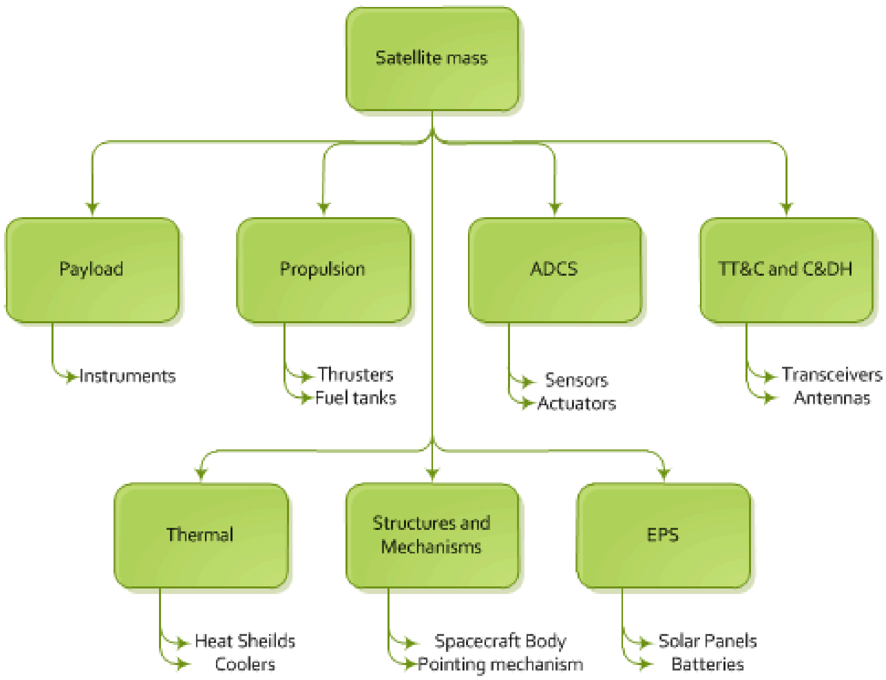

Figure 3.2: Mass Budget Breakdown Structure

SUBSYSTEM	**% OF M_{dry}**	**SOURCE**
Propulsion	2.5-7 %	[10]
ADCS (incl. Guidance, Navigation and Control (GNC))	3-9 %	[10]
Command and Data Handling (CDH) and Telemetry, Tracking and Command (TTC)	2.5-7 %	[10]
Thermal	3 %	[16]
Power	20-40 %	[10]
Structure & Mechanics	18-25 %	[10]
Payload	15-50 %	[16]

Table 3.1: Mass budget breakdown estimation of subsystem mass, in terms of dry mass

3.3 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 3.3 on page 13 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 unit. 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 [16].

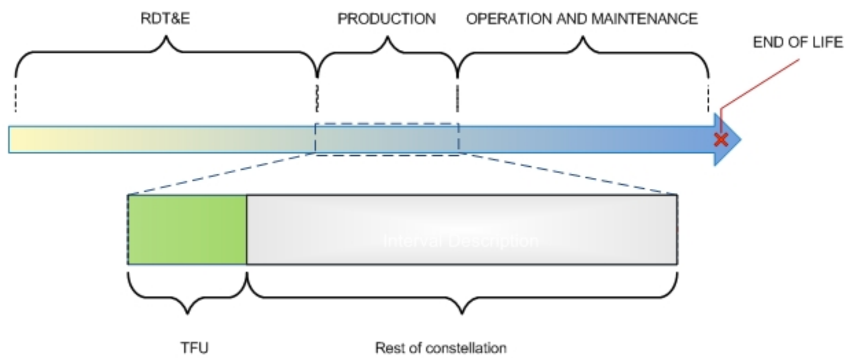

Figure 3.3: Satellite Life-Cycle

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.

The cost breakdown can be found in figure 3.4 on page 15. 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 in early estimates. Furthermore, due to the high cost of actual production of individual satellites, a swarm is hard to predict without a rough idea of the number of space platforms. It is important to note that cost is also dependent on so called heritage factors. These factors allow for reduction of costs based on already existing designs. However, if some kind of new technology is used in a subsystem then this reduction cannot be used.

Table 3.2 on page 14 contains percentage estimations of all aspects covered in figure 3.4.

3.4 Power Budget Breakdown

At this preliminary stage, the power budget breakdown is an estimation based on previous missions and on general data taken from [16].

SEGMENT	% OF PARENT	% OF TOTAL
Space Segment		30
RDTE	30	9
Payload	26	2.3
Bus	52	4.7
Structure	24	1.1
Thermal	2	0.1
EPS	22	1
Communications	38	1.8
ADCS	14	0.7
IAT	0	0
Program Level	22	2
Program Management	20	0.4
Systems Engineering	40	0.8
Product Assurance	20	0.4
System Evaluation	20	0.4
GSE	4	0.4
LOOS	0	0
Software	25	7.5
Production	45	13.5
TFU	10-30	1.4-4.8
Payload	20	0.3-0.8
Bus	45	0.6-1.8
Structures	24	0.1-0.4
Thermal	5	0.04-0.12
EPS	20	0.1-0.4
Communications	18	0.1-0.3
ADCS	28	0.2-0.5
IAT	14	0.2-0.6
Program Level	13	0.2-0.5
Program Management	30	0.05-0.16
Systems Engineering	20	0.04-0.11
Product Assurance	30	0.05-0.16
System Evaluation	20	0.04-0.11
GSE	0	0
LOOS	5	0.07-0.2
Swarm	70-90	9.5-12
Launch Segment		5-10
Launcher	100	5-10
Ground Segment		20
First Ground Station	25	5
Consecutive Ground Stations	55	11
Software	20	4
Operations and Maintenance		45
Operations and support of ground stations	100	45

Table 3.2: Cost budget breakdown estimations

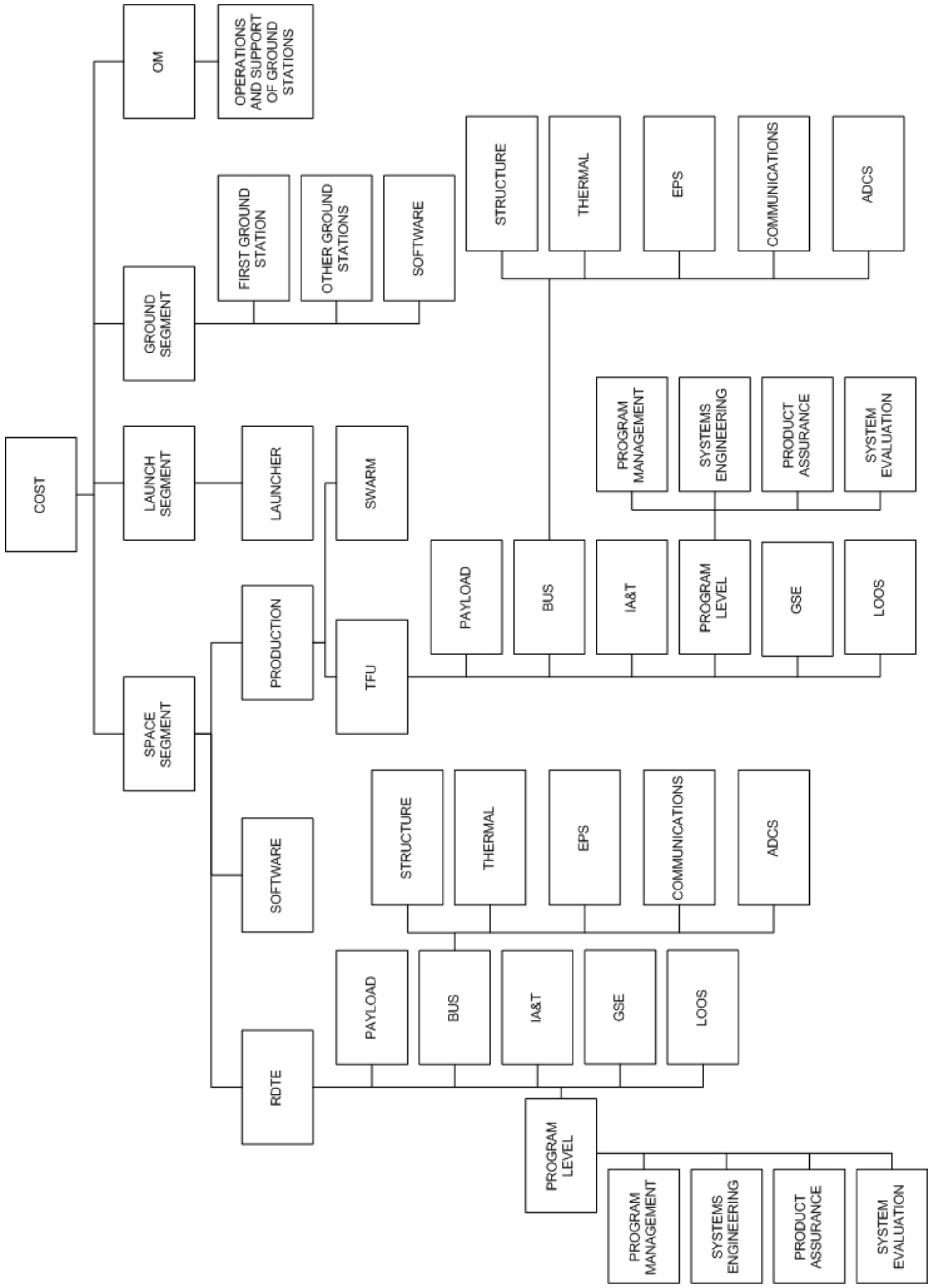

Figure 3.4: Cost Budget Breakdown Structure

The biggest power requirements come from the mission payload and the power system. The payload needs power to be able to operate the emitter and the receiver for the whole mission lifetime. About 30% of the total power will be used to operate the transmitter and 10% for the receiver. The power distribution and control will take up about 30% of all the power. Of this 25% will be needed for the regulators and converters and 5% will be lost due to wiring losses during the distribution of the power. Furthermore, the ADCS system will need approximately 15% of the total power to be able to operate. The amount of power that will go to its subsystems (sensors, controllers and processors) is an estimate based on previous missions (i.e. FireSat [16]).

The last subsystem that needs power is Communications. This system is divided further in to the actual communications and the command and data handling subsystem. Each receive about 5% of the total power. The power distribution for the transmitter and receiver was taken from values of typical X-Band, S-Band and Ku-Band communication subsystems. For the power distribution of the command and the telemetry unit, average nominal values were used [16].

The power breakdown structure and the corresponding power allocation for each system can be seen in figure 3.5 on page 17. A tabulated representation of the breakdown estimations can be found in table 3.3.

SUBSYSTEM	% **OF TOTAL**
ADCS	15
Sensor	3
Controllers	8
Processors	4
CDH and TTC	10
Communications	5
Receiver	1
Transmitter	4
Data Handling	5
Command Unit	1
Telemetry	4
Power	30
Regulator/Converters	25
Wiring	5
Payload	40
Emitter	30
Emitter	10

Table 3.3: Power budget breakdown estimation

3.5 Technical Resource Contingencies

When starting the design of a new system the technical resource budgets, like power or mass of the system, are not known yet. They must be estimated based on the information of previous missions, general guidelines and educated estimations. During the design process, the actual budgets will differ from these initial estimations. For example the mass of a satellite will change during the design process because of unexpected problems, changes in the design, etc. To ensure that a product will achieve the required performance while not overspending the budgets, Technical Performance Measurement (TPM) is used. This technique shows the difference between the actual and required performance. For example a contingency of 20% for the mass budget means that the actual mass can differ up to 20% of the required mass. TPM establishes contingencies for each technical parameter. This contingency is managed to provide a reserve to spend as the desing difficulties are discovered. The contingency is reduced as the project matures and disappears at the product

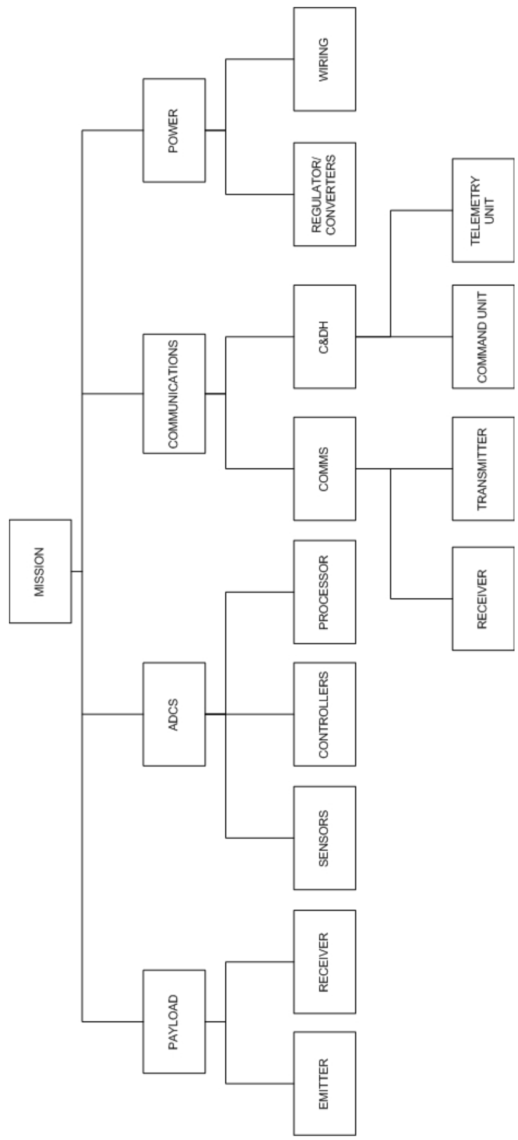

Figure 3.5: Power Budget Breakdown Structure

delivery.

Fortunately, some guidelines exist for contingencies of aerospace vehicles. The main technical resource budgets for our project are mass and power. Mass and power contingencies are given in tables 3.4 and 3.5 on page 19.

	Contingencies [%]						
Design maturity	Propulsion	ADCS	C&DH	Thermal	Power	Structures & Mechanisms	Payload
Conceptual estimate	20	20	25	20	15	20	20
Layout calculation	15	15	20	15	10	15	15
Pre-released drawings	5	5	10	10	5	5	5
Released drawings	3	3	5	5	5	3	3
Specifications (vendor/subcontractor)	5	5	5	5	5	5	5
Actual measurement qualification hardware	1	1	1	1	1	1	1
Actual measurement flight hardware	0	0	0	0	0	0	0

Table 3.4: The mass contingency allowance

	Contingencies [%]			
Design maturity	ADCS	C&DH	Power	Payload
Conceptual estimate	20	20	15	30
Layout calculation	15	15	10	20
Pre-released drawings	5	5	5	10
Released drawings	3	3	5	5
Specifications (vendor/subcontractor)	5	5	5	5
Actual measurement qualification hardware	1	1	1	1
Actual measurement flight hardware	0	0	0	0

Table 3.5: The power contingency allowance

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 top level 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 parties and research or educational facilities. Every space mission is created for at least one specific (user-demanded) requirement set by a user. 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 cannot be met and alternatives should be devised, 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 the 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 (more receivers and one emitter) 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 cannot 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 is not 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 adjust the 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 relatively 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 demonstrates relative reliabilities of different passive ADCS systems.

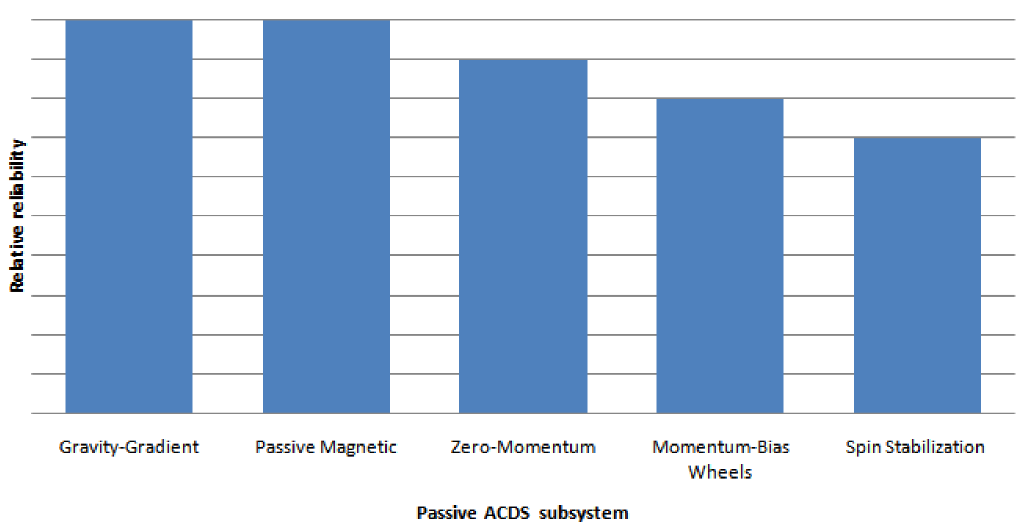

Figure 4.1: Relative reliability of passive ADCS subsystems.

F ***Active systems***

F1. Actuator

F1a. Thrusters (hot and cold gas).
Multiple-axes thruster systems are very efficient ways for controlling attitude 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 important 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 demonstrates relative reliabilities of different active ADCS systems.

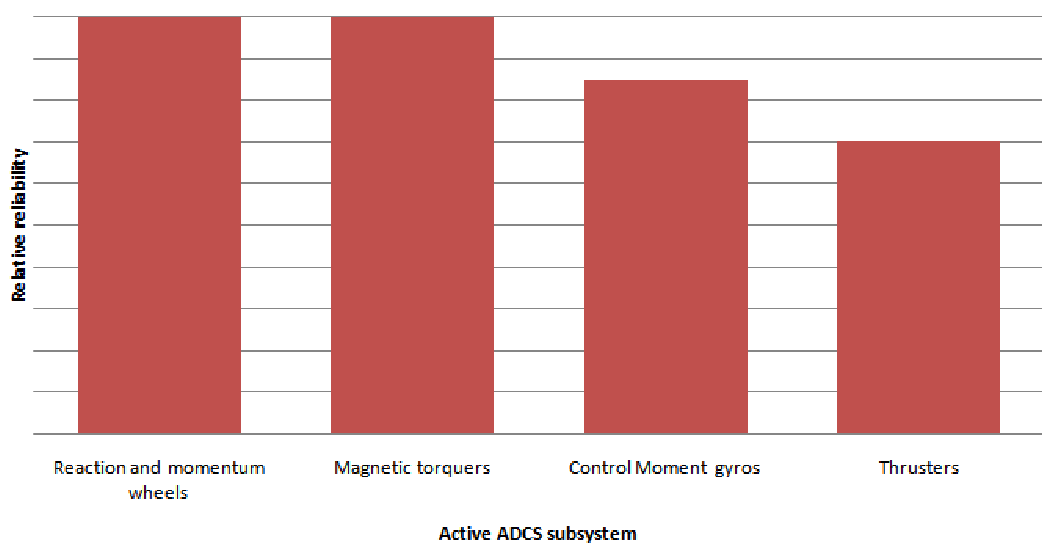

Figure 4.2: Relative reliability of active ADCS subsystems.

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 possible. During the initializing of the mission (assuming the spacecraft is in the right orbit), the solar panels need to be deployed. 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 available electrical power will occur. This makes the consequences pretty severe. Any other mechanical failure

(broken joints, internal PN-junction failure,or a loss of 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%.

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 is 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 can 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 analysis.

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 cannot 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, GLAS 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 technical readiness level, the reliability is high.

J2. Time differences
Considering a high 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, because this is not a part of the project.

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
This event is dependent on the reliability of the EPS.

M6. Launch loads
Due to large forces and vibrations during launch the structure of the satellite can fail. Since the launch loads are well known the reliability is low, the consequences can be severe.

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, however, could be very severe.

N6. Classified information (military)
Some parts of the measurement are considered 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 (LEO) one could just wait a couple of years and air drag will cause the satellites orbit to degrade to the point when they can 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 of satellite collision. The probability to no longer be able to eject the satellite from orbit depends on whether or not its propulsion system 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 ejecting them from orbit further into space. This way, new GEO satellites can take the place of the swarm. If this is not done, then 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.

4.5 Risk Control

Sometimes it is possible to decrease the failure probability. For example, A1, F1a and H4 could be the top 3 risk segment. Failure probability of A1 can be pulled down by doing detailed market analysis. In case of F1a, safe combustion performance as well as fuel consumption can be tested and modified in a laboratory environment to increase the thrusters' reliability. Laser degradation (H4) is crucial in the system, and reliable energy source should be used to prevent failure. Meanwhile, thermal control can be performed to achieve high reliability of the laser emitter.

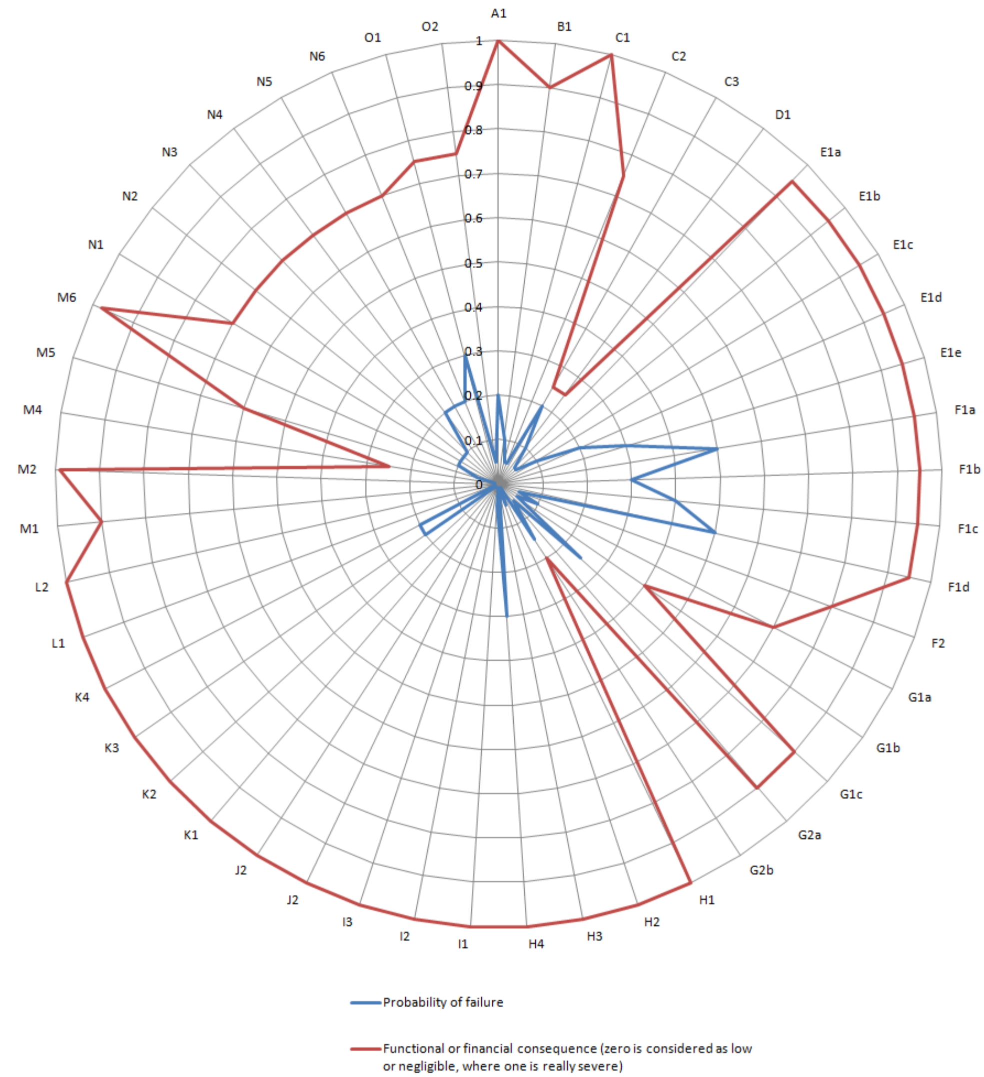

Figure 4.3: Technical Risk Map

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. This is known as an active system. 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 for the ADCS can be found in figure 5.1 on page 29. The chart displays a few properties of each design option for a quick comparison.

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 Global Positioning System (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 gases 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 stabilize 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 stabilization and passive magnetic. Due to the Earth's gravity field, gravity gradient satellites will always tend to point towards Earth with their

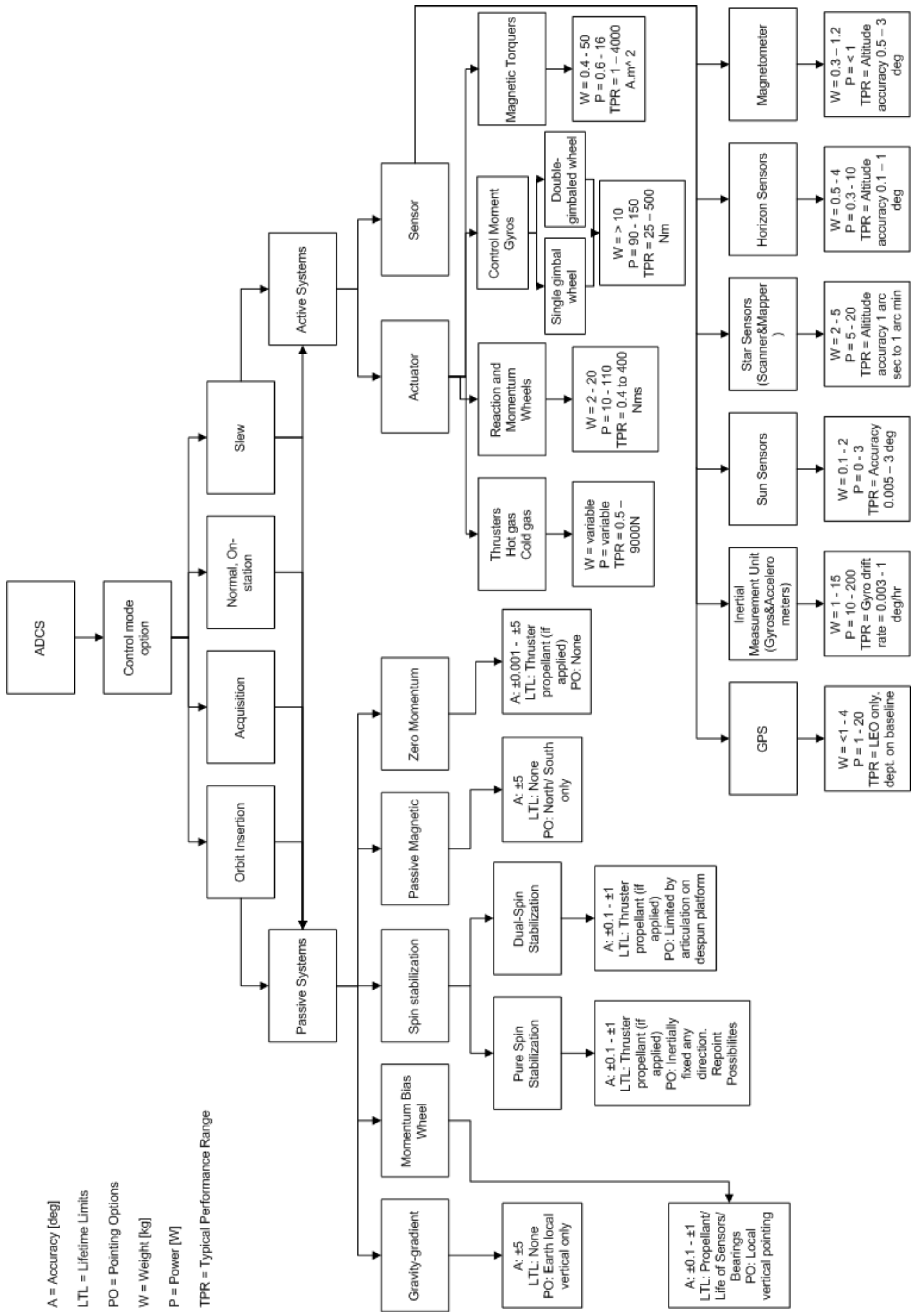

Figure 5.1: Design option tree for the ADCS

smallest moment of inertia axis. Spin stabilized spacecraft are rotating around one axis to stabilize along the other two. Passive magnetic stabilizing uses the magnetic field of the Earth and permanent magnets for its stabilization. Passive means of stabilization are often only usable for stabilization along two axis. If stabilization or even control along three axis is needed a different system is required.

5.1.3 Note on Receiver Pointing

It is important to acknowledge the fact that the receiver payload does not necessarily need to be pointed with the use of ADCS. Other options exist, for example, it can be maneuvered with the use of actuators, or some kind of mirror devices can reflect the photons into the receiver. This is a viable option since it allows the ADCS to be less complex and hence cheaper in terms of cost and mass.

Actuators can take the form of electro-motors or piezoelectric actuators. These options can move the instruments, but will add moving parts and therefore momentum to the satellites. To ensure the attitude to stay correct the ADCS needs to counteract for this.

The other option is using optics. Mirrors or lenses can bend the incoming or outgoing beam of light and point it in the right direction.

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 and 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. Figures 5.2, 5.3 and 5.4 (pp. 34-36) show the complete design option tree for the EPS.

5.2.1 The Power Source

Launch vehicles primarily use batteries as power source, because launch durations are quite short and thus batteries can be kept fairly small. For missions lasting from weeks to years however, batteries would be too large for the mission to be useful.

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 5-8%. 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 [14]. The efficiency of fuel cells can be as high as 80%, but will drop significantly at higher currents.

Photovoltaic solar cells are most common. 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 in the lab [5], but production efficiencies are around 22% [16]. 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 strong than rigid panels.

A comparative table for the different power sources can be found in table 11-35 on page 410 of [16].

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), 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 type use silver zinc, lithium thionyl chloride, lithium sulfur dioxide, lithium monoflouride 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 [16].

Lithium-Ion batteries offer a significant advantage over nickel based ones. According to [16], 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 present 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 [11].

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 peak power). Every solar panel has a peak power point, which can be seen from the panels 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 it's own current and voltage. The result of this is that the battery will change the panel's current and voltage to it's 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 it's 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 have 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 [12]. 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.

5.3 Communications

For the communications subsystem 5 different topics were investigated:

- Tracking
- Swarm satellites crosslink frequency
- D/L and U/L frequency
- Antenna configuration
- Communications architecture

5.3.1 Tracking

For tracking there are the following design options:

- Microcosm Autonomous Navigation System (MANS)
- GPS
- Tracking and Data Relay Satellite (TDRS)
- Satellite crosslinks
- Ground tracking

MANS uses observations of the Earth, Sun and Moon from a single sensor to provide real-time position and attitude data. These objects can be unambiguously identified with high reliability and low cost. Observations can be done with minor modifications to attitude sensors which are already on most spacecraft. The MANS flight software can also make use of data from a GPS receiver, star sensors, gyros and accelerometers to increase the accuracy. MANS can provide ground point look and sun directional information, which also works at any altitude in between LEO and GEO.

GPS is a system of navigation satellites which allows position determination with an accuracy of 50-100 m for non-military use. GPS can also be used to determine attitude by using multiple GPS antennas attached to a rigid element of the spacecraft which allows accuracies between 0.3 and 0.5 degrees. There must always be four GPS satellites in sight in order for the system to work, but which can not be guaranteed that this number of satellites is in sight continuously.

TDRS is a system of two satellites operated by NASA, which can provide tracking data coverage of 85% to 100% of most low-Earth orbits. The system collects mostly range and range-data from the TDRS satellite to the satellite being tracked. Angular information is available, but which is much less accurate than the range and range-rate data. If atmospheric drag effects on a satellite are small, TDRS can achieve 3σ accuracies of 50 m, which is considerably better than most ground-tracking systems.

Satellite crosslinks allow relative position determinations by using of cross link equipments. If absolute position determination is required, a separate tracking system is required. Which can also allow relative position determination, making the satellite crosslinks technique redundant.

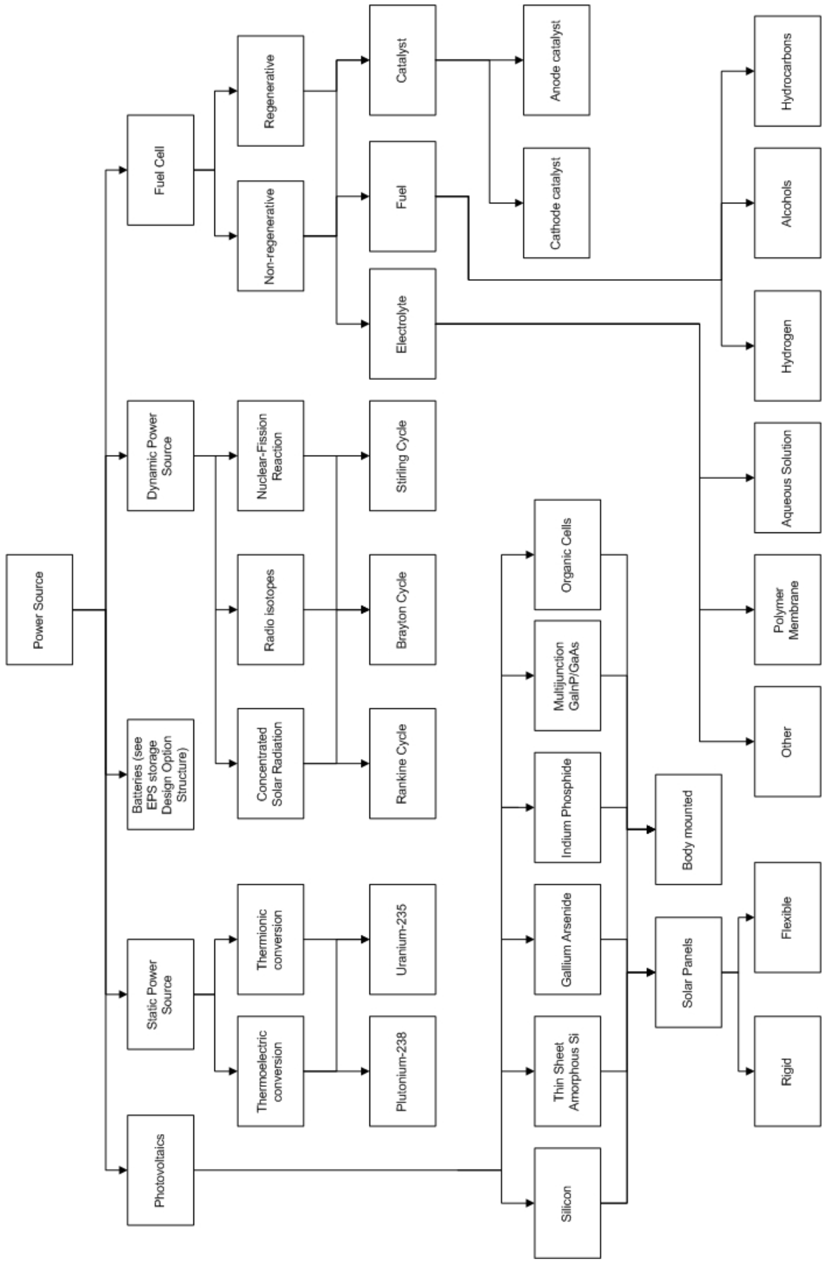

Figure 5.2: Design option tree for the power source

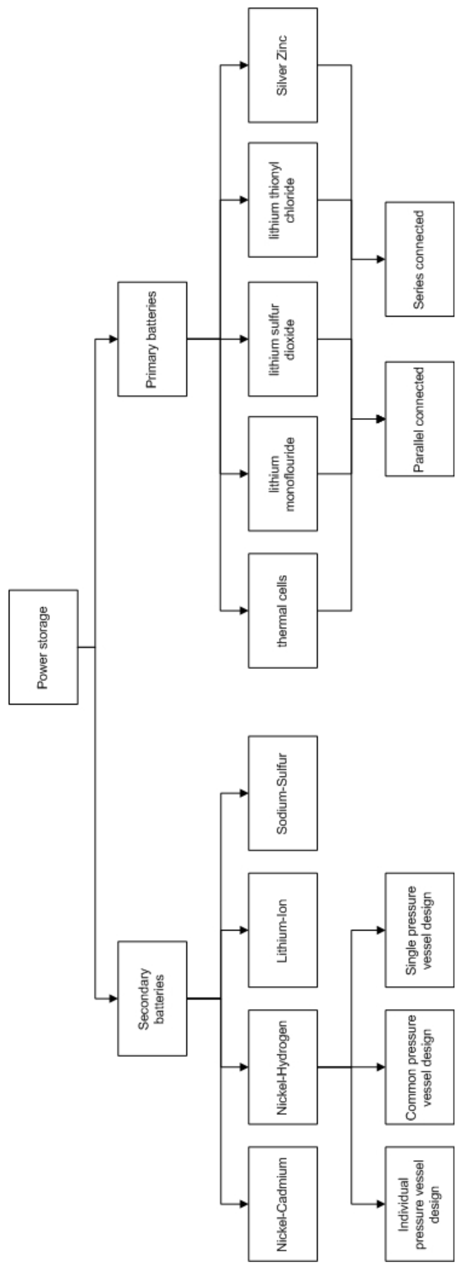

Figure 5.3: Design option tree for the power storage

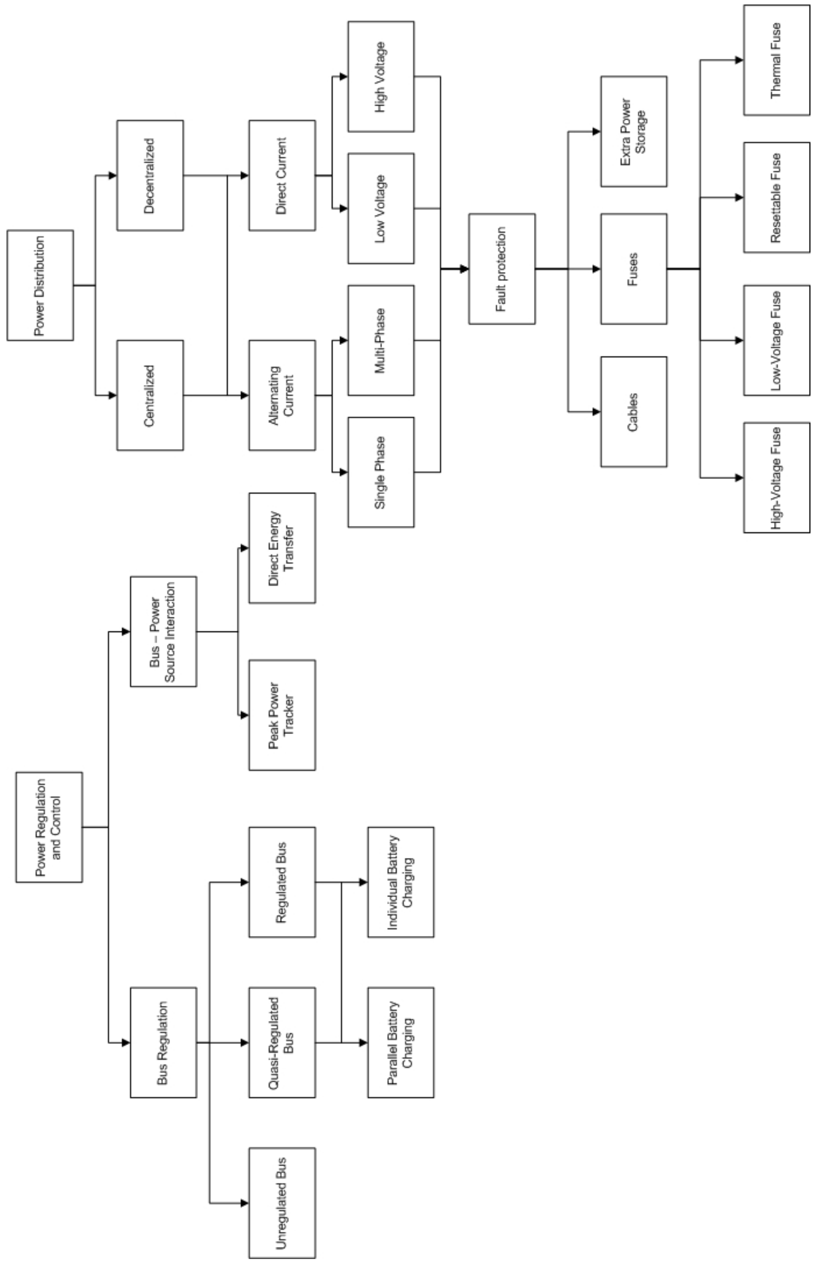

Figure 5.4: Design option tree for the distribution and regulation and control of the EPS

Finally ground tracking allows determination of range and range rate. Angular measurements are also available at times but which are typically far less accurate. Several passes over a ground station are required for orbit determination. 3σ accuracies typically are about several kilometers for LEO.

5.3.2 Swarm Satellites Crosslink Frequency

The frequency band normally used for inter satellite communications are the V-band frequency. This band lies around 60 GHz and has no limit on the power flux density.

5.3.3 D/L and U/L Frequency

Frequency bands available for upload and download of scientific data are:

- C-Band
- X-Band
- Ku-Band
- Ka-Band
- SHF/EHF-band

Details for all frequency bands can be found in [16], p. 566.

5.3.4 Antenna Configuration

The following antennas, suitable for beamwidths of less than 20 degrees, producing gains above 15dB, are considered:

- Parabolic reflector center-feed
- Parabolic reflector cassegrain
- Parabolic reflector off-set feed
- Phased array
- Lens with switched-feed array
- Parabolic reflector off-set shaped subreflector with feed array for scanning

Drawings and descriptions of these antennas can be found in [16], p. 573.

5.3.5 Data Storage

Today Dynamic RAM is almost exclusively used by satellites for mass data storage because they allow very large capacities of 1000 Gbits and more. Typically, they do not have external addressing to each RAM location but operate on a block or file basis. The block may be of the order of 1000 bytes or sometimes it will correspond to one source packet. Dynamic RAM has a simple structure: only one transistor and a capacitor are required per bit, compared to six transistors in Static RAM. This allows Dynamic RAM to reach a very high density [7].

Since this technology has recommended itself as the best around, there is no logical reason to design or trade-off other options. For this reason it is not included in the design option tree.

5.3.6 Communications Architecture

In this subsection we will discuss the possible communications architectures and the transmission power and bandwidth required for the communication subsystem in each satellite:

- Centralized architecture
- Decentralized architecture
- Extremely decentralized architecture

In centralized architecture there is one central satellite which handles all communication between the swarm and the Earth. Which means the communication subsystem of the central satellite requires a broad bandwidth and high transmission power both for the communication between itself and the swarm and the Earth. All data can be stored, organized and compressed efficiently on this central satellite. Data can also be stored on each satellite as a backup in case there is no continuous communication.

A decentralized architecture gives all satellites the bandwidth and transmission power to communicate with the ground, but communication within the swarm is also still possible. In this architecture each satellite has to store its own data itself.

In an extremely decentralized architecture all satellites communicate independently to the ground and also communication between the swarm goes through ground stations. Also for this architecture all satellites need to store their own data.

The design option tree for communications subsystem is shown in figure 5.5 on page 39.

5.4 Orbit Architecture

In this section the design options in the orbit architecture tree as seen in figures 5.7, 5.8 and 5.9 (pp. 41-41), are described.

The orbits are divided in four categories: LEO, Medium Earth Orbit (MEO), Geosynchronous Earth Orbit (GEO) and High Earth Orbit (HEO). Some orbits like hyperbolic and parabolic trajectories are not included as they are not relevant for the mission.

After the main categories there are subdivisions, which can in turn have subdivisions as well. These subdivisions list the special orbits types that are possible, sometimes there is also a block called "other". This is because special orbit types have special constraints, as will be detailed in later sections; whereas the "other" block is there to represent all the other possible orbits. It is not possible at this point to determine exact values as the orbit depends on the payload, the power subsystem and the communications subsystem.

All orbits are assumed to be Keplerian orbits. As such the orbit can be described as a plane located in 3D space. Therefore six elements will define the position of the satellite - the classical orbital elements. The elements are: the semimajor axis (a), the eccentricity (e), the inclination (i), the right ascension of the ascending node (Ω), the argument of perigee (ω) and finally the true anomaly (ν). As a reference for the last four terms, figure 5.6 may be used.

The most important of the four Keplerian elements during this part of the design are: the semimajor axis, the eccentricity and the inclination. The remaining elements are not considered until detailed design.

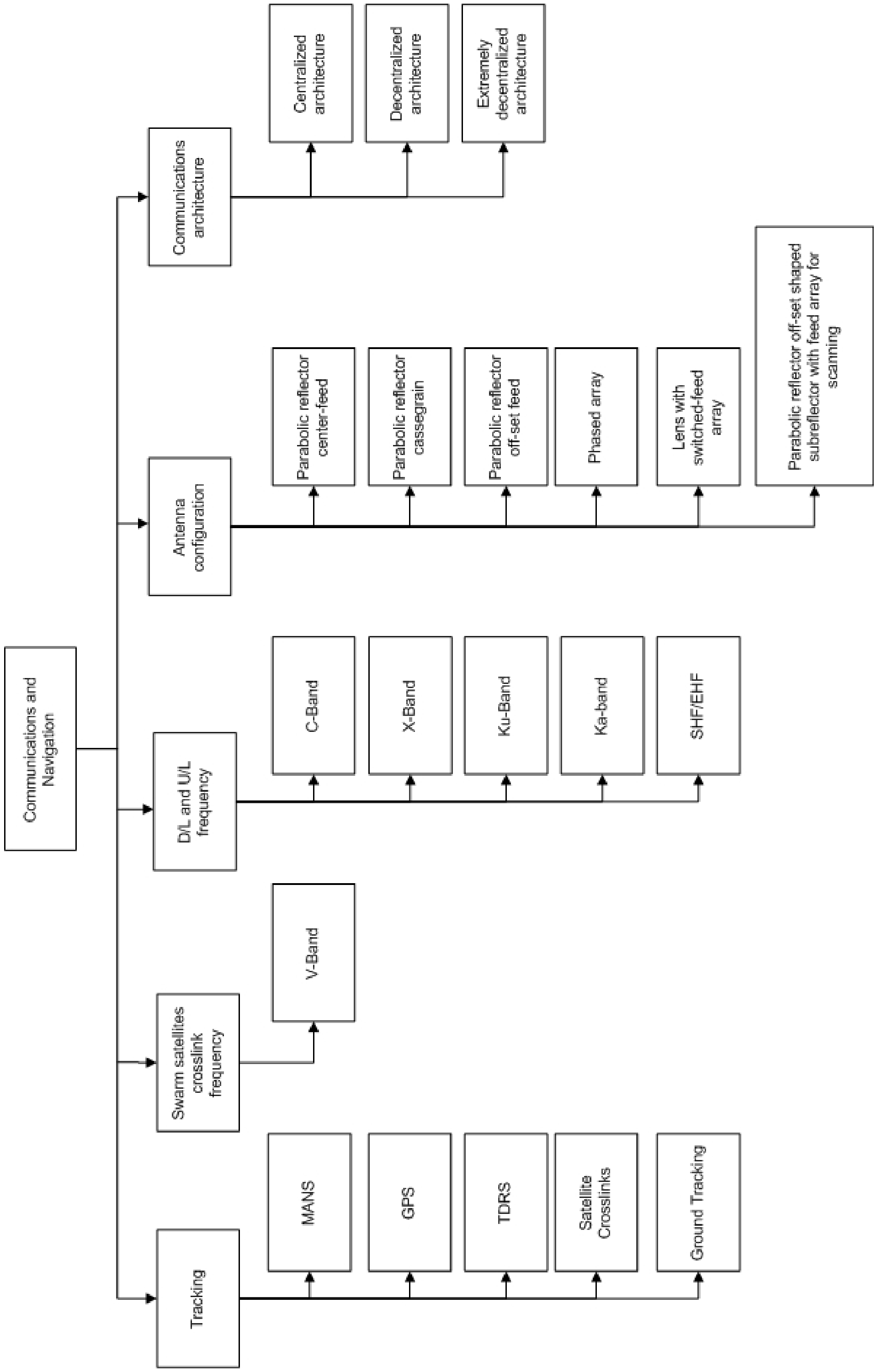

Figure 5.5: Design option tree for communication systems

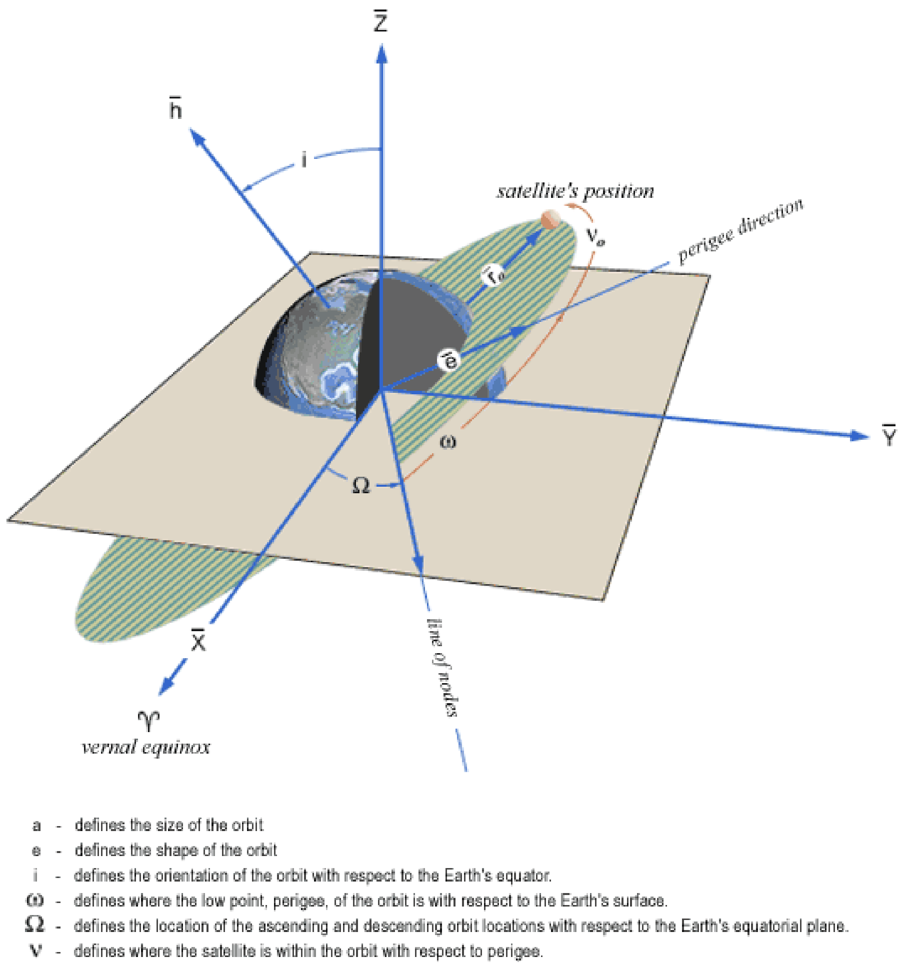

Figure 5.6: Definitions of the inclination i, the right ascension of the ascending node Ω, the argument of perigee ω and the true anomaly ν. *source: http://reentrynews.aero.org*

The following sections will each describe one of the four main categories. They are described in the order listed in the first paragraph of this section. However, first the most common orbits are discussed in the following paragraphs.

5.4.1 Common Orbits

Some special orbits are repeated multiple times in the orbit architecture trees as shown in figures 5.7 to 5.9 (pp. 41-41). These common orbits are the polar orbit, the sun synchronous orbit, the sun synchronous polar orbit, the frozen orbit and the repeat orbit.

1. Polar orbit: This orbit is unique in the sense that the satellite will pass over or closely pass over both poles during a single revolution. As such the angle of incidence i is close to ± 90 degrees. This setup allows for near global coverage, and is useful for polar ice sheet research.

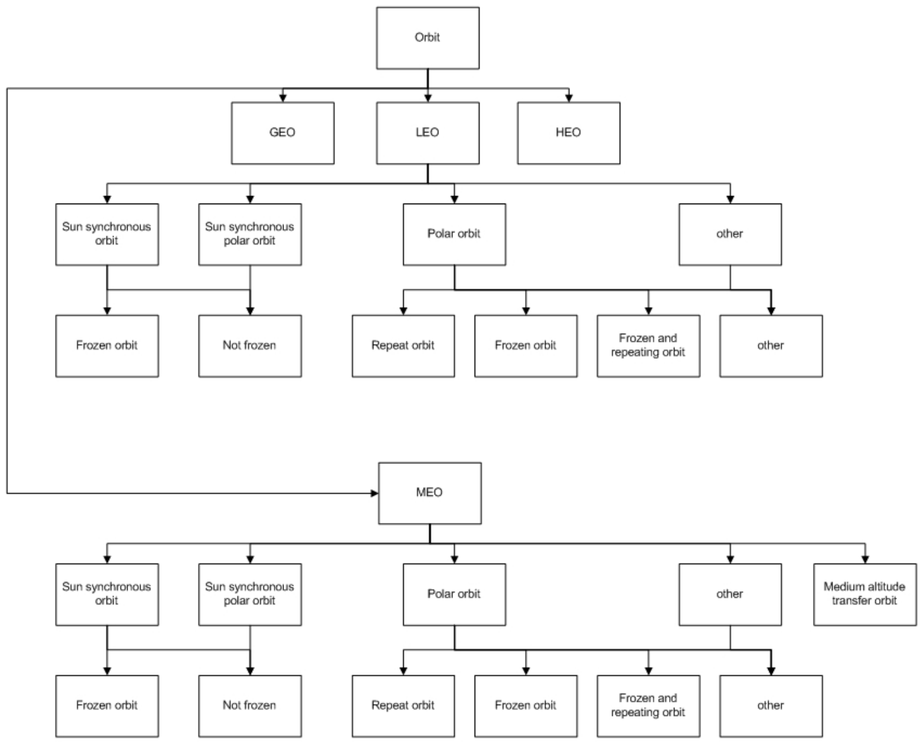

Figure 5.7: Design option tree for the orbit architecture of LEO and MEO

2. Sun Synchronous orbit: An orbit where the satellite passes over the same ground area at the same local time each revolution. At increasing altitude the inclination increases as well, so above approximately 5000 km this orbit tends to lose its usefulness [16].
3. Sun Synchronous Polar orbit: A combination of the previous two items. For this orbit to keep the same local time the altitude should be as low as possible.
4. Transfer orbit: Used to transfer the payload from one orbit to another
5. Frozen orbit: An orbit where there are no long-term changes in argument of perigee and eccentricity. The eccentricity is determined based on a given semimajor axis and inclination.
6. Repeat orbit: An orbit where the satellite comes across the same point on the ground after a integral number of revolutions. A combination of a repeat orbit and a frozen orbit also a possibility.

5.4.2 Low Earth Orbit

The altitude of a satellite in LEO ranges from approximately 80 to 2000 km [2]. The lower limit arises due to the fact of the presence of atmospheric aerodynamic drag, which reduces the velocity and places it in a different orbit. Placing a satellite in a too low orbit means many attitude corrections have to be made (to prevent de-orbit), which is undesirable. The upper limit arises from the inner Van Allen radiation belts, because at 2 to 5 km the radiation is the most intense [7]. Figure 5.7 shows the design option tree for the LEOs.

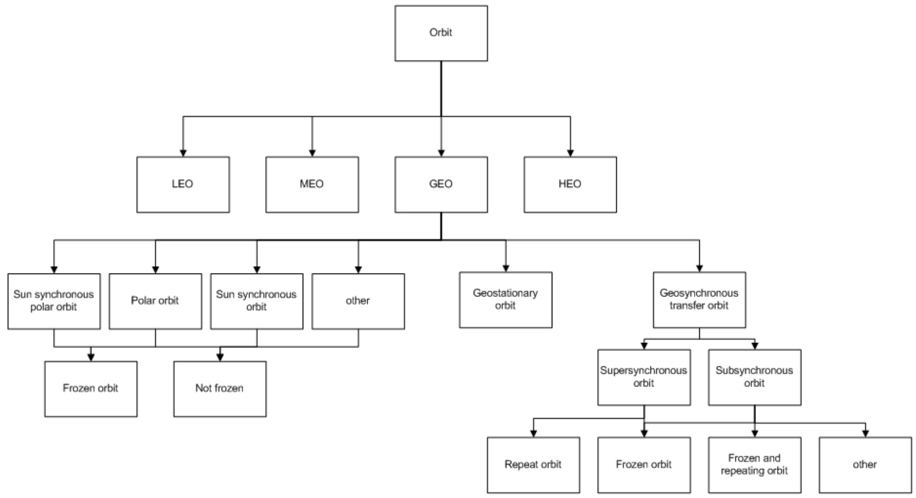

Figure 5.8: Design option tree for the orbit architecture of GEO

5.4.3 Medium Earth Orbit

The altitudes for these orbits have the range from 2000 to 35700 km, which is in between the LEO and the GEO. These are the last orbits where sun synchronous orbits are typically used. This is because of the increase in inclination angle, as mentioned in 5.4.1. Figure 5.7 shows the design option tree for the MEO.

5.4.4 Geosynchronous Orbit

Figure 5.8 shows the design options for the geosynchronous orbits. This type of orbit is characterized by the satellite completing its orbit in exactly one day. This is not to be confused with a Geostationary orbit, which is a special case of geosynchronous orbit where the eccentricity is 0 and the altitude 35786 km. For the reason mentioned in section 5.4.1 sun synchronous orbits are not used at this altitude. GEO has two special types of orbits, the supersynchronous and the subsynchronous orbits. The apogee of supersynchronous orbits is located located at significantly higher altitude than that of a GEO orbit, and both orbits are placed under the GEO category due to the relation with the GEO. The supersynchronous orbit is also known as a graveyard orbit (or other similar synonyms). As the last name suggests this orbit is used to store dead satellites. Recently another application is used, namely to send the satellite to supersynchronous orbit and change the inclination here and then place the satellite in GEO. This is a more efficient maneuver than changing the inclination at GEO [1]. Sometimes a payload is placed in a subsynchronous orbit by the rocket after which the payload has to use its own propulsion to get into GEO.

5.4.5 High Earth Orbit

Continuing the trend indicated in previous sections these orbits are located above the GEO orbits. Sun synchronous orbits are usually not used for this orbit as well. Figure 5.9 shows the design options for the HEO, meanwhile the HEllO discussed in the next subsection is part of this figure.

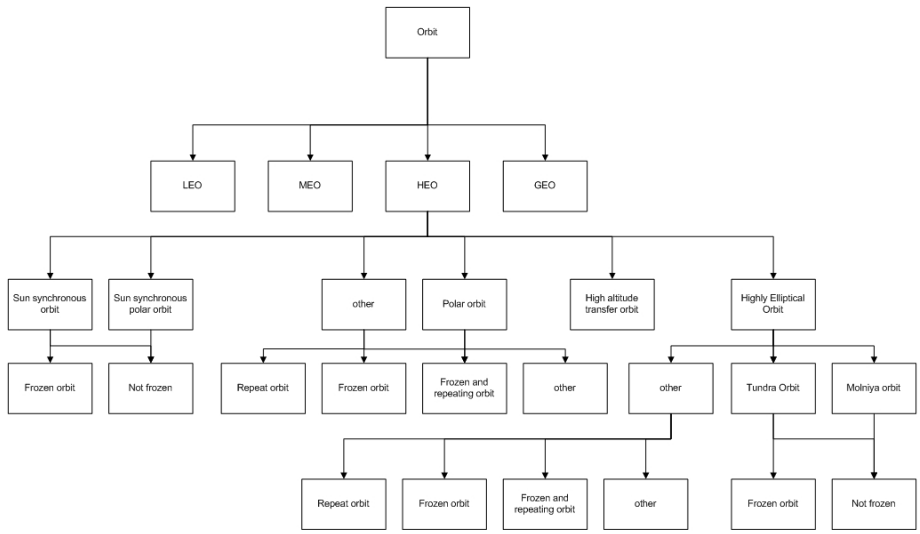

Figure 5.9: Design option tree for the orbit architecture of HEO

Highly Elliptical Orbit

The Highly Elliptical Orbit (HEllO) is defined as a subcategory of the HEO, because a satellite in one of these orbits spends most of its time at its apogee. Other than their high eccentricity, the other orbital elements can take any value. Some special orbits are the Molniya orbit and the Tundra orbit. The Molniya orbit has a period of half a sidereal day and the Tundra orbit has a period of a whole sidereal day. For both orbits the inclination often used is 63.4 degrees, this inclination ensures there is no shift or perigee along the orbit. When designing an orbit with a different incidence angle, the oblates of the Earth has to be accounted for to negate this effect.

5.5 Optical Emitting Payload

5.5.1 Introduction

Satellite altimetry missions use active remote sensing techniques. For that reason the quality of the system is dependent on the emitted electromagnetic radiation and the analysis of the returned signal. Considering the orbit of a regular Earth observation satellite to be in the order of 500 to 800 kilometers (statement based of earlier altimetry missions) and the fact that the power of the emitted radiation decreases exponentially with distance, finding a proper optical emitting device is not a simple task. Next to the decrease in energy, atmospheric absorption or scattering at specific wavelengths are also another important issues to consider, since altimetry missions usually are designed to measure the actual Earth's surface, i.e. the emitted photons need to be able to reach the surface, have enough energy to scatter and reach the receiver.

To be able to select any optical emitting device, the important parameters for optimizing the altimetry results should be revised. Several problems occur if emitting radiation is chosen as the remote sensing technique.

i First of all, general electromagnetic radiation will show isotropic behavior. This results in an effective energy loss, since most of the radiation is not pointed towards the desired position. Hence, a divergence limited source would be preferable.

ii As mentioned before, the wavelength is an important parameter since it will influence the photons actually reaching Earth. Since the Earth's atmosphere is transparent for wavelengths in the visible spectrum, it would be better to have an electromagnetic radiation source with a wavelength in this interval. Next to that, a regular radiation source (like the Sun) emits radiation consisting of a whole spectrum of wavelengths. The less the number of discrete wavelengths (preferable in the visible spectrum), the higher the quality of the analysis can be.

iii The total work done on the photons to reach the Earth's surface, scatter and return to the receiver, is generally very large. To cope with this large work, the energy of the specific energy pulses should be high.

All of these criteria are important when considering the optical emitting device. The most convenient mechanism for solving these preliminary problems can be solved by using a Light Amplification by Stimulated Emission of Radiation (laser) system. Optical emitting devices using laser technique have considerably low divergence (high energy density), discrete and known wavelength characteristics and a relative high pulse energy.

5.5.2 Laser Characteristics

The characteristics of laser systems are determined by the quantum mechanical interaction of electrons (and holes) between the conduction and valence band. Quantum mechanics predicts that electron energy levels are discrete and quantizised, hence, predicting the existences of energy gaps. The electron configuration in a given chemical element,will be distributed according to the Boltzmann distribution, assuming the element is in thermal equilibrium. Generally, the electrons will exhibit the lowest energy state. Electron excitation can take place by energy addition to an n-level energy system, where n denotes the number of quantizised energy levels. This energy can be thermal, electric or photonic.

Photonic excitation, i.e. electron excitation by an induced photon, can lead to stimulated emission, which is the starting point for the working principle of the laser technique. To achieve photonic excitation, the energy of the incoming photon should be equal to the the difference in electron energy level from ground state to excited state. Since this energy is dependent on wavelength, the resulting wavelength of the laser radiation is known as well (incoming wavelength equals wavelength laser radiation). Every chemical composition has its characteristic discrete energy levels and bandgaps, allowing different bandgap energies and hence, different wavelength characteristics.

The laser is an optical emitting device consisting of a certain aperture value. For this reason, diffraction will occur. However, due to the low divergence, the diffraction will be lower relative to isotropic radiating sources.

Radiation from the laser can be continuous or pulsed. The input power, also known as the pump power, is the power needed to ensure the right amount of stimulated electron emission in the laser cavity. This power is then redirected towards the radiation. Since many lasers are pulsed, with pulses in the order of pico- to nanoseconds with relatively high frequency, the pulse power, i.e. the power divided per unit pulse, is in the order of 0.001 to 1 Watt. Peak powers in that case can exceed 1 gigawatt [17].

5.5.3 Laser Types

There are actually many types of lasers. The main types are considered below:

1. Gas. A variety of lasers is based on gases as gain media. The laser-active entities are either single atoms or molecules, and are often used in a mixture with other substances having auxiliary functions. Most gas lasers emit with a high beam quality, often close to diffraction-limit, since the gas introduces only weak optical distortions.
2. Semiconductor lasers, also known as diode lasers, are lasers based on semiconductor gain media, where the optical gain is usually achieved by stimulated emission at an interband transition under conditions of a high carrier density in the conduction band.
3. Solid-State lasers are lasers based on solid-state gain media such as crystals or glasses doped with rare Earth or transition metal ions. They are mainly optically pumped with flash lamps or arc lamps, which will lead to high powers and low costs but also to relative low power efficiencies and moderate lifetime. [13]

Figure 5.10 on page 46 gives the final design option tree of the laser emitter.

5.6 Optical Receiver Payload

5.6.1 Introduction

A large fraction of the photons emitted by the laser are not suitable for detection since they are redirected by the atmosphere, spread due to surface scattering or they are simply absorbed by either of them. The decrease in photon quantity is really severe considering these factors. Since the number of photons actually reaching the perimeter of the Optical Receiving Device (ORD) is usually only in the order of one to ten, the ORD has to be able to detect and analyze individual small energy packets, preferably single photons.

The main objective of the ORD, in that sense, is to detect the individual energy quanta. An obvious way of achieving this is to increase the ORD receiving surface area. In this case, more photons can be detected which normally, with a relative small optical receiving area, would go past the receiver. The main disadvantage is the increase in noise, which decreases the quality of your measurements. The laser used in this altimetry mission is not the only electromagnetic radiation source in the neighborhood of the satellite constellation. Next to thermal radiation from Earth, electromagnetic radiation from other celestial objects and other spacecraft, the Sun is also a major contributor of electromagnetic radiation. All of these sources are considered irrelevant to the altimetry mission and therefore, can be categorized as noise. Increasing the ORD receiving area consequently increases the detection of noise. An optimal receiving-surface-area to noise-detection ratio should be determined.

5.6.2 Types of Optical Receiving Devices

The science behind single photon detection is a relative new branch in the field of quantum optics and at the moment, considering its capabilities and myriad applications, the amount of research done in this field is growing by the day. A number of technologies, primarily based on quantum dynamics, and devices are already in production or in development. Since the science behind the single photon detection devices is pretty elaborate, only an introduction of the techniques are given below.

i Charge-Coupled Device (CCD). A CCD is a two-dimensional array of metal-oxide-semiconductor capacitors which can accumulate and store charge due to their capacitance. When the device is properly biased, the electrons in each array element 'feel' a finite potential well in which they are trapped, assuming no quantum tunneling, i.e. no motion of electrons through potential barriers. By inducing energy to these trapped electrons, the energy can exceed the potential barrier well. If the capacitors and the electrons are

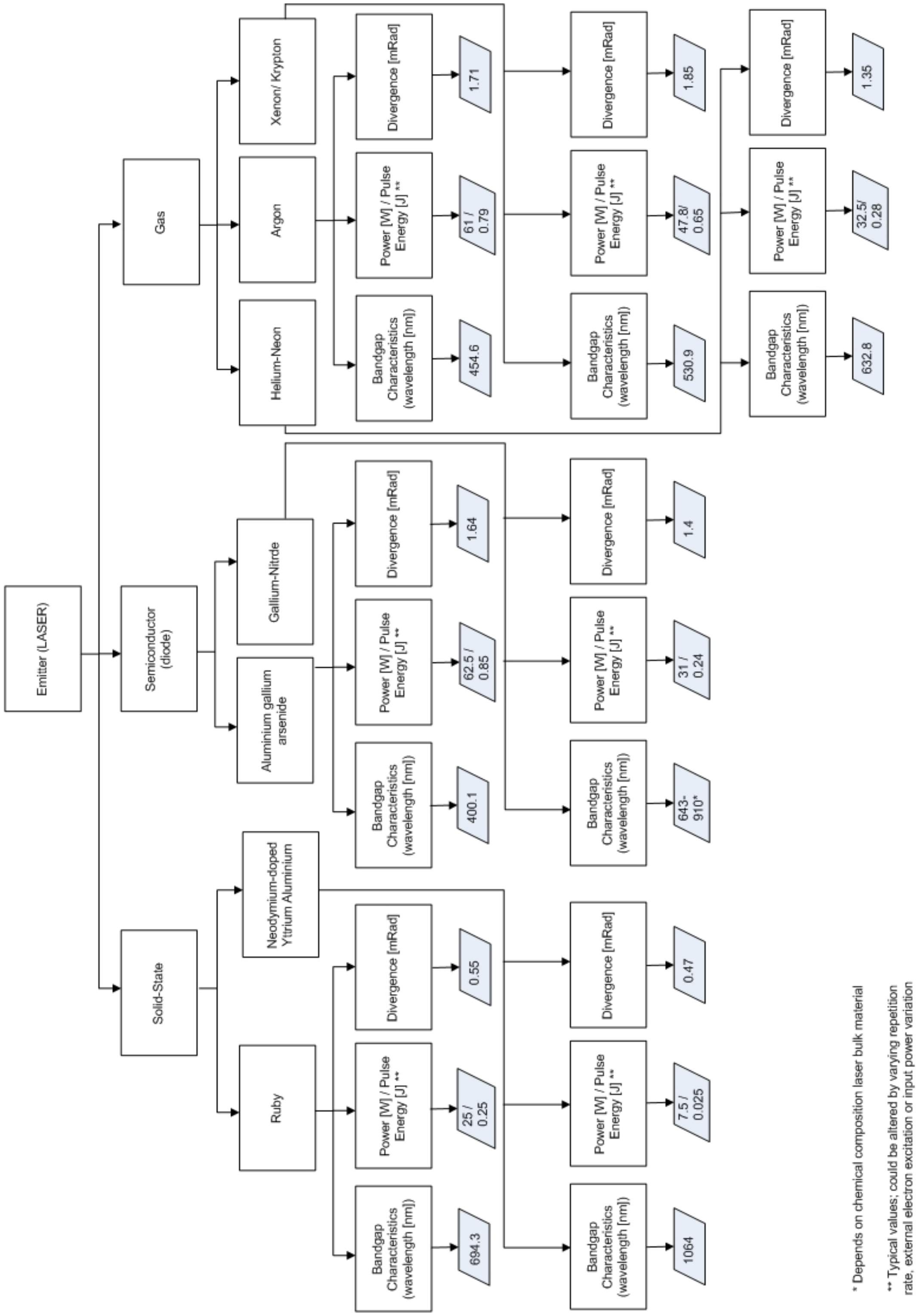

Figure 5.10: Design option tree of laser emitter. Numbers indicate typical values.

in thermal equilibrium, this induced energy should come from photonic excitation (assuming no applied external electric field). If a photon with the proper wavelength (and hence with the proper amount of energy) hits one of the capacitors, electron excitation can occur, giving it an energy level higher than the energy of the potential well. Of course, the energy of the incoming photon should be equal or higher than the energy difference of the initial electron state and the potential energy. Once the energy of the electron is beyond the potential energy, a charge is created which can be measured by applying a PN-junction structure in cooperation with the two-dimensional array.

Usually, the number of photons actually hitting one specific capacitor is not equal to one. Single photon detection however could be achieved by registering individual photonic excitations per capacitor.

The main advantage of CCDs is the high level of technical readiness and the fact that a single capacitor failure does not influence the overall performance.

ii Photomultiplier Tubes (Avalanche Processes). Photomultiplier tubes are photon detectors that are useful in low intensity applications. A photomultiplier tube is a vacuum tube consisting of an input window, a photocathode, focusing electrodes, an electron multiplier and an anode. Figure 5.11 on page 47 gives a clear view of photomultiplier tube configuration and working mechanism.

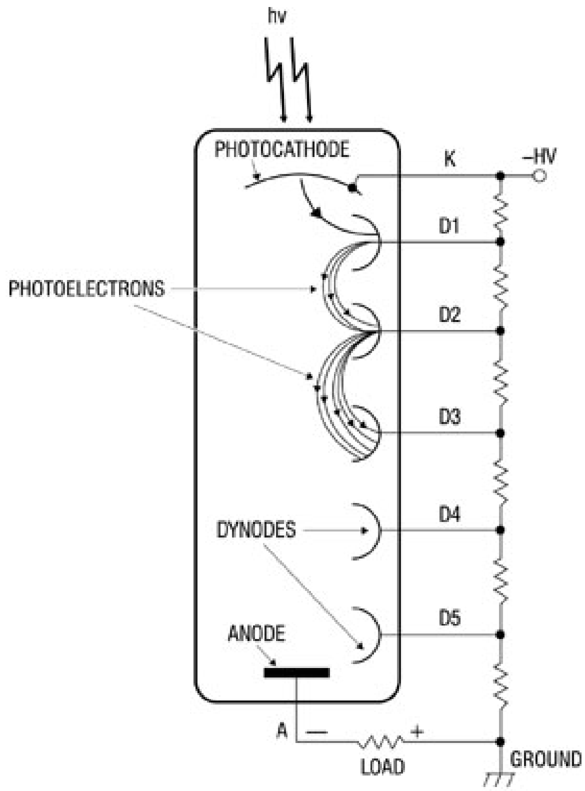

Figure 5.11: Photomultiplier tube configuration and working mechanism

The main mechanism of this device actually uses the photoelectric effect, which refers to the emission, or ejection, of electrons from the surface of, generally, a metal in response to incident light. Energy contained within the incident light is absorbed by electrons within the metal, giving the electrons sufficient energy to be 'knocked' out of, that is, emitted from, the surface of the metal. Next to this emission of the electron, secondary emission is induced due to the appliance of an electric field into the tube, giving the incoming electrons more energy. This way, the number of electrons increase with dynode bypass.

The main disadvantage is that a single tube measures one photon at a specific time. Making a array of tubes adds a lot of mass to the overall system.

iii Quantum Dots. Quantum dots are semiconductor nanocrystals having dimensions in the order of 2 - 50 nanometers. A semiconductor is a material that has a small bandgap between the valence and conduction band. An exciton pair is defined as a negatively charged electron and the positively charged hole it leaves behind when it is excited up to the conduction band. The exciton-Bohr radius is equal to the exciton distance. A quantum dot is a semiconductor so small that the size of the crystal is on the same order as the exciton-Bohr radius, giving birth to small number of quantized discrete energy levels.

Quantum dots can be used to detect single photons. The Quantum Dot Receiving Device (QDRD) is based upon a transistor structure in which the conducting channel is closely spaced from a layer of quantum dots. If the separation of the quantum dots and the channel is just several nanometers, the resistance of the transistor is sensitive to a change in the occupancy of a single quantum dot by just a single electron. This attribute allows the device to act as a detector of single photons, since the absorption of a photon creates a carrier in the semiconductor, which after capture by a quantum dot, produce a detectable change in the resistance of the channel of the resistor. [6]

MPS's single-photon detection modules (see design option tree on page 50) uses quantum dots. Figure 5.12 on page 48 gives a clear view of single photon detection using quantum dot configuration.

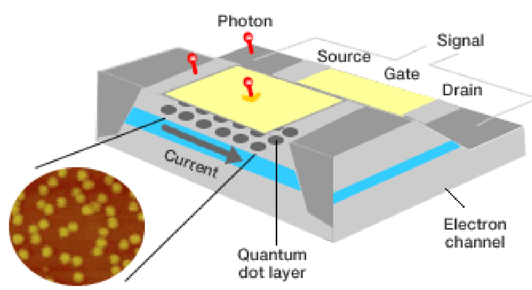

Figure 5.12: Single photon detection using quantum dot configuration

iv Quantum Computers. Quantum computers use, in contrary to regular computers, no semiconducting chips for processing. Although this technology is very interesting with myriad of possibilities, the technology behind it is not understood very well at this point. Scientist are mainly focusing on the theory behind it, without actually building a (theoretical first) model. Although futuristic, many scientists predict a bright future for quantum computers in the field of optics.

The theory behind quantum computers relies on intrinsic properties of fermions. Fermions are elementary particles with a specific value of intrinsic spin (not angular momentum about principal axis). Fermions, like electrons, have a specific intrinsic spin of 1/2. This spin can be spin up (+1/2) or spin down (-1/2). Pauli's exclusion principle states that no identical fermion can exhibit the same energy level, i.e. two electrons can occupy one energy orbit (spin-up and spin-down). Electromagnetic radiation can alter spin configurations. By evaluating spin characteristics, a binary code could be written. Using this binary code, basic computational processes could be extracted [15].

5.6.3 Important Parameters for Optical Receiving Devices

Obviously, the mass and the average power usage are important aspects for choosing the desired ORD in any satellite (constellation) mission.

Considering the fact that you want to detect as much photons as possible, another important characteristic is the Optical Field of View (OFOV). The OFOV (also field of vision) is the (angular or linear) extent to which the ORD is able to observe photons at any given moment. As mentioned earlier, the problem of noise increases if the OFOV becomes too large. However, since the OFOV is in the order of 0.001 radians, it can be stated that a higher values increases the amount of detectable photons.

The last aspect to consider when determining the ORD, is the detector quantum efficiency. It is defined as the efficiency that an incoming photon induces an exciton. It is an accurate measurement of the device's electrical sensitivity to light. Figure 5.13 on page 50 gives the final design option tree of the laser receiver.

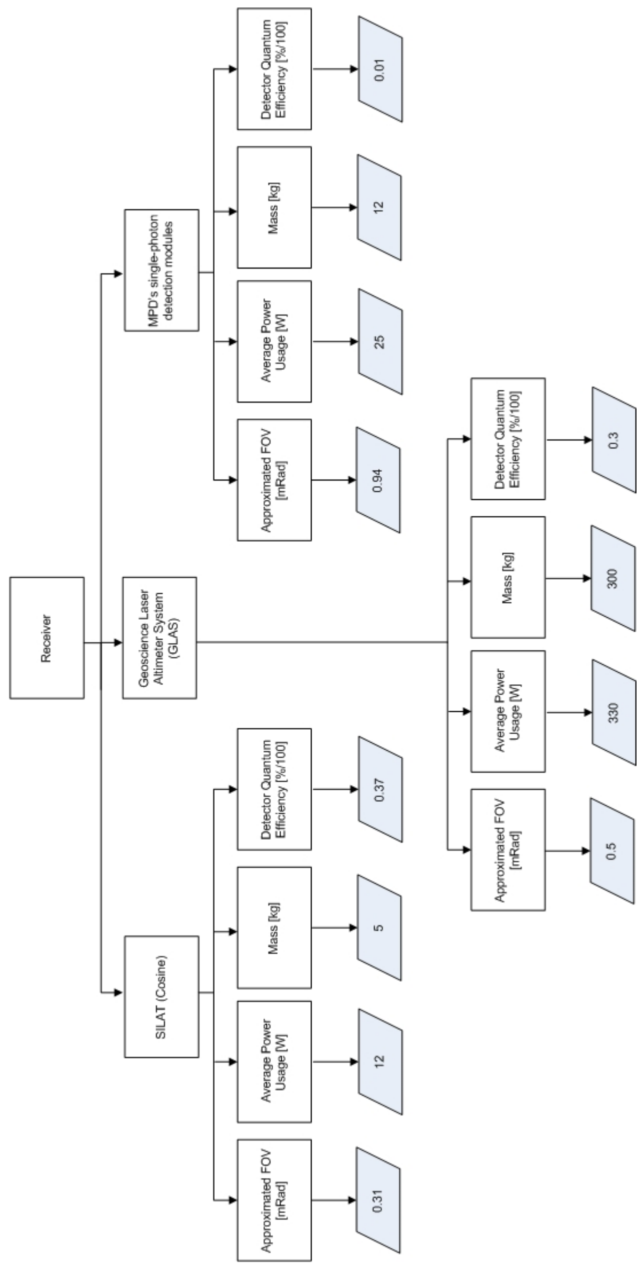

Figure 5.13: Design option tree for laser receiver. Numbers represent typical values.

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 parts: 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 aircraft to the launch site (assuming main payload is built on the same location).

Spreading the swarm, i.e. piggybagging using 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, while the other still working satellites can be reused.

6.3 End of Life

Each satellite will be at the end of its life if it can't perform its function anymore or if its onboard propellant is low. It is important that after the mission is over all satellites are removed from their orbit 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

The Return On Investment (ROI) is a typical financial performance measure to evaluate the efficiency of an investment. To be able to analyze any hypothetical investments a market analysis should be performed, indicating possible organizations, companies or individuals (which are from now on referred to as 'users') that might be interested in the delivered product, service or data. Considering that any mission is specifically user-demanded, the option to attract several users from multiple scientific fields is limited. However, regarding the high price per kilogram ($4,000 - $40,000 / kg depending on launcher and altitude) to put something in an orbit around Earth, it probably would not suffice to have a positive return on ROI using a small amount of users.

It should be noted that the ROI is usually mistakenly misinterpreted by only considering the potential financial profit. The term 'return' does not only stand for financial return. Of course, increasing the potential financial profit increases the feasibility of the mission. However, the return can also refer to scientific or educational gain, which can be seen as long term ROI.

7.1 Market Analysis

Market analysis determines the main potential users and is mainly dependent on the quality and quantity of the obtained data. There are four types of the measurement data sets and it is convenient to analyze them separately. Figure 7.1 on page 54 gives the market diagram with the different data sets. In the Market Breakdown Diagram, each data type has both science and commercial potential users. For example, in the science field the oceanographic data can be used for climate research. Scientists can study the evolution of weather patterns from the ocean system by modeling changes in the heat distribution of the ocean. On the other hand, in the commercial field maps of currents, eddies and vector winds are used in commercial shipping and recreational yachting to optimize routes. All the blocks in the diagram could be potential ROI on short or long term.

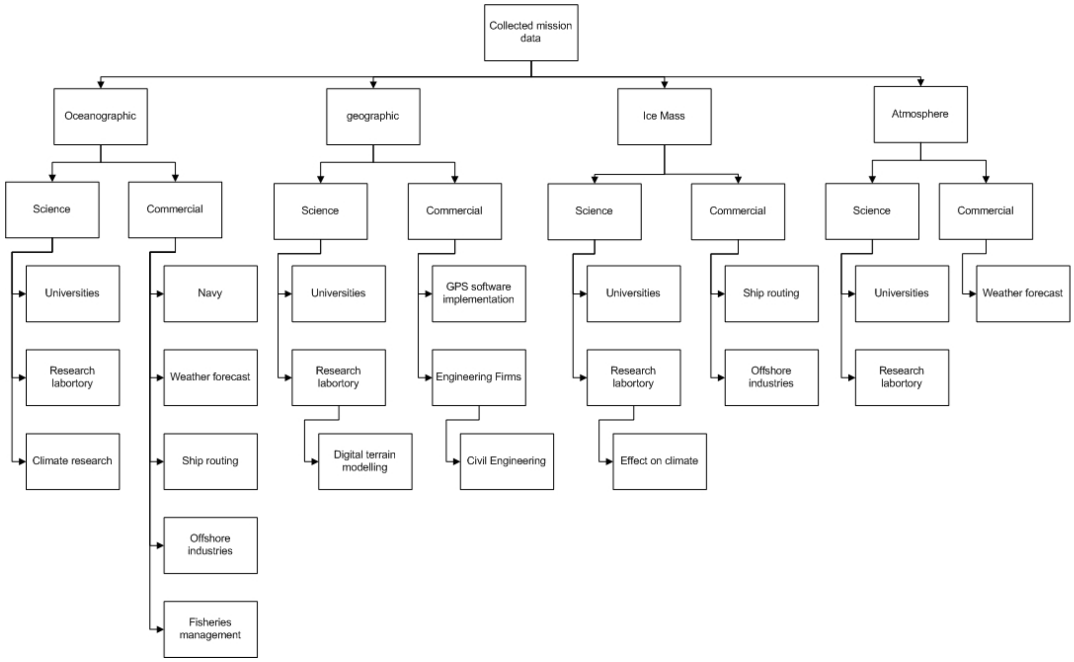

Figure 7.1: Market Breakdown Diagram with different data sets [3]

7.2 Investment and Profit Opportunities

Short term investments, in this case the investments to cover the preliminary costs as well as the costs for Integration, Assembly and Test (IAT), all the subsystems, and the implementation of the spacecraft in its orbit can be determined with a high accuracy, since their costs are non-recurring. Long term investments are communications with the satellite (constellation), personnel costs and ground station maintenance, they can be considered recurring (with a certain uncertainty). Since the risk of partial failure is severe in any space mission, the quality or the quantity of the data sets can decrease, lowering the ROI over time.

Depending on the actual operational lifetime of the satellite constellation (and especially, the lifetime of the emitter), the data volume over the years can generate a positive ROI. If the lifetime of the total system exceeds the determined lifetime upfront, the profit opportunities increase due to an increase in data volume.

Not only that, but a successful space mission could create trust in users for future missions, who will in turn be more eager to invest in long term projects.

Techniques like LiDAR are usually more attractive to the scientific community. Because of this, the potential financial profit is not that important. This laser altimetry mission is the most interesting for scientific organizations, like universities and research institutes. The ROI is then referred to as return in knowledge.

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 treated 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 57 gives a overview of subsystem contributions to satellite failures after different time span from a non-parametric analysis. A more detailed reliability analysis can be found in chapter 4.

8.2 Availability

The system is designed to be functional all the time to cover the entire Earth. To save power it is also possible to 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 are 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 considered separately as follows:

1. Preventive maintenance. During the regular system operation time, there is periodic maintenance and condition dependent maintenance. System software or simulator servicing maintenance of the ground station is mandatory and the 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 receivers is not functional, the system can relocate the rest of the receivers to decrease the influence of the failure. But if the laser emitter is broken, there is no way to perform the maintenance. Corrective maintenance is also used during measurement analysis 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.

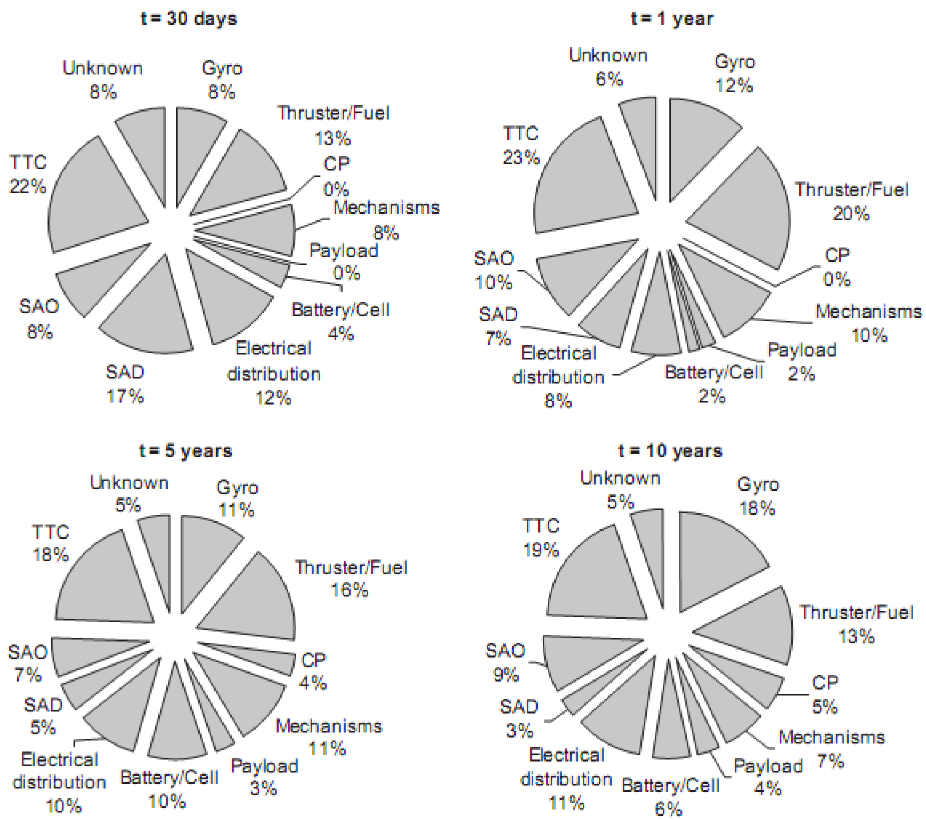

Figure 8.1: Subsystem contributions to satellite failures after 30 days, 1 year, 5 years, and 10 years in orbit [9]

Bibliography

[1] Lecture 6: Orbits.

[2] Ancillary description writer's guide. global change master directory., 2008.

[3] Ocean surface topography from space, Jul 2008.

[4] ICESat Anomaly Review Board. Independent glas anomaly review board executive summary. Electronic, 2003.

[5] Dave Doody. The basics of space flight., February 2001.

[6] M.S. Shur E. Borovitskaya. *Quantums Dots*. World Scientific Publishig, 2003.

[7] Peter Fortescue et al., editor. *Spacecraft Systems Engineering*. John Wiley & Sons Ltd., Chichester, West Sussex, England, 3 edition, January 2007.

[8] Douglas Isbell. Icesat press release c98-a.

[9] Joseph H.Saleh Jean-Francois Castet. Satellite and satellite subsystems reliability: Statistical data analysis and modeling. *Reliability Engineering and System Safety*, 94:1718–1728, 2009.

[10] W. Jongkind, Q.P. Chu, J.J. Wijker, A. Kamp, and B.T.C. Zandbergen. *Space Engineering & Techonlogy II Part B: Space Vehicle Engineering*. Delft University of Technology, September 2003. Lecture Notes.

[11] Priscilla Reale Jusef Hassoun, Stefania Panero and Bruno Scrosati. A new, safe, high-rate and high-energy polymer lithium-ion battery. *Advanced Materials*, 21:4807 − 4810, 2009.

[12] Tony R. Kuphaldt. All about circuits.

[13] Dr. Rdiger Paschotta. Photonics: Laser and optical physics, 2008.

[14] W. G. Rees. *Physical Principles of Remote Sensing*. Cambridge University Press, Cambridge, second edition, sixth printing edition, 2001.

[15] R. LaFlamme T.D. Ladd, F. Jelezko. Quantum computers. *Nature*.

[16] Wiley J. Larson & James R. Wertz, editor. *Space Mission Analysis And Design*. Microcosm Press and Springer, El Segundo, CA (Microcosm Press) & New York, NY (Springer), 3 edition, 2006.

[17] Zhizhan Xu. *Frontiers of laser physics and quantum optics:*. Springer, 2000.

Appendix A

Requirement Discovery Trees