Delft University of Technology

# Laser Swarm

Mid Term Review

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


**1 Introduction** **5**

**2 Technical Design Development** **6**

2.1 Project Approach Description . . . . . . . . 6

2.1.1 Group Procedures . . . . . . . . 6

2.1.2 Reporting . . . . . . . . 6

2.1.3 Project Outline . . . . . . . . 7

2.2 Work Flow Diagram . . . . . . . . 7

2.3 Work Breakdown Structure . . . . . . . . 7

2.4 Gantt chart . . . . . . . . 12

**3 Simulation** **13**

3.1 Introduction . . . . . . . . 13

3.2 Simulation . . . . . . . . 13

3.2.1 Orbit . . . . . . . . 13

3.2.2 Earth Model . . . . . . . . 14

3.2.3 Signal Path . . . . . . . . 15

3.2.4 Noise . . . . . . . . 16

3.3 Data Analysis . . . . . . . . 17

3.3.1 Altitude Determination . . . . . . . . 17

3.3.2 Bidirectional Reflection Density Function Determination . . . . . . . . 17

**4 Risk Management** **18**

4.1 Technical Resource Contingencies . . . . . . . . 18

4.2 Technical Risk Assessment . . . 20
4.2.1 Ground Segment . . . 20
4.2.2 During Mission . . . 21
4.2.3 Measurement Protocol . . . 24
4.2.4 Post-Mission . . . 26
4.2.5 Risk Control . . . 26

**5 Design Options** **28**
5.1 Attitude Determination and Control Subsystem . . . 28
5.1.1 Attitude Determination . . . 28
5.1.2 Attitude Control . . . 28
5.1.3 Note on Receiver Pointing . . . 29
5.2 Optical Emitting Payload . . . 29
5.2.1 Introduction . . . 29
5.2.2 Laser Characteristics . . . 30
5.2.3 Laser Types . . . 30
5.3 Electrical Power System . . . 31
5.3.1 The Power Source . . . 31
5.3.2 Power Storage . . . 32
5.3.3 Power Regulation and Control . . . 32
5.3.4 Power Distribution . . . 33
5.4 Communications . . . 33
5.4.1 Tracking . . . 34
5.4.2 Swarm Satellites Crosslink Frequency . . . 34
5.4.3 D/L and U/L Frequency . . . 35
5.4.4 Antenna Configuration . . . 35
5.4.5 Data Storage . . . 35
5.4.6 Communications Architecture . . . 35
5.5 Orbit Architecture . . . 36
5.5.1 Common Orbits . . . 36

5.5.2 Low Earth Orbit . . . . . . . . . . 37
5.5.3 Medium Earth Orbit . . . . . . . . . . 37
5.5.4 Geosynchronous Orbit . . . . . . . . . . 37
5.5.5 High Earth Orbit . . . . . . . . . . 38

**6 Sustainable Development Strategy** **39**
6.1 Production and Logistics . . . . . . . . . . 39
6.2 Operations . . . . . . . . . . 39
6.3 End of Life . . . . . . . . . . 40

**A Design Option Diagrams** **42**

**B Gantt Chart** **54**

# List of Acronyms

**ADCS** Attitude Determination and Control Subsystem

**ASTER** Advanced Spaceborne Thermal Emission and Reflection Radiometer

**BRDF** Bidirectional Reflection Density Function

**CMG** Control Moment Gyro

**DEM** Digital Elevation Model

**ECEF** Earth-Centered, Earth-Fixed

**EOL** End-of-Life

**EPS** Electrical Power System

**GDEM** Global Digital Elevation Model

**GEO** Geosynchronous Earth Orbit

**GLAS** Geoscience Laser Altimeter System

**GPS** Global Positioning System

**HEllO** Highly Elliptical Orbit

**HEO** High Earth Orbit

**JAT** Java Astrodynamics Toolkit

**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

**MTR** Mid Term Review

**OEP** Optical Emitting Payload

**ORP** Optical Receiving Payload

**PDS** Power Distribution System

**TDRS** Tracking and Data Relay Satellite

**TOD** True Of Date

**WBS** Work Breakdown Structure

**WFD** Work Flow Diagram

**WGS84** World Geodetic System 1984

# 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.

In this mid term review some options for this concept are proposed and traded off so as to find the best fit for the mission. This tradeoff is what the bulk of this report is about. Also described are the workings and results from the simulator, as well as the project and risk management. Chapters on operation and logistics, stability and control and sustainability are included too.

# Chapter 2

# Technical Design Development

In this chapter an update is given on the Technical Design Development as given in the Project Plan. All charts and documents have been updated. First the project approach description is updated in section 2.1. Work Flow Diagrams and Work Breakdown Structures are updated in sections 2.2 and 2.3, whereas the Gantt chart is revisited in section 2.4.

## 2.1 Project Approach Description

### 2.1.1 Group Procedures

The DSE project is approached by first establishing specific roles for the group members, so that every group member is assigned a clearly defined managerial and technical function. After this the group operational procedures are defined. They are as follows:

1. The Chairman will lead a 'scrum' meeting every morning upon arrival of all members to establish what everyone has done the day before and what they will be doing the day of the meeting. This is done in order to keep all members up-to-date with all aspects of the project. The meeting concludes with updates on any external communications (with organizations and teaching staff) as well as any other points relevant at that time.
2. When done, groups responsible for certain design tasks will present their results to the rest of the team.
3. The team meets with tutor and coaches at least weekly.
4. Everyone is present at The Fellowship between 09:00 and 17:00 every workday, except for a 45 minute lunch break.
5. Upon completion of a deliverable, a meeting is conducted to establish a plan for the next deliverable.

### 2.1.2 Reporting

The reporting is done in LaTeX. There is a main report file which contains the layout of the report and the references to other files that contain the chapters, sections, figures, tables and other documents required for the report. When the file is compiled and printed it will show the entire report.

This has the advantage that work can be easily divided among group members, and any change made to a file will not influence the rest of the report. The file sharing is performed using Subversion (SVN). SVN not only allows file sharing, but it automatically assigns versions to a document and keeps track of changes. The repository is hosted with Google™Code.

### 2.1.3 Project Outline

The official start of the DSE project is the establishment of the Mission Need Statement. At this point all members should be aware of the main goal of the assignment.

The design process is started by defining the tasks in the project plan, then finding the requirements and functions. From the requirements, a set of design options will be created for the Mid Term Review (MTR). In the MTR a trade-off will be made based on an extensive functional and risk assessment. After the MTR, work on the detailed design can begin. At this stage all subsystems will be given a careful consideration in terms of cost, mass and power budgets. Final decisions on detailed parameters and variables will be made. Leading up to the Final Review (FR), the feasibility study can be concluded.

Parallel to the design phase, the simulator software will be developed by a team of 3 to 4 people, depending on workload and time available. This software should be able to perform calculations accurate enough to aid the trade-off scheduled before the MTR. In this chapter the project planning will be revised. More specifically the Work Flow Diagrams (WFDs) and Work Breakdown Structures (WBSs) of the midterm and final reports and the Gantt chart. These have been updated because now it is more clear as to which tasks have to be performed for the remainder of the project. Also, the work flow for the simulator has been added to the charts. In sections 2.2 the revised Work Flow Diagram will be shown. This is followed by section 2.3, which shows the updated Work Flow Diagram. The final section gives the Gantt chart, our timeline for this project.

## 2.2 Work Flow Diagram

The tasks to be done on the simulator have been updated and presented in more detail. Some tasks have also been changed in the WFDs: they now start earlier or later so as to better describe the work flow of the project. The tasks in the green and blue boxes respectively describe the simulator design finalization and tradeoff execution in more detail. Red boxes are tasks which are explicitly needed for the Mid Term Review (MTR). The updated WFD of the MTR is given in figure 2.1, page 8.

The WFDs of the final report have also been updated. In retrospect, a very important part if the final report was not present in the WFDs: perfecting the design and the feasibility determination. Now these tasks have been added to the daigram to make it complete. As with the WFD of the MTR, all boxes in red are explicitly required for the final report. The diagram can be seen in figure 2.4, page 11.

## 2.3 Work Breakdown Structure

The WBSs have been updated like the work flow diagrams. Some extra and other more detailed tasks defined in the WFDs have also been added to these structures. Also, the layout has been changed somewhat to improve readability and correctness. The updated WBSs can be seen in figures 2.3 and 2.4, starting on page 10.

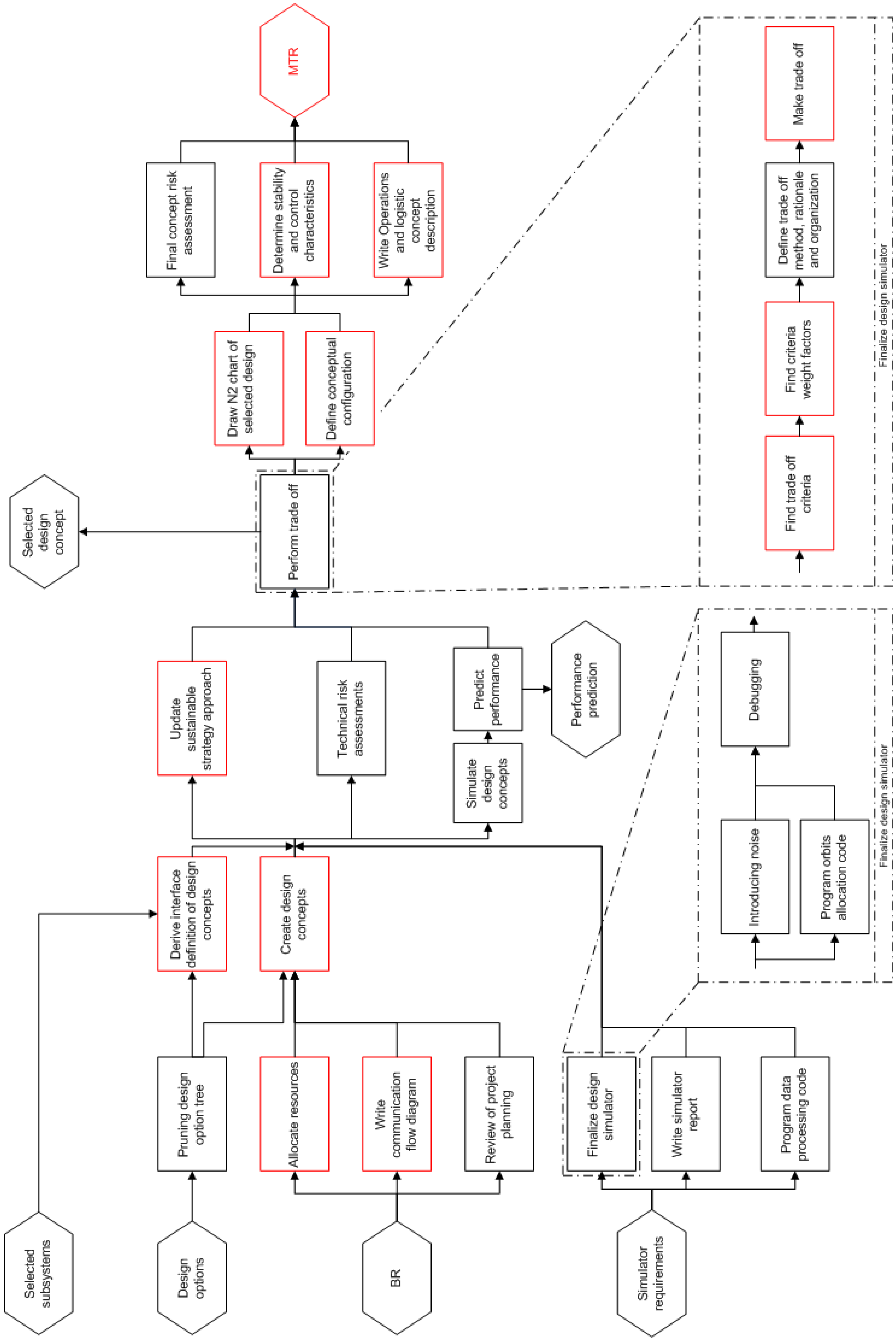


Figure 2.1: Updated work flow diagram for the mid-term report

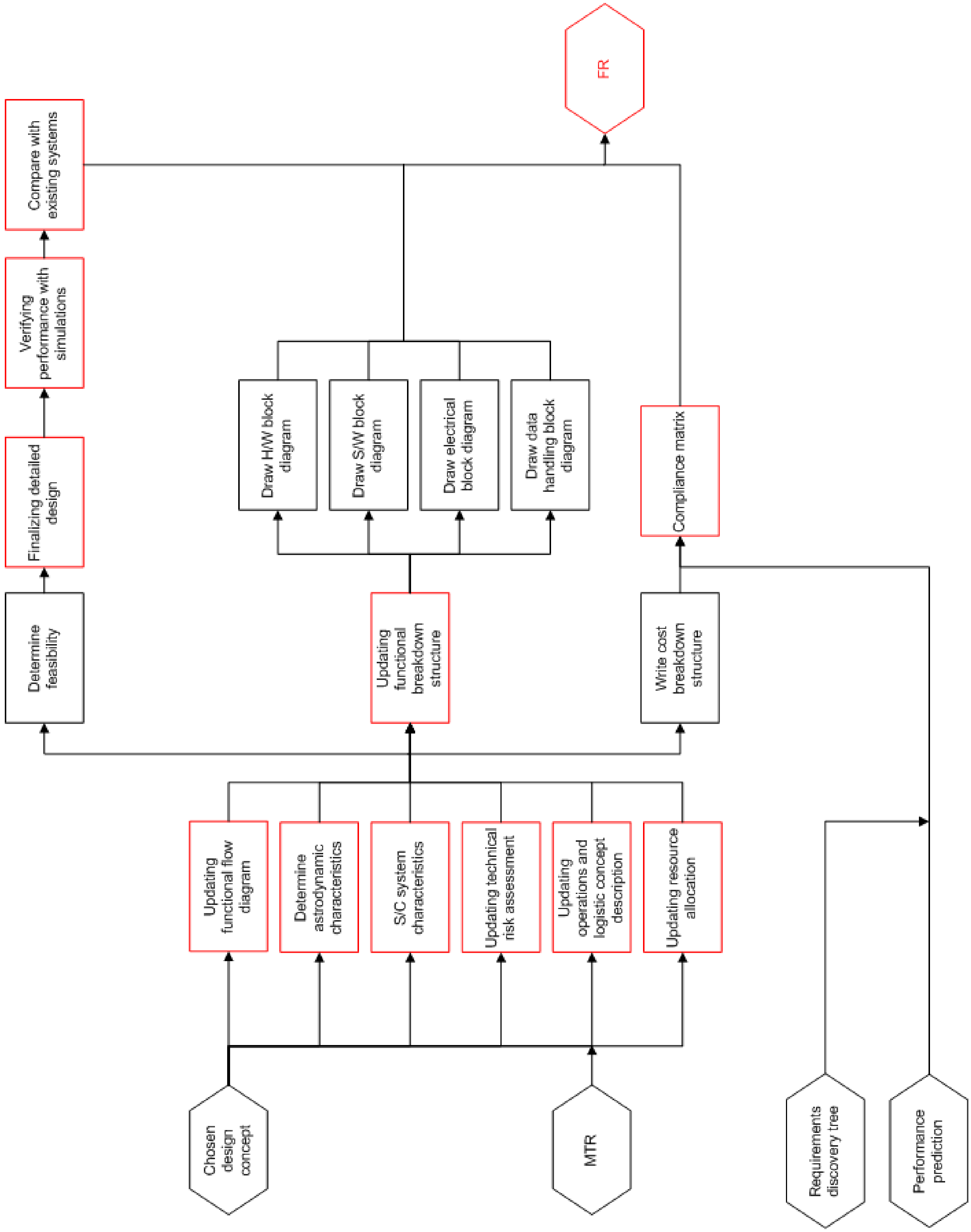


Figure 2.2: Updated work flow diagram for the final report

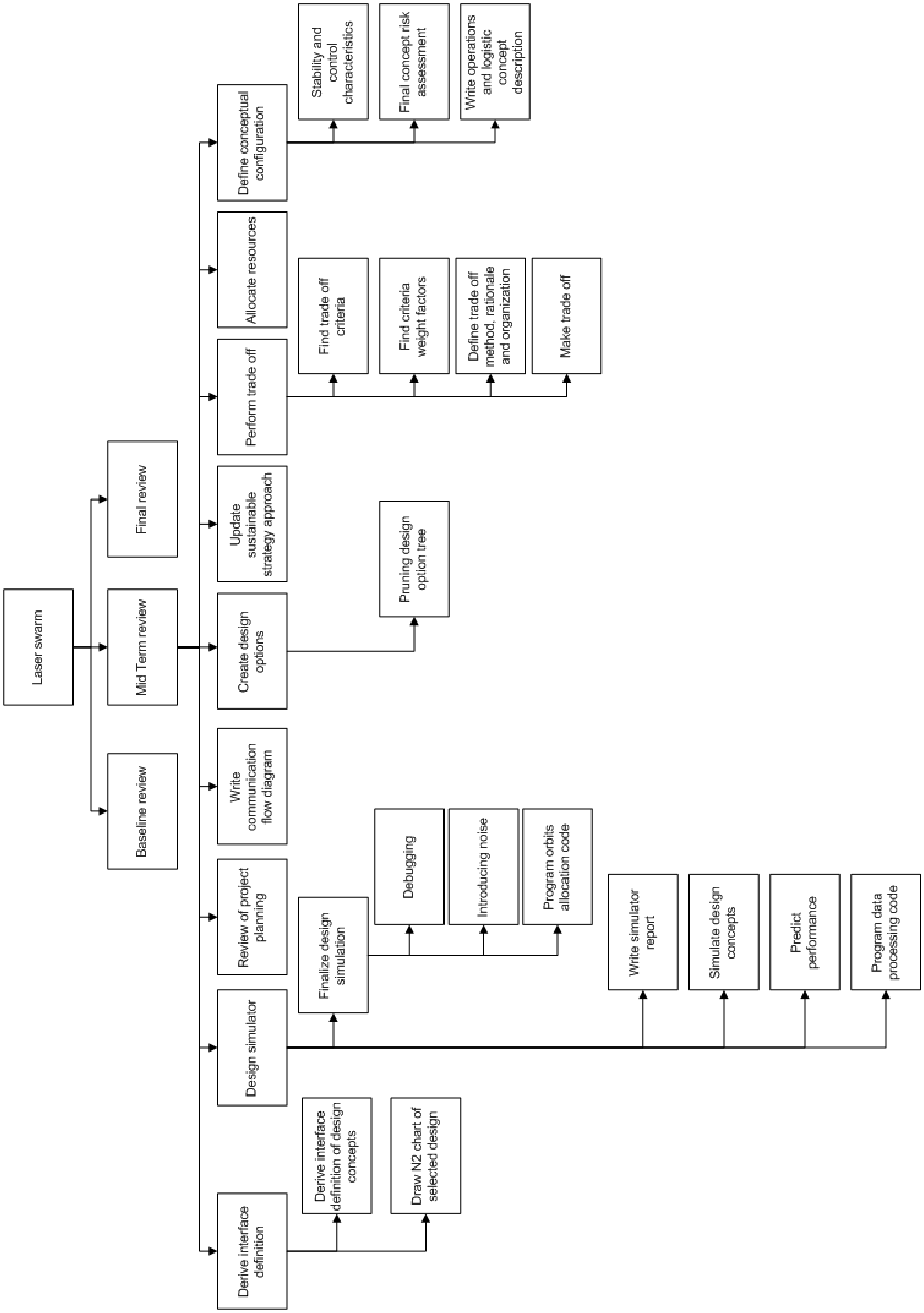


Figure 2.3: Updated work break-down structure for the mid-term report

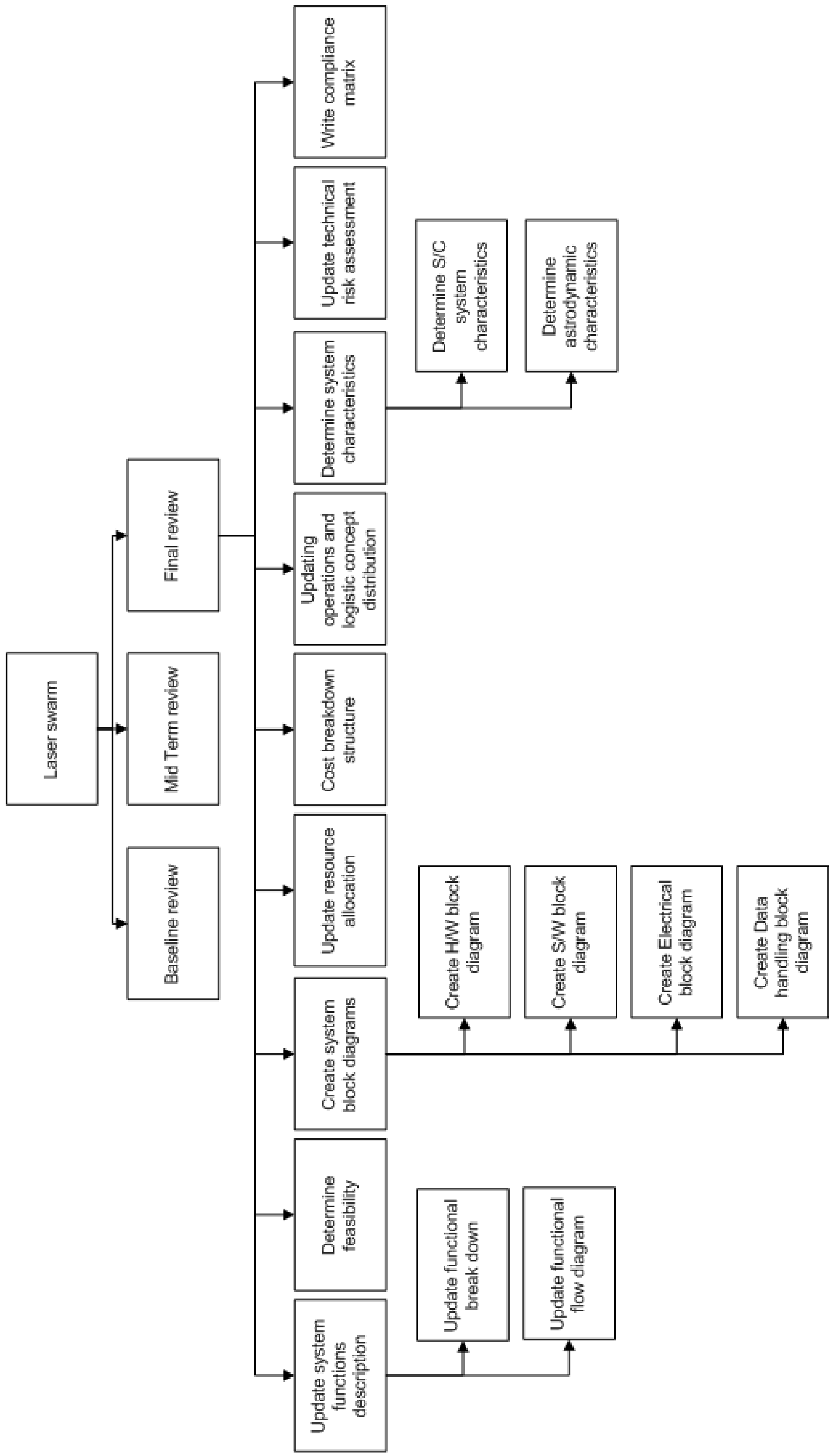


Figure 2.4: Updated work break-down structure for the final report

## 2.4 Gantt chart

As the WFDs and the WBSs have been updated, so the Gantt chart has also been updated. Now that we are further along in the project, it is easier to estimate which tasks need to be done for the mid-term and final reports. The estimated duration of these tasks also was much easier to estimate. The Gantt chart has been updated to contain all tasks set in the WFDs. Special care was taken to make sure the Gantt chart is consistent with the WFDs and the WBSs. This time the simulator tasks were not separated from the rest of the project because the simulator is more involved with the rest of the project for the mid-term and final reports. The updated Gantt chart can be found in appendix B, page 54.

# Chapter 3

# Simulation

## 3.1 Introduction

This report preliminarily discusses the design and workings of a software tool developed with the purpose of guiding and validating the results of the tradeoff done in the main laser swarm project, as well as demonstrating the feasibility of the concept of the laser swarm.

Generally speaking, the simulator program is divided into two parts. The first part is the simulation of the physical path of the photons in the laser pulse up to the point were they are received by the receiving satellites. This part has been documented in section **??**.

The second part is the analysis of the received photon data, to determine terrain height and the terrain Bidirectional Reflection Density Function (BRDF). Obtaining this data is what the software and in general this feasibility study is about. This part is described in section 3.3.

## 3.2 Simulation

In this section the simulation of the satellites, moving through space and sending and receiving laser pulses, is discussed. In subsection 3.2.1 the emitter and receiver satellite orbits are considered. In subsection 3.2.2 the way the Earth's surface was modeled is explained. In subsection 3.2.3 the path of the signal is examined, whereas in subsection 3.2.4 the introduction of noise into the signal is disclosed.

### 3.2.1 Orbit

The orbit of each satellite in the constellation is defined by means of six Keplerian elements. These define the shape of the orbit, the orientation of the orbit with respect to the center of the Earth and the position of the satellite on the orbit.

As there are a number of rotating bodies, three reference frames are used in order to facilitate the process of locating the satellites in space. The three reference frames used in the simulation are described below.

The first is True Of Date (TOD). Its X-axis points towards the direction of the vernal equinox and its Z-axis coincides with the axis of rotation of the earth. TOD takes into account nutation and precession of the earth. For practical reasons, it does not rotate with respect to the Sun.

In the second place there is Earth-Centered, Earth-Fixed (ECEF). Its X-axis points towards 0° latitude and 0° longitude. The XY plane lies in the plane of the equator. Its origin is in the center of mass of the Earth.

Thirdly there is the PQW. The P-axis points towards the perigee of the orbit, the PQ plane lies in the plane of the orbit and the W-axis is perpendicular to the plain of the orbit. The origin of the frame is at the focal point.

The program converts between the before-mentioned reference frames for the user's convenience.

The position of the satellite is defined with respect to the TOD reference frame. Kepler's equations are solved for a certain time to determine the position. The orbit is determined for every satellite in the constellation. The orbit is assumed to be perfect, meaning that its orientation and shape do not change: perturbations are not considered.

The Earth's rotation about its own axis and around the Sun is simulated in order to provide a more realistic simulation environment. From the rotation of the Earth around the Sun, the sun vector is deduced; this is used in noise calculations.

Most of the functions are adapted to the simulator from the Java Astrodynamics Toolkit (JAT) library.

### 3.2.2 Earth Model

The Earth model is the digital representation of the Earth surface. It stores a Digital Elevation Model (DEM), and the scattering characteristics of the local terrain. In section 3.2.2, the DEM implementation will be elaborated and section 3.2.2 describes the way incoming radiation is scattered.

#### Digital Elevation Model

A Digital Elevation Model is a digital representation of a topographic surface. For the ground representation in the simulator a few tiles of the Advanced Spaceborne Thermal Emission and Reflection Radiometer (ASTER) Global Digital Elevation Model (GDEM) were used. This DEM was created using the complete history of the ASTER instrument launched on the Terra satellite. This DEM has a spatial resolution of one arc-second and a vertical accuracy of $7 - 14$ meters. The elevations of the intermediate points were obtained using nearest-neighbor interpolation.

The ASTER GDEM elevations are expressed with respect to the World Geodetic System 1984 (WGS84) ellipsoid. To simplify the internal simulator, the DEM tiles were projected to the EPSG:3857 spheroid. The projection is done using the GeoTools Java toolkit [geo(2010)].

The digital elevation is used to compute the total distance (and thus the time) that the laser pulse travels. Because scattering is dependent on the terrain normal, this normal is derived form the DEM, using the two perpendicular surface gradients that can be extracted from the DEM.

#### Scattering Model

Scatter is the physical process where incident radiation is absorbed and reflected back towards the atmosphere. To this end a scattering model is used. This scattering model is a way to construct a BRDF. A BRDF is a distribution of the incident light over the hemisphere [Rees(2001), pages 47-49]. An example is shown in figure 3.2.2, page 15.

The most basic example of a BRDF is the Lambertian model [Rees(2001), pages 49-50]. This model assumes a homogeneous perfectly rough surface, causing a homogeneous scattering distribution. Apart from the index of refraction of the air, the incident radiation vector and the exittant radiation vector (which are all known), use of Snell's law is needed to find Fresnel's coefficients, thereby requiring the index of refraction of the ground.

A modification that can be made to take into account the tendency of surfaces to scatter more in the direction of the surface normal, is called the Minnaert model [Rees(2001), page 50]. It causes a more elliptical BRDF. It depends on the Minnaert parameter $\kappa$.

Another important modification is the Henyey-Greenstein term. It accounts for the tendency of surfaces to back- or forwardscatter [Rees(2001), page 51]. This rotates and deforms the elliptical BRDF obtained from the Minnaert model. The Henyey-Greenstein term is parameterized by $\Theta$. The final result is shown in figure 3.2.2, page 15.

This is the scattering model used in the program. Because there is no data from which all three parameters can be accurately determined, a coefficient map was made up. It does not matter much what the precise form of the BRDF used in the simulation is, so long as it can be retrieved.

With the help of the formulae from [Rees(2001), pages 43-51], the incidence vector can be taken and the number of photons radiated in a specific direction can be calculated. Note that the program does not integrate over part of the sphere: because the angle of the cone is very small, the BRDF is assumed to be constant over the cone. The error induced here is worth avoiding the computationally expensive integration.

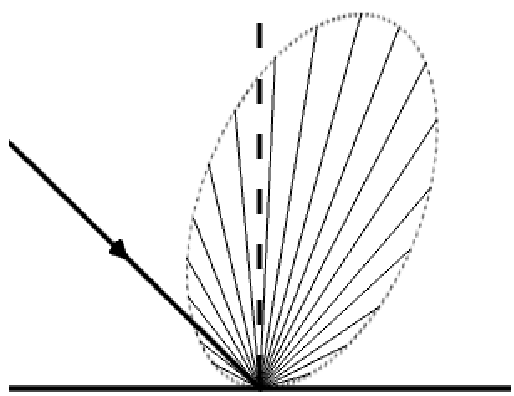

Figure 3.1: Example of a BRDF

### 3.2.3 Signal Path

The simulated path that the Light Amplification by Stimulated Emission of Radiation (laser) signal follows is visualized in figure 3.2.3 on page 15. There are three distinct phases. The first one is the travel of the signal down through the atmosphere. This is followed by the scattering on the Earth's surface. Finally the pulse has to travel back up through the atmosphere. This sequence is elaborated on in more detail below.

The signal originates from the emitter. For each pulse, the emitter position is determined from a set of Kepler elements. The energy in the pulse is found by distributing the emitter power over a constant number of pulses of a given duration.

The signal then starts to propagate through the atmosphere. The atmosphere affects the signal in several manners, but the most important one is the attenuation of the signal. Attenuation is the only disturbance by the atmosphere taken into account. The pulse energy exponentially decays with travel distance through the atmosphere. Furthermore also the optical thickness of the atmosphere is taken into account.

Then the intersection of the pulse with the DEM is computed. As a simplification in this process, the intersection of the pulse (i.e., the ray) with a sphere is computed. The sphere has a radius of the average terrain height of the DEM tile plus the Earth radius. Then the ray-sphere intersection point coordinates are converted into latitude and longitude. These are then used to find the actual terrain elevation from the DEM and the three-dimensional position.

Then the scattering characteristics are constructed. For this, the terrain normal and the inbound laser pulse vector are used. The power of the emitted pulse is now distributed over the entire footprint area of the emitter and then scattered back using the scattering technique described in section 3.2.2. The backscattered energy is computed separately for every receiver satellite.

The reflected pulse now travels back through the atmosphere. More attenuation takes place along this path. The energy received by the receivers now depends on the receiver aperture. The received energy can then be converted into photons by dividing the energy received by the energy per photon.

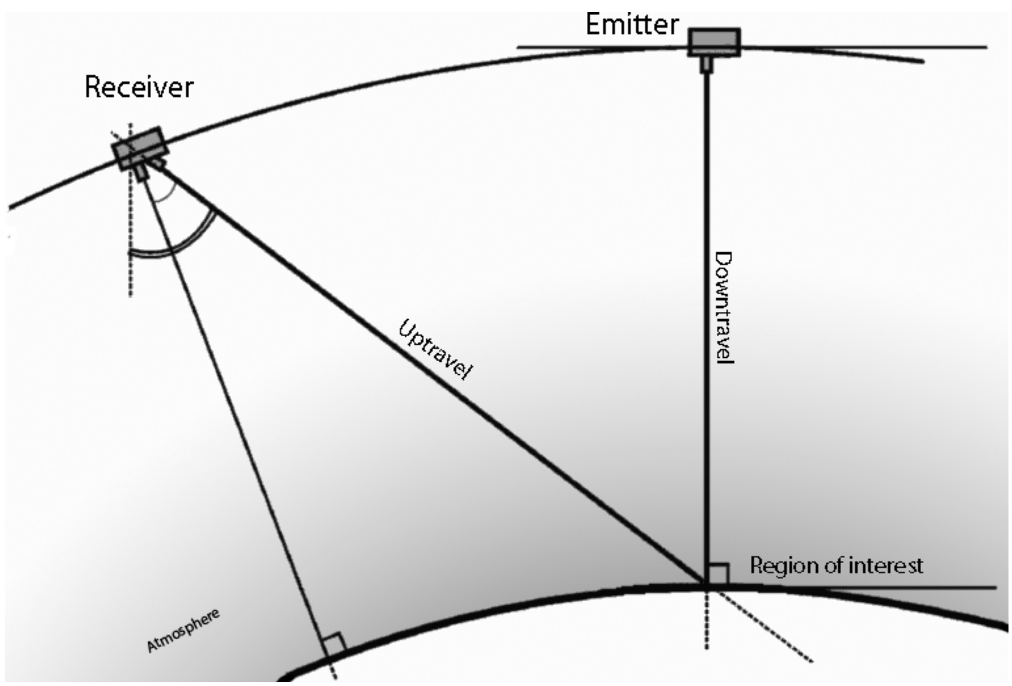


Figure 3.2: Signal path representation

### 3.2.4 Noise

The main source of noise in the system comes from the Earth's surface. This includes sources on the Earth such as lights along highways and reflected radiation, i.e. the Earth's albedo. The amount of radiation that is received by the receivers depends on the footprint of the receivers. This is an ellipse created by the intersection of the cone originating from the receiver with the terrain. A simplification is made in that the DEM is assumed to be a flat area with the elevation of the center point.

The amount of power emitted per square meter is dependent on the illumination of the Earth by the Sun. If the Sun illuminates the footprint, the emitted power is the scattered power of the Sun in the receiver detection wavelength bandwidth. This power can be found by integrating Planck's law over the detection spectrum and the solid angle the Sun subtends to the receiver footprint on Earth. If the Sun does not illuminate the receiver footprint, the only contribution to noise is the Earth's greybody radiation, which is also taken into account.

While noise propagates through the atmosphere it is also attenuated. This attenuation is computed in the same way the signal attenuation is; see section 3.2.3.

## 3.3 Data Analysis

In this section the analysis of the received photon data taken care of in the software tool is elaborated on. The analysis consists of two parts: the terrain altitude determination and the BRDF determination. The first is expounded in subsection 3.3.1, the second in 3.3.2.

### 3.3.1 Altitude Determination

In order to tackle the problem of decrypting the raw data and converting it into a coherent terrain model, various statistical techniques need to be employed. The basic principle behind the altitude determination of the terrain is as following. First the time difference between generation and reception of the pulse is registered. As the position of the emitter at the time when the pulse was sent is known, just like the position of the receiver at the time the pulse was received is, the distance to ground can be determined. In theory only one emitter and one receiver are necessary to determine the altitude, but due to various uncertainties in position and noise characteristics of the received data, more receiving satellites are necessary to obtain a reliable and complete terrain representation.

In the simulator the emitter is modelled such that it records the time when the pulse was sent. The receivers are modelled such that they register the time and intensity (in photons) of the received pulse. The problem is that not all emitted pulses are registered, and sometimes noise, which does not correspond to any emitted pulse, is registered.

One of the ways to solve this problem is to use multiple receivers. The noise could be identified and removed by means of looking for a spike in the received photons for the whole swarm within a certain time range (usually twice the time it takes for a light beam to travel the orbit altitude). This data could be filtered by means of correlating the distance of the receiver to the Earth center and the time of the received pulse. Usually, the larger the distance, the larger the time difference. This method helps to eliminate outliers. Since the footprints overlap, the measured altitudes could be further smoothed out by means of a running average.

### 3.3.2 Bidirectional Reflection Density Function Determination

The BRDF returns the ratio of the radiance to irradiance for a given angle perpendicular to the surface. In theory, it is possible to measure it by means of the received photons of all of the receivers for a given pulse, where each received photon indicates the relative intensity. From the position of the receivers the direction of the irradiance vector can be calculated. Having the direction and intensity of the reflected radiation a segment of the BRDF for a specific incident angle could be reconstructed. By making a multiple passes over the same region, a partial BRDF for different incidence angles can be recorded. If the data is interpolated, a complete BRDF can be determined.

# Chapter 4

# Risk Management

## 4.1 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 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 4.1 and 4.2 on page 19.

| Design maturity | Contingencies [%] | | | | | | |
|---|---|---|---|---|---|---|---|
| | 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 4.1: The mass contingency allowance

| Design maturity | Contingencies [%] | | | |
|---|---|---|---|---|
| | 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 4.2: The power contingency allowance

## 4.2 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.

### 4.2.1 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.

### 4.2.2 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 Attitude Determination and Control Subsystem (ADCS) systems.

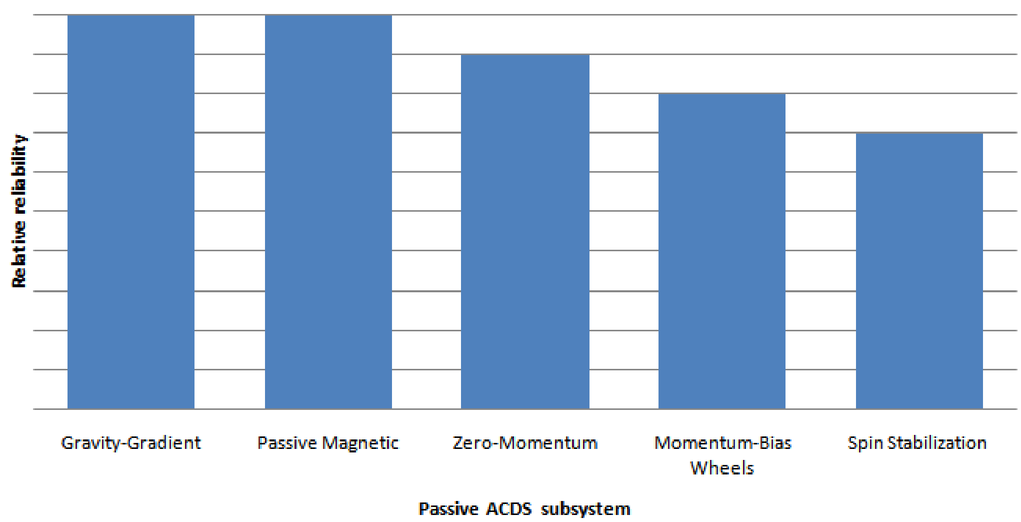


Figure 4.1: Relative reliability of passive ADCS subsystems.

## 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.

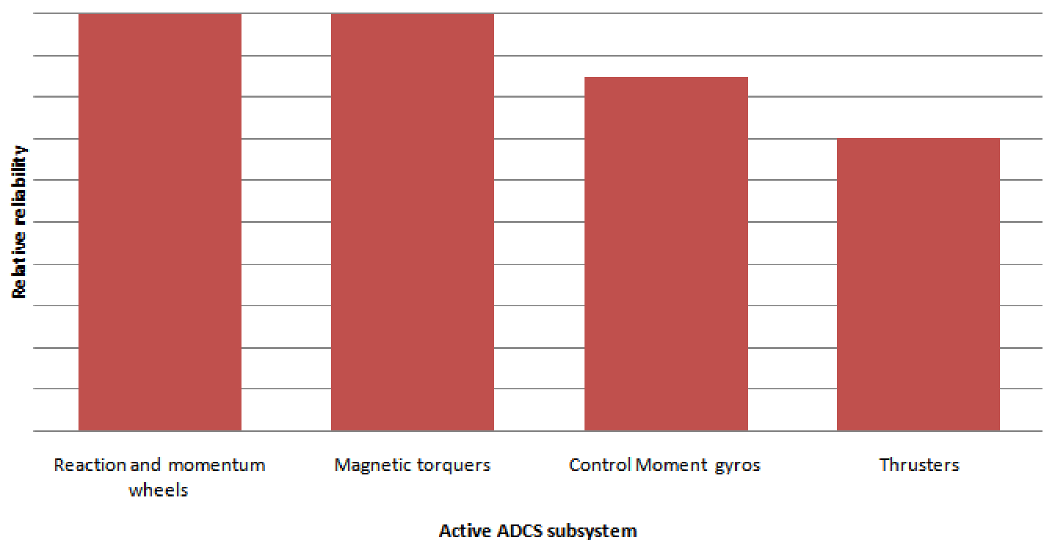


Figure 4.2: Relative reliability of active ADCS subsystems.

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.

### 4.2.3 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.

### 4.2.4 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.

### 4.2.5 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.

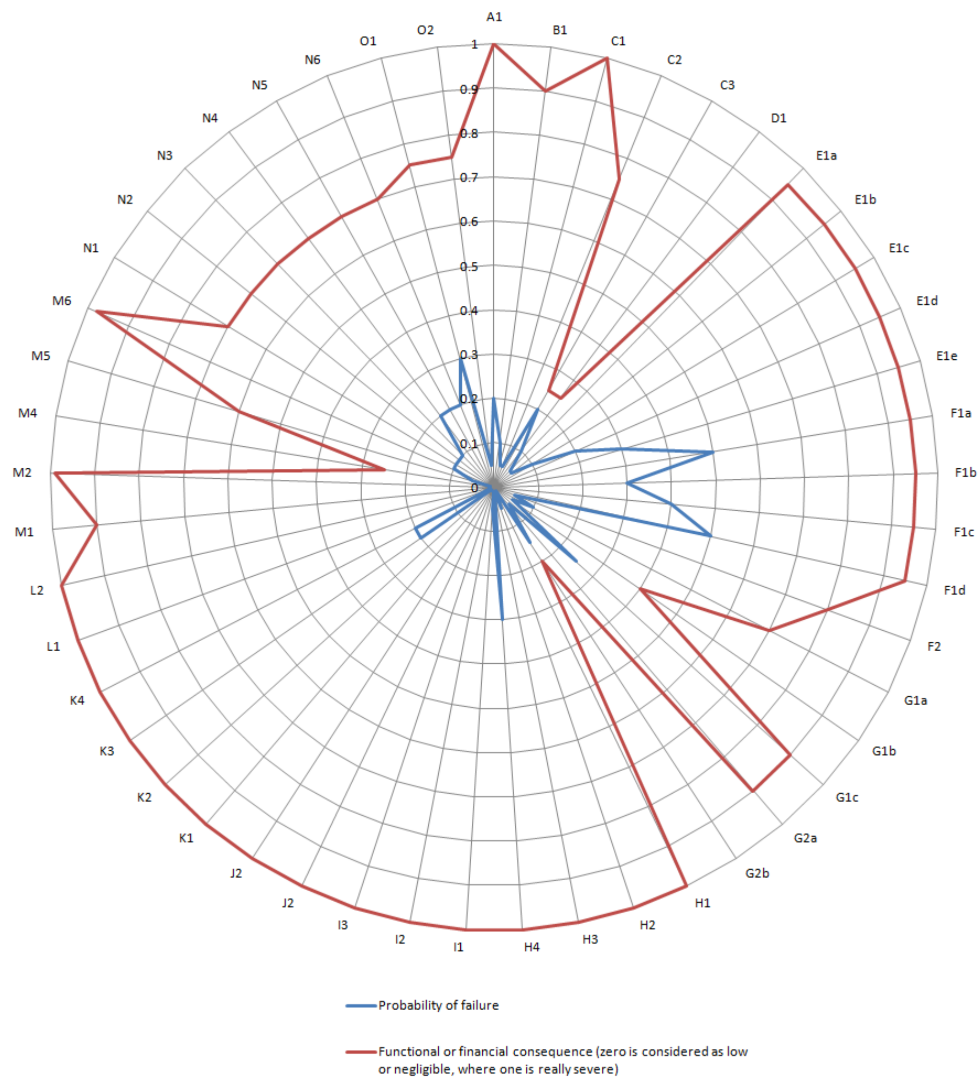


Figure 4.3: Technical Risk Map

# Chapter 5

# Design Options

In this chapter the design options are repeated from the baseline report. This is done so that the reader gets the whole picture from options to the design finally chosen. This is done for the ADCS, the telecommunications subsystem, the Electrical Power System (EPS), the orbits, the Optical Emitting Payload (OEP) and the Optical Receiving Payload (ORP), in subsections 5.1 to **??**. The full text has not been included, and the diagrams have been moved to appendix A.

## 5.1 Attitude Determination and Control Subsystem

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 A.1 on page 43. 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 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 Optical Emitting Payload

### 5.2.1 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 on 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 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.

### 5.2.2 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 [Xu(2000)].

### 5.2.3 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. [Paschotta(2008)]

Figure A.10 on page 51 gives the final design option tree of the laser emitter.

## 5.3 Electrical Power System

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 A.3, A.4 and A.9 (pp. 45-50) show the complete design option tree for the EPS.

### 5.3.1 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 [Rees(2001)]. 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 [Doody(2001)], but production efficiencies are around 22% [Wertz(2006)]. 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 [Wertz(2006)].

### 5.3.2 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 [Wertz(2006)].

Lithium-Ion batteries offer a significant advantage over nickel based ones. According to [Wertz(2006)], 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 [Jusef Hassoun and Scrosati(2009)].

### 5.3.3 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 5.3.4). 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 End-of-Life (EOL).

### 5.3.4 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 [Kuphaldt()]. 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.

## 5.4 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

### 5.4.1 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 $3\sigma$ 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.

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. $3\sigma$ accuracies typically are about several kilometers for LEO.

### 5.4.2 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.

### 5.4.3 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 [Wertz(2006)], p. 566.

### 5.4.4 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 [Wertz(2006)], p. 573.

### 5.4.5 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 [et al.(2007)].

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.

### 5.4.6 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 A.2 on page 44.

## 5.5 Orbit Architecture

In this section the design options in the orbit architecture tree as seen in figures A.6, A.7 and A.8 (pp. 48-48), 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 ($\Omega$), the argument of perigee ($\omega$) and finally the true anomaly ($\nu$). As a reference for the last four terms, figure A.5 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.

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.

### 5.5.1 Common Orbits

Some special orbits are repeated multiple times in the orbit architecture trees as shown in figures A.6 to A.8 (pp. 48-48). 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 $\pm 90$ degrees. This setup allows for near global coverage, and is useful for polar ice sheet research.
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 [Wertz(2006)].
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.

### 5.5.2 Low Earth Orbit

The altitude of a satellite in LEO ranges from approximately 80 to 2000 km [nas(2008)]. 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 [et al.(2007)]. Figure A.6 shows the design option tree for the LEOs.

### 5.5.3 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 5.5.1. Figure A.6 shows the design option tree for the MEO.

### 5.5.4 Geosynchronous Orbit

Figure A.7 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 5.5.1 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 [jer()]. 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.

### 5.5.5 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 A.8 shows the design options for the HEO, meanwhile the HEllO discussed in the next subsection is part of this figure.

#### 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.

# Chapter 6

# Sustainable Development Strategy

In this chapter a Sustainable Development Strategy is discussed in the order of production (section 6.1, page 39), operations (section 6.2, page 39) and end of life (section 6.3, page 40).

## 6.1 Production and Logistics

The design is aimed at a swarm of mostly identical satellites. This may allow for series production which is more efficient in terms of resources than a one-of large satellite 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 into two parts: transportation to the launch site and the launch from the surface to the final orbit in space. On both occasions the system can again profit from its small size. If the satellites are not launched all together, they can piggyback on another satellite's launcher.

Spreading the swarm, i.e. piggybacking using different launchers, has several advantages. First of all the emissions are lower than in case of a dedicated launcher. Also, 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, the satellite's 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 orbiting the Earth. The deployment mechanism however, which is responsible for most of the debris, is not included in this technical feasibility study. Later studies developing the ideas from this feasibility study should keep an eye on it, since more satellites could mean more deployment mechanisms and hence more waste. 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, whilst any remaining satellites can be reused.

## 6.3 End of Life

Each satellite will be at the end of its life if it cannot perform its function anymore. 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 do not 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.

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# Appendix A

# Design Option Diagrams

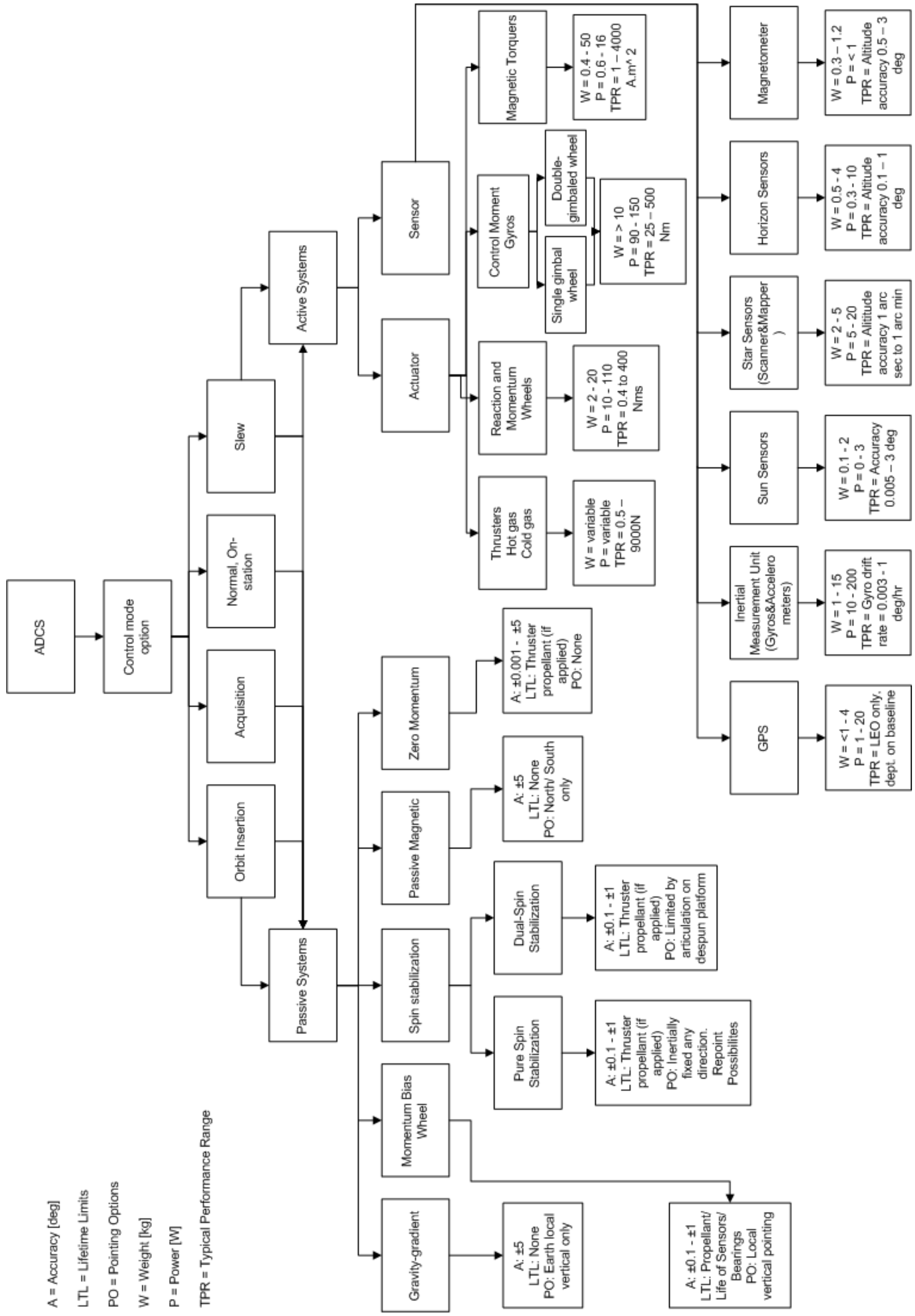


Figure A.1: Design option tree for the ADCS

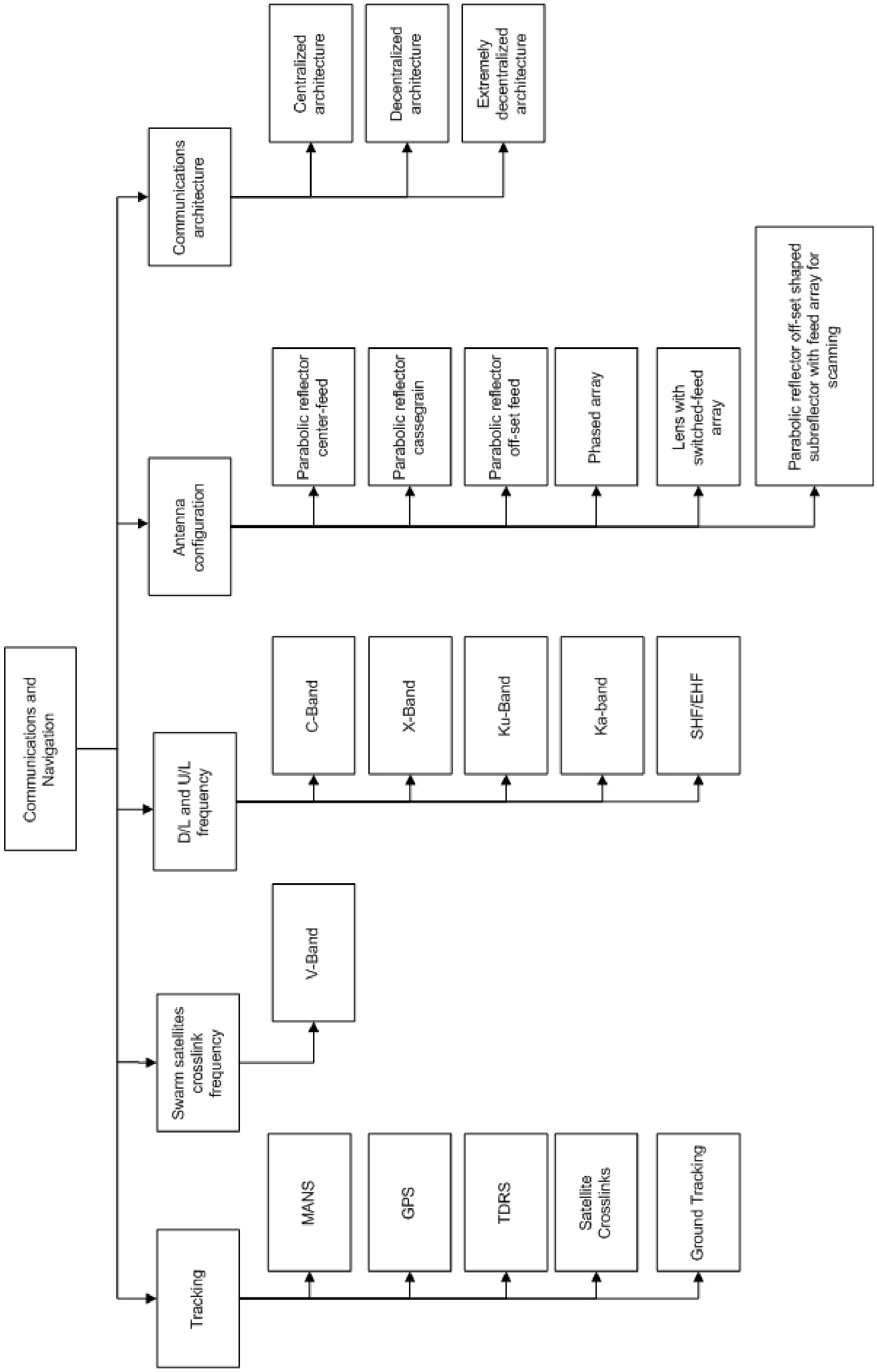


Figure A.2: Design option tree for the communication systems

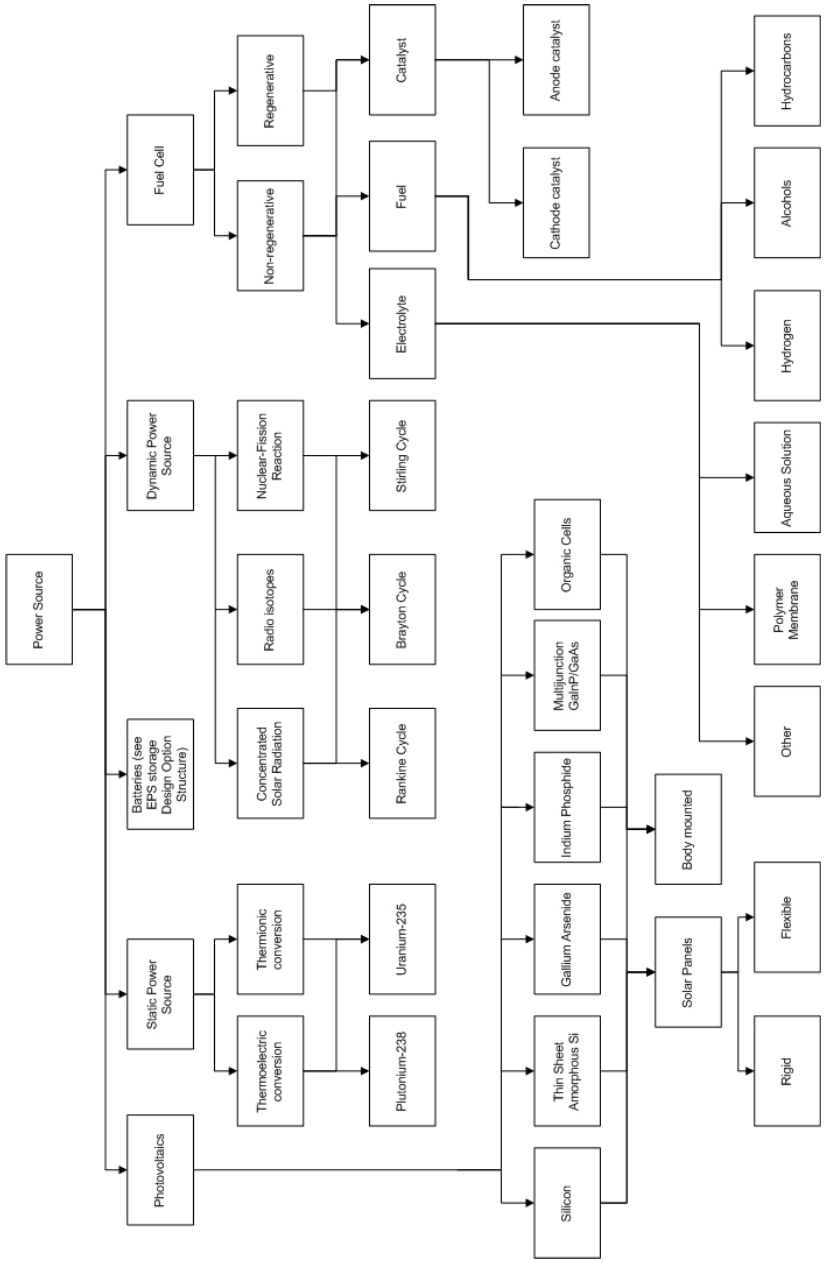


Figure A.3: Design option tree for the power source

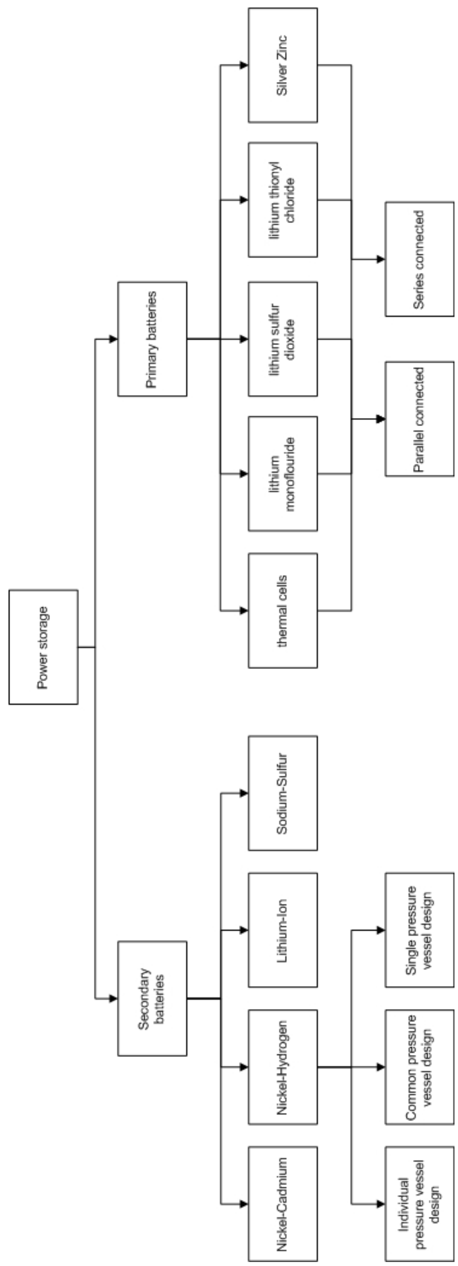


Figure A.4: Design option tree for the power storage

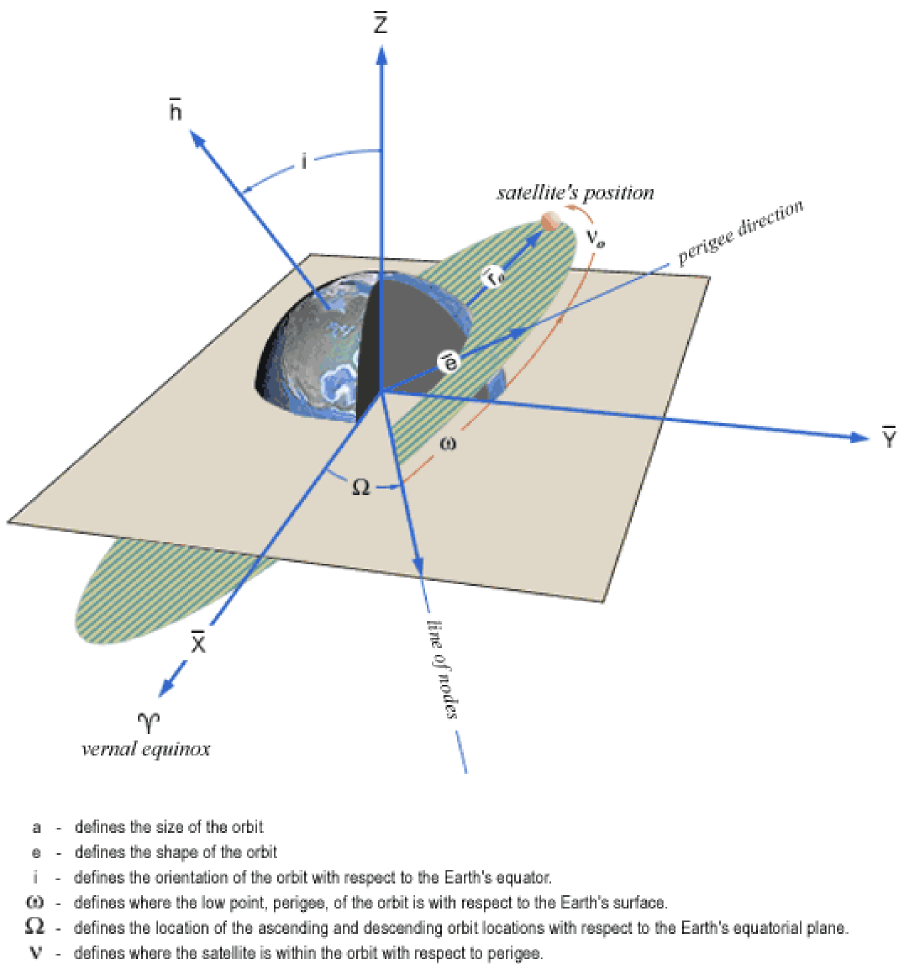


Figure A.5: Definitions of the inclination $i$, the right ascension of the ascending node $\Omega$, the argument of perigee $\omega$ and the true anomaly $\nu$. *source: http://reentrynews.aero.org*

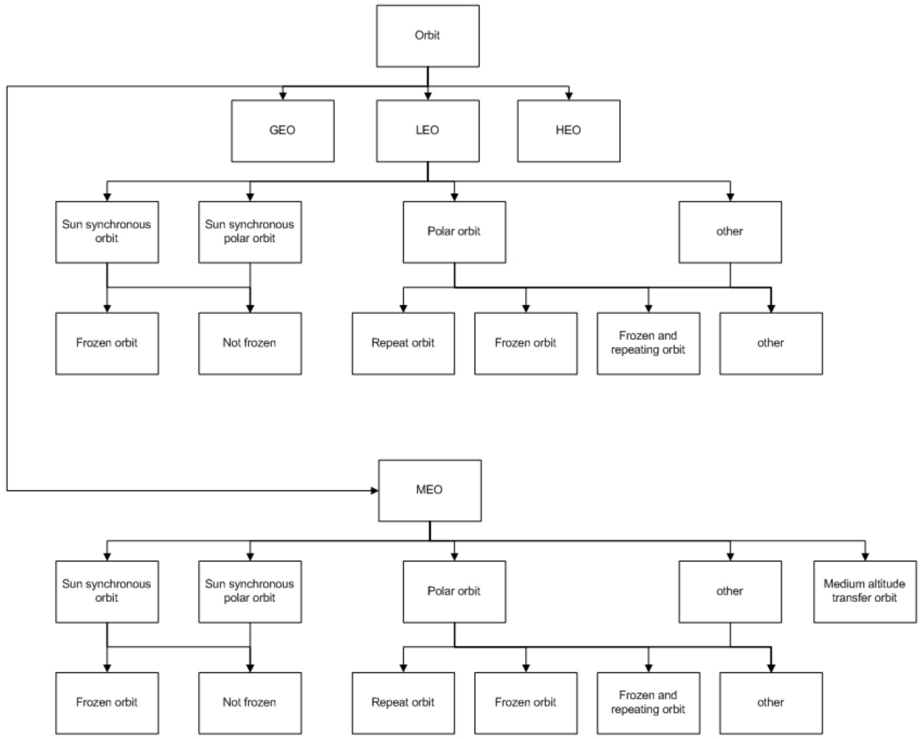


Figure A.6: Design option tree for the orbit architecture of LEO and MEO

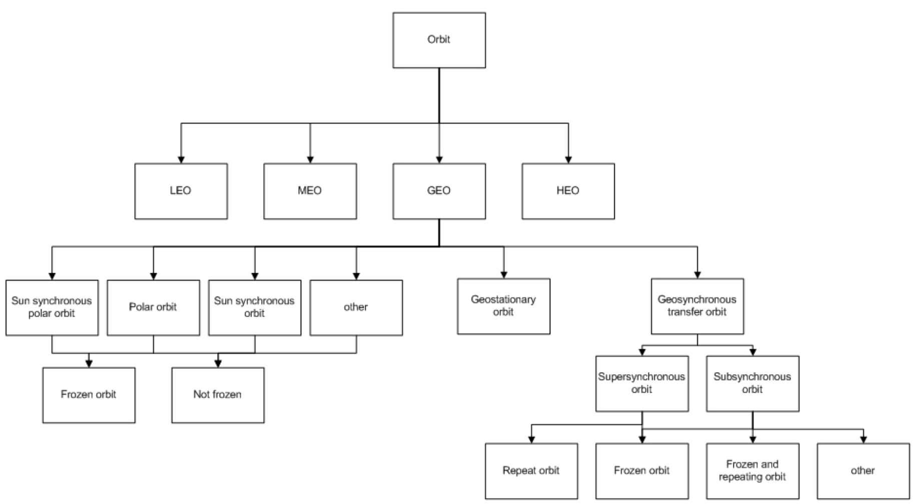


Figure A.7: Design option tree for the orbit architecture of GEO

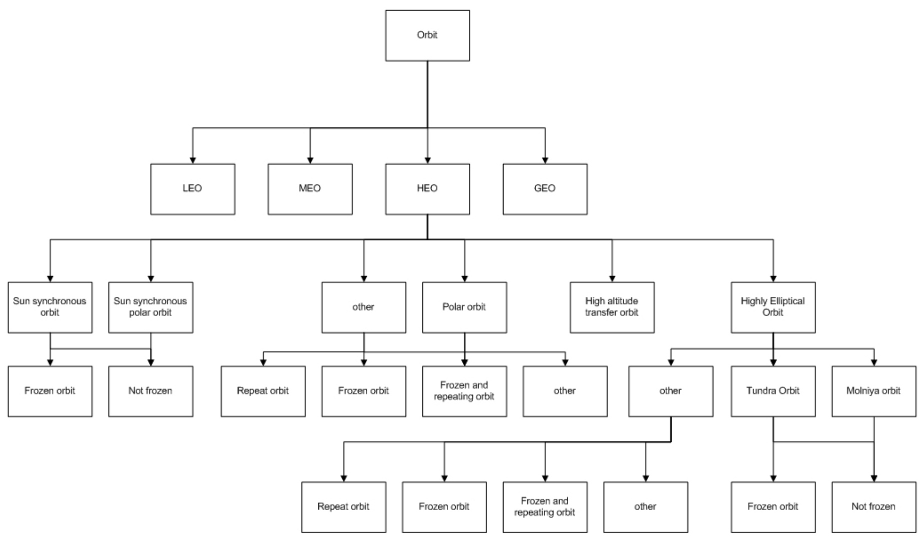


Figure A.8: Design option tree for the orbit architecture of HEO

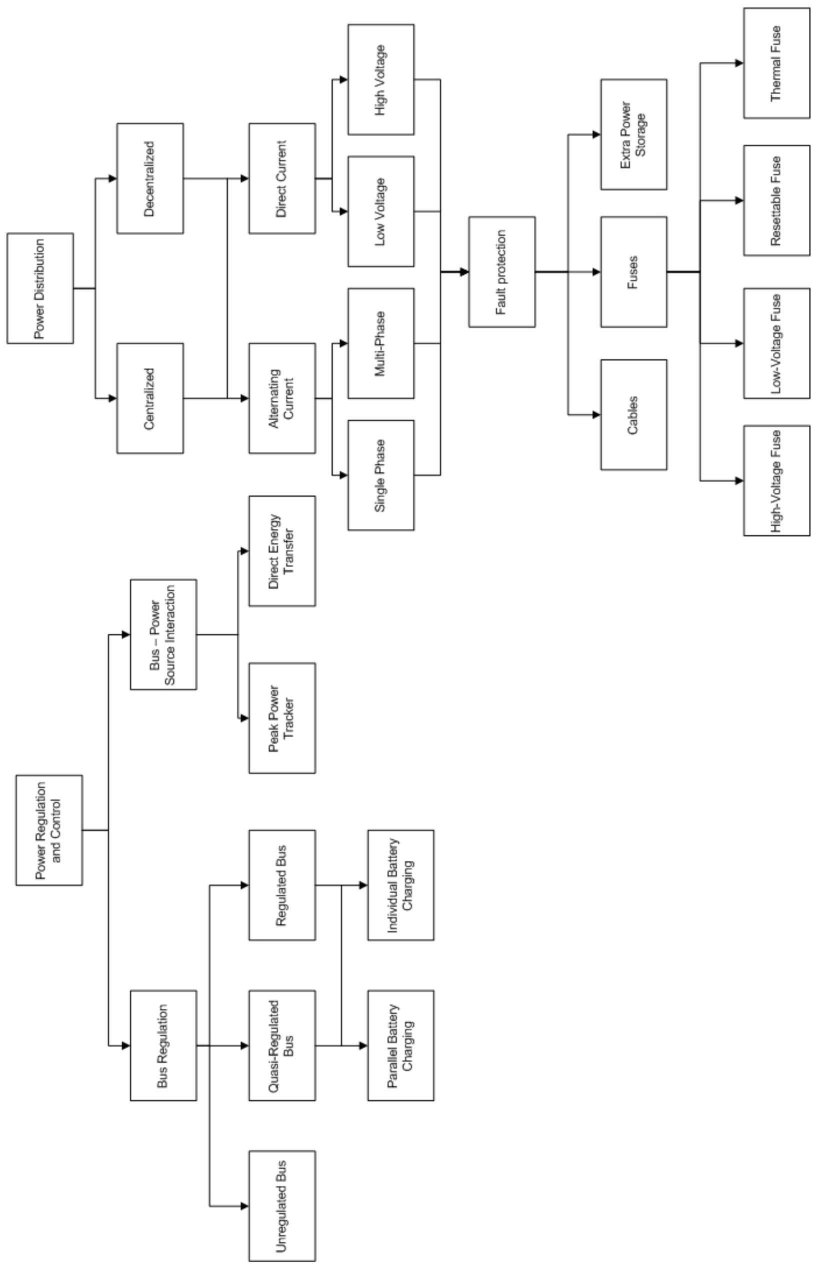


Figure A.9: Design option tree for the distribution and regulation and control of the EPS

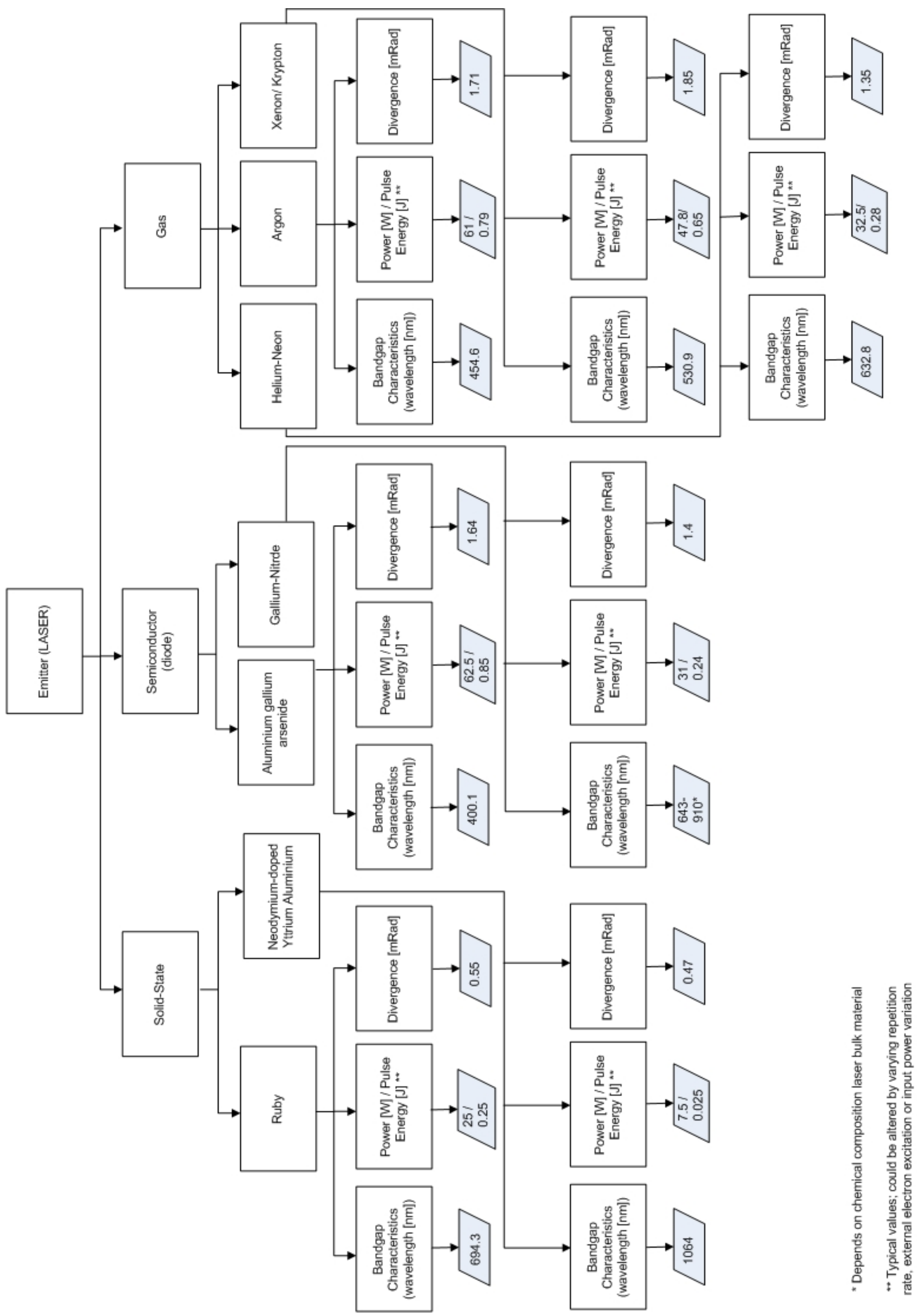


Figure A.10: Design option tree of laser emitter. Numbers indicate typical values.

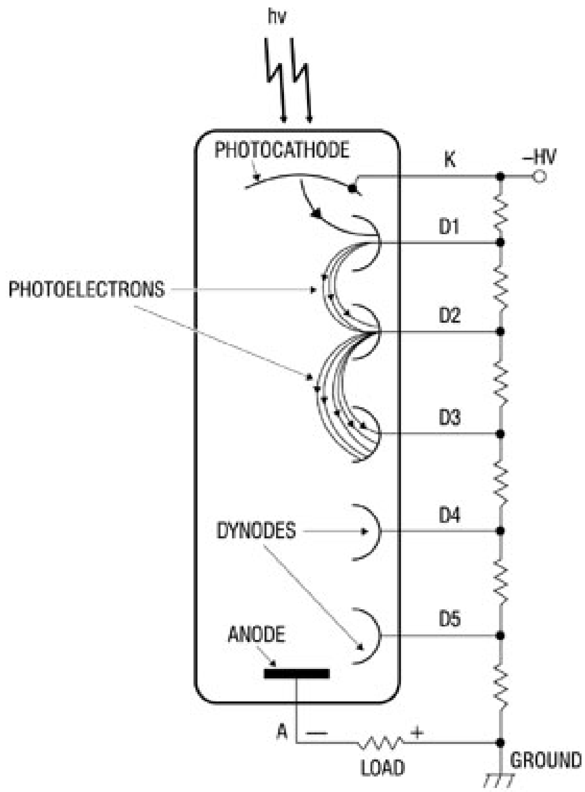


Figure A.11: Photomultiplier tube configuration and working mechanism

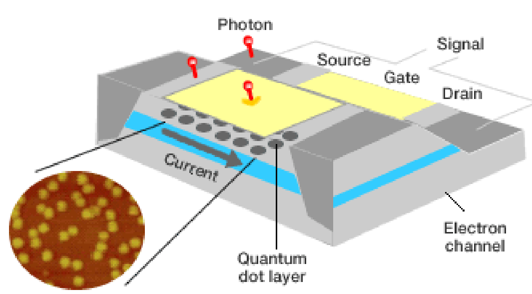


Figure A.12: Single photon detection using quantum dot configuration

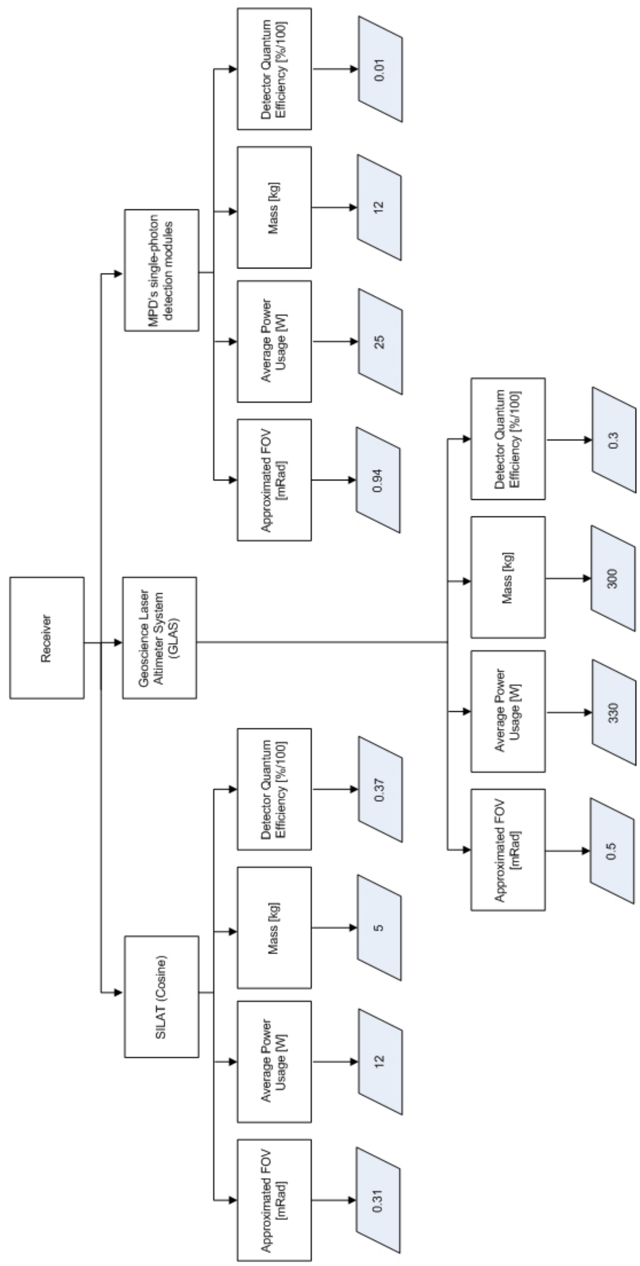


Figure A.13: Design option tree for laser receiver. Numbers represent typical values.

## Appendix B

# Gantt Chart