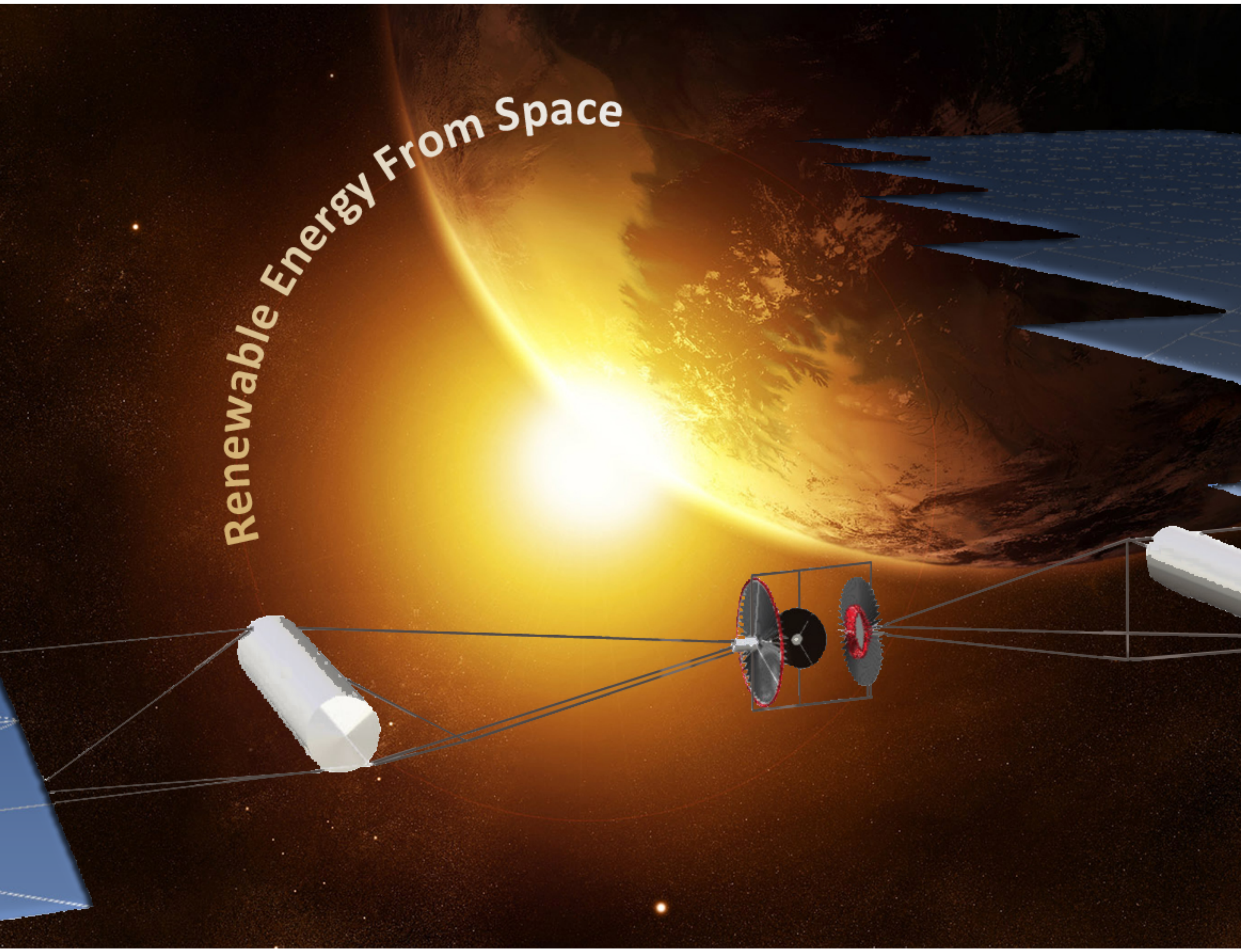
Heliodromus



DESIGN SYNTHESIS EXERCISE 2009

JUNE 17, 2009

Courier of the Sun
HELIODROMUS

TU Delft
Delft University of Technology

Final Report

Renewable Energy From Space

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Preface

This is the final report of the Design Synthesis Exercise subject: Renewable energy from space. The Design Synthesis Exercise is the final bachelor project for Aerospace engineering students at the Delft University of Technology. The purpose of this project is to provide students the opportunity to obtain design and project management experience.

For the past 10 weeks, we, a group of 10 randomly chosen students, have been working together on the subject Renewable Energy from Space. The goal was to come up with a Space Based Solar Power (SBSP) design that was both technological feasible in the near future and market viable. In order to obtain the best design possible for the given timeframe of 10 weeks, this project has been divided into four stages, namely project plan, baseline, midterm and final stage. In the project plan and the baseline report the project was started using system engineering (SE) techniques, which were really helpful to provide a good overview of the project.

In the first weeks of the project we investigated many existing SBSP concepts, designed by organizations such as NASA and ESA, in order to get a good understanding of the entire system and mission. After that, we identified the critical subsystems and in subteams we zoomed in on each of these components. At the same time we integrated the acquired knowledge by each team and came up with several rough concepts. Eventually we chose the most promising concept and worked out this concept in the final period. The results can be found in this final report.

We would like the thank the TU Delft,in specific our tutor Ir J.M. Kuiper and our two coaches, Dr. R.E.M Riva and Dr. P. Stocchi, for giving us constructive advice and support throughout the entire project.

Summary

Renewable energy sources are becoming more and more important as fossil fuel reserves are running out or are becoming too expensive. This phenomenon opens new opportunities for renewable energy sources, such as Space Based Solar Power (SBSP). The working principle of SBSP is the generation of power in space using solar cells and subsequently transmitting this power to the Earth by means of wireless power transmission. The main advantages of a space system compared to terrestrial solar conversion systems are a higher surface power density (due to a higher sunlight intensity outside the atmosphere), and a continuous power supply (no day and night cycle).

The mission need statement is formulated as:

> To design a 'green' spacecraft (series) to supply renewable solar energy to Earth, to be launched before 2025.

The project objective statement is defined somewhat different:

> Perform a market and technology feasibility study and make a conceptual design for a SBSP harvesting platform by 10 students in 10 weeks."

The project includes an analysis of the current and future electricity market, the conceptual design of a SBSP system, and an analysis on its technical performance, economical aspects and sustainability compared to Earth-based solar farms.

The mission need statement gives rise to a number of requirements for the mission. In addition, there are mission constraints, which lead to more requirements. The top level requirements of the SBSP mission are:

- Operational lifetime of at least 10 years
- Effective power output of 1 GW on Earth at the end of life
- To be launched before 2025
- Cost-competitive with other energy sources in terms of $\$/kWh$

Besides these top level requirements, numerous other requirements are defined. These requirements state the criteria on sustainability, safety and (subsystem) design.

Heliodromus, is the result from a concept study that started with three initial concepts. A trade-off between these concepts resulted in the selection of a fourth concept, as a combination of the other three.
The Heliodromus concept consist out of a constellation of ten satellites orbiting in LEO and two satellites orbiting in GEO having five mirrors each.

The number of satellites in LEO is scalable depending on the energy demand on Earth. For each added LEO satellite a mirror must be launched to GEO. The LEO satellites are in a 1400 km orbit and the height of the GEO mirrors is 35786 km.

For an end-of-life power output of 1 GW on Earth, a total surface area of 3.44 km^2 of thin film photovoltaic cells is needed per LEO satellite. The photovoltaic conversion is done with a 15% end-of-life efficiency. Thin film panels are used for collection in space, because of advantegous power density and package density. This will decrease the number of launches needed.

The generation of the laser beam for wireless power transmission is done with an efficiency of 60% and each individual laser has a power output of 1 MW, at a wavelength of 1064 nm. In total, the system of ten LEO satellites uses 4000 lasers. The choice for using laser in Heliodromus instead of microwaves transmission is made, because of the huge antennas and rectennas required to generate power in the order of Giga-Watts. Moreover a concept using laser transmission was open for fresh looks and out of the box thinking. A mirror system on the LEO satellites, using mirrors with a diameter of 9 meter and 12 meter, bundles all separate laser rays to a single beam and directs it to a GEO satellite. The LEO satellites have a permanent contact time with one of the two GEO satellites. A GEO satellite, having five mirrors of 24 m diameter, reflects the energy beam to the associated ground station, providing energy to Earth continuously. The laser beam travels through the atmosphere with an efficiency of 85% and reaches the Earth's surface at two ground stations, located in Arizona, USA and Egypt, North Arica. These areas are selected considering cloud coverage and aerosol density, seismicity and political stability in the area.

At the ground station the laser beam is converted by means of monochromatic photovoltaic cells, optimized for the wavelength of the laser. This conversion can be performed with an efficiency of 40%, resulting in a power output of 500 MW per ground station.

The performance of Heliodromus can be evaluated by looking at different aspects, namely the overall efficiency, technical readiness levels (TRL), the energy payback time (EPT) and the total costs. A performance comparison between Heliodromus and an Earth-based solar plant is made to see whether Heliodromus can be competitive with more conventional renewable energy sources.

The overall efficiency of the Heliodromus system is estimated to be approximately 2%, based on a power input and output of respectively 47 GW and 1 GW. The major losses occur during the initial energy conversion by the photovoltaic thin films in space, which have an efficiency of only 15%. The conversion of electricity to laser is done with an efficiency of 60%, as mentioned before. The conversion on ground also significantly contributes to the losses, with an efficiency of 40%. To be market viable the cost price of electricity delivered by Heliodromus must be around $\$7 - 8$ cents/kWh, as concluded from a market analysis. The total cost of Heliodromus is $\$98$ billion, of which $\$80$ billion is contributed by the laser devices and the launches. The resulting electricity cost price of Heliodromus is approximately $\$1/kWh$. The energy payback time for the Heliodromus concept is estimated using the following division: the collection system, the transmission system, the ground station and the launches. This led to a total energy payback time of approximately 6 years.
The main bottlenecks in the used technologies are the laser and the thermal subsystems. The required 1 MW continuous wave lasers are not yet available and certainly not with a lifetime of 10 years or higher. Moreover, the 60% efficiency of the laser results in 40% energy loss, i.e. waste heat. This waste heat has to be radiated by means of thermal radiators. The amount of radiating surface needed is high, thus contributing 50% to the mass and volume budgets of a LEO satellite. Therefore, improvements on either the efficiency of the laser or the effectiveness of the thermal radiators must be made. Furthermore, to increase the overall efficiency of Heliodromus, the efficiency of photovoltaic cells needs to improve.
Another major concern for Heliodromus is the assembly of the satellites in space. The dimensions of both the LEO satellites and the reflectors in GEO are of such large order that they do not fit into a single launch vehicle. For this reason assembly in space is required to create the complete structure. Since the assembly involves

thousands of parts, the assembly of these parts and the realization of the large amount of launches in a short time frame will become a big challenge.

When all the PV arrays of Heliodromus are installed on Earth, 125% more energy is generated, assuming an extended lifetime of 15 years for the space based system and 20 years for the Earth-based system (degradation of photovoltaic cells in space is higher). When comparing Heliodromus to already existing Earth based solar farms, both photovoltaic (PV) and solar dynamic (SD), Heliodromus is $5-10$ times more expensive. For an Earth-based system, the energy payback time ranges between 1 to 2.7 years, compared to the 6 years for Heliodromus.

An estimate of Heliodromus' total efficiency resulted in a value of 2%, which is not sufficient to compete with Earth-based solar systems. Heliodromus' energy payback time is 6 years, which is within the minimum lifetime of 10 years, but still longer than the energy payback times of Earth-based system. The total cost of Heliodromus is around $98 billion and at this stage it can not be price competitive with fossil or Earth-based renewable energy sources.

Finally, assembly in orbit has never been done on the scale required for Heliodromus, therefore it opens a totally new field of research and development. The total efficiency can be improved up to a factor 4 in near future, getting Heliodromus closer to becoming market viable.

List of Acronyms

Symbol	Description	Symbol	Description
AD&C	Attitude Determination and Control	MODIS	Moderate Resolution Imaging Spectrotadiometer
AM1	Air Mass Coefficient for 1 Atmosphere	MPE	Maximum Permissible Exposure
APE	Absolute Pointing Error	MSS	Magnetic Sensing System
AU	Astronomical Unit	MT	Magnetic Torquers
BOL	Begin Of Life	OM&L	Operation Management and Logistics
BTU	British Thermal Unit	PR	Progress Ratio
CAES	Compressed Air Energy Storage	PCU	Power Conversion Unit
CCGT	Combine Cycle Gas Turbine	PM&D	Power Management and Distribution
C&DH	Command & Data Handling	PV	Photovoltaic
CSS	Coarse Sun Sensors	QD	Quantum Dots
CTE	Coefficient of Thermal Expansion	R&D	Research and Development
DSE	Design Synthesis Exercise	RF	Radio Frequency
EOL	End Of Life	ROM	Rough Order of Magnitude
ESA	European Space Agency	RSO	Rate Sensor Units
ESS	Electricity Supply system	RWA	Reaction Wheel Actuators
EPT	Energy Payback Time	SBSP	Space Based Solar Power
FGS	Fine Guidance Sensors	SD	Solar Dynamics
FHST	Fixed Head Star Trackers	SEC	Solar Energy Collection
GEO	Geostationary Earth Orbit	S&M	Structures and Mechanisms
GTO	Geostationary Transfer Orbit	SPD	Spatial Debris Density
GS	Ground Station	SPS	Solar Power System
HEL	High Energy Laser	STK	Orbital Calculation Software
HTF	Heat Transfer Fluid	SWOT	Strength, Weaknesses, Opportunities and Threats
IEA	International Energy Agency	TDRS	Tracking and Data Relay Satellite
IGCC	Integrated Gasification Combined Cycle	TRL	Technical Readiness Level
LEO	Low Earth Orbit	TT&C	Telemetry Tracking & Command
mas	milliarcsecond	WPT	Wireless Power Transmission

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Chapter 1

Introduction

Renewable energy sources are becoming more and more important as fossil fuel reserves are running out or becoming too expensive. This phenomenon opens new opportunities for renewable energy sources, such as Space Based Solar Power (SBSP) system. The working principle is generating power in space using solar cells and subsequently transmitting this power to the Earth by means of a wireless power transmitter. Main advantages of these kind of systems compared to terrestrial solar conversion systems are higher surface power density, due to much more intense sunlight in space, and continuous, non-diurnal power supply.

The main purpose of the SBSP mission is to design a green spacecraft (series) to supply renewable solar energy to Earth to be launched before 2025. The spacecraft series consisted of a N number of satellites orbiting in a Low Earth Orbit (LEO) collecting solar energy and transmitting this collected energy via two geostationary mirrors to the Earth. The N number of satellites emphasizes the fact that the project is scalable, there can be 10 satellites in LEO but it can easily scaled to a 25 satellites, or even a 50, depending on the energy demand on Earth. Two satellites in Geostationary Earth Orbit (GEO) provide continuous coverage for the transmission and puts a minimum demand on the launch to GEO. The transmitted energy is collected on Earth by a solar farm which collects the energy from laser transmission continuously and also solar energy during the day. The goal of this DSE project was to perform a technological feasibility and market viability study for a SBSP system on the basis of the aforementioned conceptual design. Which considers all elements of a SBSP system from launch to operational phase where emphasis is put on the balance between all these elements.

The project was started using system engineering (SE), which were only moderately helpful to provide a good overview of the project. It was soon discovered that standard SE tools were not sufficient for this project, since it is a incredibly complex system due to its size and the need for never proven technologies. New SE tools needed to be provided to perform trade offs for several subsystems, like power transmission and energy conversion. These trade offs were based on technology readiness level (TRL), energy payback time (EPT) and of course our own interests leading to new challenging ideas. The methodology behind these new SE tools are elaborated in chapter 2. In chapter 3, the market analysis is discussed. Then, in chapter 4, an overview of the subsystem trade-off and resulting concepts established in the mid term is given. In chapter 5 the orbit and ground station location are discussed. In chapters 6, 7 and 8, the subsystems collection, transmission and ground station conversion are further analysed and developed. The integration of these (sub)systems is described and evaluated in chapters 9 and 10. Finally, further developments and conclusions are given in chapters 11 and 12.

Chapter 2

Project Definition

To properly start any project it is important to begin with defining the goal that is desired to be achieved and the mission goal. For the Heliodromus these are stated below.

2.1 Mission Need Statement & Project Objective Statement

Mission Need Statement

> Design of a market viable ”green” spacecraft (series) to supply renewable solar energy to Earth to be launched before 2025.

Project Objective Statement

> Perform a market and technology feasibility study and make a conceptual design for a SBSP harvesting platform by 10 students in 10 weeks.

2.2 Requirement Analysis

A helpful tool in the analysis and identification of the requirements of this project is the requirement discovery tree, represented in Appendix A. The mission to be performed by the space based solar power system is defined as: ”Gain solar energy from space for distribution on Earth”. Foremost, the mission needs to be technically possible. Secondly the mission needs to be performed within constraints defined by customer needs, environmental issues, governmental and political demands and the technical constraints.
To perform the mission on a technical point of view two basic aspects are needed.

- Working satellite
- Working ground station

The main aspects of a working satellite needed for performing the mission are defined as follows:

- Collection of solar energy and conversion to electrical power.
- Conversion of the collected solar energy (electrical power) for transmission toward Earth.

- Transmission of converted solar energy
- Communication with Earth

All these aspects have several requirements in common: their efficiencies, the size of the sub-systems, type of the sub-systems, orbit and structure.

For the working ground station these aspects are defined as:

- Collecting transmitted energy
- Conversion of collected energy
- Delivering of converted energy to the electricity grid
- Establish communication link

These requirements were set at the beginning of the project. From there they have evolved during the course of the project, the current requirements and whether they were attained can be seen in the compliance matrix given in 10.4.

2.3 Analysis Methodology

In this report some analysis methodologies are used to find out whether the requirements of the system are met and what the bottlenecks of the Heliodromus project are. These can be technological as well as operational or economical bottlenecks. The methodologies used are explained in the following paragraphs.

To discover the state of development of every technology used in Heliodromus, their Technology Readiness Levels (TRLs) are determined. TRL is a scaling method for technologies, introduced by the United States' government and currently used by many companies to show the maturity of new and evolving technologies. In figure 2.1 the definition of TRL used by NASA is presented. By defining the TRLs, it will be possible to quickly check which parts of Heliodromus lack sufficient technological development and require further development.

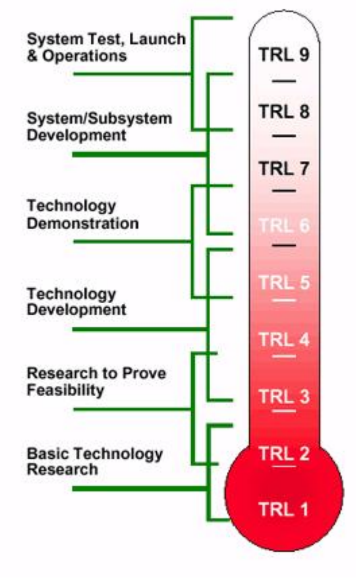


Figure 2.1: Technology Index Of NASA

Also the efficiencies for every part of the energy transmission process are given. Consequently, the parts of the process where most losses occur become visible. This gives an indication about which efficiencies should be improved.

Another method, namely defining the Energy Payback Time (EPT), is applied to determine the time it takes for the Heliodromus to generate all the energy which was used in producing Heliodromus and putting it into operation. This way, the most energy demanding elements of the project will become apparent.

Furthermore the mass, volume and power budget for each subsystem are defined. This overview gives insight into the distribution of the mass and volume over the subsystems.

Bringing mass and volume into orbit is expensive and costs a great amount of energy, therefore the subsystems with high values for these parameters should be optimized to minimize mass and volume.

The lasers consume the majority of the available power, therefore the power requirements of the other subsystems are of less importance.
At last, also the cost of the Heliodromus project is estimated. This is done by making an educated guess for the costs of Heliodromus' subsystems and the required launches to bring the system into space. When assuming a certain price for the energy delivered to the consumer, the time it takes before the breakeven point is reached can be calculated.

All these methodologies are used in every work package (i.e. defining the Orbit and Ground Station Location, Solar Energy Conversion, Transmission, Ground Station Conversion and System Integration) and are summarized in the chapter System Evaluation. In this chapter also a comparison is made with a terrestrial solar farm.

Chapter 3

Market Analysis

According to the Mission Objective Statement the entire mission should be market viable. For this the current and future energy market has to be mapped first. The following sections will provide a rough overview of the different market segments, their advantages, disadvantages, market share, market growth and the trend in terms of market share and electricity cost price in US dollars per kWh.

During research, different sources, representing different energy technologies, have been used. These sources are found to be quite different from each other with respect to market share, cost price and trends. Several reasons for this, besides the obvious fact that they are in general quite optimistic about their own technology and less optimistic about others, are: different years of publication and different energy price calculations (e.g. production cost price versus consumer price and different cost prices for different countries/areas). In order to make a fair overview, statements about current prices are estimated for the beginning of 2009 and the cost price is based on the costs of generating 1 kWh of electricity.

Besides the analyzed energy sources in this chapter, there are numerous of other alternative energy sources that take account for approximately 5% of the worlds electricity supply. These sources are amongst others: Biomass, Geothermal and Solarthermal.

3.1 Fossil Fuels

Fossil fuels are the main source of electricity generation. According to the International Energy Agency (IEA)[86], in 1973 roughly 75 % of the electricity production was gained from burning oil, coal and natural gas, against 67 % in 2006.

Figure 3.1 shows the cost of electricity generation for the main technologies. The values are based on the technology expected to prevail over the next ten years. Note that Combine Cycle Gas Turbine (CCGT) uses natural gas and Integrated Gasification Combined Cycle (IGCC) uses synthesis gas made out of coal.

3.1.1 Oil

Current Price, Market Share and Trends

It is estimated that 5% of the electricity is generated by oil[86], in the future it is expected that this contribution will decrease slightly to 4% in 2030 [87]. According to the BP Statistical Review of World Energy2008[88] the oil reserves will last for roughly 42 years assuming the current rate of oil usage and the current proved reserves. It is quite difficult to make a future projection of the oil price since it proved to be a very volatile commodity

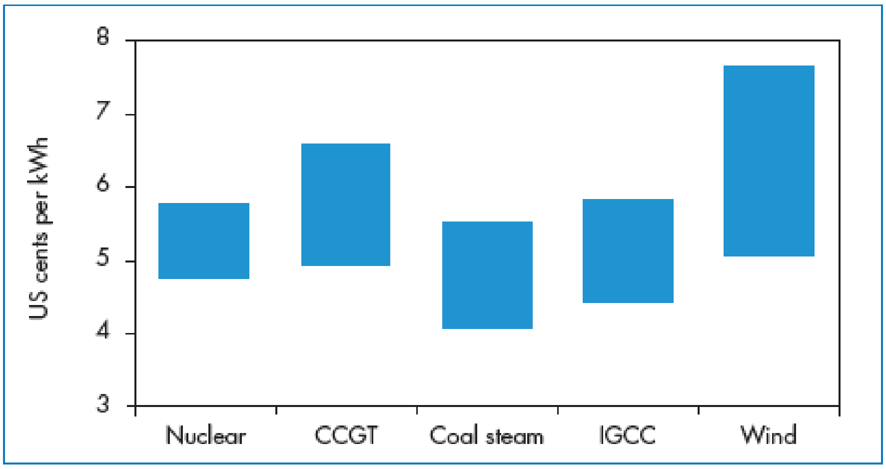


Figure 3.1: Electricity generation costs of common technologies (IEA 2006)[104]

during the last decade. It started this decade at around 30 dollars a barrel before the 2003 Iraq war, after which it rose, at an ever increasingly rapid pace, to 140 dollars in the summer of 2008. However, the barrel price droped below 50 dollar levels at the start of 2009 [88].

Table 3.1: Advantages and Disadvantages of oil as fuel

Advantages	**Disadvantages**
Easy transportation and storage	Economically recoverable oil reserves proven to only last another 42 years
Cleaner to burn than coal	Emission of carbon dioxide is much higher than for gas when burned
	Supply unevenly distributed over the world
	Non renewable

3.1.2 Coal

Current Price, Market Share and Trends

Roughly 40% of the electricity in the world is generated by coal today[86]. It is expected that this contribution will stay roughly the same in 2030[87]. Current studies[88] predict coal reserves will last for over 130 years. The current price for coal based electricity generation is approximately 4 - 6 dollar cents per kWh, the cheapest of all. When making synthetic gas out of coal, the price is slightly higher ($\pm$0.5 cents).

Table 3.2: Advantages and Disadvantages of coal as fuel

Advantages	**Disadvantages**
Long lifetime for reserves	Emission of greenhouse gases is higher than for oil when burned
Cheap	Extensive transportation system needed
Evenly distributed over the world	Non renewable

3.1.3 Natural Gas

Current Price, Market Share and Trends

It is estimated that 20% of the electricity in the world is generated by natural gas[86]. It is expected that this contribution will increase to 30% in 2030[87]. The gas reserves will last for 60 years with current reserves and usage, according to BP[88]. The current price for natural gas for electricity generation is roughly 5 - 7 dollar cents per kWh.

Table 3.3: Advantages and Disadvantages of natural gas as fuel

Advantages	**Disadvantages**
Cleaner burning than coal and oil	Difficult to transport, due to the high risk of leakage
Longer lifetime of reserves than oil	Non renewable

3.2 Nuclear Power

Current Price, Market Share and Trends

Nuclear power currently supplies 13% of the worlds electricity. The world's most nuclear dependent country is France, which gets 75 % of its power from nuclear sources. In 2009 there are a total of 436 commercial nuclear power reactors operating in 30 countries with a total of 372 GW capacity [90] for a total of 2793 TWh[86] which is expected to rise to $509 - 663$ GW in 2030.

In the past nuclear power has flourished due to several reasons in different countries. The nuclear development in France, for instance, has sustained so well as France has very little in fossil fuel resources and "recent events have only reinforced the wisdom of our choice"[89]. The United States nuclear success has been intermittent and marred with the Three Mile Island incident and in short succession Chernobyl. Nevertheless interest in nuclear power is currently undergoing revitalization.
Nuclear power's most important bottleneck is the public opinion. France continues its development and people are quite content with this while United States development has suffered a severe slump as people strongly opposed this.
According to the International Energy Association Nuclear's market share may decline to 4.8% in 2030 though it may hold on at 6.8 %[86].

Table 3.4: Advantages and Disadvantages of nuclear energy

Advantages	**Disadvantages**
Low CO_2 emissions	Greater capital costs and initial investment
Cheap energy	Threat of catastrophic failure
	Threat of terrorist attack
	Radioactive waste handling

3.3 Hydro Power

Current Price, Market Share and Trends

Hydro power currently takes care of 15 % of the World's Electricity supply, generating a total of 3121 TWh. Hydro Power has been a stable supplier of electric power for a long time. Its success is mostly due to long term stable and "free"supply.
Its problem is mostly the regional availability. Hydro power in a flat country will never be as viable as it is in a country with mountains. According to the International Energy Association Hydroelectricity will only supply 2.4 - 2.9% of the world energy in 2030[86].

Table 3.5: Advantages and Disadvantages of Hydro energy[91] [86]

Advantages	Disadvantages
No fuel	Possible high investment costs
Relatively low maintenance cost	Hydrology dependent
	Loss/Change of natural habitat

Hydro Power stations do not use fuel, which results in no CO_2 emission and low operating cost. It should be note that the investment costs only apply when the sole purpose is energy generation. When hydro power installations are developed in combination with multiple objectives and projects, like dikes, the costs can drop significantly.

3.4 Wind Energy

Modern wind energy is a relatively young renewable energy technology of which developments started 30 years ago. However, it has grown into one of the most mature renewable energy technologies, with a current market share of 1.5 % which is rapidly expanding.

Current Price, Market Share and Trends

The following three factors influence the price per kWh of wind energy the most:

- the size of the wind farm
- the wind speed at the site
- the cost of installing the turbines.

Due to these factors, the current price ranges from $3-7$ dollar cents per kWh. In order to make a good comparison with other energy sources, a weighted average price of 5.7 dollar cents per kWh for all countries is used.

The global wind market has grown over the past 5 years with an annual growth of 27.6 %. With an evaluated progress ratio of 0.82 for on shore wind turbines, a price trend can be derived from this. However, the current global crisis and the ever increasing size of wind turbines, which has increased the price per kWh hours since 2006, should be taken into consideration to make a more realistic estimation of the price trend.

Currently 1.5 % of the worldwide electricity use is supplied by wind energy. As already mentioned earlier, the market growth over the past 5 years has been 27.6 %. According to the scenarios of the European Wind Energy Association in 2005 this will rise to 5.5 % in 2010, 13 % in 2020 and 23 % in 2030. However, as already mentioned earlier, the global crisis will most likely have a negative impact on the projected growth.

Advantages

The main advantages are that it is relatively cheap, compared to other renewable sources. The fact that it is one of the lowest contributers to greenhouse gas emissions and air pollution of all energy resources, which makes wind energy a lot more competitive if one takes the additional costs for neutralizing the environmental impact into consideration.

Disadvantages

- Governmental funding dependency: At this moment wind energy still needs governmental incentives in order to meet current market prices, mainly due to it's high risk profile and thus the additional costs (e.g. higher discount rates)
- Wind turbines are made out of scarce material (i.e. steel and carbon)
- A large impact on the environment (e.g. 'landscape pollution' and large amount of fatalities among birds)

Key Success Factors Wind energy has conquered the startup problems and has entered a new, more mature phase. In order to maintain its impressive growth of 30 % a year and to obtain a significant market share in the near future, it is going to depend on two key success factors, namely:

- Continuation of cost reduction that makes funding superfluous.
- Efficient measures to integrate wind energy production in the electricity supply system (ESS).

Based on a conservative progress ratio (PR) of 0.9 (i.e. for every doubling of the cumulative market size, the unit costs decline with 10 %), the cost of wind-generated electricity will reach the break-even point with coal-generated power by 2010, based on current status of the wind technology. However the technology is constantly developing thereby reducing the effect of the PR.
Many countries have adopted grid connection rules for wind power, which list a number of technical requirements that any connected wind turbine should comply with. This regulates the local impact. Besides local impact, system-wide impacts also play an important role into the integration of wind energy production in the ESS. Significant technical measures should be taken in order to comply with the variation and predictability of wind speed and thus the wind power.

3.5 Solar Energy

The solar energy market looks to be very promising as an energy source of the future. The market has grown rapidly at about $20-25$ % per annum over the last 20 years in both the industrial and commercial segment[92].

Advantages Due to the fast growing market, the prices are decreasing significantly as can bee seen in figure 3.3 from the light blue line. The United States Department of Energy predicts a decrease of 50 % of the installation cost of a solar cell within ten years (starting from 2006)[93]. Also the fact that solar energy can be harvested on both large or small scale, making it attractive for larger commercial companies and individual consumers.

Disadvantages

The main disadvantages of solar energy is the governmental funding dependency. Solar energy is still relatively expensive. According to the Solarbuzz LLC survey[94] the current price of solar electricity is 0.3701 $\$/kWh$. In countries such as Germany a feed-in tariff has been introduced to make it more attractive for users.

Current Price and Trends

The following factors influence the price per kWh of wind energy the most:

- the unit price.
- the installation cost.
- the cost of wiring and inverters

As a result of the factors above, the energy price range between 37 dollar cents per kWh and 21 dollar cents per kWh for residential and industrial use respectively, in April of 2009 [94].

The unit price is the highest cost component of the price payed per kWh. Thus reducing this will have the largest impact. The unit price of the highly dependent on the pull of the market. It can therefor be expected that the unit price will decrease. The main drivers for this decrease are the government support, the cheaper materials and cheaper manufacturing techniques. According to research conducted by BP the price of solar energy will be approximately 0.20 $\$/kWh$ in the year 2020[95] for residential use.

Current Market Share and Trends

The market share of photovoltaic cells has rapidly increased over the last few years, as can be seen in table 3.6.

Table 3.6: The PV solar electricity production compared to the world electricity production

Year	2005	2006	2007
Solar energy production [GW]	1.8	2.5	3.8
Increase [%]	-	38	52
Percent of total electricity production [1]	0.009	0.013	0.02

[1]Based on the world energy production of 2007[96]

From table 3.6 it becomes clear that the solar electricity production only is a tiny fraction of the world energy production, though it increases more every year, at the current projections doubling PV electricity production within 1.5 years.

3.6 Market and Cost Trends of Energy Sources

In the previous sections various promising energy sources have been investigated, ranging from the conventional to the unconventional. In this section these energy sources are compared on the basis of aspects of current and future market share and cost price. One has to keep in mind though that predictions of the market share and cost price per kWh of electricity for these energy sources on a global scale for the upcoming twenty years is very hard , if not impossible, to make. There are multiple reasons for this. In the first place, different authorities and governments with different preferences and agendas publish significantly other numbers on this matter. Besides that, current renewable and non-renewable energy sources are still in a start-up phase and are expected to make significant developments, which makes it even harder to come up with predictions.

Cost Trend

In order to design a market viable SBSP system an estimate of the future energy price is desirable, since from such an analysis the available budged can be deduced. An accurate forecast for the price payed per kWh is very hard to make. Therefore an expected price range is constructed for the next decades. Using the Energy Outlook 2009 of the Energy Information Administration (EIA)[97] and the Energy Forecast 2008 of the European Photovoltaic Industry Association[98] an average price is deduced for the upcoming 21 years. The result can be seen in figure 3.2. In this calculation an inflation of 2% is used, which is a rather conservative figure based on the last 20 years[101].

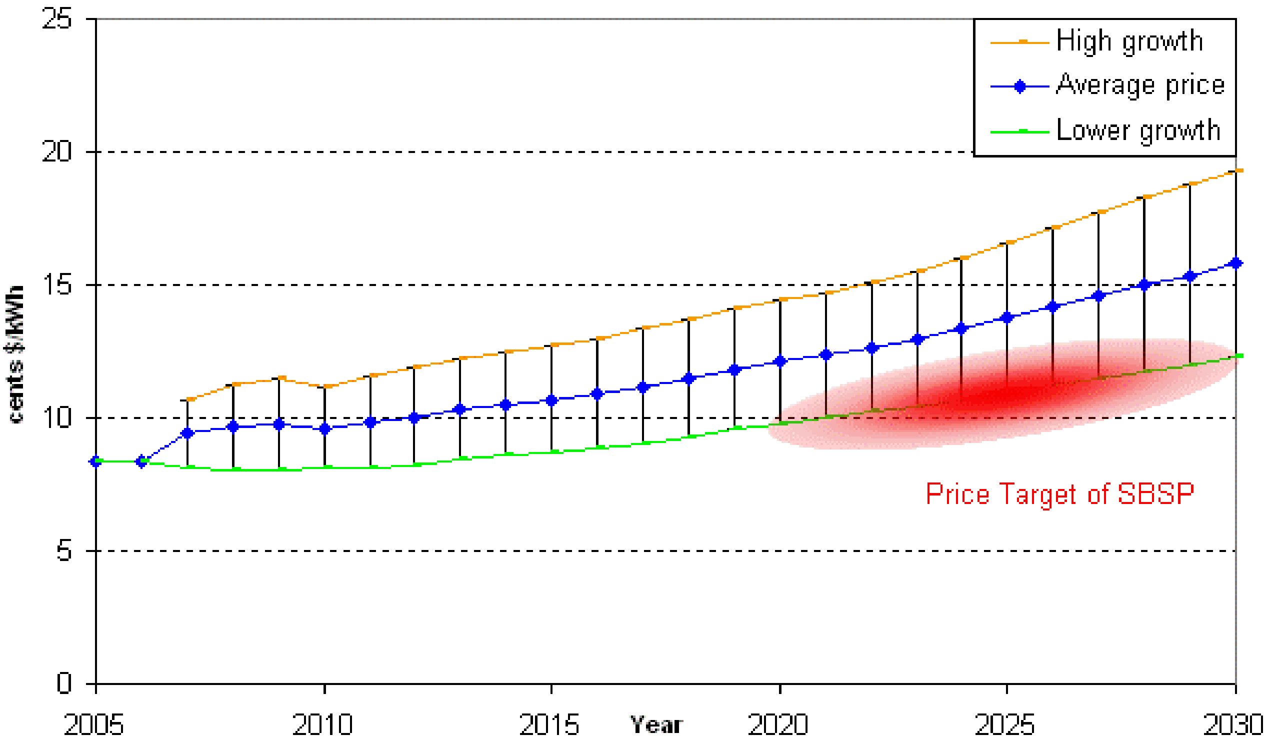


Figure 3.2: The expected energy price including inflation of 2%

In the trend a lower and higher estimate are given. The design should aim for the lower electricity price in order to be competitive. The red area indicates a possible range for which the electricity of the SBSP system should be sold in order to be competitive during its mission (note that the mission operation time is not yet fixed). Assuming the mission will take place between 2020 and 2030, the electricity price should be about $11 *cents* (in 2030 dollars). This equals $7 − 8 *cents* in 2009, taking into account the 2 % inflation.

In order to predict the market segment with which the SBSP system will compete, not only the general electricity price should be analyzed, but also the expected electricity price produced by each market segment has to be identified. Figure 3.3 the expected price per kWh of the different segments are plotted.

The red area in figure 3.3 is placed in the same position as was deduced in figure 3.2. At this price ($7 − 8 *cents* in 2009), the SBSP system will be able to compete the all energy source, even with the fossil sources.

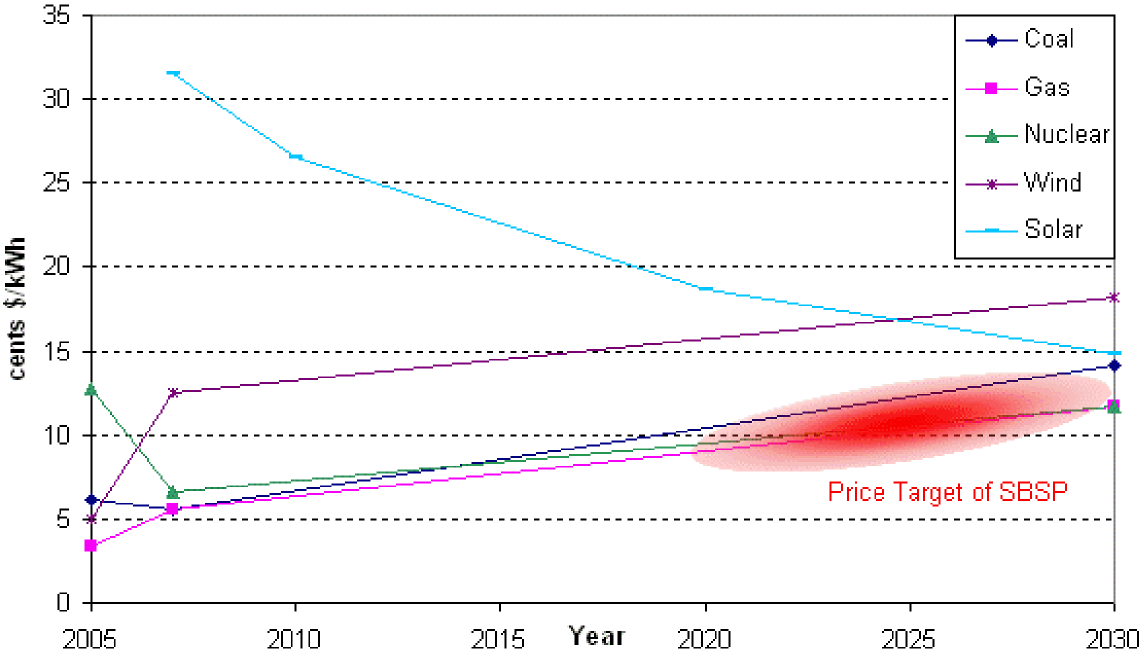


Figure 3.3: The expected energy price per market segment[99] [100] including inflation of 2%

Market Trend

A prediction of the division of the market among the available energy sources between now and 2030 is given in 3.4. From this figure, it can be deducted that the majority of the global energy will probably be supplied by fossil fuels to satisfy the energy needs in the upcoming decades.

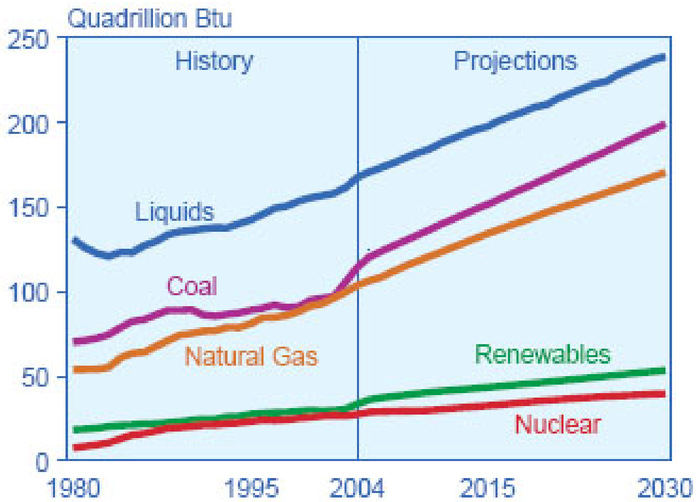


Figure 3.4: Predicted Energy Market Division [108]

3.7 SWOT

A market analysis is to determine potentially what the price range should be, however the market is continuously evolving and to understand this changes in the opportunities,threats, strengths and weaknesses are of importance. This is accessed by a SWOT analysis of the product. There are several aspects that contribute to achieving the objective which is set for Heliodromus is to deliver 1 GW of energy from space to Earth. These aspects can be categorized by identifying Strengths, Weaknesses, Opportunities and Threats. Each Strength can be capitalized, weaknesses must be improved, threats mitigated and opportunities exploited.
Identification of a SWOT is important because the next steps in the planning process of making Heliodromus feasible might be obtained from the SWOT as shown in figure 3.5.

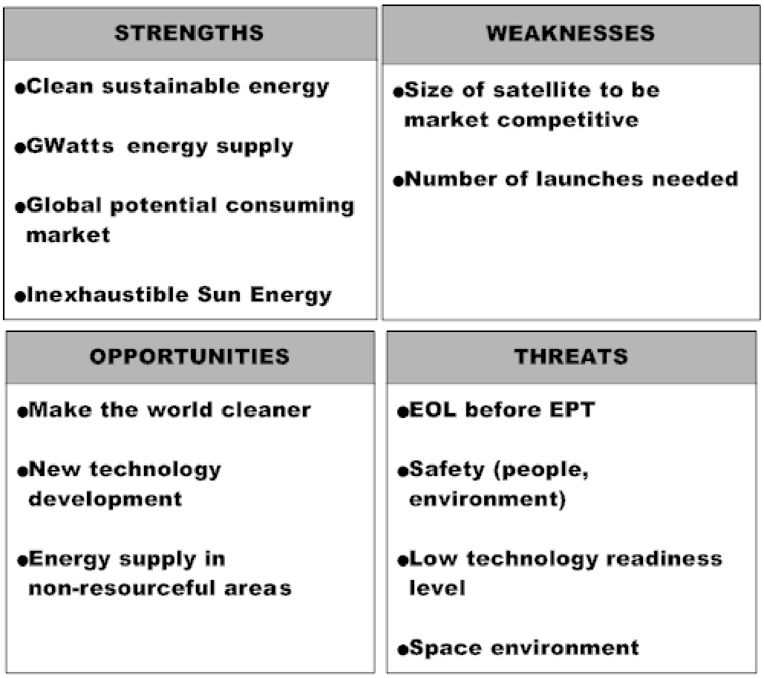


Figure 3.5: SWOT analysis

Chapter 4

Subsystems Trade-off and Concepts

Now that the requirements are set and initial literature studies have been performed the specific features that define the SBSP system are analyzed. The first feature is the conversion of solar power, the second is the transmission of the collected solar energy. The last determining feature is the stabilization of the satellite. There are several options for each feature and choice must be made, hence a trade-off was performed.

4.1 Trade-Off

The trade-off method is based on a rationale qualitative analysis and evaluates the specific features. The criteria that are evaluated of each feature are Availability, Accuracy, Mass, Volume, Cost, Risk, Efficiency, Safety and Life-Time.These criteria are seen as parameters important to the system to ensure a certain quality. All these criteria are given relevant arguments to come-up with the best possible features. The argumentation-representations for the criteria have been implemented into subsystem trade-off matrices, given in Appendices C, D and E, respectively for stabilization, conversion and transmission. The color indications in these matrices led to some conclusions.
Regarding the conversion the photovoltaic options are more efficient and leaves the solar dynamics option out for both the LEO and GEO orbit. Concerning the transmission there are little distinctive differences to conclude which is the best option. Both microwave and laser have pros and cons in each orbit option. The given argumentations and the corresponding results in the trade-off matrix show that laser transmission is more promising in LEO orbit. Stabilization method depends on the option of conversion and the orbit. There is a balance between active and passive option. The passive option for GEO can however be discarded due to low gravity gradient force. Therefor GEO must be actively stabilized and LEO has the possibility of both. The qualitative subsystem trade-off is meant to assist during the design and selection of concepts. Along with these subsystem trade-off conclusions and concepts introduced by NASA and ESA gave three different options. The concepts that are characterized are the the GEO concept in section 4.2.1, LEO concept in section 4.2.2 and finally the Space Mirror concept in section 4.2.3.

4.2 Concept Description

An impression and clear overview of the features for different concepts is indicated in table 4.1. After the table a short outline of the concepts are given.

Table 4.1: Design options of the subsystems of the design concepts

		Concepts		
		LEO	GEO	Mirror
Design Options	**Transmission**	Laser	microwave	sunlight
	Collection	Photovoltaics	Photovoltaics	none
	Orbit height	1000 $[km]$	36000 $[km]$	36000 $[km]$
	Stabilization	Active	Active / Passive	Active
	Contact time	Low	High	High
	Number of sats.	multiple (± 50)	1	1
	Number of GS	multiple (± 50)	1	1
	GS type	existing PV farms	custom built rectenna	existing PV farms
	Weather attenuation	high	low	high

4.2.1 GEO Concept

The GEO concept will consist of potentially multiple GEO satellites which are coupled to their own ground stations. For conversion of solar energy photovoltaics will be used in cooperation with solar concentrators. A microwave system is chosen as transmission method. The stabilization of the system has to be performed actively.
Photovoltaics were chosen as a conversion method. This decision was made since solar dynamics would contain too many moving parts resulting in too much complexity and maintainability issues. The solar conversion is assisted with the use of solar concentrators. These concentrators increase solar radiation that can be captured by the PV cells increasing the efficiency of the solar cells. As a consequence the amount of solar cells needed to provide for the one gigawatt power requirement, is decreased leading to lower cabling mass.
Transmission is done by a microwave system, meaning an antenna is required in space and a rectenna on the ground. The size of these parts of the system are dependent on the transmission frequency and the desired power output. Two frequencies are continuously mentioned in literature as being best suited for this task, 2.45 and 5.8 GHz. Either frequency chosen, the desired output (at least 1 GW on the ground) will make the scale of the system enormous, a scale of several square kilometers. The microwave beam will experience nearly no losses in space. It is only when they cross the atmosphere that significant losses will occur. For a microwave beam though, losses incured by weather are negligible compared to lasers.
High pointing accuracy is important. Gravity Gradient and other passive methods of stability control are ill suited for high precision in GEO unless the structure is specifically designed for this[102]. An active stabilization method is therefor chosen, but this increases the complexity of the system.

4.2.2 LEO Concept

The LEO concept will consist of a constellation of multiple satellites. For conversion of solar energy photovoltaics will be used. The type of photovoltaics remain undetermined up to this point. A laser system is chosen as transmission method. The stabilization of the system has to be performed actively.
The WPT option, laser transmission, has advantages in volume and package density over the microwave option. A second advantage is that laser beams are possibly compatible to deliver energy to both current and new Earth based solar farms. And a third advantage of a LEO orbit is the possibility to serve energy peak moments in the morning and evenings on earth. Taking into account these energy peak moments, the orbit could be tailored to fly over Earth based solar farms at these times. This is an interesting aspect for further research, how ground systems could be designed to use both laser beams and sun light power.

For the space energy conversion a PV type system is selected. A Solar Dynamic system shows promising future perspectives but mechanical life time issues make PV a better option.
Stability of a satellite is an important item because of the required pointing accuracy of the laser beam. This makes gravitational stabilization, although an easy and inexpensive stabilization method, ill suited to this concept. So active stabilization is required. An electric stabilization method is preferred as this does not require specific propellants on the satellite.
The downside of using a LEO orbit is that ground contact times are limited. The result is that multiple ground stations are needed to have a continuous link with ground stations. The disadvantages of using laser is weather attenuation. Laser beams are strongly affected by the atmosphere. However in a multiple ground station concept, this should not be a problem, since ground systems with clear sky can always be found. But this must be investigated in more detail. Another consequence of a low orbit and the limited ground times is the need for a constellation of multiple satellites either in the same or different orbits. Considerations about swarms of nano-satellites should also be taken in consideration. The field of nano-satellites is at this moment of special interest of the TU Delft.

4.2.3 Space Mirror

One of the possible concepts for SBSP is the so called space mirror concept. The space mirror concept consists of a satellite in orbit and a ground station on the Earth. The satellite is a passive mirror reflecting the solar radiation toward the ground station without converting it to other forms of energy. The concept excels in simplicity in the absence of a conversion system, transmitting devices and cabling. As a result the values for the mass and volume of the system can be made significantly lower than other design concepts.

Also, the absence of the collection, conversion and transmitting requirements removes these low efficiency functions. An artist impression of the space mirror is shown in figure 4.1.

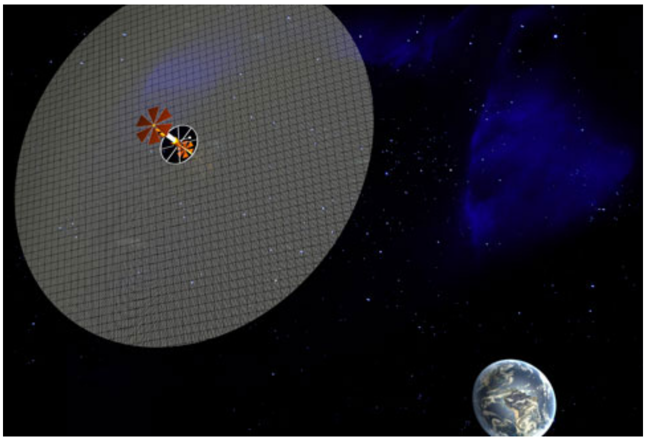


Figure 4.1: Space Mirror - an artist impression[103]

For the space mirror concept different system and orbital configurations can be considered. A single large mirror satellite or multiple smaller mirror satellites can be used and a trade-off must be made between LEO and GEO. For convenience the mirror satellite is assumed to have a single mirror, reflecting the sunlight direct to Earth. The mirror satellite would require multiple reflectors or lenses involving advanced optics, which makes the system complex and expensive.

4.3 Concept Choice

One concept needs to be chosen for further development. While having several strengths the mirror concept is dropped by simple viability problems. The solar incidence difference caused by the distributed nature of the sun (e.g. not a point-illumination) provides an insurmountable obstacle. The only way to surmount this, is by making use of a Soller slit. This is a device that filters the solar radiation in such a way that only parallel sunrays pas through. This way the illuminated spot on earth is reduced but so is the total received energy. However, this solution would completely negate the simplicity angle that made it attractive.

The chosen concept is indicated in the trade-off matrices by the purple indications in the Appendices C, D and E.

Having a LEO orbit, one faces disadvantages such as the low coverage and availability. On the other hand for a LEO orbit, the launch height will be reduced significantly, which is a major advantage with respect to the total cost of the system.

The LEO concept uses laser as WPT method, which has advantages in pointing accuracy and volume above the GEO concept which makes use of microwaves. Also the possible use of Earth based PV or SD farms make the laser system an interesting option. From the stabilization matrix the LEO option will result in the need for the more expensive active stabilization method. Using the subsystem trade-off tables no clear technological choice of a subsystem can be made.

The GEO concept resembles several prospects suggested by NASA allowing for less out of the box thinking. It should once again be mentioned that excluding either LEO or GEO is not based on technological in-feasibility of the chosen subsystem. The trade-off matrices do not provide a conclusive best option, nevertheless provided an rough indication of the pros and cons. Therefore the choice is made mainly based on interest and promising new opportunities.

To recapitulate, the LEO concept can be synergized with Earth based solar farms, leading to both a cheaper and larger concept which can be operational earlier than other concepts. Hence the LEO concept was recognized to be more interesting and providing more opportunities.

However, along with this concept, some obstacles arise. Namely, the disadvantageous orbital properties, such as the low coverage and contact times. Also the pointing accuracy of the laser beam would become a challenge in the LEO concept. To solve these problems a LEO-GEO concept shown in figure 4.2 was created and named after the courier of the Sun in Mithraic ceremonies: Heliodromus.

In the next chapters, Heliodromus is further evaluated. The concept exist of 10 satellites in a sun synchronous low earth orbit and 2 Geostationary mirror satellites. The choice of the number of satellites in LEO is partly a starting value and partly due to initial redundancy considerations. It is also assumed that linear scaling can be applied on the number of satellites. The LEO satellites generate a laser beam, using energy collected by photovoltaic cells.

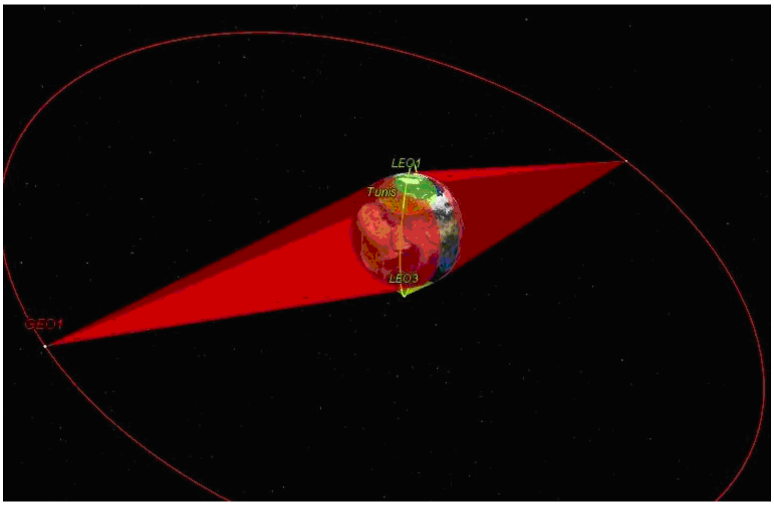


Figure 4.2: Initial Heliodromus Concept Sketch

The motivation for the number of GEO mirrors is the requirement to continuously beam the collected energy from the LEO satellites to the GEO mirrors and subsequent to the ground stations. In chapter 5, these aspects will be discussed in more depth.

The laser is focused using mirrors to one of the two mirror satellites in Geostationary Earth Orbit. The mirror in GEO will reflect the laser beam to the ground station below it where the energy is converted to electrical energy. The specific orbit characteristics and ground station location will be described in chapter 5. Parameters of the photovoltaic arrays will be stated in chapter 6. The complete WPT system is described in chapter 7, and finally the characteristics of the ground stations are discussed in chapter 8.

Chapter 5

Orbit and Ground Station Location

The configuration of the orbits and the location of the ground stations are important parameters, both having an influence on the overall performance of Heliodromus. Therefore it is essential to study, analyze and eventually select the most promising options for the orbit and ground station. As a result of the type of LEO and the number of satellites, the contact and eclipse times are evaluated. Together with the period that the LEO satellites are in line of sight with the Sun, these are decisive factors in the choice of orbit.

In section 5.1, the orbital configurations, such as the orbit type and altitude are discussed. Furthermore the contact time and eclipse time are covered in section 5.1.1. The drag and debris are discussed in section 5.2 and finally all the aspects related to the selection of the ground station location are elaborated on in section 5.4.

5.1 Orbital Configuration

The LEO Sun-synchronous satellites in the Heliodromus concept can be configured in different ways. There are various types of Sun-synchronous orbits and the orbit altitude of the LEO satellites can be adjusted to fulfill certain requirements with regard to contact time and coverage. This also holds for the GEO satellite. The distance between the GEO satellites is a parameter which is entirely dependent on the location of the ground stations. For the proper functioning of the mirror, it is essential to investigate the variation in distance between the LEO satellites and the GEO satellite. This is discussed in subsections, 5.1.1.

5.1.1 Constellation and Configuration

To ensure continuous power delivery, the satellite in LEO should be in sunlight continuously. This criteria determines both the orbit altitude and inclination. First of all, to satisfy the requirement mention above, the LEO should be sun synchronous. In a Sun-synchronous orbit the satellite passes over any given point on Earth at the same local solar time. In other words, the angle between the orbital plane of the satellite and the orbital plane of the Sun, remains constant during the entire mission. The ascending node of the Sun-synchronous orbit is chosen to pass the equator at 06:00 local time. As a consequence the satellite orbit will face the Sun. This will reduce the number of eclipses and the duration of the eclipses. To eliminate eclipses entirely the orbit should be high enough. The minimum height is dependent on the inclination. In general it can be stated that with a higher orbit, the inclination should also be higher. Using simple mathematics the minimum required height is calculated. Figure 5.1 gives a schematic representation of the required orbit height such that no eclipses occur.

In figure 5.1, β is the obliquity of the Earth's rotational axis, the angle between the celestial equator and ecliptic plane. β has a maximum of 23.43 degrees both ways. From figure 5.1 it becomes clear that the angle between the ecliptic plane and the orbital plane can not be more than the sum of the orbit inclination i and the obliquity β. The geometrical relation for the orbital height is given in equation 5.1.

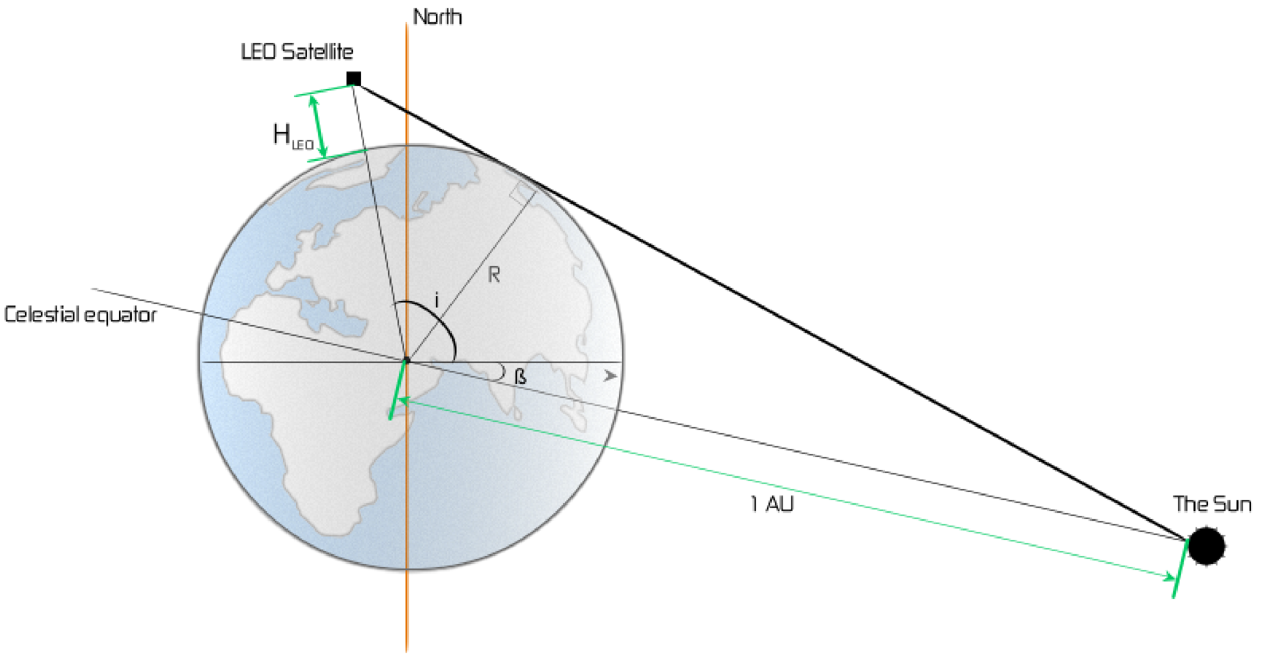


Figure 5.1: Determination of the minimum orbit height

$$H_{LEO} = R\left(\frac{\sin\left(\arctan\frac{1\ AU}{R}\right)}{\cos\left(\arctan\frac{1\ AU}{R}\right) - i - \beta} - 1\right)\ [km] \tag{5.1}$$

where i and β are in radians, R is the Earth's radius (6378.14 $[km]$) and 1 AU is the astronomical unit (149597000 $[km]$). This results in a minimum orbital height of 1396.16 $[km]$. Note that the inclination is determined using orbit simulation software STK[53], in which the inclination depends on the height; making it an iterative process. Since the size of the LEO satellite will be in the order of kilometers, the orbit is chosen at 1400 $[km]$ to prevent partial eclipses.

Figure 5.2 was generated using orbit simulation software STK[53] and shows the days of eclipse per year that a satellite would experience for different altitudes. It confirms the fact that the orbit height of 1400 $[km]$ will suffice. Though it has to be mentioned that there are also eclipses due to the Moon's presence. However these eclipses are short, about 88 minutes per year, that it can easily be back up with a small battery unit in the satellite.

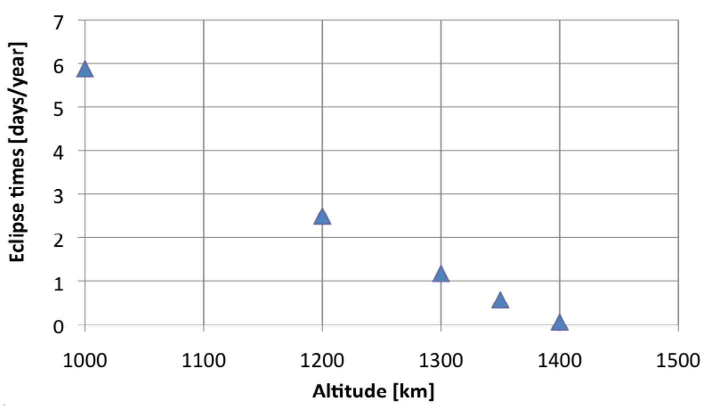


Figure 5.2: Days of eclipse per year vs. orbital altitude

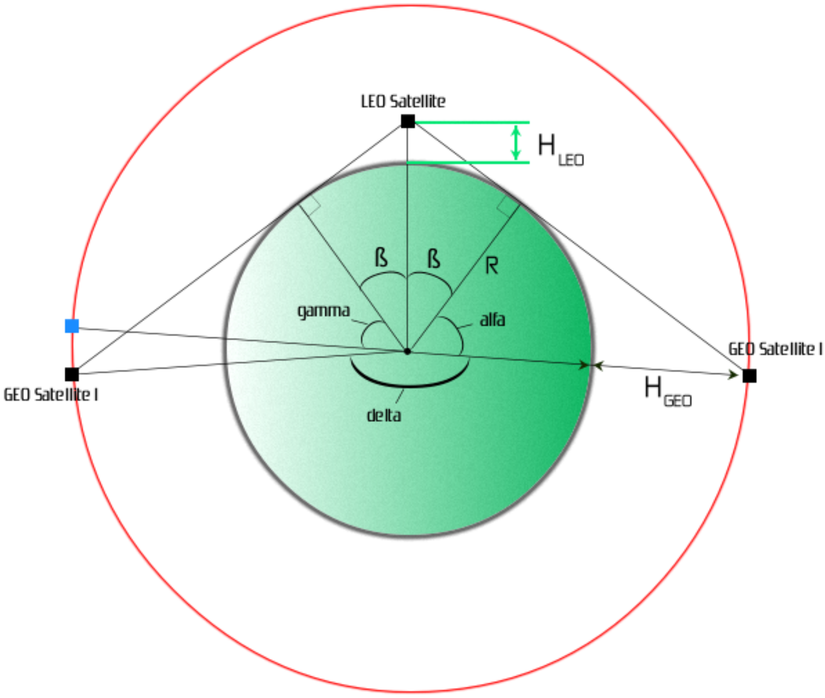


Figure 5.3: Minimum angle between GEO mirrors

The second requirement is that the LEO satellite should be able to transmit its received energy at all times. This means that the there should be at least two mirrors in a GEO and two laser in LEO. The minimum angle for the separation of the mirrors in GEO is again a geometrical problem, dependent on the orbit altitude of both the LEO and GEO satellites. Figure 5.3 illustrates the geometrical relations from which the separation angle (difference between the longitude of ascending node of the GEO satellites) can be determined.

The location of the LEO satellite, as shown in figure 5.3, can differ, though the case depicted in figure 5.3 is a worst case scenario, where the LEO satellite switches from one mirror to the other. The minimum angle is determined using the geometrical relation shown in equation 5.2.

$$\delta = 360 - 2\left(\arccos\frac{R}{R+H_{LEO}} + \arccos\frac{R}{R+H_{GEO}}\right)\ [deg] \tag{5.2}$$

where R is the Earth's radius (6378.14 $[km]$), H_{LEO} is the orbital height of the LEO satellite (1400 $[km]$) and H_{GEO} is the orbital height of the GEO satellite (35785.86 $[km]$). Using the values the minimum angle is 127.6 $[deg]$, this angle is also known as the mean anomaly. Taking this angle to be 130 $[deg]$ allows for margin, such that the laser beam will not be traveling through the thickest parts of the atmosphere, but at an altitude of 84 $[km]$ above the ground. Once again the geometrical approach is checked by means of the orbit simulation software STK. Figure 5.4 is the result of four simulations for different separation angles, where the number of blind spots were counted. A blind spot is a period of time in which the LEO satellite can not send its laser to either one of the two GEO satellites. It becomes clear from figure 5.4 that the separation angle of 130 $[deg]$ will not give rise to any blind spots.

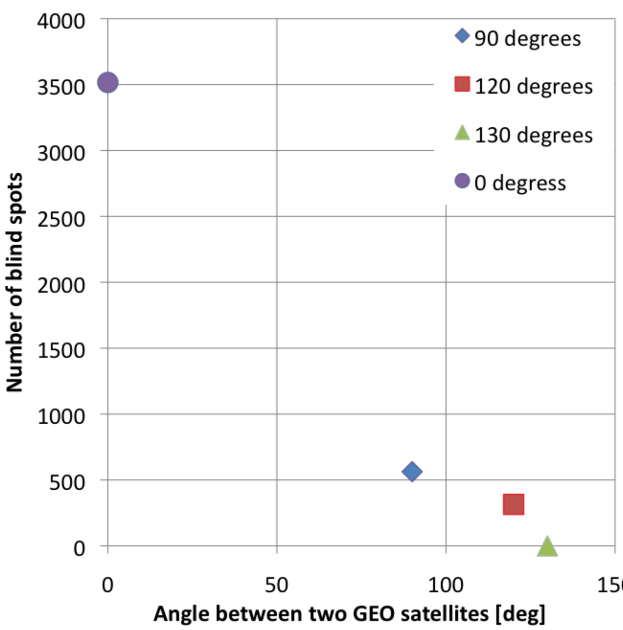


Figure 5.4: Number of blind spots due to relative satellite position

In table 5.1 an overview of the orbital parameters is given

Table 5.1: LEO and GEO orbital parameters overview

Parameters	**Value**		**Unit**
	LEO	**GEO**	
Inclination	101.432	0	[deg]
Height	1400	35786	[km]
Eccentricity	0	0	[deg]
Eclipse time	88	n.a.	[min/yr]
Min. true anomaly	$360/\#\ sats$	130	[deg]
Min. $\#\ sats$	2	2	[-]

Now that the configuration of both the LEO and GEO orbit is known, the minimum and maximum contact time and distance between the satellites is calculated using STK[53]. These parameters are important for the determination of the mirror size, discussed in section 7.2.3. The results are summarized in table 5.2. As described in section 4.3, 10 LEO satellite and 2 GEO mirrors will be used.

Table 5.2: LEO and GEO orbital parameters overview

Parameters	**Max**	**Min**	**Unit**
Distance	46164	34388	[km]
Contact time	3 : 12	1 : 13	[hours:min]

5.1.2 Shadowing effect

In section 5.1.1 the possibility of eclipses for the LEO satellite were investigated. However, there is also the possibility of eclipse on Earth due the enormous structure of the LEO satellite. In section 9.5 the structure of the LEO satellite will be discussed. Based on the artist impression in section 9.5 it is assumed that the satellite will have 3.9 $\times$ 5.0 km as its outer dimensions. If the satellite is to pass straight in front of the Sun, an observer from Earth would see the situation presented in figure 5.5.

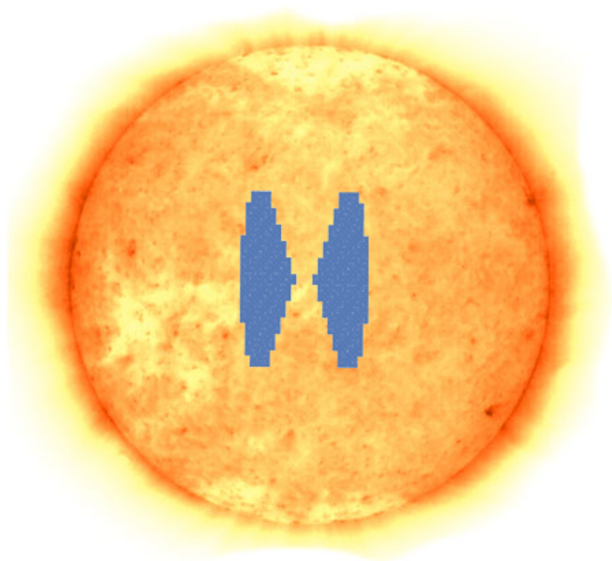


Figure 5.5: The LEO satellite in front of the sun, on scale seen from Earth

The view shown in figure 5.5 is actually very rare. Since the LEO satellite is in a Sun-synchronous orbit it will always have the ascending node on the same local time. Only at dusk or down in mid summer or mid winter, near the poles (depending on the observers location) such a view would be possible to see. This situation is illustrated in figure 5.6, where $i = 101.4\ [deg]$ is the inclination and $\epsilon = 23.5\ [deg]$ is the obliquity of the Earth's rotational axis. It is assumed that the sunlight is parallel. Note however, that the shadowing effect is not noticable with the naked eye, since only 2.5 $\%$ of the sunlight is blocked. A total eclipse is therefor excluded. Moreover the passing of the LEO satellite in front of the Sun will only take 1.8 seconds at most, since the orbital velocity of the LEO satellite is 7.16 km/s and the equivalent solar disc diameter 13 km. All in all, the shadowing effect would not have a significant effect on the environment.

5.2 Drag and Debris

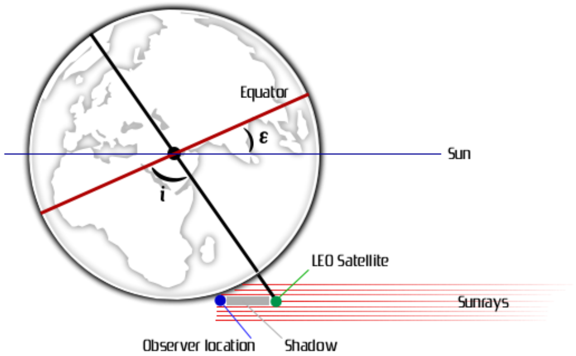


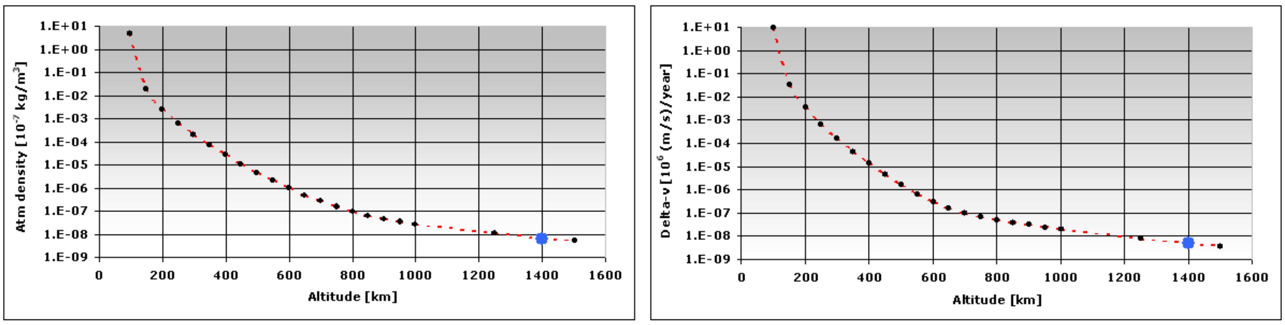
Figure 5.6: The shadow on Earth due to a LEO satellite

Aeordynamic drag forces the satellite to move out of its orbit. It is of importance to evaluate the effect of drag on the satellites.

The aerodynamic forces in GEO are neglected, since it is far removed from Earth's atmosphere. LEO orbits do have to take atmospheric drag into consideration. The debris concentration in space is also an issue which has to be taken into account, since the objects in space have, although small in size, enormous velocities, capable of causing significant damage to the satellite system.

5.2.1 Atmospheric Density and drag

Getting further away from Earth's atmosphere results in lower atmospheric density. The atmospheric density in space approaches zero 'far away' from the atmosphere of the Earth. LEO orbits are relatively close the Earth and therefore the atmospheric density should be taken into account. The LEO satellite of the Heliodromus concept will be operating at an altitude of 1400 km. This number was calculated in section 5.1.1. At an altitude of 1400 km the atmospheric density is negligible. Plot 5.7a shows the behavior of the density with respect to the altitude.



(a) Atmospheric density vs. altitude [4]

(b) ΔV required to maintain altitude [4]

Figure 5.7

The low, negligible drag force of operating at 1400 km means a low ΔV budget is required to maintain the altitude. This is visible in plot 5.7b, which is identical to the density plot. This is logical since drag is the driving factor regarding the required ΔV budget for the maintenance of the altitude.

5.3 Van Allen belt

However there is a downside to the 1400 km altitude, namely the the Van Allen Belt. The Van Allen radiation belt is an area of charged energetic particles around the Earth. The Van Allen belt maintains its shape and position due to the Earth's magnetic field [4][57]. It is known that the magnetic field is not uniformly distributed around the Earth, thus the concentration of the energetic particles in the Van Allen belt is also not uniformly distributed. However the belt does have a characteristic form, as can be seen in figure 5.8. The belt consist of an inner and an outer part.

The charged particles of the Van Allen belt are of significance as it degrades electronics. The GEO satellite does not suffer as much from this, as there is very little electronics. Therefor the shielding of the GEO satellite will be less of an issue. A minor difference in comparison to the LEO orbit is that the GEO satellite will be orbiting in the outer part of the Van Allen Belt. This means that the GEO satellite will be in the radiation belt constantly, as opposed to the LEO satellite, which passes through the inner part of the radiation belt on 2 occasions during one orbital period.

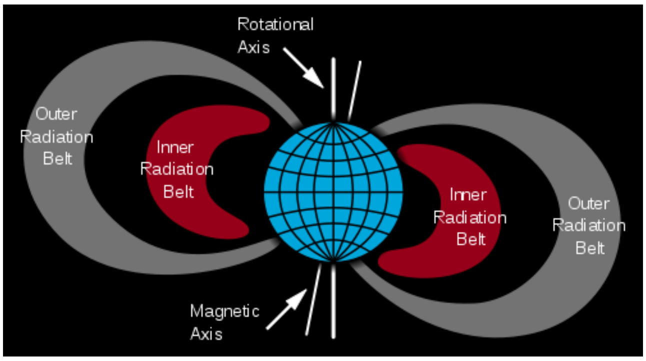


Figure 5.8: Van Allen radiation belt [58]

At 1400 km, the satellite operates just inside the inner part of the radiation belt, where high concentrations of energetic protons are trapped by the strong magnetic field of the Earth. The high concentration of energetic protons can drastically damage the electrical equipment on board of the satellite. In order to avoid this, one has to shield all the electrical components of the satellite system. This will increase the mass of the satellite and therefore the cost of the system. However it will be compensated by the cost reduction of needing less ΔV due to the low atmospheric drag at 1400 km altitude.

5.3.1 Space debris in LEO and GEO

There are a great number of objects flying in space, with velocities in the order of kilometers per second. Although the size of these objects is not always significant, their high velocity could cause drastic damage to space systems due to the high kinetic energy. The two graphs in figure 5.9 show the spatial densities of space debris, i.e. the number of objects per volume of space. The upper graph shows the spatial density around GEO orbit and the lower graph shows the spatial density in LEO orbits.

From these graphs it can be concluded that the spatial densities in GEO orbits are approximately 10 times less than in LEO orbits. In others words the probability of a collision with a big sized object in a LEO orbit is bigger in comparison to a GEO orbit. However in GEO the probability of being hit by relative small objects is bigger compared to a LEO orbit[4]. Therefor different requirements apply to the GEO satellite than for the LEO satellite at 1400 km. There are active and passive means of debris protection which can be applied for the GEO mirror and the collection system in LEO orbit. Shielding and redundancy are examples of passive means, as opposed to active means such as collision avoidance.

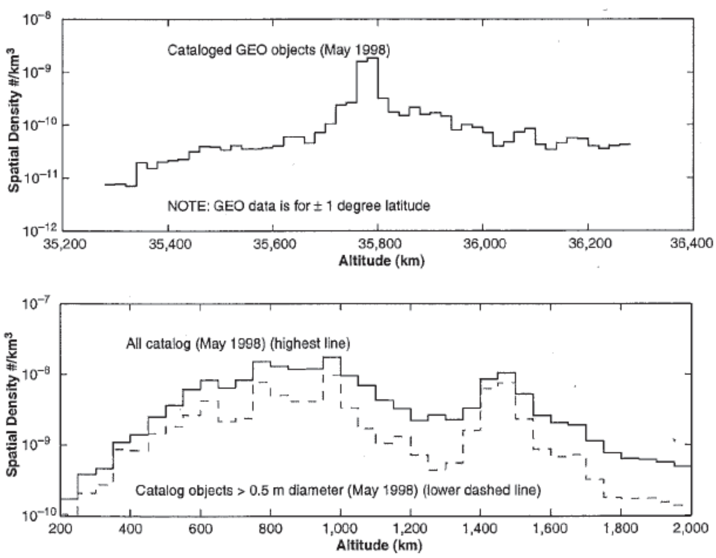


Figure 5.9: Debris spatial density values for GEO & LEO [4]

Using the spatial density, a rough estimate for probability impact with debris can be calculated. Equation 5.3 approximates the probability of a piece of debris impacting a space system using the kinetic theory of gases[4]:

$$P = 1 - e^{-SPD \cdot AC \cdot T \cdot V_{REL}} \tag{5.3}$$

where *SPD* is the spatial density of debris, *AC* is the cross-sectional area, *T* is the mission duration and V_{REL} is the relative velocity between the satellite and the debris object. The cross-section area is a defined area viewed from one orientation and approximated by $1/4$ of the total surface area. The value for SPD can be read from figure 5.9, knowing the orbit height. The value V_{REL} is typically between 9 to 10 km/s for LEO and 0.1 to 0.5 km/s for GEO. Using the value above and the known surface area the impact probability can be derived, as shown in table 5.3:

Table 5.3: Debris impact probability per year for each LEO and each GEO

	LEO	**GEO**	**Unit**
Height	1400	35786	$[km]$
SPD	$5.00\ 10^{-9}$	$2.00\ 10^{-9}$	$[\#/km^3]$
AC	0.86	$7.2\ 10^{-4}$	$[km^2]$
T	3.15^7	3.15^7	$[s]$
V_{REL}	10	0.5	$[km/s]$
P	75.2	$2.27\ 10^{-3}$	$[\%/yr]$

Table 5.3 shows that the impact probability in LEO for a large satellite is a major issue. After 15 years the impact probability will be 99.99999 %. Though this probability is the impact probability for any debris object small and large, where the likelyhood of small object colission is the largest. Therefor the LEO satellite needs to have shielding, redundant systems and collision avoidance for the larger debris objects.

5.4 Ground Station Location

Determining the exact locations of the ground stations depends on various factors. These factors have to be taken into consideration during the trade-off phase. Some weigh heavier than others. The laser beams are strongly affected by the weather attenuation. Therefore cloud cover is the most decisive aspect of the trade-off process. However the elevation of the ground station with respect to the equatorial plane is also a significant factor. On top of these factors, the seismicity maps are studied to avoid a ground station location at which seismic hazard are often encountered. Ultimately the generated power from space has to be transported to the end consumer. Therefore considering the logistic and commercial aspects of the ground station location is not entirely superfluous for this project.

5.4.1 Specific location Ground Station

The LEO satellites should always be in view of the GEO mirror. The minimum angle for which this requirement is fulfilled is calculated to be 130 degrees in section 5.1.1. This means that certain combinations of ground station locations should be excluded to fulfill the 130 degrees requirement between the GEO mirrors. Potential locations have also been analyzed, to make sure local geographical conditions permit building a ground station. Furthermore, commercial aspects have been taken into consideration. Although during the trade-off, some aspects weighed more than others. For example the annual cloud coverage weights more than logistical aspects, since the efficiency of the laser system is strongly affected by weather attenuation. However political aspects should not be underestimated with respect to the location of the ground stations, since the ground stations are preferred to be in a politically stable country. The reason for this is that the Heliodromus project is very complex and concerning the large scale of the different space and ground components, the project requires to be designed in a internationally cooperative program.

Finally, two locations for the ground stations have been selected. One in the state of Arizona in the United States of America and a second one in Egypt in North Africa. The geographic coordinates are approximately "33^o latitude, $112,5^o$ longitude" for the ground station in Arizona and "30^o latitude, 32^o longitude" for the ground station in Egypt. Regarding the logistics and commercial considerations, the distances to the cities in the vicinity have also been taken into account during the trade-off. Table 5.4, shows the biggest cities in the vicinity of the ground stations with the regarding population.

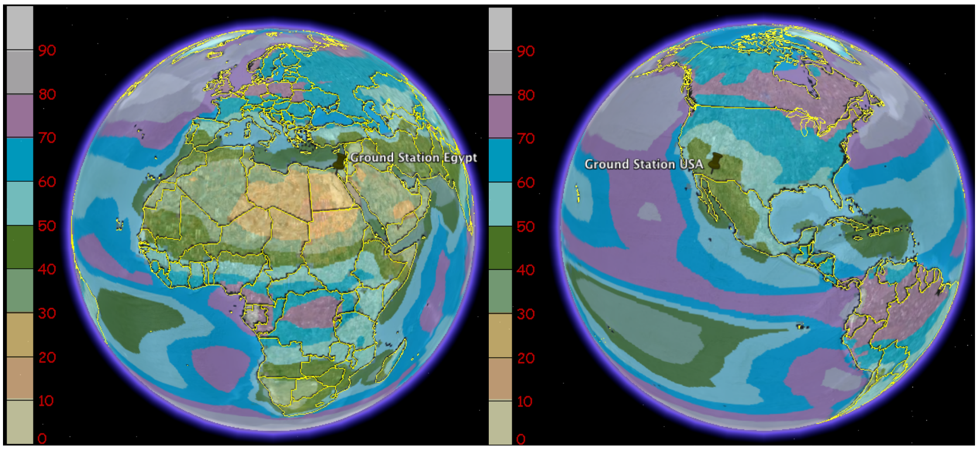
Table 5.4: LEO and GEO orbital parameters overview

US Ground Station			**Egypt Ground Station**		
Cities nearby	**distance**	**Population 10^6**	**Cities nearby**	**distance**	**Population 10^6**
Phoenix	145 km	1.55	Cairo	68 km	7.95
Tucson	256 km	0.53	Suez	61 km	0.48
San Diego	344 km	1.35	Ismailia	90 km	0.75

Detailed argumentations for the determination of the mentioned ground station locations are discussed in subsections 5.4.2 and 5.4.3.

5.4.2 Clouds and Aerosols

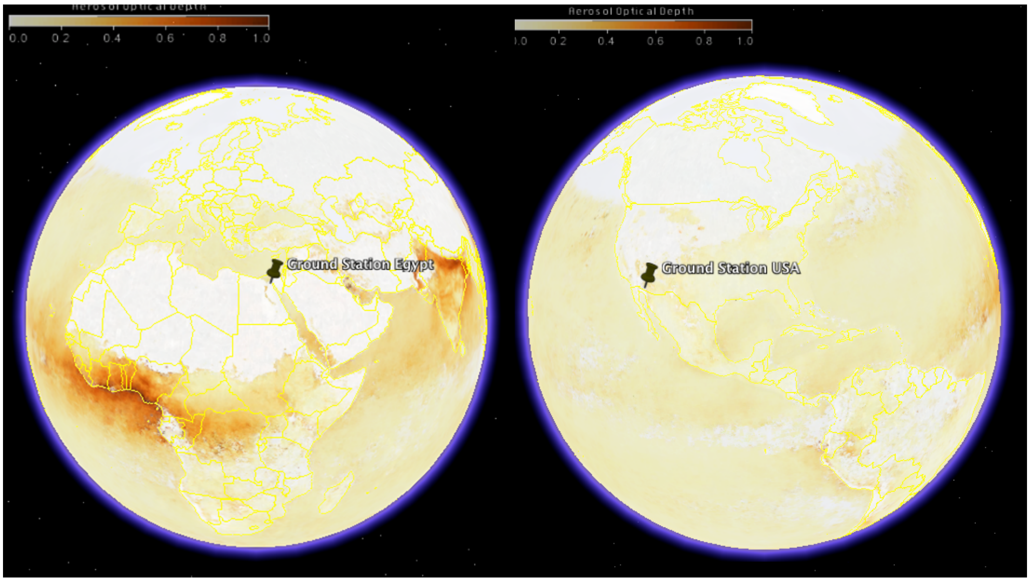
Cloud coverage The laser is highly sensitive to weather attenuation. Therefore, a minimum amount of cloud cover is desired. The International Satellite Cloud Climatology Project[75] provides data about the annual average cloud coverage percentage over the period of twenty years (until 2001). From figure 5.10 the mean annual cloud coverage is shown for both ground stations. It can be seen in figure 5.10a that the ground station in Egypt is optimally located with respect to cloud coverage, as where the ground station in the USA is positioned not optimal. Maneuvering the ground station in the cloud cover related optimum area results in mountainous terrain, which would not be desirable. It is important to keep in mind that the ground stations should be separated by the same angle as the mean anomaly between the GEO mirrors (130 $[deg]$), in order to be positioned directly beneath the GEO satellites.



(a) Cloud coverage Egypt (b) Cloud coverage USA

Figure 5.10: Mean annual cloud coverage $[\%]$

Aerosols Aerosols are solid and liquid particles suspended in the atmosphere, such as dust, sea salts, volcanic ash and smoke. The data on aerosol levels come from the MODIS sensor on NASA's Terra satellite[78]. The aerosol optical depth product, as indicated in figure 5.11, is a measure of how much light airborne particles prevent from passing through a column of atmosphere. Aerosols tend to absorb or reflect incoming laser light, thus reducing visibility and increasing optical depth. An optical depth of less than 0.1 (palest yellow) indicates a crystal clear sky with maximum visibility, whereas a value of 1.0 (reddish brown) indicates very hazy conditions[77]. Figure 5.11 give the aerosol optical depth for november 2008, which is representative for the annual aerosol optical depth for the region of interest. From figure 5.11 it becomes clear that none of the ground stations suffer from high aerosol optical depth values, meaning that the amount of laser energy lost due to aerosol will be kept to a minimum.



(a) Aerosols Egypt (b) Aerosols USA

Figure 5.11: Aerosol optical depth [%]

5.4.3 Seismicity

Determining the location of the ground station requires a broad research in various aspects, which sometimes do not show any interrelations.

The analysis of the seismicity of a potential ground station location is one such an aspect. The ground station is a valuable component of the whole system. Therefor natural hazards such as earthquakes should be taken into account and avoided. During the trade-off for the location of the ground station, the seismicity levels have been taken into consideration, as can be seen in figure 5.12 and 5.13 where pale indicates low hazard, progressing through green and yellow to red, which indicates high threat.

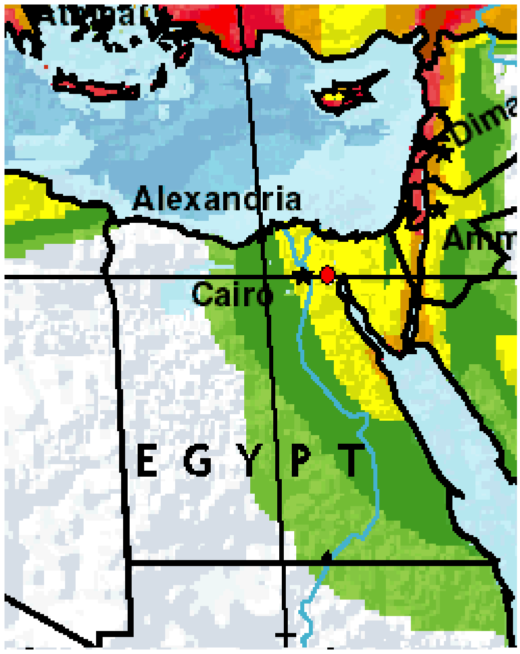


Figure 5.12: African seismic hazard map [80]

For the determination of the most optimum location with respect to seismicity levels, first the global hazard map is investigated, then the local maps for the potential locations is analyzed. For the ground station in Egypt the local hazard map in figure 5.12 has been studied carefully. The ground station is indicated with the red dot. In combination with the local Clouds & Aerosols maps Egypt, North Africa was shown to be the best location.

Concerning the exact location of the ground station in the United States of America, the same procedure as with the ground station in Africa has been performed. Performing a trade-off, looking at the local Clouds & Aerosols maps of the regarding region and the local seismicity maps shown in figure 5.13.

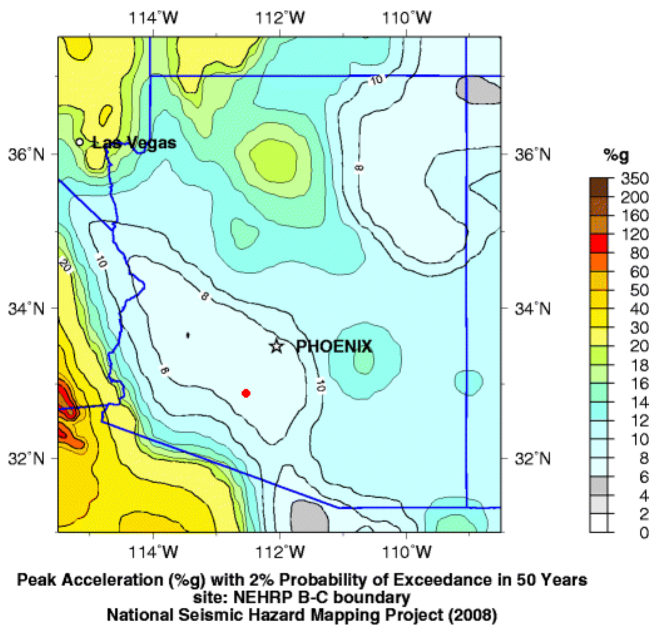


Figure 5.13: Seismic Hazard Map Arizona, USA [81]

As can be seen in figure 5.13, the seismicity levels are relatively low in vicinity of the Arizona ground station. The ground station is indicated with the red dot in figure 5.13.

Chapter 6

Solar Energy Conversion

This chapter deals with the collection and conversion of solar energy into electric energy. Making it ready for transmission to Earth, which is covered in chapter 7

There are two options that need some further evaluation for the Solar Energy Conversion (SEC), namely Concentrated Photovoltaics and Multi Junction Thin Film. These two options are discussed in section 6.1. Then a system description, with earlier set requirements, is given in section 6.2. After that the attitude control will be considered in section 6.3, thermal management in section 6.4 and the actual design of the collection structure in 6.7.

6.1 Concentrated Multi Junction vs. Thin Film

In the following sections the two options for SEC are discussed and based on this discussion a decision is made between those two options.

6.1.1 Concentrated Multi Junction

Concentrated Multi Junction Devices are very promising in this context, because these type of cells have the highest efficiencies of all current available cells. However, the worldwide yearly production is low and the cells are expensive and difficult to make, however these disadvantages can be compensated using concentrators.

For the Concentrated Multi Junction option, the Lattice mismatch GaAs Solar Cells are considered with kapton mirrors as concentrators. With an efficiency of around 35 %, these type of cells have one of the highest efficiencies at this moment. Kapton mirrors are considered to complement the solar cells. Kapton is a light, foldable material and has a great heat tolerance and is already widely used in space applications, for instance in space suits.

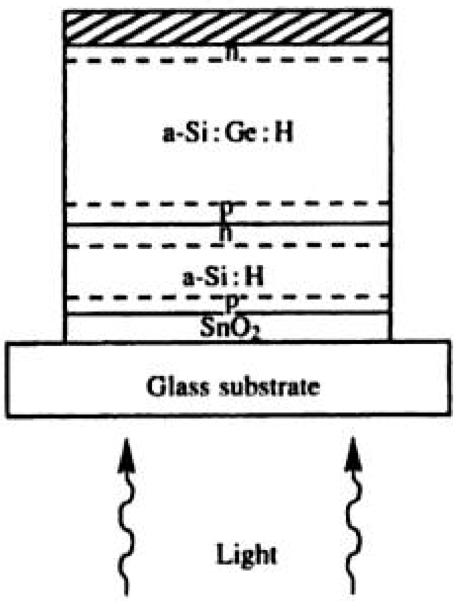


Figure 6.1: Lay up of the triple junction thin film.[76]

6.1.2 Thin Film

Thin film is the other possible option for the solar energy conversion subsystem. With an efficiency of only 12 $\%$ this option is at first glance not as promising as the Concentrated Multi Junction one. Further investigation shows however that this option also has a number of advantages. Compared to the previous option it is relatively cheap, flexible, light and has a lot of room for further improvements.

Triple junction thin films composed of Hydrogenated Amorphous Silicon ($a - Si : H$), Hydrogenated Amorphous Silicon Germanium alloys ($a - Si : Ge : H$) and Tin Oxide (SnO_2) are considered for the final design in section 6.1 This type of cell is very well suited for this project. For medium earth orbit and geostationary orbits the lifetime of solar cells is mainly limited by charged-particle irradiation. Hydrogenated amorphous silicon ($a - Si : H$) solar cells show good irradiation resistance properties, which may enable the application of these solar cells in space missions, especially in high-radiation environments. It has been determined that thin-film silicon solar cells have greater radiation tolerance compared to conventional solar cells based on crystalline Si or GaAs. Furthermore, $a - Si : H$ cells exhibit low-temperature annealing behavior, meaning damage induced by irradiation is reversed when subjecting the cells to a temperature in the range of 100^o C. In addition, thin-film silicon solar cells can be produced on flexible substrates,inexpensively and lightweight, because no cover-glasses are needed to protect the cells from irradiation. At this stage, the Thin Film has an efficiency of around 12 $\%$, but in the near future efficiencies of above 20 $\%$ are expected[76].

The options have now been narrowed down to the Lattice mismatch GaAs Solar Cells with kapton mirrors as concentrators and the triple junction Hydrogenated Amorphous Silicon Germanium alloys. Both options have their pros and cons: the main advantage the concentrated PV have is the high efficiency and a relatively small amount of expensive PVs are needed. The Thin Film on the other hand, have a great power density, is cheap and has a lot of room for improvement. An overview of the important properties of each type are given in table 6.1:

Table 6.1: Solar Energy Conversion Properties

	Concentrated Multi Junction	**Triple Junction Thin Film**	**Unit**
TRL	7:Technology Demonstration	4: Proof of Feasibility	
Begin of Life Efficiency	35	20	[%]
End of Life Efficiency	25	35	[%]
Power Density of Complete Array	400-600	3000-3500	[W/kg]
Surface Power Density	3400489	200-275	[W/m^2]
Cost	5-10	2-4	[$/W]
EPT	1-2	1-2	[yrs]

6.1.3 Conclusion

It must be said that both options have their advantages, however in terms of market viability, the thin film option has the best credentials. Thin film is not only cheaper to produce, but also has a larger power and package density. Consequently, less launches will be needed. Another advantage of thin film is that it is easy to assemble and maintain, reducing overall costs. The main drawback of thin film is the large surface area needed, with the accompanying amount of cables. This puts a lot of extra weight to the system, but it is considered that the total cost of thin film will be lower than Concentrated PV. Hence, the triple junction thin film will be used in the final design.

6.2 Subsystem Description

In the previous section, it was decided to use triple junction thin films composed of Hydrogenated Amorphous Silicon (a-Si:H), Hydrogenated Amorphous Silicon Germanium alloys (a-Si:Ge:H) and Tin Oxide (SnO_2) for the final design. In this section, properties of this solar energy conversion type is investigated and a final design is made.

Requirements The main requirement is market viable energy delivery to Earth, together with an output of 1 GW. This requirement led to the following initial requirements for the SEC system. The most simple requirement is that the deployed surface area should be as small as possible. Related to this is the requirement on the configuration of the system to limit the amount of launches needed, which is dependent on the volume and mass. At this stage, an update of these requirements can be given. The output requirement still holds, but can be made more specific. The SEC system should have a 7 GW output in space in order to still have a 1 GW output on earth assuming an end of life efficiency of 15 %.
In addition, the total costs (i.e. production costs, launch costs, assembly costs, operational costs and maintenance costs) should be as low as possible, but this is a more general requirement. The requirements influence the specific characteristics of the SEC system and are mentioned in section 6.6. But first two important aspects have to be investigated, namely Thermal Management and Attitude Determination and Control.

6.3 Attitude Determination and Control

In order to make efficient use of the sun and its energy, the solar arrays need to be pointed toward the sun during operation. This is done by the Attitude Determination and Control subsystem that is connected to a solar array drive mechanism.

Depending on the incidence angle of the sun, the SEC device will generate a certain power output. This is a wide range, so compared to the ADC subsystem of the transmission device, the ADC for the SEC subsystem does not have to be that accurate. Therefore an optimum has to be found between costs, weight and accuracy. The SEC device has to be three-axis stabilized, so that its orientation is fully controlled relative to three axis.

For the attitude determination of the solar arrays, sensors are used. In total eight sensors are implemented:

- **Four rate gyros**, these are two-axis rate gyros and two of them are enough to provide angular rate measurements along the three axes of the array. The two others are thus used as back-ups.
- **Two digital Earth sensors**, one nominal and one redundant, are used to measure angular displacement about the pitch and roll axis.
- **Two digital Sun sensors (SSD)**, one nominal and one redundant, are used to measure angular displacements around the yaw axis.

  For the attitude control, actuators are used.
- **Three magnetic-bearing reaction wheels (RRPM)** are used to apply torque to the satellite and thus to rotate it about one of the X, Y or Z axes.
- **Two magnetic torquers (MAC)** create a torque, through interaction with the Earth's magnetic field, which are used to control the speed of the reaction wheels.
- **Two types of thrusters (burning hydrazine)** each producing a force to adjust the position.

6.4 Thermal Management

In this section the thermal management of the SEC system is covered. It was already established that, in order to have 1 GW energy output on Earth at EOL, an output of 7 GW is required. With an end of life efficiency of 15 %, this means that the SEC system receives a 46.5 GW solar energy input.

The efficiencies of most solar cells are related to the temperature. For most PV cells the optimum temperature is around 25 degrees celcius, and the operation temperature range is between zero degrees to 80 degrees. However, Hydrogenated Amorphous Silicon has great heat properties, as can be seen in figure 6.2

With ten LEO satellites 4.65 GW incoming solar energy per satellite is needed. With an absorption of 0.75 [4] of silicon cells and an energy efficiency of 0.15 a total of 60 % of the incoming energy is turned into heat, i.e. 2.7 GW. In order to get rid of this waste heat, the energy has to be rejected into cold space. The required radiator area can be calculated using the Stefan-Boltzmann law given by equation 6.1. Q is the radiated power, ϵ the emissivity factor, σ Stefan's constant ($5.6703 \cdot 10^{-8}$ W/m^{2}K^4), A the radiating area and T the temperature difference between the panels and outer space. It is assumed that the backside of the solar panels can be used as effective radiation surface pointed to outer space with equal size (3.44 km^2), covered by black paint with an emissivity value of 0.9.

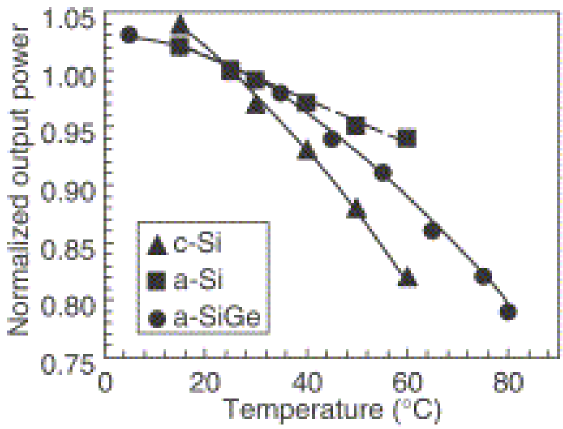


Figure 6.2: Solar Cell Efficiencies versus Temperature.[111]

$$Q = \epsilon \cdot \sigma \cdot A \cdot T^4 \tag{6.1}$$

Using equation 6.1 and assuming the temperature in outer space is 0 K, the value for T is 352 K or 79 $°C$. This value is reasonable for solar panels, hence no additional radiators are needed.

6.5 Technology Readiness Level

This triple junction thin film is already used in practice. However, these cells have a begin of life efficiency between 10 to 12 %. Begin of life efficiencies of around 20 % and are at this stage lab tested only., However is expected to be ready in 10 years from now, setting the TRL level currently to 4 [76].

6.6 Resource Budgets

In order to determine the costs and dimensions of the SEC system in the final design, specific characteristics of the triple junction thin film have to be mapped first.

Efficiency Budget The cell efficiency at the end of life is influenced by various aspects. The proposed triple junction thin film shows good irradiation resistance behavior. They also exhibit low-temperature annealing behavior, meaning damage induced by irradiation is reversed, when subjecting the cells to temperature in the range of 100 degrees Celsius. It is assumed that these cells will degrade rapidly in space, but will stabilize after a short period of time due to their annealing behavior. It is therefore estimated that the Thin Film cell, with a begin of life efficiency of 20 %, will have an estimated efficiency of around 15 % after 10 years.

Density Budget The power density is the power output per unit mass. In this case, the entire mass of the subsystem is considered (i.e. solar cells, cables, support structure, etcetera). According to the NASA study about Space Solar Power Using Ultra-Lightweight Arrays[82], the power density is 2023 Watt per kilogram for an end of life efficiency of 8 % to 9 % . Assuming that the mass of our subsystem will be the same, but with an end of life efficiency of 15 %, the power density of 2023 Watt per kilogram is multiplied by a factor 1.6.

Surface Power Density The surface power density is the power output per unit area. The average Solar Constant in space is 1358 Watt per square meter. With an begin of life efficiency of 20 % and an end of life efficiency of 15 %, the surface power density will be between 205 and 275 Watt per square meter .

Energy Budget The energy payback time of single layer a-SI thin film with an efficiency of 6 %, is 1.4 years [83]. In our case, triple layer thin film cells are used, which of course will result in an energy increase during production. However, it is assumed that this increase will be canceled out with the much higher begin of life efficiency of 20 % of the triple junction thin film. Therefore the energy payback time will remain approximately 1.4 years.

Cost Budget Various sources report that within a few years the price of thin film modules will reach the 1 dollar per Watt price [84]. Currently, complete a-SI modules are already sold for 1.85 $/W by Aten Solar[85]. The triple junction thin film in our system consists of multiple layers and it is therefore expected that a 1 $/W price cannot be reached. Also, in space applications additional costs have to be taken into account, such as costs for extra support structures. On the other hand, future trends show cost reduction, due to learning curves, high demand and supply, and optimizations of the processes. The costs of the entire subsystem is estimated to be 2 $/W in 10 years from now. These (end of life) results are given in table 6.2.

Table 6.2: Properties Triple Junction Thin Film

Triple Junction Thin Film		**Unit**
TRL	4	[-]
Cell efficiency	15-20	[%]
Power Density of Complete Array	3000-3500	[W/kg]
Surface Power Density	200-275	[W/m^2]
Cost	2-4	[$/W]
EPT	1-2	[yrs]

With these data, the dimensions and specifications of the collection design can be determined.

6.7 Collection Design

The solar cells are deployed using a 150 by 150 meter carbon fiber reinforced polymer boom structure. These booms are deployed from a center position of the array. In the center a box structure is placed where the booms and the thin film solar cells are stored during launch. These thin film solar cells will then deploy between the booms. A graphical representation of the solar panel is represented in figure 6.3.

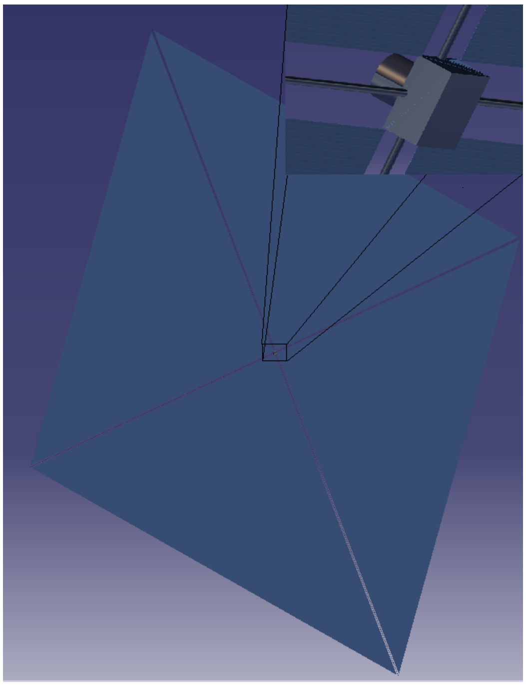


Figure 6.3: Artist impression of a 150 m (CFRP) boom structured solar array

At this moment a 20 meter and 50 meter boom structure has already been qualified and tested for deployment on the ISS. ESA is now developing a 150 meter boom structure for deployment on the Sail Tower solar power system which is expected to be ready in 2012[82].

Table 6.3: Mass and volume considerations

Mass Considerations			Volume Considerations	
Type	Qualified	Developing	**Characteristic**	**Value**
Dimension	20×20 $[m]$	150×150 $[m]$	Total area required	34.4 $[km^2]$
Area	400 $[m^2]$	22.500 $[m^2]$	Energy required	7 $[GW]$
Storage volume	0.108 $[m^3]$	6.1 $[m^3]$	# panels	1530
			Power output per panel	4.6 $[MW]$

The amount of solar cells needed for the entire mission is calculated using the solar area from and the area per panel, given in table 6.3. This leads to 1530 solar panels needed for the entire mission, as can be seen in table 6.3. One solar panel therefore needs to produce 4.6 MW, when divided by the specific power this leads to an allowable mass of one solar panel of 1417 kg.

Chapter 7

Transmission

A disadvantage of Heliodromus is the complexity of the transmission system. The energy must be transferred via a long route, which is described in section 7.1

7.1 Subsystem Description

Before the energy is received on Earth it has already traveled for over 70000 km. The journey starts at the LEO satellite where all the solar energy is transformed into various laser beams containing 1 MW of power. Each of these different laser beams will hit a small parabolic mirror that will concentrate the slightly diverging laser beams. The mirrors should be constructed in such a way that all laser beams are focused into the same focal point. This focal point will also correspond with the focal point of another mirror which will be significantly larger in size and weight and will concentrate all laser beams into one single beam. The previous described constellation of mirrors and lasers will act as one big platform that is oriented indifferently through time.

In this way the large laser beam can be directed perpendicular to the geostationary orbit plane at all times, which is important since otherwise the laser beam would not be able to be directed toward the GEO satellite without being interfered by the structure of the LEO satellite itself. In order for the laser beam to hit the geostationary satellites, it has to be deflected one last time by another mirror that has its own stabilization system independent from that of the platform. From now on this mirror will be referred to as the 2^{nd} LEO mirror. At geostationary altitude the laser beam must hit a mirror once again in order to radiate in its final direction toward the ground station. There will be one such mirror for each LEO satellite since the mirror has to be directed at an angle that depends on the exact location of the ground station as well as the location of the LEO satellite.

Requirements The system's lifetime has been set to 10-15 years. This lifetime has been set early in the project and is primarily based on the expected lifetime of the PV cells. However, the lifetime of the transmission components has not been treated before. The mirrors are expected to have a minimum lifetime as long as the total system's lifetime. The lasers, as they are currently made, are exposed to enormous amounts of heat and resulting degradation and usually have a life expectancy of only a few years. A common lifetime for a relatively low power laser is 10000 hours, i.e. 1 year[64]. As the lasers can not be replaced, one of the requirements that must be set is that the technology of the lasers must improve and a lifetime of 10 - 15 years must be guaranteed, for the system to be feasible. Moreover, power output of the lasers over its lifetime must be constant.

The most important requirements for the transmission components involve the accuracies of the optical components. This includes the pointing accuracy of the lasers and the mirrors, the perfection of the mirror shape and the reflection of the mirrors. An early stated value of 0.1 deg pointing accuracy of the optical components has been changed into a much more accurate value, because over a distance of 36000 km this already gives a

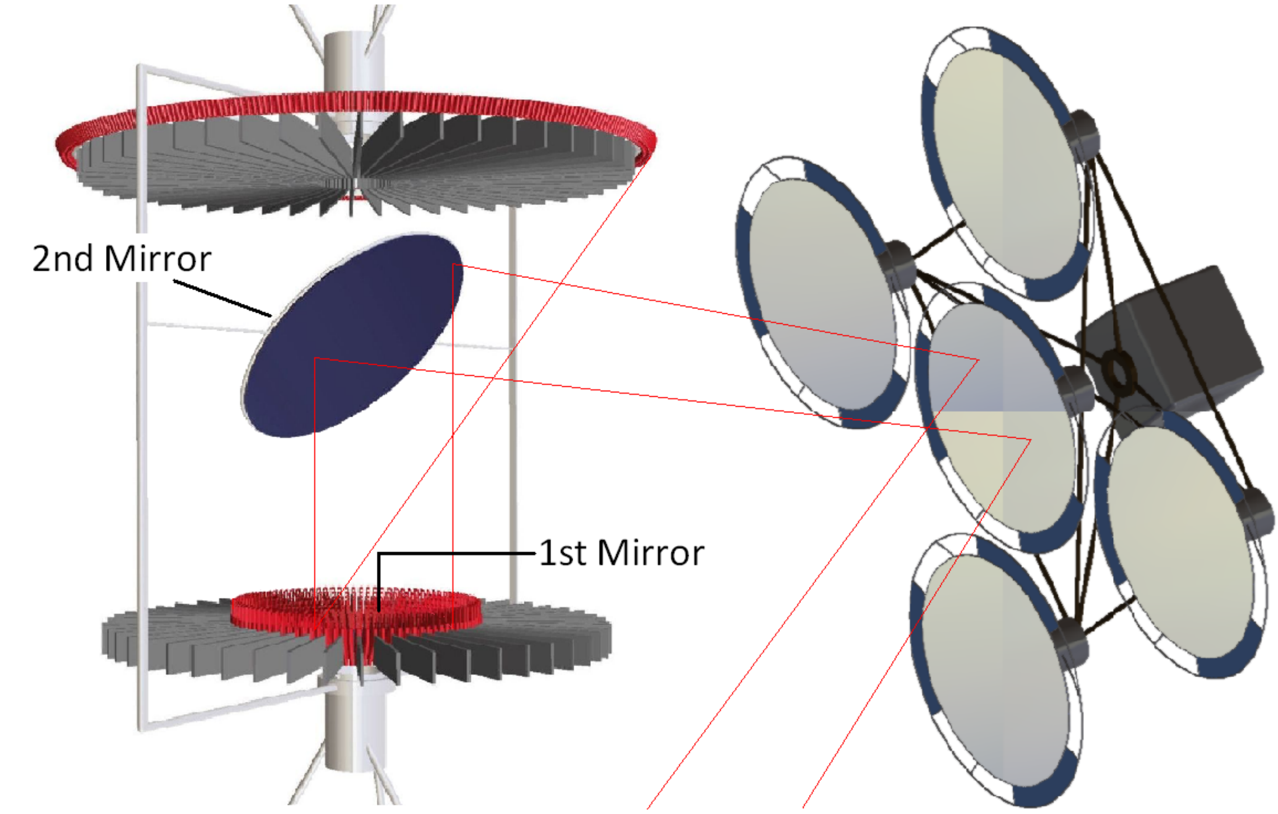


Figure 7.1: The laser transmission system with mirror controlled laser beams in LEO(left) and GEO mirrors(right)

position offset of 65 km. The updated value for the pointing accuracy has been set to 3 times less accuracy than the Hubble telescope, i.e. 30 mas (milliarcsecond). It is expected that a pointing accuracy better than the Hubble telescope is unreasonable, since the Hubble only tracks static objects, while Heliodromus tracks dynamic objects. Lasers and mirrors should be manufactured such that the 30 mas pointing accuracy can be achieved. Moreover, the stabilization system must account for the same accuracy or better. The mirrors should be optimized for both shape and reflectivity. It is assumed that the mirrors are perfect spherical mirrors, hence no accuracy losses occur when the laser hits the surface, except for optical aberrations. Mirrors should thus be manufactured with negligible errors. In the calculation of efficiency a reflective efficiency of the mirror of 99.5 %[52] has been assumed, hence for the system not to fail this reflectivity factor must be set as a requirement. Material used for the mirror and shape optimization must guarantee this value.

Requirements on mass, volume and costs are not set for the mirrors and the lasers. These parameters were kept variable. Mass, volume and costs must be optimized with improving knowledge about the technology and must be kept as low as possible. No quantative restrictions can be stated.

7.2 Transmission Design

In the following subsections various laser, mirror and other optical considerations are presented. First the choice of the laser is outlined in section 7.2.1. Then the angular coverage will be considered in subsection 7.2.2, which relates to the dimensions of the mirror given in subsection 7.2.3.

7.2.1 Laser Design

In this section the design of the laser array is described. When the requirement of 1 GW on earth has to be met and 10 LEO satellites are considered the output for the laser system on one LEO satellite has to be 0.4 GW. To generate the laser beam, 1 MW lasers are used. Although there is not one source that describes 1 MW lasers with an efficiency of 60 %, both specifications are made seperately and it is assumed that already or on short term a system with such specifications will be available [12]. To gain 0.4GW of output power multiple laser systems are needed. This requires 400 lasers per LEO satellite. To make this separate lasers function as one some interference problems have to be encountered. In figure 7.2 an overview of the effects of interference of two electromagnetic waves with the same wavelength is presented. When two waves are exactly 180 degrees out of phase the complete wave front is zero. But when all laser operate in phase, the output power is at maximum. This requires phase locking of the separate lasers. The result of phase locking is that it makes several lasers act as one spatial coherent laser source. Phase locking is already used in laser arrays, however the locking between different arrays is a new technological challenge. When the lasers get out of phase, large losses can occur. This also sets challenges to mirror design because here the phases difference between the waves occurs when the beams are reflected. At the mirrors also an other optical loss occurs, which is described in section 7.2.5.

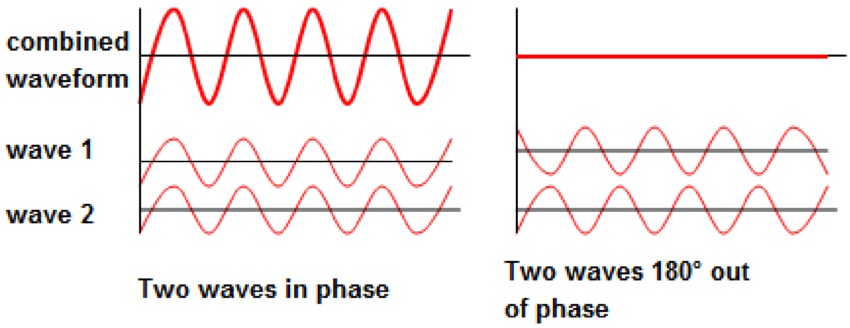


Figure 7.2: Combined Waves

7.2.2 Angular coverage

In order to guide the beam toward the ground station, both the GEO as well as the 2^{nd} LEO mirror must be able to cover a range of different angles.
First of all the range for the 2^{nd} LEO mirrors will be established. From figure 7.3 it can be seen that the mirrors have to deflect the laser with an angle, depicted as α, which will stay between $90 \pm \arctan(\frac{7770}{42164})$ degrees. This results in an angle between 79.5 and 100.5 degrees at all times. In order to do this, the mirrors will need to be inclined with an angle half of those values. Hence maximum inclination with respect to the laser beam will be 50.3 degrees and minimum inclination will be 39.7. All angles in between will be adopted in a cyclic manner. This cycle will have the same period as the orbit time of the LEO satellite. The maximum angular velocity will be reached when the inclination amounts to 45 degrees. At that time the speed of the LEO satellite perpendicular to the GEO orbit plane will be approximately 7 $\frac{km}{s}$. Therefore the angular velocity will be $1.66 \cdot 10^{-4}$ $\frac{rad}{s}$. On top of that, the2^{nd} LEO mirror must also be able to make a 360 degree turn around its principal axis. The period of this movement will be equal to the orbit time of the GEO satellite, namely one day. This results in an angular velocity of $7.27 \cdot 10^{-4}$ $\frac{rad}{s}$. However, due to the fact that the LEO satellite is not always in sight of the same GEO satellite, the mirror will be obliged to make a near 180 degree turn as fast as possible. This situation occurs twice a day.
Next to the 2^{nd} LEO mirror, also the GEO mirrors have to change their orientation throughout time. This

is dependent on the ground station and the position of the LEO satellite. It was already determined that the angle between the LEO satellite and the equator as seen from a GEO satellite will be between 10.5 and -10.5 degrees. The ground stations are located at a northern latitude of 32.5 and 29.8 degrees. Seen from geostationary altitude these ground stations are located at an angle of 5.5 and 5 degrees respectively compared to the equator. Therefore it can be stated that GEO satellite 1 will need to orientate its mirrors between 98 and 86 degrees compared with the geostationary orbit plane and that GEO satellite 2 will need to orientate its mirrors between 97.75 and 85.75 degrees compared with the geostationary orbital plane. Compared to the Earth's rotating axis both GEO satellites will have to adopt angles between 80 and 100 degrees.

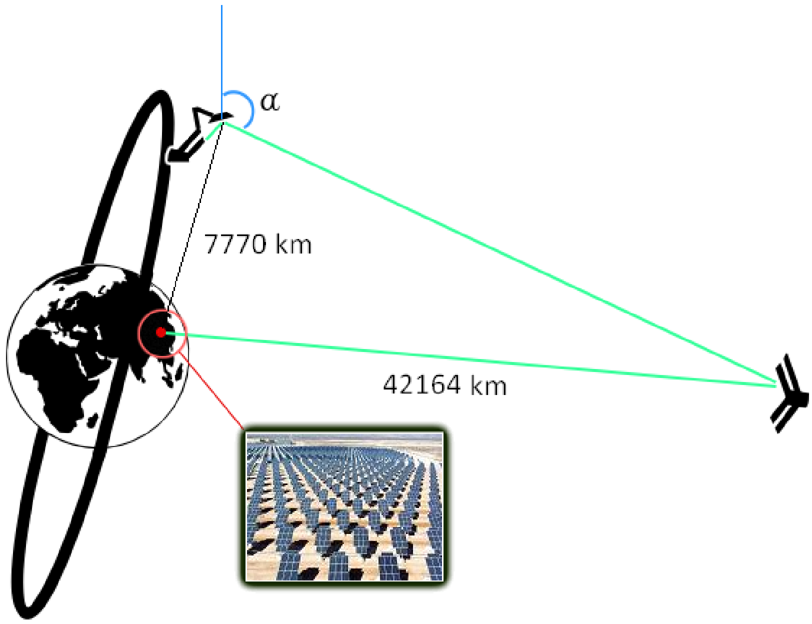


Figure 7.3: Angular coverage indication

7.2.3 Mirror Design

The entire system depends on energy transfer that is being controlled by massive mirrors. For each LEO satellite to operate properly, 3 mirrors will be needed (2 on the LEO and 1 on a GEO satellite). The dimensions of these mirrors are dependent on each other and can therefore be optimized. In this section the mirror system will be optimized, considering launch costs. The costs of the mirrors are constant per volume, hence costs can be minimized by getting more mass in LEO. One of those three mirrors will be a mirror of a GEO satellite which reflects the laser to the ground station. And since one mirror cannot reflect several beams with different incoming angles to the same spot on earth, one such mirror is needed in GEO for each LEO satellite. The system also needs a mirror integrated in the LEO satellite itself, this to effectively increase the laser system aperture. On top of that, the system needs a second LEO based mirror to track a GEO satellite and guide the laser to that satellite. This 2^{nd} LEO mirror must be able to be maximally inclined with an angle of 50.3 degrees to the laser beam as stated in section 7.2.2. While doing that, it is important that the mirror still reflects all of the laser beam preventing further efficiency losses. This means that the 2^{nd} LEO mirror needs to be almost 1.6 times as big as the one that combines the different laser beams into one beam since:

$$cos(50.3) = 1.6$$

As already mentioned the mirrors have a relation, namely the following:

$$r_{spotsizeGEO} \cdot r_{mirrorLEO} = 0.61 \cdot \lambda \cdot Distance$$

Where λ represents the wavelength of the laser, which was chosen to be 1064 nanometers. $Distance$ is taken to be 46163 kilometers, i.e. the farthest the two satellites will ever be separated from each other. $r_{spotsizeGEO}$ is the radius of the spot of the laser beam on a GEO mirror and where $r_{mirrorLEO}$ is the radius of the 2^{nd} mirror that effectively increases the system aperture. Working this out leaves the following:

$$r_{spotsizeGEO} \cdot r_{mirrorLEO} = 30$$

It can be stated that both mirrors will need the same radius when this radius is taken to be 5.5 meters. This will be taken as a starting point while optimizing the most optimal radii of the mirrors. In order to determine how big both mirrors really need to be, several other aspects have to be taken into account. For instance, it is easier to lift a heavy mirror to LEO than to GEO. Secondly, an other parameter of great importance for the GEO based mirror is the pointing accuracy. This was taken to be 30 mas for the mirrors, which makes it three times less accurate than the Hubble Space Telescope. This translates into a radius of 6.7 meters larger then previously calculated for the GEO located mirror. Note that if the LEO mirrors radii increases with a factor X, the spotsize in GEO needs to decrease with that same factor in order for the aforementioned equations to hold true. At this point the surface of all mirrors can be calculated, however one variable remains.

$$\begin{aligned} A_{mirrorGEO} &= (6.7 + \frac{5.5}{X})^2 \cdot \pi \\ A_{mirrorLEO1} &= (5.5 \cdot X)^2 \cdot \pi \\ A_{mirrorLEO2} &= (5.5 \cdot 1.6 \cdot X)^2 \cdot \pi \end{aligned}$$

It was already mentioned that LEO orbit is easier to achieve. Therefore the kinetic energy needed to inject both mirrors into their desired orbit should be kept as low as possible. The formula of kinetic energy is:

$$E = \frac{1}{2} M V^2$$

Where mass is taken to depend linearly on the area of the mirrors. The speed is taken to be 9500 $\frac{m}{s}$ for the LEO mirrors and 13500 $\frac{m}{s}$ for the GEO mirror, corresponding with the delta V needed for these orbit injections. The total energy needed will therefore be linearly dependent on the following formula:

$$((5.5 \cdot X)^2 \cdot \pi \cdot 9500^2) + ((5.5 \cdot 1.6 \cdot X)^2 \cdot \pi \cdot 9500^2) + ((6.7 + \frac{5.5}{X})^2 \cdot \pi \cdot 13500^2)$$

From this it follows that the lowest value of total energy needed for injecting both mirrors into their orbit will occur when the variable X has the value of 1.07, leading to a radius of 5.9 meter for the smaller LEO mirror, a radius of 9.5 meter for the bigger LEO mirror and a radius of 11.9 meter for the GEO mirror. In comparison, the James Webb Space Telescope, often seen as the replacement of the Hubble, will have a mirror of 3.3 meter radius.

The mirror in GEO redirects the laser from LEO to the Earth. The distance to the focal point, the focal length, is a function of the curvature of the mirror. When the mirror has its focal point on the Earth the spot size is determined by the Airy disc described in section 7.2.5. The photovoltaic cells on Earth have a maximum intensity they can handle. The focal length and hence the spot size of the mirror is designed on this parameter. The spot size on the ground can be calculated in a linear way. The other way around, the focal length of the mirror can be determined with a given input. D is the diameter of the mirror and h_{GEO} is the height of the mirror in GEO, i.e. 36000 km.

$$d_{Spot} = D \cdot \frac{h_{GEO} - f}{f} \tag{7.1}$$

Using equation 7.1, the 24 m diameter mirror and a ground station with 500 m diameter (see equation 8.2) imply a focal length of the mirror of 1650 km.

7.2.4 Mirror Material

A choice can be made between several materials. P.S. Carlin[33] groups mirror materials in 6 classes. Most important characteristics for mirror are weight, smoothness of surface, CTE (Coefficient of Thermal Expansion), stiffness and thermal conductivity. In table 7.1 the most important characteristics for different mirror material types are stated. This table shows that low density and stiffness result in decreasing reflectivity and smoothness of the mirror. In *Lightweight Mirror Systems for Spacecraft*[33] a combination between composite and metal is proposed. It states: "Borosilicate glass is a mass produced glass with a relatively low working point ($1200^o\ C$). Borosilicate glass is used extensively in small telescope applications because of its availability and low cost. However, its CTE is higher than desired for large mirror applications and its stiffness, like other glasses, is low. Borosilicate glass may be a good choice as a matrix material for a glass matrix composite with increased stiffness and tailored zero CTE."[33].
For the surface coating, most used coatings are made of aluminum, silver or gold. Aluminum coating hardly gets a reflectivity higher then 95 $\%$ at 1064 nm wavelength. Since our systems needs mirrors that reflect approximately 99.5 $\%$ of the incoming radiation, aluminum coating is not an option. Also gold coating, although favorable due to its great thermal radiation qualities, cannot be used because it absorbs a lot of solar radiation over the entire spectrum [48]. Silver comes out as a clear winner from the three but nonetheless its properties would likely still not be sufficient for this system. A type of coating that could be up to the task is the Laser-Line MAX-R Coatings^{TM}. These coatings have a minimum reflectance of 99.0 $\%$ at an incidence of 0 ± 15 degrees for a wavelength of 1064 nm[69]. The laser damage threshold of this coating is 17.7 $\frac{J}{cm^2}$ for a 20 nanosecond pulse, way below the 365 $\frac{J}{cm^2}$ that occurs on the smallest mirror. For a continous wave laser however this damage threshold would increase with a factor of 10 to 100, enough for the Heliodromus system[70]. Lifetime of the mirror and the coating material would therefore probably not be influenced by pure laser radiation but by thermal damage mechanisms associated with the average intensity or space environment related factors. Additional research about this subject remains to be done.

Table 7.1: Mirror Material Selection

Material Group	**Advantages**	**Disadvantage**
Metals/Inter Metallics	Smooth, lighter then glass	High CTE
Carbon	Low density, Low CTE, easy to machine	coating required
Ceramic	Very High Stiffness	Expensive
Glass	Smooth Surface	Heavy Weight
Composites	Low mass, high stiffness, mixing of materials possible and so possible CTE of 0	Highly irregular surface
Polymers	low cost, lightweight	Very High CTE, low stiffness

7.2.5 Optical Aberration

A laser beam from LEO hitting a mirror in GEO will encounter so called optical aberration, disturbances created by the optical instruments. These aberrations have an influence on the spot size of the beam on Earth. Five different errors will be discussed in this section: *Diffraction, Spherical Aberration, Coma, Astigmatism, Pointing Errors and Atmospheric Attenuation.* Most optical errors are concluded to be relatively small, as will be shown. The pointing accuracy of the laser in LEO is an exception.

Diffraction Diffraction of the laser is caused by the interference of the electromagnetic waves when hitting the edge of a mirror. This edge can be the actual edge of the mirror or the edge of the beam hitting the mirror. Both the real and virtual edge cause the same diffraction, hence diffraction can not be prevented. A consequence of diffraction is that the beam diverges and forms a sharp inner circle containing $84\ \%$ of the energy and vague outer circles containing the rest of the energy. A mirror focusing the laser in the focal point can not create a single point of light, but creates a blurred spot. The diameter of this spot is given by formula 7.2, in which only the inner circle, the *Airy disc*, is counted for:

$$d_{Airy} = 1.22 \cdot \frac{\lambda f}{D} \tag{7.2}$$

In equation 7.2, λ is the 1064 nm wavelength of the laser, *f* is the focal length of the mirror and *D* is the mirror diameter. The focal length of the mirror to create the minimum ground spot required on Earth ($1650\ km$) has been calculated in section 7.2.3. For a mirror with $24\ m$ diameter, the size of the Airy disc is approximately $9\ cm$. This size of the laser can be found along the beam at the focal point, i.e. at $1650\ km$ from the mirror. More important is the fact that the spot on Earth consists of only $84\ \%$ of the initially sent energy (excluding atmospheric losses), covered in the Airy disc. This adds to the total efficiency loss.

Spherical Aberration Every spherical mirror encounters spherical aberration. This defect prohibits the mirror to focus the light as a single spot on the focal point. Therefore the image of the laser on the ground will be slightly blurred. The diameter of the spot in the focal point caused by this error, *d*, can be approximated by equation 7.3. *D* is the diameter of the mirror and *f* is the focal length of the mirror as was used in the previous calculations.

$$d = \frac{D^3}{32f^2} \tag{7.3}$$

Using equation 7.3, the diameter of the laser in the focal point at $1650\ km$, caused by the spherical aberration, is $1.6 \cdot 10^{-10}\ m$. When assuming the error grows linearly with distance to the ground, the error will be roughly $3.5 \cdot 10^{-9}\ m$. This optical defect is very small and therefore negligible in the design of the ground station.

Coma Coma aberration occurs when an incident wavefront is being tilted with respect to the optical surface, or is axially displaced. In practice this means that when a laser beam hits a GEO mirror from LEO it has a certain angle with respect to the axis of the GEO mirror. The resulting coma error will create a blurred ground spot on Earth. In figure 7.4 a schematic drawing of the coma effect for a lens is shown, which is similar for mirrors.

When assuming the GEO satellite points the laser to Earth perpendicularly, the maximum offset angle of the laser hitting the GEO mirror will be θ, as shown in figure 7.5. This maximum angle gives a maximum coma error. With a simple geometrical calculation the angle can be determined, in which h_{LEO} is the height of a LEO satellite (1400 km), h_{GEO} is the height of a GEO satellite and R is the radius of the Earth respectively given by 36000 km and 6400 km:

$$\tan 2\theta \approx 2\theta = \frac{h_{LEO} + R}{h_{GEO} + R} \tag{7.4}$$

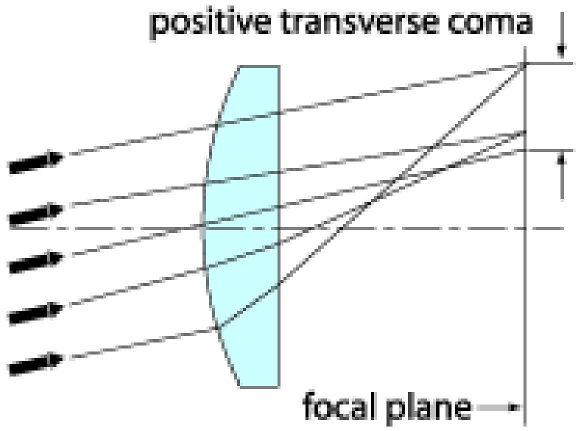


Figure 7.4: Optical Coma [52]

Using equation 7.4 the angle θ is determined to be 0.092 rad. The approximation of the length of the image caused by the coma aberration is given by equation 7.5, in which D and f have the same values as given in section 7.2.5.

$$Lc = \frac{3D^2\theta}{16f} \tag{7.5}$$

For a mirror of 24 m in diameter the error length in the focal point caused by coma is calculated to be roughly 6.0 10^{-6} m using equation 7.5. Assuming linearity, the error on Earth will be 1.3 10^{-4} m. This error is very small and hence negligible.

Astigmatism As with the coma aberration, the astigmatism aberration is also caused by an off-axis light source. The input for astigmatism is again the angle $theta$, as mentioned in section 7.2.5. The approximation of the length of the image on Earth caused by the astigmatism error can be determined using equation 7.6.

$$La = D\theta^2 \tag{7.6}$$

The error length caused by astigmatism is roughly 0.20 m in the focal point and 4.4 m on the ground assuming the mirror's 24 m diameter. This error should be considered, but is still relatively small compared to the 500 m diameter size of the ground station.

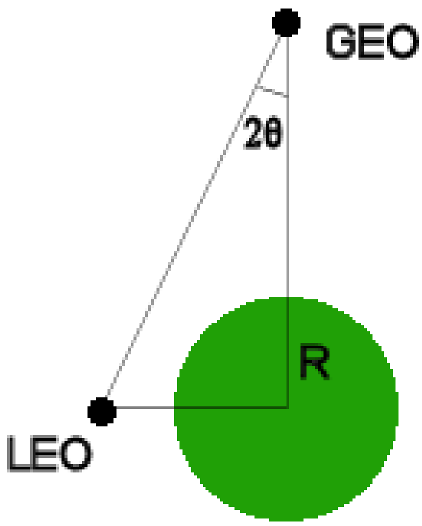


Figure 7.5: Optical Coma Angle

Pointing Errors The laser beam in LEO has a pointing accuracy of 30 mas resulting in a 6.7 m offset in GEO (see section 7.2.3). When the laser hits the mirror with this offset it will encounter a different angle than was expected and will therefore be directed to a different point on Earth. Figure 7.6 shows a schematic view of a laser reflected without and with an error offset of 6.7 m.

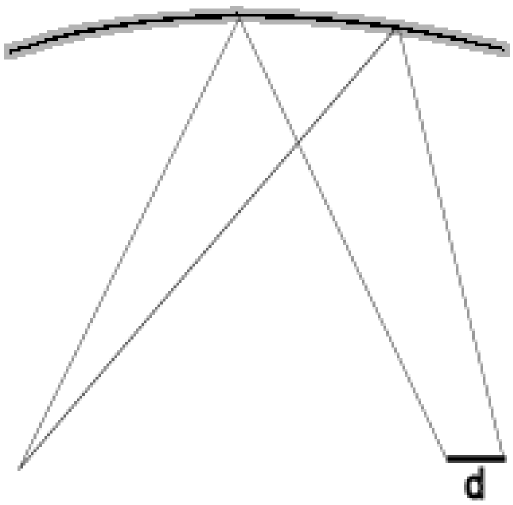


Figure 7.6: Optical Offset

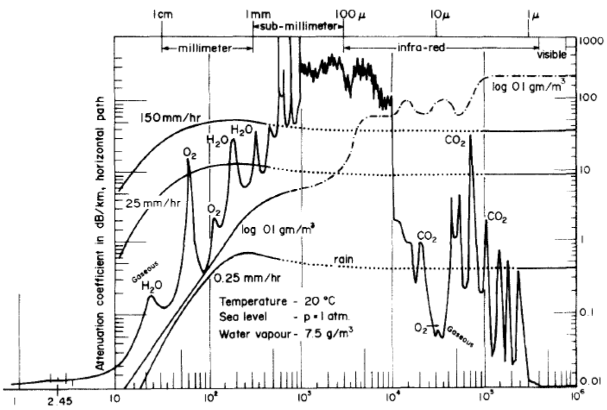
The mirror has a focal length of 1650 km. For a spherical mirror the focal length is half the radius of the mirror's curvature, i.e. the mirror has a radius of curvature of 3300 km. The total circumference of the virtual sphere made by the mirror, s, is 20.7 10^6 m. The change of the angle of the mirror at a distance of 6.7 m offset is then given by:

$$\delta\theta = 6.7/s \cdot 2\pi \tag{7.7}$$

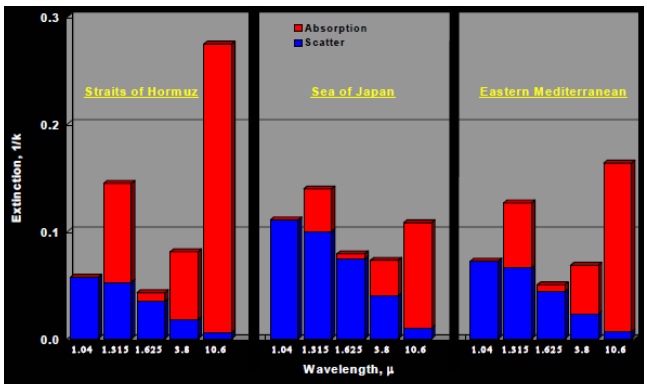
Equation 7.7 gives a $\delta\theta$ of 2.0 10^{-6} rad. An offset angle of 2.0 10^{-6} rad results in a position offset on ground of 150 m over a distance of 36000 km. The size of the ground station has to account for this error and its diameter has to be enlarged by twice the value of it, i.e. 300 m.

Besides the pointing error of the laser also a GEO mirror has a limited accuracy. The accuracy of the mirror is 30 mas. When a mirror has this angular error, it will affect the ground spot position with an offset of 5 m. The ground station has to account for twice the value of this offset , i.e. the diameter has to be increased by 10 m.

Atmospheric Attenuation Atmospheric attenuation includes scattering, reflection, absorption, refraction and diffraction of the laser. The effects of atmospheric attenuation is the loss of energy of the beam, decreasing the overall efficiency and increasing the divergence of the beam. The influence of the atmosphere is assumed to be present only in a range of 1500 km from the Earth. Above this height the density of the particles is too low to affect passing waves. The wavelength for the laser was chosen to be 1064 nm. The decision of using this wavelength is mainly due to the low atmospheric attenuation at this wavelength. In figure 7.8a the drop of attenuation of the beam can be seen clearly. In figure 7.8b it becomes clear that the attenuation at 1.04 μm, close to the chosen wavelength, is mainly due to scattering of the light particles and not by the absorption.



(a) Attenuation due to gaseous constituents and precipitation for transmissions through the atmosphere [39]



(b) Average Extinction (50% probability of occurrence)[34]

Figure 7.8

To give a quantitative analysis of the divergence of the laser induced by the atmosphere is difficult. The divergence angle of the beam in the atmosphere will be roughly 4 μrad [22]. Over a distance of 1500 km this will add 14 m to the beam diameter.

According to research of the Boeing Company, the atmospheric attenuation of vertical laser beams results in a transmission efficiency in the order of 73 % – 84 % [34]. But already 99.5 % efficiency can be reached when the receiving platform is placed at 12 km altitude and beam quality is improved [34]. So raising the platform could result in improving performances of the system. The assumed atmospheric efficiency set is an average of 85 %, as proposed by Kare et al.[27].

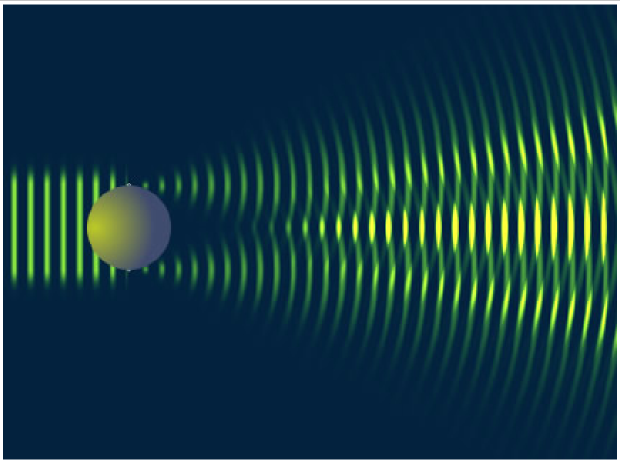


Figure 7.7: Atmospheric diffraction[65]

To summarize, all optical aberrations and their inherent effects are tabulated in table 7.2.

Table 7.2: Summary Optical Aberrations

Optical Aberration	**Effect**	**Value**
Diffraction	Effective Spot Size	84[%]
Spherical	Offset on Ground	$3.5 \cdot 10^{-9}$ [m]
Coma	Offset on Ground	$1.3 \cdot 10^{-4}$ [m]
Astigmatism	Offset on Ground	500 [m]
Pointing errors	Offset on Ground	10 [m]
Atmospheric attenuation	Addition Beam Diameter Efficiency	14 [m] 85 [%]

7.3 Attitude Determination and Control

The laser generated by a LEO satellite has to be pointed via a LEO and a GEO mirror on a spot on earth of 0.2 km^2 as can be read in chapter 8. The required accuracy for a LEO and GEO mirror is comparable to the accuracy of high performance space telescopes like for example the Hubble as stated in section 7.1. The Hubble telescope uses 5 different types of sensors and two types of actuators to reach a pointing accuracy of 0.01 mas. This is an accuracy which is assumed to be sufficient for the Heliodromus project as well. The sensors are used to accurately determine the position and orientation of the system. The sensor types used by Hubble are Fine Guidance Sensors (FGS), Coarse Sun Sensors (CSS), a Magnetic Sensing System (MSS), Rate Sensor Units (RSU) and Fixed Head Star Trackers (FHST). The FGS keep track of 2 stars and hold the telescope in the same position relative to these stars. The CSS, MSS and FHST are used to determine the orientation of the satellite. The CSS measure the orientation relative to the Sun and the MSS relative to the magnetic field of Earth. The FHST uses an electro-optical detector device to map parts of the sky and use this as an orientation reference. The RSU use two gyroscopes to obtain stable pointing. For the Heliodromus project, a "mirror tracker" should be included for both LEO and GEO mirrors. This way accurate pointing with respect to the mirrors can be guaranteed. Also the addition of a GPS system would increase the precision of the system.
Subsequently, the information coming from the sensors is used to steer the actuators and point Hubble, in the Heliodromus case a LEO and GEO mirror, into the preferred direction. One type of actuators are the Reaction Wheel Actuators (RWA). The RWA use a flywheel which can be spun up to 3000 rpm. The change in angular

momentum can then be used to point the telescope to the wanted position. The other actuators used by Hubble are Magnetic Torquers (MT), these are used primarily to control the reaction wheel's speed.

In table 7.3 mass and power estimates can be found for the AD&C subsystem. The FGS is by far the largest component of the AD&C subsystem and has a volume of 1.25 m^3 [110]. The volume of the AD&C system will be in the range of 1.5 to 2 m^3, its mass will range from about 650 to 1000 kg and the power required to operate this part of the system ranges from 145 to 1235 W. A short summary can be found in table 7.4.

Table 7.3: Mass and power estimates for the AD&C subsystem [4],[110]

	Single Unit Mass $[kg]$	**Single Unit Power** $[W]$	**Number Units Used** **[-]**	**Total Mass** $[kg]$	**Total Power** $[W]$
FGS	216	19	3	648	57
CSS	$0.1\ to\ 2$	$0\ to\ 3$	4	$0.4\ to\ 8$	0 to 12
MSS	0.3 to 1.2	< 1	2	$0.6\ to\ 2.4$	< 2
RSU	$1\ to\ 15$	$10\ to\ 200$	3	$3\ to\ 45$	$30\ to\ 600$
FHST	$2\ to\ 5$	$5\ to\ 20$	3	$6\ to\ 15$	$15\ to\ 60$
RWA	$2\ to\ 20$	$10\ to\ 110$	4	$8\ to\ 80$	$40\ to\ 440$
MT	$0.4\ to\ 50$	$0.6\ to\ 16$	4	$1.6\ to\ 200$	$2.4\ to\ 64$

Table 7.4: Short summary of the volume, mass and power AD&C subsystem [4],[110]

AD&C	**Estimation**
Volume	$1.5\ to\ 2m^3$
Mass	$650\ to\ 1000kg$
Power needed	$145\ to\ 1234W$

Although the AD&C subsystem needed for the Heliodromus is comparable to the Hubble telescope in terms of pointing accuracy, an important difference is present. The rate at which the mirrors of the Heliodromus have to turn are higher than the rate at which the Hubble has to turn. The Heliodromus has to turn with a rate of about $1.66\ 10^{-4}$ rad/sec or 0.0095 deg/sec as stated in chapter 7.2.2, while the Hubble tracks static objects. It is expected that the necessity to provide this fast pointing will result in a higher mass, volume and power required. Hence the values for the Heliodromus will be closer to the upper limits determined in table 7.4 or maybe even higher. An important remark on the AD&C system is the fact that it consists of many moving parts like gyroscopes and flywheels. The lifetime of these parts will influence the lifetime of the mission significantly. Adding redundant systems might be an option to extend the lifetime. Repairing in space is also a possibility, but is very expensive.

7.4 Radiation Pressure

One of the issues for maintaining the orbit is the so called 'solar sailing' influence on a GEO mirror. Solar sailing, the propulsion of a craft by the pressure of light from a laser or star has been the object of some interest over the last few decades. Research into solar sailing has reached the point that several satellites[37] have been launched to test deployment of a solar sail and it has been used as a method of attitude control[38]. The initiatives of using it as primary propulsion system are still in its infancy, but it is important to recognize the disturbance it introduces to a GEO satellite containing 5 mirrors reflecting a 1.7 GW laser.

$$F_S = C_R \cdot \frac{I}{c} \cdot S = 1 \cdot \frac{1.7\ 10^9}{300 * 10^6} = 6[N] \tag{7.8}$$

6 Newton of force in space with nothing to push back can be dangerous. Assuming a total mass of a GEO satellite of 45 tonnes the resulting acceleration is $1.4\ 10^{-4}[m/s^2]$.
As the vector of this force will always be away from Earth, a slightly lower orbit than GEO will create the balance of forces needed for the mirror to remain in place. At first sight this change seems negligible but one must bear in mind that the gravitational acceleration in geostationary orbit ($42164\ km$) is already diminished, due to the distance from the earths center, to only $0.22421\ \frac{m}{s^2}$. Thus total acceleration that has to be kept in balance due to the centripetal force will be:

$$0.22421 - 0.00014 = 0.22407\frac{m}{s} \tag{7.9}$$

The only requirement of a geostationary orbit is that it, as the term indicates, stays stationary with respect to the earths surface. Therefore it should have a period of 86164 seconds. Furthermore the formula of centripetal acceleration is:

$$a = \frac{v^2}{R} \tag{7.10}$$

Where R is the distance to the Earth's center and where v is the speed of the satellite which is equal to $\frac{2 \cdot \pi \cdot R}{86164}$. Now equation 7.10 can be rewritten as:

$$a = \frac{4 \cdot \pi^2 \cdot R}{86164^2} \tag{7.11}$$

Knowing the value of a, which is $0.22407\ \frac{m}{s^2}$ found in equation 7.9, we now find the value of R to be around $42138\ km$, a $26\ km$ change. Therefore it should be pointed out that radiation pressure can cause a significant change in orbit height. This would certainly be the case if the power of the system would be scaled up significantly.

7.5 Thermal Management

Although the mirrors and laser are around $99.5\ \%$ and $60\ \%$ efficient respectively, waste heat is generated due to the high power levels that are required. The thermal management for the transmission system is split up into a laser and a mirror.

7.5.1 Lasers

Laser operation efficiency of $60\ \%$ should be possible in the near future, and $70\ \%$ is already reached in laboratory conditions [22]. When using 10 LEO satellites and lasers of 1 MW [34], the input to one LEO satellite's laser subsystem is 0.7 GW according to figure 10.1. This gives a total of 280 MW of waste heat, which has to be removed or could be used for powering an additional heat engine. To remove this heat, radiation of the heat into cold space is needed. A first estimation of the required radiator area is done using the Stefan-Boltzmann law given by equation 7.12. Q is the radiated power, ϵ the emissivity factor, σ Stefan's constant ($5.6703 \cdot 10^{-8}$ W/m^{2}K^4), A the radiating area and T the radiator temperature:

$$Q = \epsilon \cdot \sigma \cdot A \cdot T^4 \tag{7.12}$$

With a radiator temperature of 400 K and an emissivity factor of 1 for an ideal radiator, this results in 0.2 km^2 radiator area. Using a typical honeycomb structure of $3.3\ kg/m^2$ [4] results in a mass of 660 tonnes. Before this amount of heat can be rejected and this structure can be assembled in space, radiator design has to be pushed to new limits due to the high temperature. It should also be noted that new cooling methods can be used such as a liquid droplet radiator. This type of cooling uses a waste heat fluid that is generated as submillimeter droplets that radiatively cool as they pass through space. Later they are recirculated in the system. This type of radiator promises to decrease radiator mass significantly, however this concept was conceived in 1978 and up to this day not much happened in that particular field. An other proposed cooling method is direct hydrogen cooling, where onboard hydrogen is used to cool the system. Most likely the system will also use its excess of heat to increase the pressure of certain onboard gasses that can then be used as a propulsion method.

7.5.2 Mirrors

The mirrors on the GEO and LEO satellites face the largest energy density, because the lasers focus their energy on these mirrors. This section will only focus on some first order estimates of the waste heat in these mirrors.

With two GEO satellite constellations (consisting of 5 mirrors each) the received energy on a GEO satellite is around 1.7 GW. With an efficiency of 99.5 % this results in 8.4 MW of rejected energy. Using again the Stefan-Boltzmann law (equation 7.12) and an emissivity of 0.9, this results in a radiator surface of at least 6400 m^2, which is a significantly large area. For the typical honeycomb structure the mass for the two GEO constellations combined would be 42 tonnes. The mirrors are kept at $400\ K$.

For the mirrors in LEO the incoming energy is $400\ MW$ per LEO of which 0.5 % is absorbed, i.e. $2\ MW$. The radiator area for each LEO is therefore $1530\ m^2$. The mass of this radiator structure will be roughly 5 tonnes. Since the LEO satellite contains 2 mirrors, the radiator size and mass is doubled, hence $3100\ m^2$ size and 10 tonnes.

7.6 Technology Readiness Level

In this section the technology readiness levels of the technologies required for WPT by means of lasers are described. The required technologies are subdivided in three parts. The laser technology will be described in section 7.6.1, the technology of the mirrors for the reflection of the laser beam in section 7.6.2. Finally the readiness of the technologies for the required pointing accuracy are in the final section 7.6.3.

7.6.1 Lasers

Lasers are widely used in cutting application and distance measurements. Large efforts are made in the efficiency of lasers. Research on high power laser is also a large field of interest for defensive applications. Several MW-laser systems, called HEL (High Energy Lasers), are developed by NASA and Boeing[34], but little information about these lasers is open to the public. The Space-Based Laser Integrated Flight Experiment (SBL-IFX) is planned for 2012. The project claims to have made large advances in mass, size and power of lasers for space use[60]. It is difficult to review the technology readiness level of laser for wireless power transmission. Cooling of the lasers is also a bottleneck in the laser technology. Development of removing optical distortion of high energy solid state lasers should make it possible to reach average outputs of 1 MW [31]. Advancements are made in diffusion cooled laser. Designs of radial laser arrays have specifications; diameter 45 cm, 2 m long and weighing 150 kg with a power output of 100 kW according to H.J.J. Seguin[32]. With improvements 1 MW with 1m diameter are possible. This diffusion cooled laser would reduce required radiator sizes. To give a TRL for the laser required for WPT is difficult. The conclusion is that laser is at a TRL of 3 at this moment. Theoretical feasibility is, however demonstrations of laser for WPT are not performed. The technologies for more powerful lasers and weight and size reductions are now the main focus of the research done in the field of lasers.

7.6.2 Mirrors

Recently, the Herschel telescope was launched with a primary mirror of 3.5 m diameter. In 2013 the launch of the James Webb telescope is planned which will have a mirror of 6.5 m diameter. In the design of the SBSP system the mirrors have dimensions varying between 12 m and 24 m. The existing space mirrors have a modular design. By extending these designs, large enough mirrors can be created. The assembly of modular mirror parts is a bottleneck for applying large mirror systems, because of the limited technology readiness level of robotic assembly. Nevertheless, subsystems of large mirrors have already been demonstrated in several space telescopes, hence the required mirror technology is set at a TRL of 6.

7.6.3 Pointing & Stabilization Systems

The Pointing & Stabilization system needs to be very accurate to provide high pointing accuracies of the lasers. The pointing accuracy requirement for the laser system is important. Serious safety issues can arise when the beam will illuminate populated areas. As is stated in section 7.3 the pointing accuracy achieved already in the Hubble space telescope is sufficient for Heliodromus. Although technology is available the Hubble is a far more compact system so advances need to be made in the application of the available systems. The conclusion is that the TRL of the required pointing system is 7.

7.6.4 Thermal Management

As described in section 7.5 thermal management is an important item. The amount of heat that has to be removed from lasers and mirrors is large. Available cooling methods have significant mass and volume disadvantages. Research into small and compact cooling methods is required. This will result in total new requirements for heat removal and thermal management systems. The TRL of this system will be low, probably around 4.

7.7 Resource Budgets

The system budgets for the laser system are needed to get a good overview on the performance. Budgets are made for the cost, the efficiency, the energy, mass and volume.

Cost Budget To create a total cost overview of the entire system, all the individual system parts should have a cost estimation. For the transmission subsystem the cost estimation is made for the lasers and for the mirrors in LEO and GEO.

The satellite system consists of ten 24 m diameter mirrors in GEO and 10 mirror pairs in LEO of 19 m and 12 m diameter. Hence, the total area of the large reflecting surfaces is roughly 8500 m^2. The 4000 small mirrors for the individual lasers are not officially dimensioned, but it is assumed that the total surface of all reflectors is approximately 10000 m^2. NASA is planning for future telescopes having mirrors with an areal cost of < 0.1 $M\$/m^2$ [55]. For convenience the cost of the mirrors for the use of transmission is taken 0.1 $M\$/m^2$. A 10000 m^2 mirror system will therefore cost approximately $1 billion dollars.

The 1 GW required on Earth implies the lasers on the LEO satellites to produce 4 GW of electromagnetic energy to compensate for the transmission losses. This is done by 4000×1 MW lasers, which add to the total system manufacturing costs. The cost of laser arrays is in the range of $20 - 50$ $\$/W$[36]. Since a total of 4000 lasers have to be manufactured, the overall costs will decrease as a result of learning curves. It is expected that the costs of the lasers will be 10 $\$/W$ when the production in large quantities starts, as proposed by Kare[27]. For the 4 GW system a total of 40 $B\$$ is needed to produce the lasers.

The costs of the lasers are of such order that it make the costs of the mirrors insignificant. A decrease of the specific laser price is necessary to lower the costs for the transmission system and have an increase in the viability of Heliodromus.

Energy Budget The energy payback time is the time it takes for a power plant to produce the amount of energy that was needed to get the power plant operating. This is an important parameter which will now be briefly discussed for the transmission system.

First off all the energy needed for the manufacturing of the lasers and mirrors will be addressed. The combined surface of the mirrors per LEO satellite is $(6^2 + 9.5^2 + 12^2) \cdot \pi = 849m^2$, leading to a total mirror surface of 8490 m^2, covering the large mirrors on the 10 LEO satellites and the 10 mirrors on the GEO satellites. In addition to this there are 4000 small mirrors that are used to deflect the 4000 laser beams into 10 large laser beams. Therefore it is assumed that a total of 10000 m^2 of mirror is needed. Assuming an areal density of 5 kg/m^2[20] the total mirror mass will be 50 tonnes. These mirrors could exist of materials such as metallurgical silicon with an energy requirement of 141 $GJ/tonne$[29]. This would result in a total energy requirement for the mirrors in excess of 7 TJ.

The lasers of the system are assumed to be made primarily out of titanium and are assumed to have a mass of 150 kg each [32], which add 600 tonnes to the system. Due to the fact that titanium has an energy requirement of 920 $GJ/tonne$ this gives a total energy requirement for the lasers of 552 TJ.

For the system to work, it needs cooling. It was already determined that the cooling of the LEO satellites would take a cooling mechanism totaling 6600 tonnes. In addition the two GEO satellites would need a combined cooling mechanism of 42 tonnes as shown in section 7.5. The honeycomb structure is assumed to be made out of aluminum which has an energy requirement value 173 $GJ/tonne$. The total cooling system would therefore require of 1149 TJ of energy to be build.

A total structural mass of 1823 tonnes is assumed. This corresponds with 20 $\%$ of the total mass (average of the $15 - 25\%$ range stated in chapter 9.2) of the laser mirror platform in LEO, as well as the 10 mirrors and cooling on the two GEO satellites. 103 TJ of energy is needed to produce that amount of stainless steel.

In total, for the entire transmission system, 1811 TJ of energy is needed. At a rate of 1 GW it would take over 21 days until that amount of energy is paid back.

Mass Budget The mass budget forms, together with the volume budget and the orbit, the main parameters that have an influence on the amount of launchers needed. Therefore it is of critical importance.

As already addressed in section 7.7 the total mass of the mirrors will be around 50 tonnes. Of those 50 tonnes approximately 11.25 tonnes will be located in each of the GEO satellites. The rest of it will be spread evenly amongst the LEO satellites of which each will carry 2750 kg of mirrors. The majority of the mass of the transmission system is accounted for by the laser systems. As stated in section 7.7 one 1 MW laser is assumed to have a mass of 150 kg each and from section 7.7 it is known that the system needs 4000 of those lasers. This translates into a total laser mass of 600 tonnes or 60 tonnes per LEO satellite. It was already established that the cooling system is calculated to have a mass of 6642 tonnes.
Next to these three most basic components of the transmission system, a great deal of the total mass of the

system will depend on the structural mass already assumed to be 1823 tonnes in section 7.7. This leads to a total system mass of 9115 tonnes of which 65 tonnes is located in GEO orbit. It should be noted that these values are assigned without an attitude and control system taken into account.

Volume Budget For the 4000 1MW lasers it is assumed that they will be two meters high and have a diameter of 45 cm[32]. The volume of one such laser will need inside a launcher is assumed to be a little bit bigger, i.e. a beam with dimensions $0.5 \times 0.5 \times 2$ meters, resulting in a volume of 0.5 m^3. With 4000 of such lasers needed to be transported the total volume needed is 2000 m^3.

For the mirror system it is assumed in 1 m^3 of payload volume, 5 m^2 of mirror can be folded up. This assumption is based on numbers of the Dobson Space Telescope project of the TU Berlin that claims to be able to even surpass such a feat[63]. It was already pointed out in section 7.7 that the total area of the mirrors would be about 10000 m^2 resulting in a payload volume of 2000 m^3.

The density of both the structure mass as the cooling system is taken to be 79 kg/m^3, the average of a group of US spacecraft launched between 1978 and 1984[4]. Such an assumption leads to a volume of 12800 m^3. As a result the total volume needed for the transmission system would be about 16800 m^3.

Efficiency Budget The transmission subsystem can be split up into the lasers and the mirrors. Efficiency losses occur when the energy is converted from electricity in the satellite to electromagnetic waves in the laser device. This efficiency is 60 % (70 % only for laboratory conditions)[22]. The mirror construction on the LEO satellite redirecting the laser to GEO is assumed to include a loss of 0.5 %, i.e. an efficiency of 99.5 %, caused by the non-perfect reflectivity of the mirrors. Diffraction causes the laser to spread and only the Airy disc will be reflected by the GEO mirror, having an efficiency of 84 %. The mirror in GEO has a reflectivity of 99.5 %. The energy moving from GEO to Earth will have losses when passing through the atmosphere. For the used 1064 nm laser the losses are 15 %[27]. Furthermore the PV cells on the ground will only cover the Airy disc, including an efficiency of 84 %. Overall efficiency of the transmission system is roughly 36 %.

Table 7.5: Transmission resource budgets of Heliodromus.

Budgets	**Value**	**Unit**
Cost	40	$[B\$]$
Energy	1811	$[TJ]$
Mass		
LEO	9050	$[GJ/tonne]$
GEO	65	$[GJ/tonne]$
Volume	16800	$[m^3]$
Efficiency	36	[%]

The total system efficiency is determined in section 10.1.

Chapter 8

Ground Station Conversion

All of the research of this project so far has been conducted for the space based solar cells. As the beaming down of energy to Earth will be done with a laser, there are a couple of opportunities to be studied. One of the possibilities is thermal energy collection and is described in section 8.1. A different way of catching the energy is the photovoltaic option which is explained in section 8.2. In section 8.3 these two methods of collecting energy are compared and the subsequent Costs, EPT, Grid Interaction and Storage techniques are discussed in sections 8.4, 8.5, 8.6 and 8.7

8.1 Solar Dynamic Laser Energy Conversion

Several options explored in the early phases for energy collection in space, together with some additional options are looked at in this section. Solar Dynamics' poor performing areas in space were weight and moving parts. These are no obstacle for Terra-based operations where lubrication and maintenance are possible and lack of launch requirement renders weight an aspect only relevant to maximum lifting loads of the construction. Besides the Solar Dynamic system, in the category of Solar Thermal, several systems are possible on Earth.

Laser Thermal Energy collection is a term derived from Solar Thermal Energy collection, which indicates harnessing of solar energy for thermal energy. In this case not the Sun, but laser provides the energy. Many of the manners to use laser energy to generate thermal energy exist already, for this reason rather than looking the possibilities of the theoretical realm we will be looking at developed products.

Existing products include:

- Suncatcher^{TM}
- Fresnel Reflectors
- Parabolic Trough
- Solar Tower

These products are all tailored to deal with sunlight reaching Earth. Modifying these for higher intensities in the form of a laser is possible, in some cases improving, in other cases diminishing the performance.
Considering the locations for the optimum down-beaming of energy are all deserts, an additional factor to consider is water use. A system capable of receiving 500 MW will require a lot of cooling, methods that use water for this purpose will need it imported in large quantities, adding significant costs. Open systems that release water vapor will cloud the transmission area resulting in lower energy received.

The Suncatcher^{TM} uses a dish concentrator design in combination with a Stirling engine based Power Conversion Unit (PCU). The dish concentrator heats one side of the PCU with a solar intensity of $13.10^6\ \frac{W}{m^2}$[10]. As the Stirling engine is optimized to function at the temperature provided, scaling the concentrator area to provide this intensity at higher than solar beam density will be the surest way to provide reliable results. Alternatively, increasing the pressure at which the stirling PCU operates and increasing the size, are ways of scaling the system. Neither of those are linear though and do not scale as well.
The Suncatcher^{TM} produces 25 kW of energy per 12 meter diameter dish[50]. With the aim of getting 1 GW on Earth with the Heliodromus option one ground station has to collect 500 MW for distribution. With 25 kW output per unit this requires 20000 units. Assuming a $0.5\ km^2$ spot size this leaves $25m^2$ of concentrator per unit. With the proven 31.25%[71] efficiency of the Suncatcher^{TM} this results in a $3167\frac{W}{m^2}$ intensity.
It is important to understand 31.25% is only reached at optimum conditions: clear sky and cold air temperatures. Deserts have as a beneficial characteristic that while hot during the day, they hold no heat and therefore cool down significantly during the night. Colder air temperatures will increase efficiency, as can be derived from calculation 8.1, leading to higher efficiencies during nighttime usage than the published 31.25%. The ability of Heliodromus to continuously sending energy down to a ground station during the night will benefit from this.

$$\eta = 1 - \frac{T_c}{T_h} \tag{8.1}$$

The Suncatcher^{TM} has as advantages: Modular buildup. Every unit being independent of the others marginalizes the impact of errors/breakdowns significantly. Low water use. Water is not used for cooling (A radiator system is used instead) leaving only the water used to clean the mirrors. And the Suncatcher^{TM}'s high solar power extraction per area.
Fully adapted to our system the Stirling Engine Dish design, compared to the Suncatcher^{TM}, would have a smaller concentrator area as the driver and alignment system are omitted, (needing only one constant direction to point to) allowing for denser packing and structural (but not systemic) cooperation between units. With closer packing the opportunity for designing for better cooling through wind-funneling arises.

Figure 8.1: Suncatcher^{TM} in Terrestrial solar configuration

There are only two disadvantages to this method. The cost, Stirling engines are expensive. In the Suncatcher^{TM} the stirling engine is the major cost driver and for Heliodromus, with the elimination of driver, tracker and alignment systems and a smaller concentrator area, this cost will still be there. The other disadvantage is the need for maintenance, the impact of this disadvantage is contained by the modular design.

Figure 8.2: Parabolic Trough

A Parabolic Trough system uses a Parabolic Trough to concentrate sunlight on an absorber tube running the length of the trough. The absorber tube contains a Heat Transfer Fluid (HTF), usually water or oil, to absorb the concentrated sunlight. This fluid then drives e.g. a turbine generator to generate electrical power. The system allows for scaling in a number of ways allowing it to be used for more concentrated energy densities. Using a bigger absorber tube with more fluids is possible or using smaller concentrator mirrors can also be an option to scale this system up. Changing nothing outside of the material of the absorber tube will lead to a far hotter HTF in which case a different generator/different HTF combination is used for higher efficiencies[9].
The major problem with a Parabolic Trough system is that it struggles to reach overall efficiencies over 20%. With the higher intensity beams hitting the absorber, allowing for a higher efficiency in the heat-to-electricity conversion, it will not approach the efficiency available to a Stirling Engine Dish design resulting in a larger space segment to send the required energy to Earth.
A Trough system is more economic than the afore mentioned dish system per capacity but the gained economic relief does not weigh up to the added cost of the larger space segment.

Fresnel Reflectors are very similar to Parabolic Trough systems, but they differ in the way they focus their energy and the location of the absorber tube. The mirrors used for reflection are flat and have a Fresnel pattern to concentrate the light on the absorber tube. Instead of using a parabolic trough it uses a series of these mirrors to concentrate the sunlight and the absorber tube is suspended above the mirrors. The advantage of this for use with solar energy is the ability to focus more energy on the tube bringing the temperature of the HTF to higher levels allowing for the use of more efficient generators. This may not be necessary with higher level beam intensity. Another advantage of this is that the absorber tube is stationary, whereas the tube in a parabolic trough design has to move, removing complexity and cost from the system.

Figure 8.3: Fresnel Reflector solar energy collection system

The system holds some advantages over the Parabolic Trough system it was derived from; better economic models for instance. But the thermal performance of the parabolic system is 70%[11], requiring an even larger space segment. Though economically more viable on Earth it is not a valid option for use in conjunction with Heliodromus.

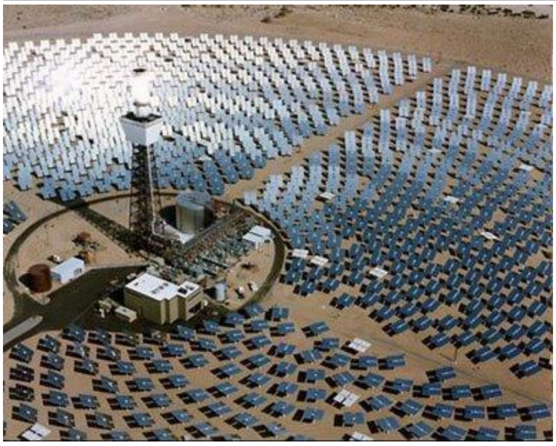


Figure 8.4: Solar Two, a solar tower in operation

The Solar Tower principle has been built in several different configurations and several others are being developed. The system is based on concentrating far larger amounts of energy than the previous mentioned systems on a single point. The Solar Tower, as the name implies, concentrates the energy on a tower. The top of the tower houses a thermal receiver which uses the energy received to run a generator. The higher amounts of energy redirected to the solar tower mean that it already operates in the higher efficiency range for generators. To adjust this collection system for use with Heliodromus would mean a scaling down of the concentrator area or a scaling up of the solar tower.
Advantages of this system is the higher operating temperature and no need for a level surface to build this system on. Many of the disadvantages of the system based on direct solar energy are removed when designing it for Heliodromus. There is no need for all the mirrors to have dual axis control, they can simply be aligned to the stationary GEO position and the stationary tower. In this respect it would transform in an upscaled version of the dish design. The e-solar[51] is an example of this system currently used for commercial power generation. The current design produces 46 MW per 0.6 km^2 based on solar energy. Taking the same intensity as in the stirling engine dish design, 3.167 times higher than solar energy at its peak performance, and assuming linear scaling, this would result in 208 MW per 0.6 km^2. Indicating an efficiency far lower than the Stirling Engine Dish.

Solar Dynamic conclusions In conclusion, there are different techniques available to collect energy with solar thermal energy collection methods. All of the techniques have a market proven concept, hence the technology readiness level is at 9. Of course the techniques are still being improved upon realization of more efficient power conversion. However, the high efficiency of the Stirling Engine Dish design is of such importance to the entire system that the lower development/assembly/building cost of the other systems are easily recovered by cost reduction of the space section. The customization of the Stirling Engine Dish design is therefor seen as the best candidate as a solar dynamic option.

8.2 Photovoltaic Laser Energy Conversion

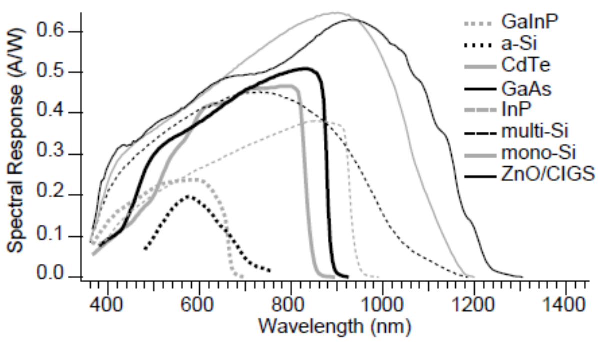


Figure 8.5: The spectral response characteristics of several solar cells[47]

The total conversion of photovoltaic cells is different from the space solarlight conversion, because the power that must be converted is monochromatic light instead of full spectrum. A laser sending monochromatic light drastically changes the configuration of a photovoltaic cell.
The conversion of the laser emitted monochromatic light approaches the theoretical limits of 90 % in the optimum case[16].

To approach the theoretical limit a design strategy is to match the band gap of the semiconductor to the energy of the incident photon. This way less heat is impared in the photo-

voltaic process. Thus the photovoltaic cell must be tuned such that it absorbs the desired photon of the sent wavelength.
Minimizing the solar cell temperature, maximizing the bandgap energy and the laser photon energy, and matching the energy of the incident laser light closely to the photovoltaic band gap energy are the steps that need to be taken to benefit optimally.

Matching the wavelength of the laser with the bandgap energy is the first thing considered. Taken into account the weather absorption of certain wavelengths and other advantages, as described in section 7.2.5, 1064 nm is considered to be a good option. The question remains how photovoltaic cells react to this wavelength. As can be seen from figure 8.5 the best options seem mono-Si[5] and ZnO/CIGS, since they have the highest spectral irradiance at 1064 nm wavelength.

It is important to select a wavelength near the optimum value of a photovoltaic cell. A laser sends a specific wavelength, hence a receiving solar cell can be optimized for this monochromatic illumination. Near the optimum wavelength, the response of a solar cell to monochromatic illumination is much higher than the efficiency produced by the broad solar spectrum. This is primarily due to two factors:

1. The sun produces a wide-band spectrum, thus not all photons can be used efficiently using a single bandgap. Photons with energy less than the bandgap will not be absorbed, and for photons with energy greater than the bandgap the energy will be lost.

2. A solar cell will only have high quantum efficiency over a limited range. Quantum efficiency refers to the percentage of photons that are converted to electric current. A photovoltiac cell can be tuned to a wavelength where the quantum efficiency is close to unity.

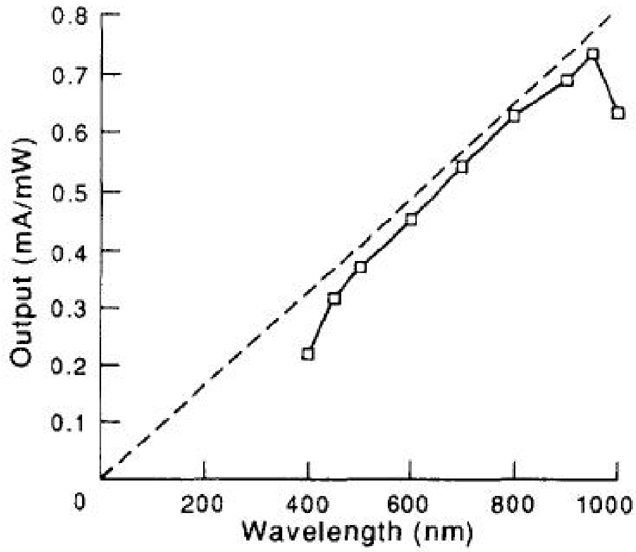


Figure 8.6: Measured output of a standard silicon solar cell as a function of incident wavelength. The dashed line indicates the ideal (unity quantum efficiency) spectral response [22].

These two factors together result in the fact that the efficiency of a solar cell under monochromatic illumination at a wavelength near the spectral response peak will be more than double the solar efficiency[26]. The spectral response peak for Silicon cells can be seen in figure 8.6. Existing solar cells have peak response to monochromatic illumination at about 850 nm for GaAs cells and about $950 - 1000$ nm for Si cells[5][22]. For wavelengths shorter than the peak, the efficiency will decrease roughly linearly with wavelength as is indicated by the last point in figure 8.6. For longer wavelengths, the efficiency will rapidly drop to zero. Figure 8.7 shows the monochromatic light efficiency as a function of light intensity for three different wavelengths of which one is the 1064 nm. A peak efficiency is observed at about $1\ \frac{W}{cm^2}$ illumination level with peak efficiency of 45.1% for monochromatic light with a wavelength of 1020 nm[25]. Efficiency decreases at higher intensities due to resistance losses. The efficiency at 1064 nm reaches a maximum value of $38.8\ \%$ at $1.3\ \frac{W}{cm^2}$[25]. When reflection losses are optimized, light trapping improved, as well as choice of optimum substrate thickness and reducing the resistivity the efficiency is expected to increase above 50% for wavelengths in the $1025 - 1064$ nm range with intensities in the $1 - 5\ \frac{W}{cm^2}$ range[25].

Improvements and other issues There remain many unsolved problems in increasing the efficiency of solar cells for monochromatic light. For many applications it is desirable to design the cell for both laser and solar conversion. This allows both the use of the solar energy as well as the laser energy. Many technical issues must be addressed before high efficiency photovoltaic receivers of laser power can become practical. To improve the use of monochromatic light at 1064 nm, photovoltaic cells require light-trapping or new material[26]. Light-trapping involves a light trap of a high optical quality array, consisting of micro-lenses as depicted in figure 8.8. Concave lenses are used to trap the light. The radius of curvature of the micro structures is tuned to provide the required focal length of the desired final polymer micro-lenses. By using the light trapping element on top of the photovoltaic cells, an increase in cell absorption is obtained. By recycling reflected photons that otherwise would be lost, thinner films with more beneficial electrical properties can effectively be deployed. The light trapping element enhances the absorption rate of the solar cell and increases the photocurrent by as much as 25 %[23].

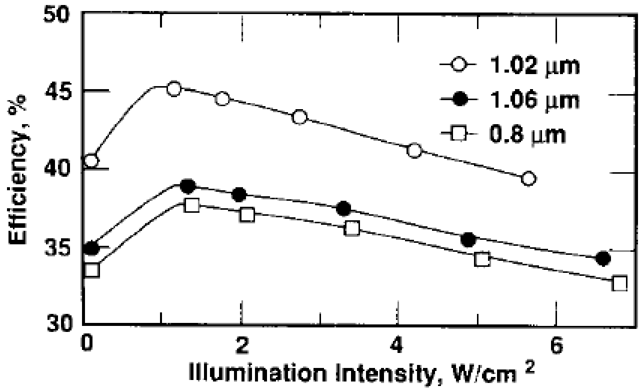


Figure 8.7: Monochromatic light energy conversion efficiency as a function of illumination intensity at three different wavelengths for silicon cells. Measurements courtesy of Sandia National Laboratories[25].

New materials that can be considered are highly complicated structures. The most common triple junction solar cell structure is the InGaP/GaAs/Ge structure with theoretically and experimentally obtained efficiency values of around 33%[24]. By increasing the spectral use between the GaAs bandgap and Ge bandgap efficiencies toward 50% or even higher could be obtained. On the other hand, Quantum Dots(QD) are attractive since the electronic properties of QD depend on its size, shape and surrounding matrix. The QDs therefor can make optimum use of a desired wavelength they are engineered for[24]. Upon optimization of the InAs QD layers, the QDs can be integrated in the existing layers. Consequently, this approach will have the potential to reach an efficiency of 45%[24].

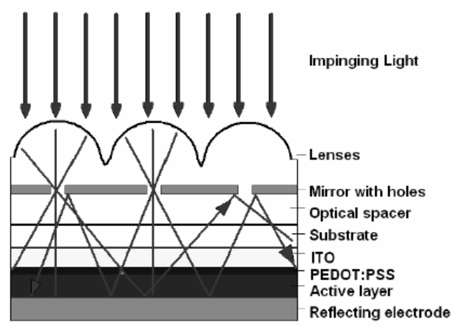


Figure 8.8: Operational principle of the light trap and its constituents[23].

As the temperature increases, the efficiency of conversion decreases[16][26]. The operating temperature also limits the highest possible intensity. The intensity must be high to deliver the 1 GW to Earth on a small spot. The spot size must be physically feasible in both the conversion and transmission perspective. The efficiency would be expected to increase with increasing illumination intensity due to increased open-circuit voltage, if the cell temperature is maintained constant at 25 ^oC and cell series resistance is not excessive[16]. The linearity in figure 8.9 implies that the performance at high power is not affected by the series resistance. An efficiency of 38.8% at $1.3\frac{W}{cm^2}$[25] has been obtained which comes down to 13.000 $\frac{W}{m^2}$. Figure 8.9 shows measurements performed at an even higher intensity up to 54.000 $\frac{W}{m^2}$, although illuminated by monochromatic light at a wavelength of 826 nm. But the optical-to-electrical power conversion efficiency ranges from $55-59\%$[21]. Thus photocells are capable of handling these high intensity powers but extensive cooling of the system is required to maintain the desired 25 ^oC[21].

In conclusion some final remarks. Firstly, the efficiency of photovoltaic cells in response to monochromatic illumination by lasers show a promising future. However, interest in this field of investigation was lost after the early nineties. Research by, for instance G.A. Landis and M.A. Green, has been performed to prove the feasibility, but has not been developed to a further stage. This results in a relatively low technology readiness level of four, TRL 4, and thus needs further development to lower the technology risk.

Secondly, to give an estimation of the ground station size and in relation the laser intensity can be determined as follows. Currently the highest proven laser intensity at 1064 nm is $13.000\frac{W}{cm^2}$[25]. Considering the requirement of delivering 1 GW and a conversion efficiency of roughly 40% at this intensity the laser beam should contain 2.5 GW. Considering the intensity this will result in a minimum required area as shown in equation 8.2.

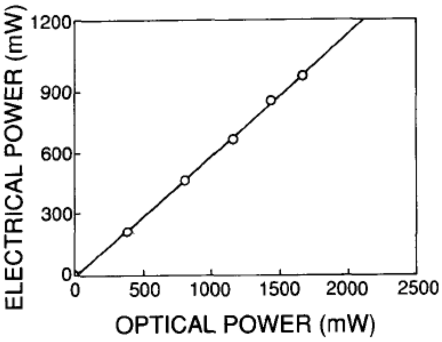


Figure 8.9: Maximum electrical power generated by photocell versus incident optical power. Least squares fit to data gives 59% optical-to-electrical power conversion efficiency[21]

$$\frac{P_{laser}}{I_{laser}} = \frac{2.5 \cdot 10^9}{13.000} = 192307.7m^2 \approx 0.2km^2 \tag{8.2}$$

This would result in a ground station with a diameter of about 500 m. This is too small a spot to point the laser beam to and restricts the beam accuracy tremendously as described in section 7.2.5. This section considers various influences on the laser accuracy and leads to an addition of 350m in diameter, which makes the effective ground station diameter 850m. The laser beam is characterized by the diffraction pattern leading to a non-uniform energy distribution. The non-uniformity results in a energy distribution outside the spot and thus outside the ground station effective area. For safety purposes the ground station is required to have a clearance area. A non-uniform distribution[73] and the harmful exposure times of the 1064 nm laser must be accounted for. When looking at an unlimited exposure time a suggested 16 W/m^2 is required as the maximum permissible exposure (MPE) for the 1064 nm wavelength[74]. Then looking at the distribution and assuming a Gaussian pattern which creates different rings surrounding the spot size which have decreasing intensities. The third ring contains a maximum of 0.078 % of the first maximum of 13000 W/m^2 which means a 10 W/m^2 [73]. This is below the MPE limit hence sufficiently small to guarantee the safety. The Gaussian pattern therefor determines the size of the clearance area. The linear relation between the m and the size of the considered ring determines the clearance area size as indicated by equation 8.3. Where $m_{firstring}$ has a value of 1.220, which accounts for the Airy disc. The $m_{thirdtring}$ has a value of 3.238 and accounts for the Airy disc plus two outer fainter rings maxima.

$$\frac{m_{thirdring}}{m_{firstring}} \cdot D_{spot} = \frac{3.238}{1.220} \cdot 500 \approx 1300m \tag{8.3}$$

To be more flexible in the accuracy and for safety purposes a larger ground station is suggested with a diameter of 1300 m. This allows for the possibility to implement solar sensitive photovoltaics or more laser sensitive photovoltaics catching the low intensity laser in the outer rings. The efficiency just considered is an improvement compared to efficiencies reached by solar illumination, which has a current maximum of 40.2%[7]. Although this efficiency is higher, it is highly dependant on the fluctuating solar intensity. The intensity of the sun is very dependant of the weather and solar light is only available for an average of 8 hours a day at most places on the equator[54]. However the laser system is available 24 hours a day at a constant intensity, which makes the 38.8% at 1064 nm very attractive. In the future the attractiveness increases since high efficiency GaAs cells can produce over 50% efficiency under laser illumination, although not at 1064 nm, and conventional Si cells over 40%[25][26]. Thus an improvement in the energy absorbtion but requires a specific kind of cell. Still a solution of combining both solar and laser illumination would be more optimal.

8.3 Solar Dynamic vs. Photovoltaic

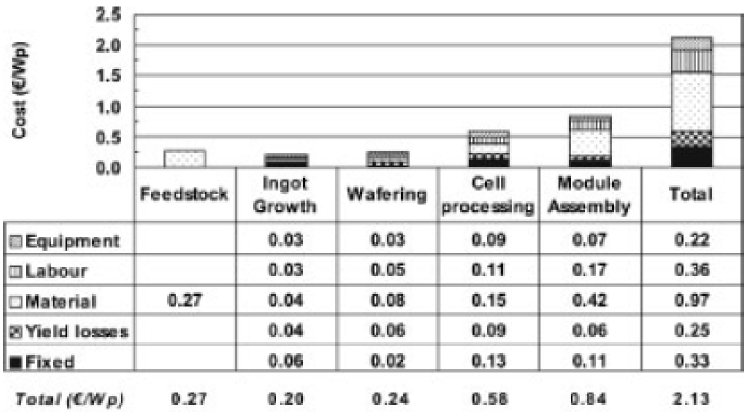
Having looked at the different options to convert the laser energy to useful energy for Earth applications some conclusions can be made. The Stirling engine is the best solar dynamic option with a conversion efficiency currently at 31.25%[71]. The specific Silicon Cell $2255 - 1$ proved to be the best photovoltaic option with a conversion efficiency of about 40 %[25]. Thus the photovoltaic option shows a better conversion efficiency. Also the fact that photovoltaics are used more than heat engines and production is implemented faster make photovoltaic favorable. The cost estimation is hard to compare since photovoltaics are highly dependent on material price, but both experience a learning curve effect as an explosive growth in the production rate is required. The investigation of combining the laser illumination and the use of the solar spectrum leaves room for improving the efficiency even further.

8.4 Cost

The PV market has expanded enormously in the last decade with annual growth rates in the range of $40 - 50$ %[14]. The PV market is largely based on the crystalline silicon technology, which has a share of about 90 %[14][15]. In the future the question remains if the feedstock of highly purified silicon for solar cells will be sufficient[15]. The feedstock of electronic grade polysilicon is at an average of 30 €/kg in 2008 [14]. Taking all steps of the production process into account of which the cost breakdown is shown in figure 8.10, it comes down to a cost of about $2 - 2.3$ € per Watt.

When manufacturing is performed on a 1 GW scale, the total cost can be reduced by 30%[14]. Mainly because of the reduction in complexity, the cost of installation and reduction of module manufacturing cost decreases as the production line is scaled up. The cost reductions are tabulated in table 8.1.

The technological improvements can thus reduce the costs by 28 to 35 % and with these improvements the 1 € per Watt goal is within reach provided large scale production is implemented. Applying this to a 1 GW system with an efficiency of about 40 % would lead to a 2.5 billion dollar investment to provide the ground station with enough solar cells.



	Feedstock	Ingot Growth	Wafering	Cell processing	Module Assembly	Total
Equipment		0.03	0.03	0.09	0.07	0.22
Labour		0.03	0.05	0.11	0.17	0.36
Material	0.27	0.04	0.08	0.15	0.42	0.97
Yield losses		0.04	0.06	0.09	0.06	0.25
Fixed		0.06	0.02	0.13	0.11	0.33
Total (€/Wp)	0.27	0.20	0.24	0.58	0.84	2.13

Figure 8.10: Breakdown by process step and by cost category, in €/Wp.[14]

8.5 Energy Pay-back Time

Any type of energy production, including solar, generate pollutants when their entire life cycle is accounted for, since the manufacturing process takes power provided by (currently) the burning of fossil fuels and the expulsion of greenhouse and other polluting gasses. These emissions must of course be smaller than the total energy produced by the energy generation. The Energy Pay-back Time (EPT) is a good measure for this. The EPT of the photovoltaic ground station is somewhat hard to estimate, since it involves photovoltaics that are

Table 8.1: Reduction in $\%$ from the cost model indicated in figure 8.10 due to large scale production[14].

	Ingot $[\%]$	**Wafer** $[\%]$	**Cell** $[\%]$	**Module** $[\%]$
Equipment	$-20/-30$	$-20/-30$	$-20/-30$	$-20/-30$
Labor	$-30/-40$	$-30/-40$	$-30/-40$	$-30/-40$
Materials and consumables	Crucible $-30/-40$ Rest $-10/-20$	$-10/-20$	Pastes $-30/-40$ Rest$-10/-20$	$-10/-20$
Yield	$-1/-2$	$-0.5/-1.5$	$-1/-2$	$-0.5/-1.5$
Fixed costs	$-40/-50$	$-40/-50$	$-40/-50$	$-40/-50$

not commonly used. To adjust the solar cells to the monochromatic light, a different manufacturing process will be required although many general things remain the same such as energy needed to obtain raw material. The electricity generation of a photovoltaic system has an environmental impact during its life-cycle. A life cycle starts from the mining and processing of materials of which solar cells are made and ends with their final decommissioning, disposal and/or recycling. A Performance Ratio of 75 $\%$ accounts for the effects of shading, snow cover, heat loss and DC-AC conversion loss needs to be considered. The value of this ratio might be somewhat conservative with respect to the size, location and environmental aspects of the ground station[13].

The production of all photovoltaic cells for the ground station will take a numerous amount of years and combined with that a high amount of energy. An estimation of the ground station EPT can be performed based on the minimum required area set in section 8.2. The effective area was set to have a 850 m diameter (567450 m^2). A configuration based on a mono crystal silicon photovoltaic cell the properties are described in table 8.2 is assumed.

Table 8.2: Specifications of PV system [19]

	Value	Unit
Area of a cell	142 (pseudo square)	cm^2
Thickness of a cell	350	μm
Packing factor	0.82	
Density of silicon	2.3	g/cm^3
Mass of a single cell	11.43(square shape)	g
No. of cells in an array	36	
Total mass of cells	7.245	kg
Area of module	0.624	m^2
Effective cell area in module	0.9 $\%$	

Solar cell fabrication entails a sequence of high temperature diffusion, oxidation, and deposition and annealing steps. The energy required to prepare 1 m^2 of silicon cell is in the range of 100-290 kWh depending on the type of processing[19]. Taking the maximum 290 kWh/m^2 and relating this to the specific module it will require, $0.624 \cdot 290 = 182$ kWh per module. The number of modules needed is related to the spot area, this number is:

$$\frac{effectivearea[m^2]}{areamodule[m^2] \cdot effectivemodulearea[\%]}$$

Which comes to $1 \cdot 10^6$ cells. The total amount of energy required to produce these cells is $182 \cdot 10^6$ kWh. The converted energy will be 1 GW, with the considered 40% efficiency. Thus the amount of time required to regain the fabrication energy is $\frac{182 \cdot 10^9 [Wh] \cdot}{1 \cdot 10^9 [W]} = 182[h]$. This time is negligable, but this time is only the time

to produce to fill the effective area with photovoltaics. The actual EPT will be much larger since the life cycle includes mining, transportation and assembly as well. But these energies are hard to predict, hence a small comparison of existing Earth based PV systems is considered. Under average Southern-Europe insolation, 1700 $kWh/m^2/yr$, a complete rooftop PV systems based on multicrystalline silicon has an EPT of, respectively 2.2, and 2.7 years for multi-, and mono-Si technology[19]. However, if illuminated by the laser, creating an intensity of 13.000 W/m^2 instead of 1700 $kWh/m^2/yr$, which is about 200 W/m^2, and thus a factor 65 times smaller, the EPT will also be reduced due to the intensity increase, although not by a factor 65. This is because the EPT is not linearly related with the illuminated intensity. It can thus be concluded that the 182 hours can not be seen as a useable figure and the rooftop comparison gives a better indication of the EPT.
The infrastructure and facilities required to integrate the PV system with an existing or setup a new grid will also require energy and needs to be taken into account for the entire ground station. The value of this energy can not be determined since too little is known about the specific facilities that are required.

8.6 Grid Integration

The converted energy must be delivered to the consumer, hence a distribution method must be facilitated. In grid-connected inverters for PV applications, a number of different approaches have been developed and used over the last 20 years. For example in India, 31 grid interactive projects of a capacity of 2.5 MW have so far been installed for voltage support in the remote areas of weak grids with a total efficiency of 98 %[19]. Grid connected PV systems involves a DC-AC converter, cables and some module support materials. To be small, light and low cost, module-integrated converters generally use high frequency switch mode technique[35]. By using the converter-per-panel approach the efficiency is even increased to 100 %, when a temperature of 25^{o}C is assumed[35]. DC-AC converters make it possible to transport the electricity over long distances and are done in two general ways: overhead and underground transmission. The overhead method involves the high power wires that are integrated within the scenery. These wires, almost always made of Aluminium Alloy, are conductors which are not covered by any insulation. This alloy is weaved into a wire and usually reinforced with steelstrands. Copper is sometimes used for overhead transmission but aluminum is lower in weight for equivalent performance, and much lower in cost as can be seen in table 8.3. Improved conductor materials, such as superconductors, and more optimized shapes are regularly used to allow increased capacity but bring along high costs since long distances are required. Conductor sizes range from 12 mm^2 to 750 mm^2[18], with varying resistance and current-carrying capacity.

Table 8.3: The specific resistance of various conductor types[4]

Material	Resistance (ohm-cm)
Aluminum	$2.650\ 10^{-6}$
Brass	$7.0\ 10^{-6}$
Carbon (amorphous)	$3.8 - 4.1\ 10^{-6}$
Constantan	$45.38\ 10^{-6}$
Copper	$1.678\ 10^{-6}$
Semiconductors	$Variable$
Electrolytes	$Variable$
Superconductors	$0(exactly)$

Voltages are usually considered to be 110 kV or even higher inside the powergrid. Voltages less than 33 kV are usually used for distribution. Voltages above 230 kV are considered extra high voltage and require different designs compared to equipment used at lower voltages.

8.7 Energy Storage

In autonomous systems a storage method is required, since solar cells cannot store the energy themselves[19]. Many different methods for electric power storage are in use. Integration of PV with large energy storage systems will stabilize the energy supply of a network. One option that is currently being considered is represented by electrolytically produced hydrogengas. The latter could be used as an energy buffer whereby the extra energy generated can be stored, only to be converted back to electricity by means of fuel-cell devices when the need arises. Another available energy storage option is pumped hydro electric and compressed air energy storage(CAES)[17]. CAES maybe the soonest available viable large-scale energy storage solution in the short term. The most efficient method currently available to store large amounts of energy is by transforming electricity to potential energy using a fluid, water most commonly used. This is a technique mostly used where a natural formation is already present to provide the height difference needed for this. Considering the locations of the Ground Stations, however, this is not a viable, economic option. A most promising option is Flywheel technology for energy storage. This technique uses electric energy to spin up large discs to store kinetic energy. This method is actually more efficient than using water for potential energy but is limited to smaller systems[49]. The technique is in wide use and has a distinguished track record but lacks application on larger scale.

CAES is the most viable system for energy storage currently available, but with research into large scale flywheel systems this may be a serious contender.

Chapter 9

System Integration

The Heliodromus is a project existing of three main parts: 2 GEO satellites, 10 LEO satellites and 2 Ground Stations. In the end, all these parts will have to cooperate and function as one single system. How this will be achieved is discussed in this chapter.
In the chapters 6 to 8, the Collection subsystem, the Transmission subsystem and the Ground Station are discussed separately. The main focus of Heliodromus is on these subsystems. The Power Management & Distribution, the Command & Data Handling and the Telemetry, Tracking & Command are discussed in section 9.1. However, while the focus of the project is not on these subsystems, they are still important for the system to function as one single unity.
In section 9.2 a summary is made of the mass, dimension and power budget of a single LEO and GEO satellite. How the system will function as a whole and the interrelation between different subsystems is presented in section 9.3. Section 9.4 will discuss how the LEO and GEO satellites will be brought and assembled in space. The last part of this chapter, section 9.5, shows artist impressions of the system and discussed the conceptual design. These are meant to give the reader an impression of how the system could look like.

9.1 Subsystem analysis

In chapters 6, 7 and 8 the collection, transmission and groundstation were already discussed. These were the subsystems on which the focus was during the Heliodromus project since they are different from other space projects. For the LEO and GEO satellites the thermal control, section 6.4 and 7.5, and Attitude Determination & Control, section 6.3 and 7.3, were already defined. This chapter discusses the subsystems not presented yet and which are similar to most space missions. These are the Power Management & Distribution (PM&D), the Telemetry, Tracking & Command (TT&C) and the Command & Data Handling (C&DH) subsystems.

9.1.1 Power Management and Distribution

For the LEO satellites, the power provided by the thin film photovoltaics will be approximately 0.7 GW or 700 MW as determined in section 6.2. The biggest part will be used to generate the laser which transports the energy towards earth via the GEO satellites. A very small portion (a few kW compared to 700 MW is almost nothing ($< 0.001\%$)) will be used to provide the power of the other subsystems. In chapter 6 it was calculated an area of 3.44 km^2 is needed to provide these 700 MW.
For the GEO satellites the power requirement is estimated to be about 2 kW. How this number is obtained can be seen in section 9.2. This number is rounded off to 2 kW, taking redundancy of the separate subsystems into

consideration. To supply this power to the GEO satellite, solar arrays are placed around or integrated in the mirrors and convert part of the laser beam to the 2 kW energy needed by the GEO satellite. Each LEO satellite has a mirror in GEO onto which it sends its laser beam. When the laser beam arrives at a GEO mirror, which has an area of 440 m^2, it contains 0.4 GW of power as described in section 7.7. This gives an average intensity of 900 kW/m^2. Using laser optimized Si solar cells reaching an efficiency of 40% as assumed in section 8.2, only 55 cm^2 of solar cells are needed to provide 2 kW of power to the satellite. Since each GEO satellite has 5 mirrors, 11 cm^2 of solar cells have to be integrated in every mirror.
The LEO satellites orbit in a sun synchronous dawn/dusk orbit. This means no eclipses will occur caused by Earth. However, there are some eclipses caused by the Moon. The longest of these eclipses will last for 20 minutes (section 5.1.1). Energy storage to keep the subsystems working during this time should be included. Providing about 2 kW for 20 minutes means about 700 Wh has to be provided by the battery. These batteries will need to be rechargable. When Nickel-Hydrogen batteries are chosen, which have a specific energy density of $43 - 57$ Wh/kg[4], a mass of $12 - 17.5$ kg is obtained for each LEO satellite.
The GEO satellites do not need energy storage because it receives its energy from the lasers. However, it is recommended to increase the above defined 11 cm^2 per mirror slightly. This way the GEO satellite will continue working even if one of the lasers loses contact with a mirror.
The cables which interconnect the different subsystems contain a big part $(10 - 25\%)$ of the electrical subsystem's mass[4]. For the LEO satellites this means a mass of about $30,700 - 92,000$ kg. The cabling mass for the GEO satellite is much lower and has a mass ranging from $70 - 330$ kg. These cable masses were calculated using the numbers of table 9.4 and 9.3 of section 9.2 and are shown below.

$$\begin{aligned}
Cable\ mass\ LEO\ (10\%) &= 0.1 \cdot (Cable\ mass\ LEO\ (10\%) + Laser\ mass\ LEO + Thin\ Film\ mass\ LEO) \\
&= 0.1 \cdot (30700 + 60000 + 216000) = 30670\ kg
\end{aligned}$$

$$\begin{aligned}
Cable\ mass\ LEO\ (25\%) &= 0.25 \cdot (Cable\ mass\ LEO\ (25\%) + Laser\ mass\ LEO + Thin\ Film\ mass\ LEO) \\
&= 0.25 \cdot (92000 + 60000 + 216000) = 92000\ kg
\end{aligned}$$

$$\begin{aligned}
Cable\ mass\ GEO\ (10\%) &= 0.1 \cdot (Cable\ mass\ GEO\ (10\%) + AD\&C\ mass\ GEO) \\
&= 0.1 \cdot (70 + 650) = 70\ kg
\end{aligned}$$

$$\begin{aligned}
Cable\ mass\ GEO\ (25\%) &= 0.25 \cdot (Cable\ mass\ GEO\ (25\%) + AD\&C\ mass\ GEO) \\
&= 0.25 \cdot (330 + 1000) = 330\ kg
\end{aligned}$$

Of course PM&D systems are used in every satellite. However, satellites having such a substantial power requirement of 700 MW have never been constructed. That is why further research in the PM&D system of such a high power satellite will have to be carried out. As a result a TRL of 7 is given to the PM&D subsystem.

9.1.2 Command & Data Handling

Command & Data Handling is an essential part of every satellite system, hence also for the Heliodromus. According to SMAD[4], the C&DH subsystem has to perform two major functions:

- Receive, validate, decode and distribute commands to other spacecraft subsystems.
- Gather, process and format spacecraft housekeeping and mission data for downlink or use by an onboard computer.

Sometimes it also includes functions such as timekeeping, computer health monitoring (watchdog) and security interfaces. These should also be included in the Heliodromus satellites. Since the Heliodromus project requires such an high accuracy and fast pointing, complex computer systems are necessary. High reliability is required, therefore redundant systems will also be needed. These facts will result in a C&DH system with a relatively large size, weight and power requirement. Using table $11 - 29$ of SMAD[4] an estimation is made for the Heliodromus C&DH system and is shown in table 9.1. It is assumed that Heliodromus will need a complex C&DH system using combined telemetry and command systems for the reasons stated above.

Table 9.1: Mass, Power and Dimension estimate for the C&DH subsystem

	Single Unit Mass [kg]	**Single Unit Power** [W]	**Dimensions** [10^3 cm^3]
Complex, Command & Telemetry Combined C&DH system	$9.5 - 10.5$	$15 - 25$	$13 - 15$

The C&DH system for the Heliodromus project requires high data rates since fast pointing and a high accuracy are required. C&DH systems having this kind of complexity are maybe not widely used but exist. Therefore development of such a system should not offer major problems and a TRL of 9 is assigned to this subsystem.

9.1.3 Telemetry, Tracking & Command

In the case of Heliodromus there is no payload data which needs to be communicated to the ground station. Still TT&C is required to pass the operator commands. Especially for safety reasons it is important to be able to shut the entire system down as fast as possible. Therefore a continuous communication link has to be established between the LEO satellites, the GEO satellites and the ground stations. To provide this continuous communication link between the two ground stations and the LEO satellites, all signals will have to pass by the GEO satellites and/or make use of the Tracking and Data Relay Satellite (TDRS) system of NASA. The following five functions are defined as being part of the communication system[4]:

- Carrier tracking (lock onto the ground station signal)
- Command reception and detection (receive the uplink signal and process it)
- Telemetry modulation and transmission (accept data from spacecraft systems, process them, and transmit them)
- Ranging (receive, process, and transmit ranging signals to determine the satellites position)
- Subsystem operations (process subsystem data, maintain its own health and status, point the antennas, detect and recover faults)

The TT&C system for Heliodromus is not different from TT&C systems of existing satellites and will also have to carry out the functions stated above. Mass, power and dimension estimates are made using SMAD[4] p394 and are depicted in table 9.2. These are based upon a Ku-Band Communication Subsystem, which provides a high data rate. It may also be possible to provide so called optical communication using the laser beam which is already send by the LEO satellites. However, the weather may then cause problems.

Table 9.2: Mass, Power and dimension estimates for a typical Ku-Band communication subsystem

	Single Unit Mass [kg]	**Number Unit Used**	**Total Mass** [kg]	**Single Unit Power** [W]	**Dimensions** [cm]
Transponder	4.45	2	8.9		$17 \times 34 \times 9$
- Receiver				4.3	
- Transmitter				20	
Filters/Switch	1.2	1	1.2	0.0	$8 \times 19 \times 4$
Antennas					
- Earth cover	0.5	1	0.5	0.0	$4.0\,dia \times 2$
- Parabola	2.0	1	2.0	0.0	$60\,dia \times 22$
- Waveguide	0.7	1	0.7	0.0	$125\,cm\ long$
Total			13.3	24.3	$245,650\,cm^3$

TT&C systems of this kind already exist. This is why a technology readiness level of 9 is assigned to this subsystem.

9.2 Technical Resource Budgets

In Tables 9.4 and 9.3 a summary is made of the mass, dimensions and power budgets of the different subsystems of a LEO and a GEO satellite. These subsystems are defined as Payload, Power Management and Distribution (PM&D), Attitude Determination & Control (AD&C), Telemetry Tracking & Command (TT&C), Command & Data Handling (C&DH), Thermal Control and Structures & Mechanisms (S&M). How the values used in the table are derived, can be read in the following paragraphs.

For the LEO satellite, the Payload is divided into the Laser part and the Mirror part. The mass and volume of the laser and the mirror are given in section 7.7. The PM&D subsystem is subdivided in the parts collection, storage and distribution. Values for the collection are based on thin film photovoltaics and are determined in section 6.7. As a storage method, Nickel−Hydrogen secondary type batteries are used as defined in section 9.1.1. The distribution, meaning cabling, is about $10 - 25\%$ of the total mass of the PM&D subsystem. Both the values for the storage and the distribution method are calculated in section 9.1.1. In section 6.3 and 7.3 the numbers for the AD&C can be found. The values for TT&C and C&DH are given in section 9.1.3 and 9.1.2. The total values for the Thermal control subsystem are a combination of the Thermal control needed for the lasers, the mirrors and the collection method. The values found for the Thermal control for the laser and mirrors can be found in section 7.5, the values for the collection can be found in section 6.3.

The GEO satellites' payloads are 5 mirrors. In section 7.7 the values for the GEO mirror can be found. Because the GEO satellites receive their energy from the laser beams, they do not need storage. The values for the conversion and the distribution are found in section 9.1.1. The values for the AD&C, TT&C and C&DH subsystems are the same for both the LEO and GEO satellite. The values given for the thermal control for the GEO satellite are obtained in section 7.5.

Table 9.3: LEO satellite Mass, Volume and Power budget

LEO satellite					
Subsystem		**Mass** [kg]	**Dimensions**		**Power** [W]
Payload	Laser	$60,000$	200	$[m^3]$	$700\ 10^6$
	Mirrors	2750	110	$[m^3]$	–
PM&D	Collection	216000	3.44	$[km^2]$	–
			933.3	$[m^3]$	–
	Storage	$12 - 17.5$	1	$[m^3]$	–
	Distribution	$30,700 - 92,000$	$388 - 1165$	$[m^3]$	–
AD&C		$650 - 1000$	$1.5 - 2$	$[m^3]$	$145 - 1235$
TT&C		13.3	0.25	$[m^3]$	24.3
C&DH		$9.5 - 10.5$	$0.013 - 0.015$	$[m^3]$	$15 - 25$
Thermal Control	Transmission	660000	0.2	$[km^2]$	–
			8354	$[m^3]$	–
	Mirrors	$10,000$	3100	$[m^2]$	–
			127	$[m^3]$	–
S&M		$1.73 - 3.47\ 10^5$	$2150 - 4392$	m^3	–
Total		$1.21 - 1.74\ 10^6$ *	$14.9 - 19.1\ 10^3$ *	$[m^3]$	

* Margin of 5% and of 25% added to lower and higher value, respectively.

The mass and volume required for the LEO and GEO satellite S&M are very difficult to estimate. This is the case because the exact layout of the entire system is not known yet. The mass of the S&M is estimated to be between $15 - 25\%$ of the total mass of the system[4]. This results in a mass range of $173,000 - 347,000$ kg for a LEO satellite and $5,824 - 11,204$ kg for a GEO satellite. Finally, a total mass can be calculated. This adds up to a mass range of $1,153,000 - 1,389,000$ kg for a LEO satellite and $38,825 - 44,817$ kg for a GEO satellite. A margin percentage off $5 - 25\%$ should be taken on the total mass, volume and power depending on the design maturity. When this is done for the total mass found for the LEO satellite, its mass range increases to $1,210,500 - 1,736,500$ kg. For the GEO satellite this becomes $40,765 - 56,021$ kg.

The same can be done for the volume calculation. The total volume including a $15 - 25\%$ margin results in a $14,920 - 19,105\ m^3$ size for the LEO satellite and $833 - 1080\ m^3$ for the GEO satellite.

9.3 Functional Diagrams

Functional Diagrams are useful tools to create an overview of breakdowns and tasks in a system. This section features the Functional Flow Diagram in subsection 9.3.1, the Functional Breakdown Structure in subsection 9.3.2 and subsection 9.3.3 describes how the subsystems of the system interact.

9.3.1 Functional Flow Diagram

Before Heliodromus supplies energy on Earth, many steps have to be taken first. These steps are described in the Functional Flow Diagram in figure 9.1. The first level consists of nine functions. The subsystems of the satellite have to be launched using multiple launches. In a first phase, one satellite has to be build in orbit. This phase holds the looping of the first three items in the functional flow diagram. When a satellite is finished the next step to item four is started. The function is to check if all subsystems on the satellite are functioning

Table 9.4: GEO satellite Mass, Volume and Power budget

GEO satellite					
Subsystem		**Mass** [kg]	**Dimensions**		**Power** [W]
Payload (mirrors)		11250	450	$[m^3]$	–
PM&D	Collection	*unknown*	55	$[cm^2]$	–
	Distribution	$70 - 333$	$1 - 4$	$[m^3]$	–
AD&C		$650 - 1000$	$1.5 - 2$	$[m^3]$	$145 - 1235$
TT&C		13.3	0.25	$[m^3]$	24.3
C&DH		$9.5 - 10.5$	$0.013 - 0.015$	$[m^3]$	$15 - 25$
Thermal Control		21000	0.0064	$[km^2]$	–
			266	$[m^3]$	–
S&M		$5824 - 11204$	$74 - 142$	$[m^3]$	–
Total		$40.8 - 56.0\ 10^3$ *	$833 - 1080$ *	$[m^3]$	–

and if pointing is possible. A communication link with the ground station is established and information about position of the mirror, laser and the satellite is shared. The ground station transmits new position requirements and adjustments to the mirror positions. The systems start the collection of solar power and the beaming of lasers. The energy is received on Earth while of the positions of all satellites and the pointing of the energy beam are constantly monitored. These functions are stated in the levels stated under number 5 till 9.

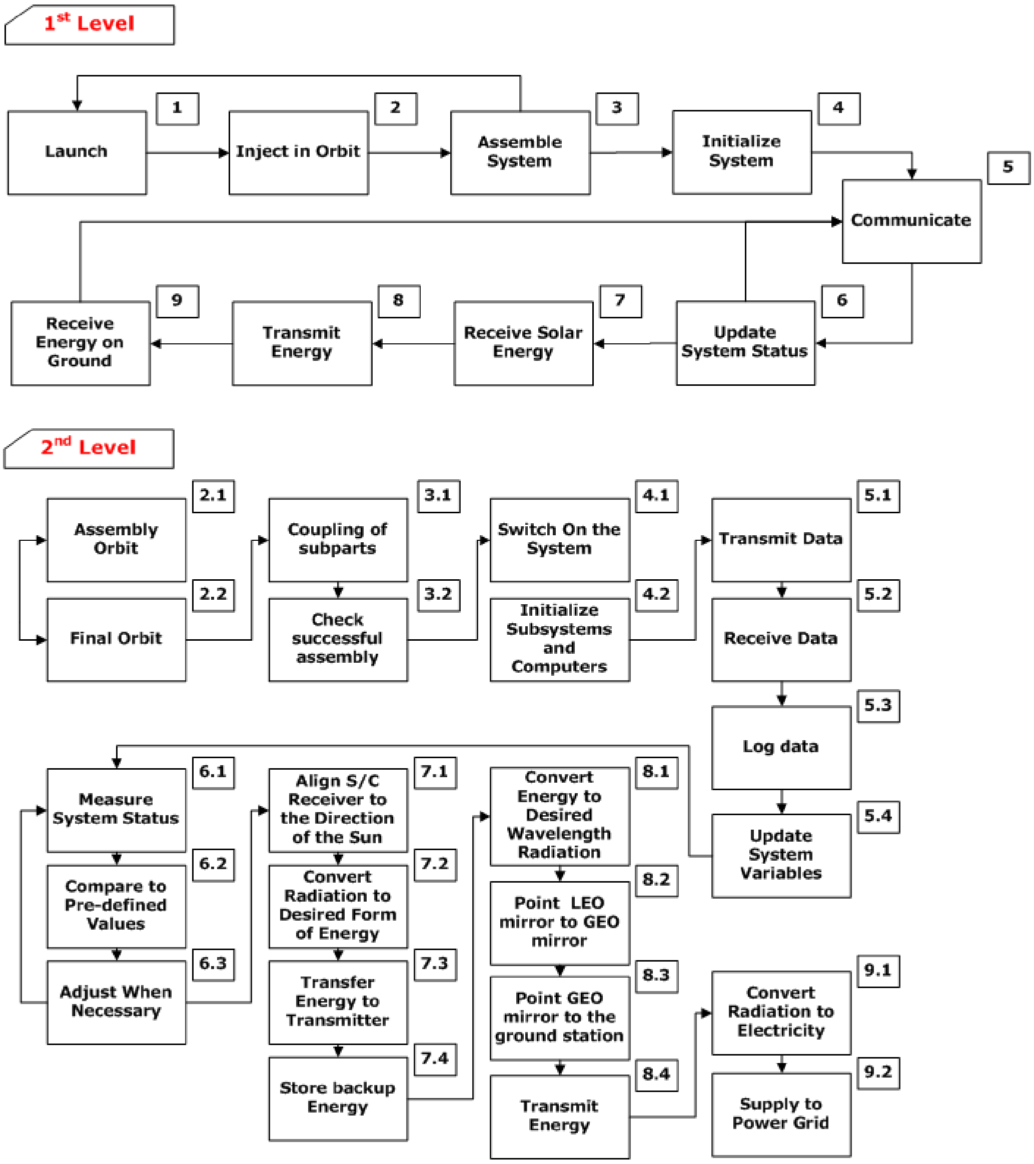


Figure 9.1: Functional Flow Diagram

9.3.2 Functional Breakdown Structure

In figure 9.2 the Functional Breakdown Structure is given. In the Functional Flow Diagram the time frame and following up of activities of the the Heliodromus concept is given. The Functional Breakdown Structure gives a clear overview of the required functions, which can be coupled to required subsystems on board the satellites or at the ground station.

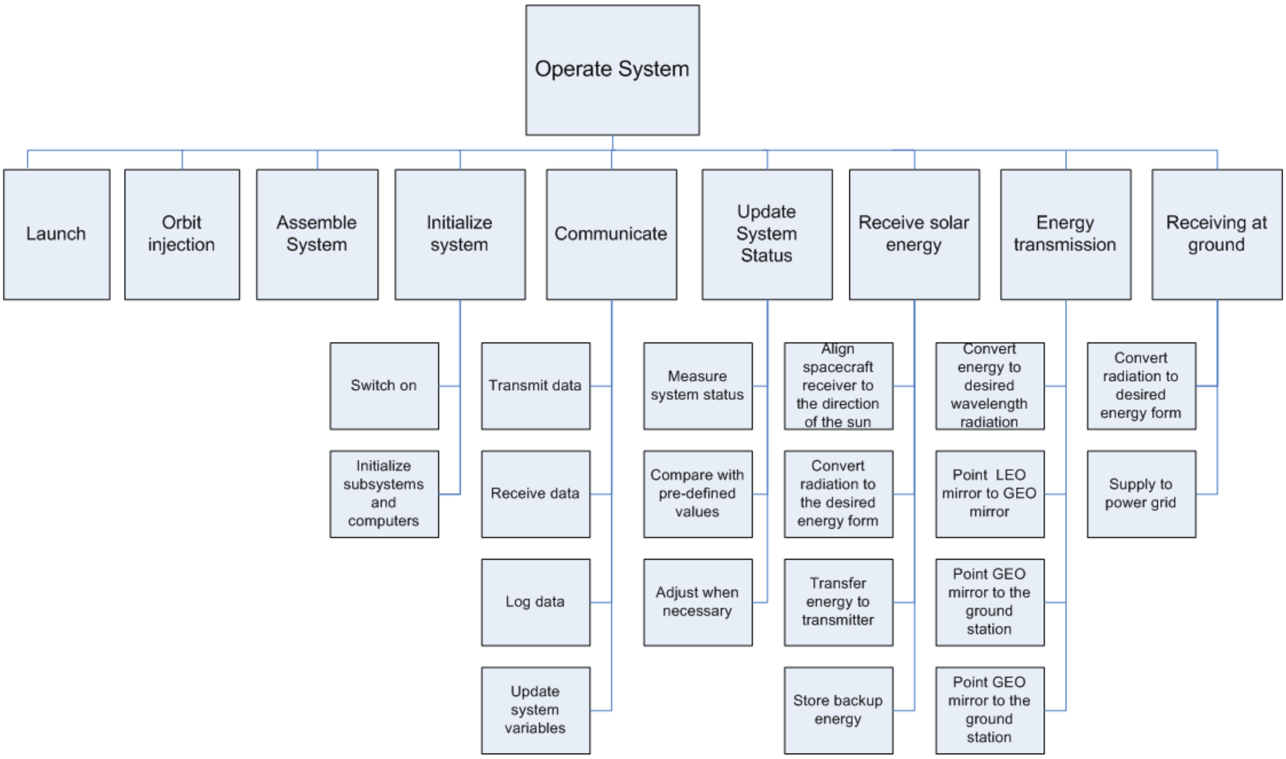


Figure 9.2: Functional Breakdown Structure

9.3.3 Subsystem Interrelations

A satellite exist of many different subsystems which each have a different function. In figure 9.3 the interrelations between Heliodromus' subsystem are shown. A division in this figure is made between the systems in GEO and in LEO because of their different subsystems. For the functional diagrams also a subdivision could be made but these figures enclose the entire system, LEO and GEO satellites.

9.4 Space Assembly & Launch

One of the major bottlenecks of Heliodromus is the assembly of the satellites in space. The dimensions of both the LEO satellites and the reflectors in GEO satellites are of such an order that they do not fit into a single launch vehicle. For this reason an assembly must take place in space to create the required structure. Since the assembly must take place for thousands of parts, the assembly of these parts and the realization of the large amount of launches in a short time frame will become a big challenge. The concept of Heliodromus has a big advantage compared to other concepts, which is the choice of putting the heavy satellites in LEO instead of GEO. The parts with the highest mass only have to be launched to LEO and hence require less launch vehicle fuel.

The assembly of the satellite components will be covered in section 9.4.1 and the launch challenges will be explained in section 9.4.2.

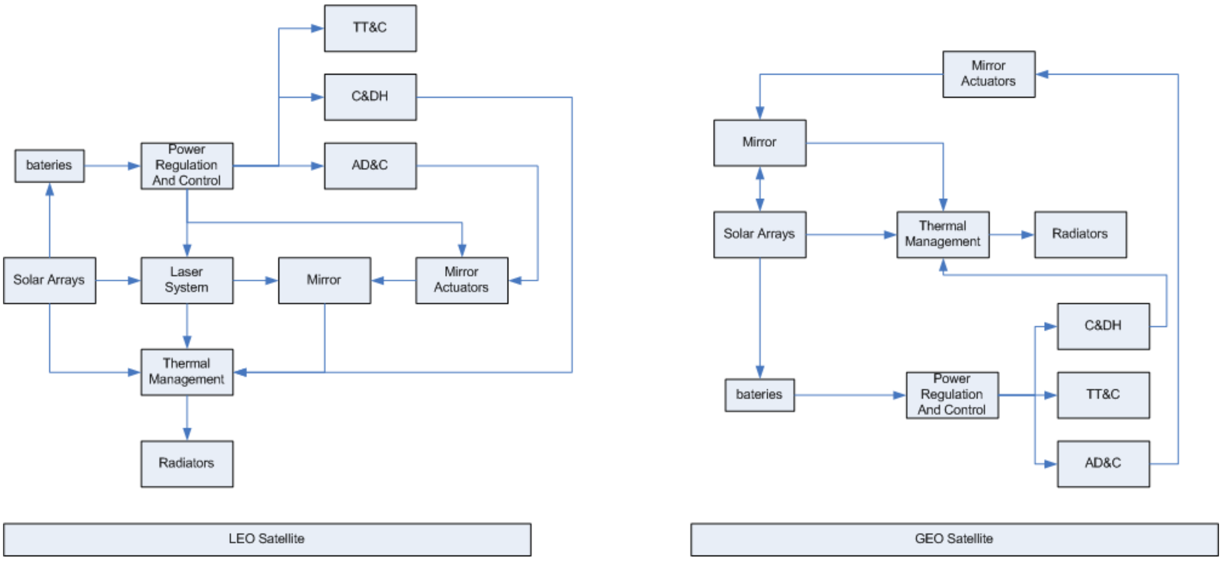


Figure 9.3: Subsystem

9.4.1 Assembly in Space

A possible scenario for the assembly of Heliodromus is described by W. Whittaker et al.[30]. The report explains the possibilities and challenges of the assembly of such a large space structure by robots. Assembly will likely take place by autonomous robots, because manned assembly involves too many risks and requires many different astronauts to get in space over a long period of time. The robots should have a long lifetime and should be able to cope with fragile parts.

As the system assembly requires a large robot operational station, it is logical to put the docking station in LEO, avoiding the launch of large mass quantities to GEO. Consequently, the LEO satellites can be built around the docking stations. Robots designed to build up the assembly station in LEO should be launched first. Additional robots specialized in assembly of the LEO satellite should be launched and added to the docking station. After launch, the construction and maintainance of the laser satellite in LEO can begin, followed by the construction of the GEO satellites. For the GEO satellite, this means that the assembly takes place in LEO first and the assembled structure has to be moved to GEO by thrusters. The 10 LEO satellites can be constructed at a single assembly station or every satellite can be constructed around seperated assembly stations.

Heliodromus' facilities will be larger, more complex, more fragile, and less accessible than the satellites of today. Delays, errors, and losses are inevitable and will result in chaos unless the process can be rescheduled quickly and efficiently. Current technology can not realize the assembly and maintainance of such large structures and a TRL of 3 is appropriate. It is clear that the technology for assembly and maintainance of the total Heliodromus system is far from available, hence research and quality guarantees on the subject are needed before Heliodromus can be constructed at all.

9.4.2 Launch to Space

The components of the LEO and GEO satellites have a high total mass and volume, as shown in section 9.2. Moreover, the robots and structural parts needed to create the assembly station will likely increase the total mass and volume by some factors. Since no values are available for the mass and volume of such an assembly operation, the total assembly will be excluded from the launch calculations, i.e. only the mass and volume budgets of Heliodromus are included. However, conclusions about the viability of the total system should be taken with care.

In section 9.2 the total mass of Heliodromus' GEO satellites has been determined to be $81,530 - 112,042\ kg$. Throughout the report the Ariane V launcher is taken as reference point for the launch related aspects of our system. It would be evident to use the Ariane V as a reference for the number of launches and the launch cost estimation. The total mass a single Ariane V launch vehicle can bring into GEO is 10 tonne or $10,000\ kg$[61]. For the GEO system a total of $9 - 12$ launches is needed, considering mass. The volume of the GEO satellites has been determined to be $1666 - 2160\ m^3$, as discussed in section 9.2. A single Ariane 5 launch vehicle can bring roughly $160\ m^3$ of component volume in LEO, hence $11 - 14$ launches are needed to get the satellites components into space, considering volume.

In section 9.2 the total mass of Heliodromus' LEO satellites has been determined to be $12,105,000 - 17,365,000\ kg$. The total mass a single Ariane V launch vehicle can bring into LEO is 21 tonne or $21,000\ kg$[61]. For the LEO system a total of $576 - 827$ launches is needed, considering mass. The volume of the LEO satellites is $149,200 - 191,055\ m^3$, as discussed in section 9.2. Therefore $933 - 1194$ launches are needed to get the satellites components into space, considering volume.

The minimum amount of launches with the Ariane V launch vehicle is limited by the volume, i.e. the amount of launches for the total system is equal to roughly $950 - 1200$.

In various sources it is mentioned that the launch cost of one Ariane V is 120 million dollars[68]. This would turn out to have a very significant impact on the total system cost. Fortunately, the size of this mission makes the economies of scale kick in. As a rule of thumb the decrease of the launcher unit price is taken to be $20\ \%$ per doubling of production. The current production of the Ariane launcher, in the years 2007 and 2008, was six launchers per year. Assuming that Heliodromus is to be launched in a time span of 4 years, the production of the Ariane launcher will increase from 6 to 306 per year. Or, put differently, a 51 fold increase in production. This means that the total production needs to double 5.67 times, since:

$$2^{5.67} = 51 \tag{9.1}$$

The effect on the unit cost of an Ariane V launcher due to this production increase can be calculated to be:

$$120\ 10^6 \cdot 0.8^{5.67} = 33.85\ 10^6 \tag{9.2}$$

Thus, at 33.85 million dollars per launcher, total launch costs amount to 40.62 Billion dollars.

The launch vehicles contribute to the total energy usage of the system through manufacturing of the vehicle and launch propulsion as will be covered in section 10.4. Improvements on the expenses and energy payback for the launch can be made by the introduction of newer, better launch vehicles. The economical boost the launch vehicle production site gets due to building the Heliodromus system is enormous and it is therefore likely technological advances in the launch vehicle occur. More effective launchers can deliver more mass and volume in orbit and increase overall viability. Further improvements can also be made in the research on component packaging. Satellite parts have to be packed in a dense way, e.g. by means of folding or by the use of inflatable structures. Smaller packages require less launch vehicle volume and therefore increase overall project viability.

9.5 Artist Impression

The artist impression is used to give an indication of the general outline of both the LEO satellite and the GEO satellite. CATIA is used as the CAD software to create three dimensional models of these satellites which provide a good overview of the designed construction.

9.5.1 LEO Satellite

The LEO sattelite consists of two extremely large solar arrays, both $1.72\ km^2$ in size. These arrays are build up from 76 separate deployable thin film solar panels discussed in section 6.7. These arrays are positioned in such a way that they will never obstruct the laser beam. The laser beam has a maximum deviation of about 10 degrees, hence the arrays form an angle larger than 10 degrees. 9.4.

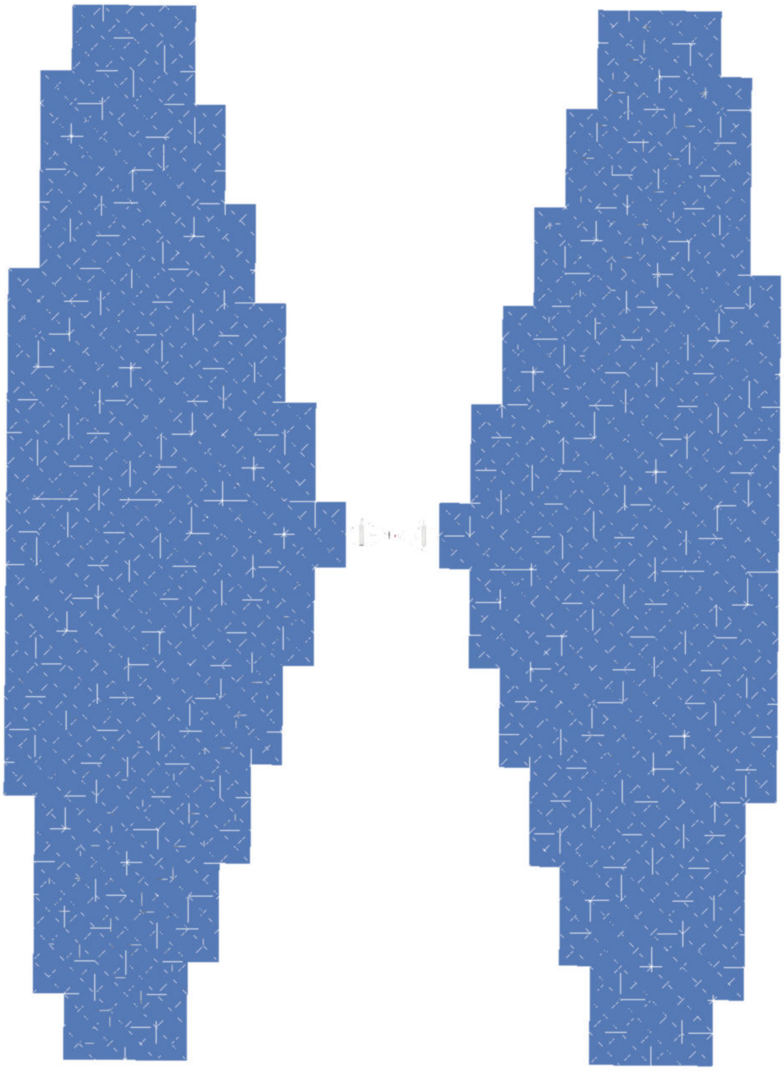


Figure 9.4: Configuration of the solar panels on the LEO satellite

Two arrays are chosen in this design to limit the amount of cabling needed to transfer the energy collected by the solar arrays to the laser system positioned in the center of the satellite, as represented in figure 9.5. Each array is connected to a separate laser constellation consisting of 200 lasers, discussed in section 7.1. The upper laser system differs from the lower one since it can point the lasers directly towards the focal point of the first mirror. The lasers are placed under an angle in a large circle to provide clearance for the rotating second mirror and still be able to beam the lasers through this focal point. For the upper lasers the cooling can be located on the inside of the laser system attached to the laser support structure. On the lower side the cooling for the lasers can be located in a similar manner only here on the outside of the lasers. The arrays and the laser constellation are connected by the support structure of the rotating mirror. Since this structure is rotating along with the mirror the arrays can also rotate with respect to each other, hence both arrays need their own attitude determination and control system to be able to point both arrays towards the Sun. This can be easily integrated in the trusslike support structure of the solar arrays, where also standard control systems needed for satellite operations can be accomodated.

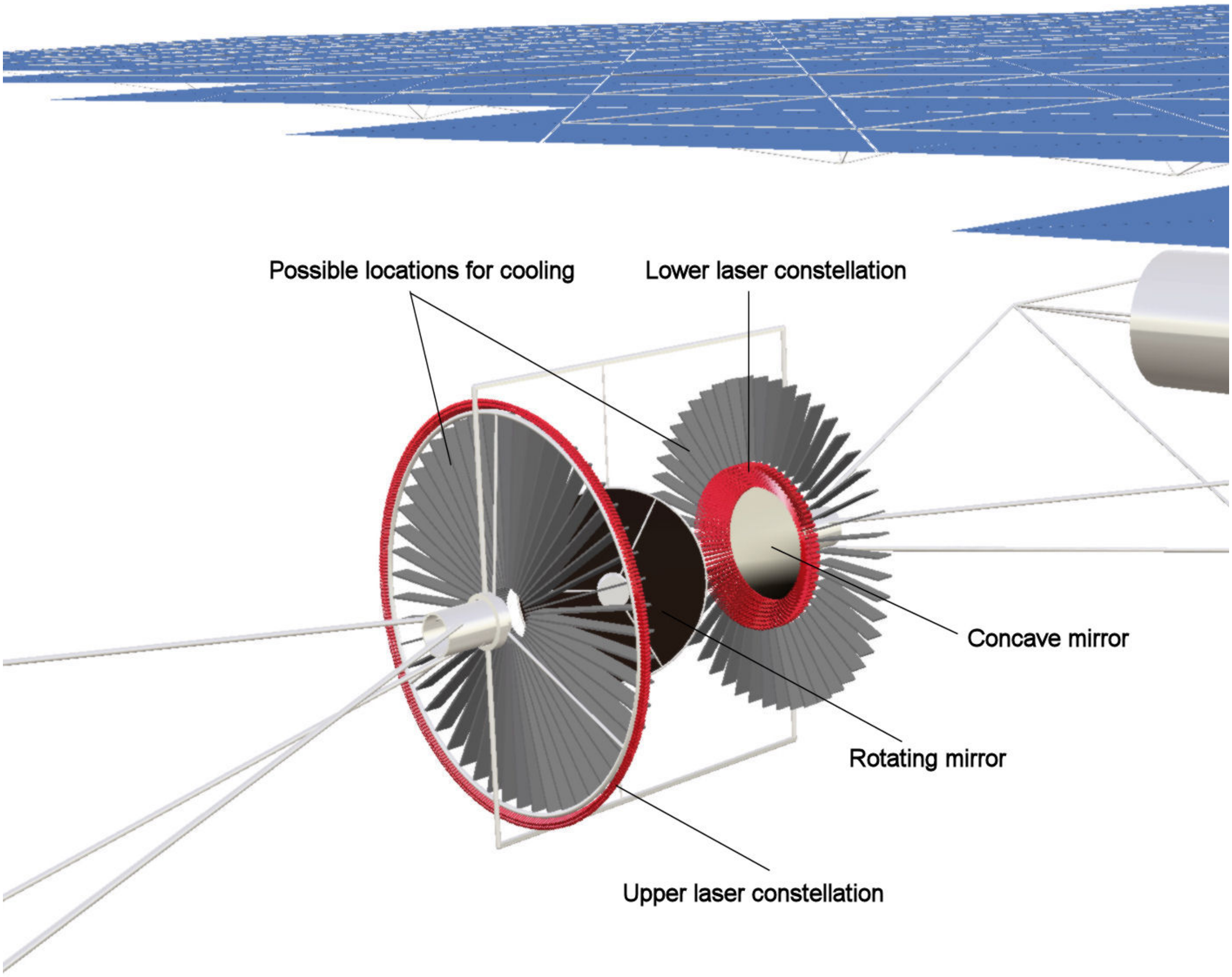


Figure 9.5: Laser system configuration

9.5.2 GEO Satellite

The GEO satellite consists of five separate mirrors which need to be able to follow a LEO satellite individually. Therefore it is needed for every mirror to have an attitude determination and control system. The mirrors are connected by a truss like structure to the central part of the GEO satellite, which accomodates all the necessary subsystems and which also includes another attitude determination and control system. The power supply for these systems is obtained from the laser beam by using solar cells located on the sides of the mirror. A graphical representation is given in figure 9.6.

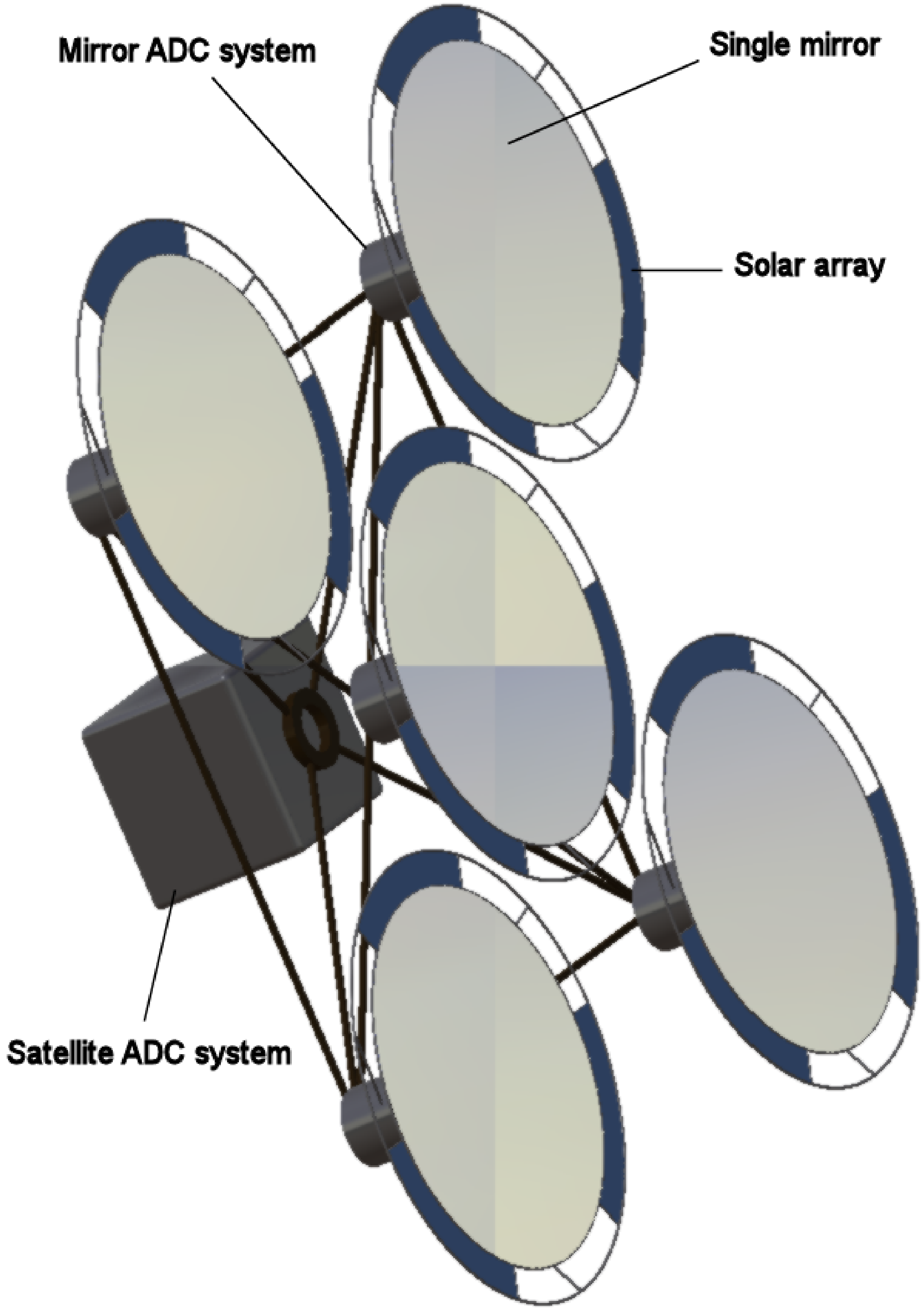


Figure 9.6: GEO mirror configuration

Chapter 10

System Evaluation

Chapters 5 to 8 dealt with several parts of the system and how the system will function and look like. In this chapter an evaluation of this system is made. This analysis is carried out to find out how the system performs. In section 10.1 the efficiency of the system is evaluated. Section 10.3 contains a risk analysis of the system. In section 10.4 the Energy Payback Time for the system is presented. Following this section, a cost analysis is made. In section 10.6 a compliance matrix shows to what level the requirements are satisfied. The last section of this chapter, section 10.7, compares the Heliodromus to an Earth−based Solar system.

10.1 System Efficiency

In this section the overall efficiency of the system is evaluated. In figure 10.1 a complete overview of Heliodromus is presented, per specific subsystem. A quick estimate of overall efficiency results in $1GW/47GW = 2\%$ complete efficiency. The 47 GW is the total amount of energy that can be received on 34.4 km^2, assumed a solar radiation of 1350 W/m^2. The 1 GW is the energy output of the ground system and has been the starting point for the calculation of the required input of Heliodromus. In figure 10.1 the efficiencies of all subparts are stated. The starting point is 1GW output power. Heliodromus is build up from five main conversion or energy transfer steps. Below rough estimates for the different parts are made.

Ground conversion, during which due to diffraction 16% of the energy is lost before it is received at the ground station (see section 7.2.5) with a conversion efficiency of 40% (section 8.2) this results in an ouput of;

$$Energy\ on\ Ground\ station = 1/(0.4 \cdot 0.84) = 2.98GW \approx 3\ GW \tag{10.1}$$

Due to the Earth's atmosphere only 85 $\%$ (section 7.2.5) of the energy will reach the earth surface;

$$Energy\ in\ Atmosphere = 1/0.85 = 3.51GW \approx 3.5\ GW \tag{10.2}$$

The beam is reflected from a GEO mirror. On this mirror again due to diffraction only 84% of the energy is received. The reflection has a efficiency of 99.5% ;

$$Energy\ from\ GEO = 3.5/(0.84 \cdot 0.995) = 4.19GW \approx 4\ GW \tag{10.3}$$

For the LEO satellite laser transmission system, an efficiency of 60% (section 7.7) is reached and again a mirror efficiency of 99.5% can be assumed which results in;

$$Energy\ from\ LEO = 4/(0.6 \cdot 0.995) = 6.7GW \approx 7\ GW \tag{10.4}$$

The final conversion is the total conversion of solar radiation to electrical energy in space. This conversion is done with a efficiency of 15% (section 6.2);

$$Energy\ on\ Solar\ Cells = 7/0.15 \approx 46.5\ GW \tag{10.5}$$

This requirement determines the minimum required amount of solar array area. This estimation of the power levels is a very rough fist order estimate which gives a little insight in the enormous amounts of energy flows.

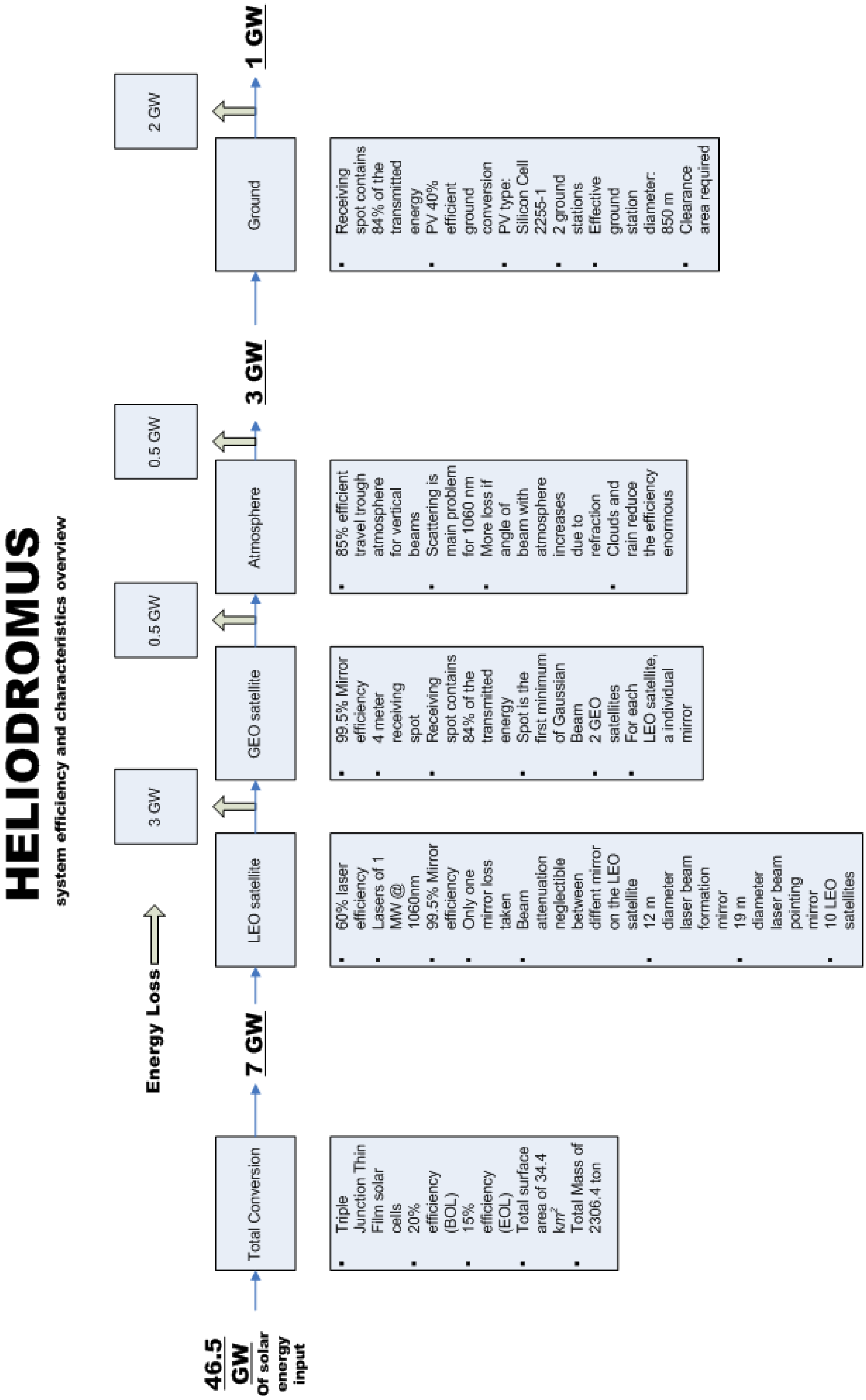


Figure 10.1: Heliodromus Characteristics overview

10.2 Sustainability

In the mission need statement we specified that: "'Design of a market viable *green* spacecraft (series) to supply renewable solar energy to Earth to be launched before 2025"'. The word *green* can be seen here as a synonym for sustainable. The reasoning behind this system is to harvest a renewable source of power (the sun) more efficiently. However, if harvesting the power of the sun in a more efficient way requires the investment of more natural resources than energy gained this would not result in a "'green"' system.
It is important in projects such as these that the system boundary is correctly defined. There are many manufacturers on the market that define their products boundary incorrectly, claiming environmental impacts irrelevant to their product. Looking at our products operation time the CO_2 equivalent contribution is limited to the output of secondary systems of the ground station. The impact of getting the satellite into space however, is quite significant.
Sustainability is viewed from four different perspectives; *Manufacturing Sustainability*, *Launcher sustainability*, *Transmission Sustainability* and *End of Life Sustainability*.

10.2.1 Manufacturing Sustainability

The manufacturing of this system is not very sustainable. The construction of a system of this magnitude requires a large amount of resources and a lot of energy to construct. However, the construction is largely similar to the same system designed for Earth based operations besides some hardening of electronics and different protections of the solar cells (UV vs dust). The biggest difference in manufacturing sustainability between our system and an Earth based system is the manufacturing impact of the launchers. The resources going into a launcher are quite significant. The scope of this project did not include the choice of a launcher preference. Ariane 5 is used as example for this chapter. An Ariane 5 ES launcher (incl 2 P230 solid rocket boosters) weighs 93.8 ton[56] (dry mass) and can lift 21 ton to LEO orbit. This does not add anything to the mass budget, but for manufacturing sustainability this means we will require 4 tonnes of launcher for every 1 tonne of payload.

10.2.2 Launcher Sustainability

In the Midterm Report[2] the launcher sustainability was calculated for a generalized case. With the design specifications known, as they are now, the exact impact can be calculated.
One launch of an Ariane 5 launcher uses 480 tons of solid propellant to get its payload in orbit[41]. The liquid fuel main booster uses Oxygen and Hydrogen to create water, its environmental impact limited to the generation of these parts. 480 tons of solid propellant (consisting of ammonium perchlorate, aluminium powder, polybutadiene) generates 495.3 tonnes of exhaust gases, built up according to figure 10.2[42]. Using the Global Warming Potential (GWP) of the exhaust gases an impact of 58883.7 tonnes CO_2 equivalence can be calculated per launch[43].

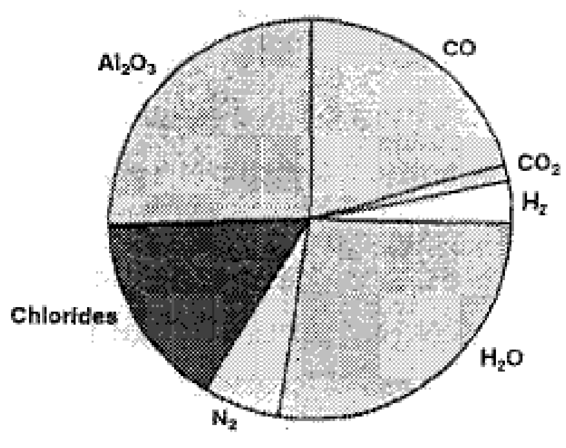


Figure 10.2: Exhaust gases emission contribution [109]

The 950 − 1200 calculated required launches (see section 9.4.2) would then result in 56 − 71 Megatons of

CO_2 equivalent greenhouse gases added to the atmosphere.
In order for the launching of this system to be worth this addition to our atmosphere, the energy provided needs to be greater than the energy provided by a non-renewable energy source that would add the same amount of CO_2 to the atmosphere. Coal, the least clean fuel currently in large scale use, produces 0.95 kg/kWh[8]. To compete with coal the SBSP system needs to provide $60 - 75$ TWh. For the 1 GW this means the exhaust payback time caused by the launch is approximately $7 - 9$ years.

10.2.3 Transmission Sustainability

The transmission of the energy from a satellite to the ground station is done by electromagnetic waves. These waves can affect the composition of the atmosphere, air quality and the environment on Earth, including living beings[28].

Microwaves have a relatively low frequency and are less harmful to the environment than high frequency radiation. The frequency of operation should be chosen in a range that the energy is barely absorbed by water vapor. This way the waves will not be absorbed by the atmospheric vapor or the water in organic material, thus preventing them from heating[44]. Research has shown that birds, monkeys, insects and humans are not adversely affected by microwave radiation at low intensities. Care should be taken when increasing the radiation intensity to values far above the safety guidelines[45].
In the range of possible wavelengths for WPT the 1064 nm laser is in the more dangerous range for living tissue. The laser should be divided into multiple small lasers to increase safety of humans and animals. A disadvantage of laser waves is the higher atmospheric absorption. The amounts of energy that is lost in transmission, which is influenced by weather conditions, will in some way create heat or chemical reactions in the different atmospheric layers[45].
To decrease the chance of negative impact on the ground environment the reception area should be closed for humans (and larger animals). Another safety feature should be that when the mirror aims at the wrong place the spot size of the beam should be increased to reduce the intensity to safe levels. 1 GW per 200 km^2 is significantly lower intensity than what the sun supplies in that wavelength bracket making it safe for human and animal. Mis-aiming by the satellite will mean missing the GEO mirror shooting the laser in deep space.

10.2.4 End of Life Sustainability

Definition of Sustainability:

> *Something that meets the needs of the present without compromising the ability of future generations to meet their own needs*[67].

This extends to the end of life strategy. Leaving a system of this magnitude in its operating orbit will seriously interfere with the ability to exploit that region of space in the future. And an object in space of such mass and volume presents a serious risk to Earth if handled incorrectly. Removing the structure from its orbit in a manner that does not create hazards, requires breaking up the structure and an additional ΔV budget. A structure this size might not burn up completely upon reentry, attempting to do so will send a several hundred meters of super-heated structure plummeting to Earth. The Tunguska event[46] showed the impact an object several tens of meters across can make, though most likely that structure was more dense. Reentry in a single piece is therefore probably not an option. The satellite will have to be launched in different pieces and a simplistic assembly is desired. These couplings should be engineered in such a way that they can be released when reentry is needed. Reentry of objects of single launch size is widely documented and known how to make it

burn up completely.
The mirror end-of-life is a different matter, sending a satellite to a degrading orbit from GEO is not as easy as from LEO, though still possible. The customary end of life strategy for GEO satellites is re-orbiting in a higher orbit. This would require an additional ΔV budget. As the mirror satellite has independent solar cells to power its systems, a function no longer required at end-of-life, this could be used to power an ion drive. A different solution is to push the satellite out of GEO and into a higher orbit by laser propulsion. However, the force the 3.5 GW laser exerts on the GEO vessel is only 6 Newton, as calculated in section 7.4. This force is insufficient on its own, thus it will not be feasible to use it for this application.

10.3 Risk Analysis

The project involves numerous risks in different aspects, which will be treated in this section. First the technological risks of the various subsystem in terms of the readiness level is analyzed. Also assembly risk and other risks that are an issue throughout the design phase and operation phase of Heliodromus are the matter of discussion in this section.

10.3.1 Technology Readiness Levels

Heliodromus uses a lot of state of the art technologies which often have not been implemented in existing products already but are achieved in laboratory conditions. As can be seen in table 10.1 all technologies are not ready for short term implementation into a functional system. Demonstration and pilot missions have to be initiated to design fully working subsystems. The main bottlenecks in technology readiness levels are laser technology and thermal management. Which are related because when the waste heat during the conversion is reduced, less advances in thermal management are required.

Table 10.1: Overview of subsystem TRL levels

	The Technology	**TRL**	**section**
Wireless Power Transmission	Mirror	6	7.6.1
-	Laser	3	7.6.2
-	Pointing & stabilization system	7	7.6.3
-	Thermal management	4	7.6.4
PV in Space	Thin Film	4	6.2
PV on ground station	Si cells	4	8.2
PM&D	-	7	9.1.1
C&DH	-	9	9.1.2
Assembly	-	3	9.4.1

10.3.2 Assembly Risk

During assembly there are two important risks, impact with meteorites or with other already assembled parts. The second risk is wrong assembly with the result that parts get damaged so that they have to be replaced or fixed. With the amount of assembly required the chance that damage occurs is large, and damage during assembly will automatically cause more costs.

10.3.3 Operation Risk

During the operational time of Heliodromus there are two main risks. The first is collision with another space object and secondly, malfunction of a complete subsystem. Both risks will have enormous impact on the operation of Heliodromus and will most of the time result in a partly or complete inoperative system.

10.3.4 Market Risk

Before Thin Film PV can be applied the technology have to be further developed. The goal of Heliodromus is to be competitive with Earth-based energy sources, when solar cells become less expensive the Earth-based PV system will also be lowered in price. For Heliodromus to become competitive with land based system, progress in launch and spacecraft cost have to made. Or efficiencies of subparts has to be increased dramatically. When it comes to reliability the risk of a system in space is larger than a system on earth in terms of maintenance and operation. Therefor Heliodromus faces some large market risks which will make it difficult to find investors to initiate pilot mission outside governmental organizations.

10.3.5 Safety Risks

The power densities of the laser beam on the ground is $13000 W/m^2$ (section 8.2). This requires no go areas and safety zones around the receiving site of the laser beams. Technologies to provide quick defocussing and steering of the beam is necessary to act when dangerous situation occur.

10.4 Energy payback time

In order to determine and calculate the energy pay back time(EPT) for the LEO-GEO laser power system, the period is calculated separately for the transmission system, the collection system, the ground bases and the launching system.

10.4.1 Launching system

An entire Ariane V launcher weighs approximately 775 tonnes which consists of 675 tonnes of fuel, 80 tonnes of structural mass and 20 tonnes of payload[59]. The spacecraft that will launch the SBSP system is assumed to have the same characteristics of the Ariane V but the fuel will, unlike that of the Ariane V, exist of a mixture of 1/7 liquid hydrogen and 6/7 liquid oxygen[106]. The energy requirements of these two types of fuels are 492 and 6 GJ/T respectively[106]. Furthermore the structural mass consists primarily out of aluminum with an energy requirement of 173 GJ/T[106]. The table below (10.2) shows the calculation and also the rough estimates about the energy density of the launcher. The energy density of the entire Ariane V launcher is thus approximately 68 TJ.

10.4.2 Total energy payback time

Besides the launcher, the energy requirements of the other parts of the system, namely the ground station, the thin film solar cells, the laser system, the GEO mirror, the laser cooling system and the launcher structure have to be considered for the calculation of the total energy density of the system. The table below (10.3) shows the different components of the system and the regarding energy densities.

Table 10.2: Energy payback time Ariane V launcher 7.7,[106],[107],[105]

	Mass [$tonne$]	**Energy** [GJ/ton]	**Energy** [GJ]
Structure(Al)	80	173	13840
Fuel(LH2)	96.43	492	47442.86
Fuel(LOX)	578.57	6	3471.43
Launcher equipment(Al)	20	173	3460
Total per Launcher	775	844	68214

Table 10.3: Energy payback entire system 7.7[106]

	Energy [GJ/m^2]	**Surface** [km^2]	**Total energy** [GJ]
Ground station	15.8	0.567	$8.96\ 10^6$
Thin film	3.1	34.4	$1.07\ 10^8$
	Energy [$GJ/tonne$]	**Mass** [$tonne$]	**Total energy** [GJ]
Laser	920	600	$5.52\ 10^5$
Cooling laser	173	6600	$1.14\ 10^6$
Mirrors	141	50	$7.05\ 10^3$
Structure	57	1810	$1.03\ 10^5$
Launcher	775	844	$8.14\ 10^7$
Total			$1.99\ 10^8$
		EPT	**6.02** [$years$]

The values used in the table 10.2 above for the total surface of the ground, the total surface of the thin film photovoltaics, total weight of the laser system and the cooling system of the laser system are taken from subsection 7.7. Table 10.2 provides the total energy per launcher. The average number of launches is 1064 giving rise to the total energy value of $1.99\ 10^8$ GJ, as can be seen table 10.2. This leads to a total energy payback time of approximately 6 years, with a margin of 10 % due to the uncertainty in the number of launchers.

10.5 Cost Summary

The price tag of the system is of critical importance, after all it is often being said that it is money that makes the world go around. And since it was mentioned in the project objective statement that the system should be market viable, it is also a key parameter for the SBSP system.

In the market analysis it was pointed out that the energy the system provides would have a price of around 8 dollarcents per kWh, it must be noted that this estimate was based on a comparison with non renewable energy sources. Furthermore cost forecasts over a prolonged period of time have often proven to be inexact.

The system is designed to have an end of life output of 1 GW. Due to the nature of laser and thin film solar cell efficiency the output is assumed to remain quite constant. This leads to a total energy output of 87.6 TWh over the ten year lifetime. At 8 cents per kWh this leads to a total budget of more then 7 billion dollars.

In the chapter 11.1 it can be seen that the conservative estimate of the total system efficiency can be surpassed with a factor of 3.5 when all technologies are fully optimized. This means that without significant increases in costs of the system, it could generate an output of 3.5 GW. Leading to a budget of almost 25 billion dollars. Not to mention the possibility of extending it to a 15 or 20 year lifetime.

The costs of the system is dependent on a lot of different factors. In section 6.2 it was estimated that the collection subsystem, which would generate 7 GW of power, could be operational at a cost of 2 dollars per Watt, totalling to a collection cost of 14 billion dollars.

The transmission system has two main components that play a major role in the system costs. The mirrors, costing a total of 1 billion dollars, and the lasers, with a contribution of 40 billion dollars.
The ground segment was estimated to cost about 2.5 dollars per Watt. Since both groundstation need to produce 0.5 GW, total ground segment cost is estimated to be 2.5 billion dollars.
Another important cost aspect is the launch cost. It was established in section 9.4 that maximally 1200 launches are needed for Heliodromus to be lifted into space. Assuming a launch cost of 33.85 million dollars, this amounts to 40.62 billion dollars.

This leads us to a combined system cost of 98.12 billion dollars, note that this figure is based on the most pessimistic mass estimate and therefore the highest launch costs. However it should be noted that this price tag only accounts for the manufacturing costs. Additional development cost will probably negate the effect of the pessimistic mass estimate. As a conclusion it can be said that for this system to be cost competitive with fossil fuels, the efficiency must be optimized as shown in chapter 11.1 and the energy cost need to quadruple at the very least. In comparison, the Apollo project, adjusted for the consumer price index, costed 161 billion contemporary dollars.

10.6 Compliance Matrix

The SBSP system has one major requirement: being competitive with other energy resources, hence being market viable. The design concept described in this report has proven to be not market viable and therefore the requirement is not met (see section 10.5). This main requirement brings in more subrequirements to fulfil. Table 10.4 gives the requirements for the total system and the subsystems that follow the main requirement. Some requirements were quantitatively set, while others were kept as a design parameter. The table shows the requirements, eventually the target value, the actual value met by the system, a notification if the requirement has been met and a reference is made to the related section.

Table 10.4: Compliance Matrix

Initial Requirement	Requirement Value	System value	Req. Met?	Section
Market Viable	$7 – 8 cents/kWh	$101 cents/kWh	No	10.5
Power Output	1 GW	1 GW (EOL)	Yes	10.1
Life Expectancy	10 – 15 year	1 year	No*	7.1
Energy Payback	52.6 TWh	88 – 132 TWh	Yes	10.7.1
Safety**	16 W/m^2	10 W/m^2	Yes	8.2
LEO Eclipse Time	-	88 min./year	-	5.1
Collection Efficiency	-	20 – 10 % BOL-EOL	-	6.2
Power Input	-	45.4 GW	-	6.4
PV Area	-	34.4 km^2	-	6.7
Transmission Efficiency	-	36 %	-	7.7
Laser Efficiency	-	60 %	-	7.7
Mirror/laser Pointing Accuracy	-	30 mas	-	7.1
Mirror Reflectivity	-	0.995	-	7.1
Reception Efficiency	-	40 %	-	8.2
GS PV Diameter (Spot)	-	500 m	-	6.2
GS PV Diameter (Accuracy)	-	850 m	-	6.2
GS PV Diameter (Safety)	-	1300 m	-	6.2
Grid conversion Efficiency	-	98 %	-	8.6

* Low laser life expectancy
** Laser Intensity Outside Clearance Area

10.7 Space vs. Earth

The Mission Need Statement that the designed concept must be market viable. In order for Heliodromus to be market viable we need to be able to provide a service for a competitive price. In the market analysis[1][2] the required price for a SBSP concept was already compared with currently available energy production methods. In order to draw a contrast, this section describes what happens if the solar cells that will be put in space, would be placed somewhere on Earth. A comparison will be made for power, costs and an energy payback time for both space and Earth-based systems will be determined.

10.7.1 Power

The location of this fictional system will be in a low cloud cover area within latitudes between 30^o north and south latitude, not too far away from an area that can use a large input of energy and willing to pay for its construction.

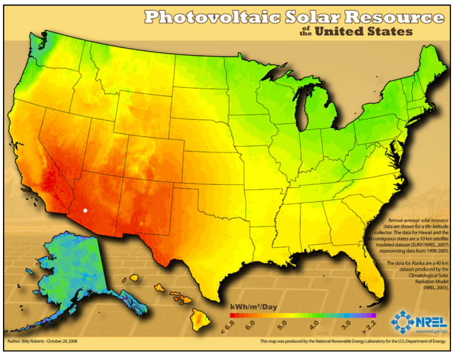


Figure 10.3: Annual Average Solar Intensity in the USA [62]

Taking figure 5.10 again as an indication this leads to California/Arizona/Nevada/New Mexico, Egypt, the Arabian peninsula, Australia and Spain. One of the two ground stations was chosen to be in Arizona, USA near the city of Phoenix, as indicated in section 5.4.1. Further comparison calculations will be related to this ground station.

The PV cells that are used for power collection in space are of the thin film type, have a collection efficiency of 20 % (begin-of-life), 15 % (end-of-life) and cover a 34.4 km^2 area in space, as stated in section 6.7. On Earth this provides us with 24/7 of 1 GW energy (Laser is limited to 1 GW throughput). That translates to 8.8 TWh (end-of-life) delivered power per year. For the SBSP system with a lifetime of 10 years this results in a total energy generation of 88 TWh. For a 15 year lifetime this increases to 132 TWh.

Figure 10.3 shows the annual average solar intensity for the USA from the National Renewable Energy Laboratory (NREL). The ground station location is indicated in white. The average solar intensity for the ground station is roughly 6.8 $kWh/day/m^2$, which is equal to an average of 280 W/m^2. A comparison of the Earth based solar plant to the space based solar plant can be made. The area of solar panels used in space for the SBSP system is equal to 34.4 km^2. Over the total area the solar plant will then have a solar intensity of 9.6 GW. Using the same PV cells with an efficiency of 20 % (start of life) and 15 % (end-of-life) a total of 14.8 TWh(end-of-life) power will be delivered per year. The lifetime of an Earth based solar plant is approximately 20 years. This differs from the space based solar panels, as the panels in space degrade at a higher rate. A 20 year lifetime results in a total generation of 296 TWh of electric energy. Previous calculations are done with the end-of-life power per year, both for the Earth and space based solar plants.

Currently this means that the system could generate approximately 125 % more energy, assuming a lifetime of 15 years for the SBSP system, by just taking the thin film cells used for energy generation in space and put them in Arizona, USA. This removes the need for the rest of the space structure and all the launches. Note however that the Earth based solar farm uses a 34.4 km^2 area on ground and the space based solar system uses only a 1.5 km^2 area on ground.

10.7.2 Cost

Where, in subsection 10.7.1, the same area of solar cells was used to make a comparison between the power generated over the lifetime of the product, in this section we will compare Heliodromus with existing power plants in terms of investment costs of the total system.
For comparison the investment cost of several existing earth based solar dynamic and photovoltaic installations.

Heliodromus is capable of delivering almost a continues $1\ GW$ which results in $24\ GWh$ per day and approximately $8800\ GWh$ per year. In section 10.5 the cost for the system with a lifetime of about 10 years are 98 billion dollars.
In table 10.5 the comparison with some earth based systems is given. Although this is a simple evaluation the results are clear; Heliodromus delivers up to a factor of 100 times more energy but costs a 1000 times more. Heliodromus is a factor of 10 times more expensive then already existing Earth-based system. Where the cost of Earth-based systems will continue to decrease before Heliodromus can be operational.

Table 10.5: Energy Cost Comparison between Heliodromus and Earth-based energy systems

	Average Output	**Costs**	**Lifetime**	**Output**
	$[GWh/year]$	$[million\$]$	$[years]$	$[\$/kWh]$
Heliodromus	8800	89000	10	1.01
Moura Spain PV	93	325	20	0.17
Waldpolenz Germany PV	40	170	20	0.21
Andasol Spain SD	180	390	20	0.11
Nevada Solar One SD	134	266	20	0.10

10.7.3 Energy Payback Time

In section 10.4 the energy payback time of Heliodromus is calculated and is around 6 years. For a life time of only 10 years the total sustainable advantage of Heliodromus is only for 4 years. For earth based systems the energy payback times ranges between 1 to 2.7 years[13]. With a life time of 20 years the sustainability of Earth-based system is higher then for Heliodromus.

Chapter 11

Future development

Before Heliodromus can be realized and become a functional system, many things still need to be done. In this chapter some of these are covered.
Section 11.1 highlights the most important challenges in developing the Heliodromus concept. Section 11.2 deals with the Logistic and Operations Management of further development. Section 11.3 details the process needed for the entire project, going into what specific actions need to be undertaken and describing where the last blind spots in the project need to be tackled.

11.1 Design Challenges

It is always the objective to come up with a concept which fulfills all relevant requirements, but also a practical concept without too many bottlenecks. However bottlenecks are inevitable. Doing a proper trade-off enables one to make choices which are the least bad, resulting in least amount of obstacles. This is a complex procedure, as one has to make assumptions about the technology readiness levels. Moreover the subsystems have interrelations which make it harder to judge and perform an optimum trade-off. Therefore sometimes the trade-off process tends to become iterative or even a trial and error process. The Heliodromus concept is in fact a concept which demands state of the art technology and also a concept which at current times lacks economic viability. Comparing Heliodromus with a current Earth-based solar energy system points out that the concept, at this point, is not able to compete with current Earth-based solarsystems, as shown in section 10.7. To be able to compete at a future time recognizing and considering potential future solutions for the occurring bottlenecks is of high importance for the success and realization of the concept.

11.1.1 Efficiency Improvements

The overall efficiency of Heliodromus is calculated to be around 2 %, as shown in section 7.7. Table 11.1 shows the efficiencies of different parts and the overall efficiency of the system. The efficiencies are not rounded, which explains the minor discrepancy between the value calculated in section 7.7. The table also shows future efficiencies for the components where improvements are expected to occur. The future efficiency will approach 8%.

The thin film solar cells currently have an efficiency of less than 20 % as discussed earlier in section 6.2. However looking at the technology readiness level of thin film solar cells, it is shown that the efficiency above 20 % will be feasible in the future[76].

Concerning the Ground photovoltaic subsystem, an efficiency of 40 % has been assumed. By beaming the

Table 11.1: Overall efficiency of overal system

Component	**Efficiency** [%]	**Future Efficiency**
Thin Film	15	30
Laser	60	70
LEO Mirror	99.5	99.5
GEO Mirror	99.5	99.5
Airy Disc Spot Mirror	84	84
Atmosphere	85	85
Airy Disc Spot Ground	84	84
Ground PV	40	60
Overal Efficiency	2.14%	7.48%

laser on a different wavelength the efficiency in lab conditions is already 60 % [21]. It can be expected that for the 1064 nm wavelength laser, future developments will rapidly occur.

11.1.2 Economic Viability

The fact that Heliodromus, at this point in time, is not able to compete with a current Earth-based solar energy system, as has been elaborated on in section 10.7, essentially states that the concept is economically inviable. Heliodromus requires a large amount of launches to put the solar cells, mirrors and lasers together with other subparts of the system into the required orbit. It should be noted that a system with the size of Heliodromus would redefine the landscape of various markets such as the thin film solar cell market, the high power laser market and the commercial launcher market. To project how these markets will respond in the event of such a soaring demand is an impossible task. However, the price estimate of 98.12 billion dollars elaborated in section 10.5 should be viewed as a worst case scenario. But even then, Heliodromus is meant to become viable and attractive for the future markets, as the fossil fuel reserves start to run out and great technological improvements are being made.

11.1.3 Subsystem bottlenecks

The subsystems of Heliodromus also show some bottlenecks. For instance the cooling of the laser system is one such a bottleneck (See section 7.6). Taking into account the scale of the laser system and the energy output, a great amount of heat will be generated. This heat needs to be removed, either by an active cooling system or by capturing the energy waste and reusing it. Another issue is the fact that the laser technology for the application in energy transmission is still in an immature phase. The low technology readiness level of laser means high research and production costs. Concerning the LEO satellite's solar cells, the assembly is expected to become an uphill task, mainly due to the unprecedented scale of the solar cells.

11.2 Operation Management and Logistics

Operation Management and Logistics (OM&L) is a multidisciplinary field that finds the balance between product development, quality management, logistics, information systems and human resources management. OM&L covers the whole supply chain, from the gathering of raw materials, through production, to the point of consumption, in other words everything that is not designed yet necessary to build or operate the spacecraft.

Running a project of this scale and impact, will require a lot of effort and proper communication. The subdivisions made in this system are described in section 11.3 and the time lines for their respective development described in Appendix B. All these subdivisions will have different development times and will suffer different delays. Coordination between these subdivisions will be the main task of OM&L.
Indicated in the Gantt chart is the Team development in which assignment of general managers is shown. OM&L is usually a small part of every managers job, however in big projects a designated manager is put on the problem. In this case, this manager will be appointed in this phase.

Important Logistic aspects
The construction of many different parts of Heliodromus is done all over the world and the launches will have to be done from a number of different space centers to accommodate the number of launches required. This means extensive transportation of parts, assemblies and subsystems.

The current world wide capacity of produced thin film solar cells is 2.8 GW[40] and all manufacturers report having problems meeting demands. The current trend in capacity expansion for thin film PV is quite spectacular, but market demand is expected to increase at a same or higher pace[40]. Considering the bulk of required solar cells it will not be possible to have one manufacturer make all required solar cells in one batch, leading to batch and manufacturing differences which have to be incorporated into one product. This is the case for several different parts of the system. It will also be impossible for the 4000 lasers required to have completely similar characteristics due to these reasons. All these discrepancies need to be handled and problems arising from that analyzed and mitigated if needed.

Licensing all the parts of this system presents a problem. The licensing of most subparts of our system will be within normal parameters. But on a large scale, our system will consist of putting a high power laser in space, where space is designated to *"not to place in orbit around the Earth any objects carrying nuclear weapons or any other kinds of weapons of mass destruction, install such weapons on celestial bodies, or station such weapons in outer space in any other manner"* according to Article IV of The Outer Space Treaty of 1967[66]. While not specifically built as weapon it can be used as such and this will certainly have impact on the licensing procedure. Besides licensing this will certainly create problems within the realm of public opinion.

11.3 Production Plan

This chapter will cover the production plan, describing the work packages and time frame needed for the further development and building of the Heliodromus concept.
The mission need statement did not contain a specific time frame when the product needs to be operational, so the product development time scale can be properly developed according to the knowledge available now.
To actually build Heliodromus is yet a major challenge. Described herein is a fair amount of detail regarding the type of subsystems and the reasoning behind it. In order to have this system operational in space, more detailed designing needs to take place. This process will take many years, the production of the subsystems needed for this system will also take years, the amount of launches again will take many years resulting in quite a long lead time of this project.
A basic workflow overview is given in section 11.3.1 and a more detailed overview is presented in the Gantt chart in Appendix B.

As displayed in the Gantt Chart Heliodromus will be fully developed and operational in the early 2040s. In some cases the size of the subsystems will exceed the capacity of a single company. The amount of solar cells required for the space segment alone for instance is half the current annual output of all solar cells worldwide[40]. Besides further development of Heliodromus a further development of these market areas are also required for those segments that do not have the capacity to supply the Heliodromus system. Whether these parts of the industry need to expand their capacity temporarily or permanently is left to the market flow. Many projects that may not be viable now may be viable when the economic impact of Holiodromus has changed price and availability of products in these fields.

The project to construct Heliodromus starts with a detailed analysis of this report and further detailing of the technology required. The resulting subsystem report will then be used to attract and select companies for the next phase of development. The further development of subsystems and creating of blueprints and construction manuals is then handled by specialized researchers and companies. Communications between the different sections is important, but also communication with the launcher development needs to be maintained as launcher customization is possible.
At the end of the development of the different subsystems, certification will have to be performed and documented so licensing and applications of permits can be done. Prototyping will take place during the time it takes to get the licenses required and manufacturing follows that. Subsystem assembly will be done at a/various specialized complex(es) with close access to a for example a large airport.
Launching all parts simultaneously is not a possibility. This means assembly will be spread out over the time required to launch everything, which means several years. Testing of the assembly as it grows is done continuously with every addition. The modular structure of the LEO satellite means that downscaled power output can be realized before the completion of the system. Parallel to this is the political and financial development of the project. Political support is important as it is required to have satellites operate in their ”airspace” as well as building a ground station on their soil. Next to that politics may provide a large part of the financial push needed. In the light of security requirements some hostile nations, when not agreeable to Heliodromus using their airspace, could destroy the system quite easily.

More detailed scheduling of Gantt-defined subsystems will be handled by the teams responsible for those subsystems.

11.3.1 Work Flow Diagram

The work flow of the project is depicted in figure 11.1

This figure is a representation of the basic work flow required to build Heliodromus. The milestones are depicted by diamonds while the processes are represented by rectangular blocks. Many of these blocks (e.g.Technology development) can last for years and in case of parallel development some of the blocks can happen at the same time.

11.3.2 Project Gantt Chart

The Gantt chart is shown in Appendix B

A project this size needs parallel development of lot of different parts, these are subdivided in:

- Logistic Development •Political Development
- Financial Development •Technical Development

The Technical Development in its turn is further subdivided in *Laser*, *Solar Cell*, *Groundstation* and *Satellite* Development.
In the case of the Technical development several of the technologies need to get more refined before they can be applied to Heliodromus. The products with sufficient TRL need to be customized to our system.

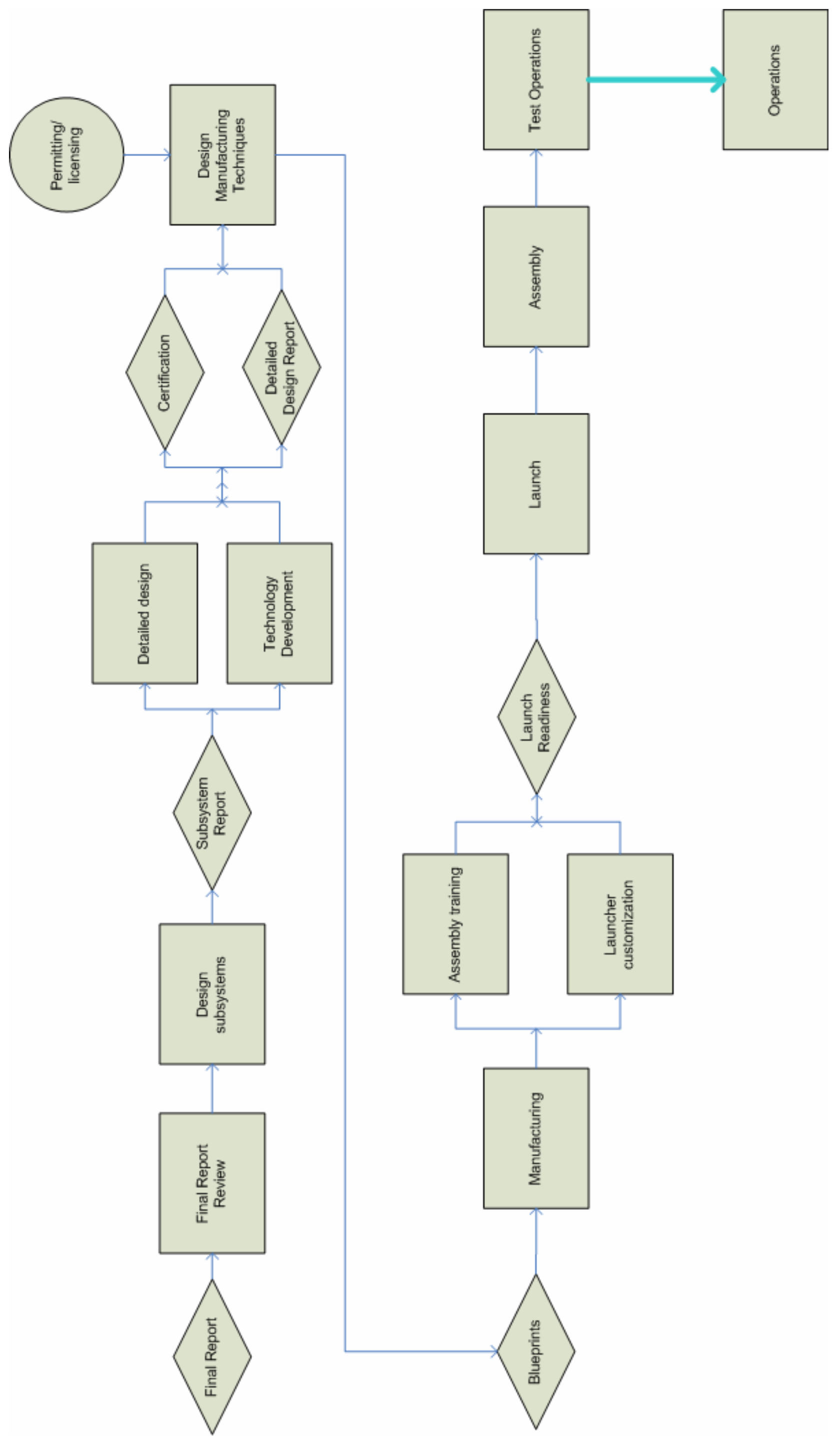


Figure 11.1: Work Flow Diagram

Chapter 12

Conclusions and Discussion

This report shows a broad analysis of the complete Space Based Solar Power system Heliodromus. The goal of the project was to give an overview of recent technologies and design possibilities. The total system from launch to the operational phase of Heliodromus has been addressed in this report. Trade-offs between different technologies for different subsystem have resulted in a complete concept.

Heliodromus is the result of a concept exploration. Analysis of the required subsystems for Space Based Solar Power resulted in the proposition of three concepts. Trade-off and analysis between the three different concepts resulted in the selection of Heliodromus. This stage has been the starting point for further analysis on the required subsystems for Heliodromus.

One requirement of the study was a concept which would be competitive with Earth based power supply facilities. The market analysis resulted in a price of $0.07 - 0.08$ \$ per kWh for fossil fuels around 2030. This price range is the goal for Heliodromus to become a full competitive concept with current systems.

Heliodromus is a constellation desgined to initially consist of 12 different satellites, though it is scalable. These satellites are divided over 2 different orbits. The orbit selection followed after a trade off between contact times of satellites and the ground station. Specific trade off characteristics for debris and drag resulted in a LEO height of 1400 km and the reflection satellites are put slightly below a GEO orbit. The location for the ground station was determined by evaluating clouds and aerosols densities, seismicity and political stability of the region. This resulted in the selection of two regions; Arizona - USA and Egypt - North Africa.

The first step in the chain to beam down energy from space is the conversion from solar light to electrical energy. From the concept trade-off, multi junction and thin film cells have been selected for further exploration. A trade-off on price, volume, mass and the TRL resulted in the choice of thin film solar cells for total conversion in space.

The choice for laser was already defined in the concept trade-off. The main reasons were: possible compatibility with already existing ground stations and a new fresh look besides already existing studies. Preliminary design characteristics for the laser system were given and the resulting requirements on pointing accuracy, size and cost were determined. The pointing accuracy was assumed not to be a bottleneck. The mirror sizes were set: for the two mirrors in the LEO satellite have resp. $12\ m$ and $19\ m$ diameter. The mirror in GEO has a $24\ m$ diameter. For each LEO satellite a mirror in GEO is required. Large bottlenecks are in the thermal management of the lasers, since large cooling system are required. Main problem of the laser system components are the low technology readiness levels. Large technological developments are required before the complete system is ready to deliver energy to Earth. These improvements are possible for the laser system.

Photovoltaics are also the most desired option for the conversion on Earth. Main reason for the selection of the photovoltaics is the possibility for optimization for specific wavelengths allows for higher efficiencies.

Heliodromus has also been evaluated as a total system. Budgets for mass, volume and power were made resulting in the following numbers. One single GEO satellite has a mass between $41 - 56$ tonnes and a volume of $833 - 1080$ m^3. While a LEO satellite would weigh $1210 - 1740$ tonnes and have a volume of $149\text{-}191 \cdot 10^3$ m^3. Volume is the limiting factor for launching with a Arianne 5 launcher which results in a total number of $11 - 14$ launches for the GEO system and 933-1194 launches for the LEO system.

An estimate of Heliodromus' total efficiency resulted in a value of 2%, which is not sufficient to compete with Earth based systems. Heliodromus' energy payback time is 6 years, which is within the minimum lifetime of 10 years. The total cost of Heliodromus is abount 98 billion dollars. At this stage, Heliodromus can not be competitive in price with Earth based fossil fuel power sources. When the all the thin film arrays used on Heliodromus is installed on Earth this system delivers 125% more energy than Heliodromus and can deliver energy up to a factor 10 less expensive than Heliodromus. Modern day renewable energy is sold at around 10-20 dollar cents compared to the 1$ per kWh which Heliodromus can deliver. In this price estimate the cost for the assembly in space is not even accounted for.

Serious bottlenecks arise when Heliodromus is implemented in the projected time frame. The lack of technological development of different technologies require large investments in the development of these technologies. Even currently available technologies have never been used on a scale required by Heliodromus. The major technological bottlenecks are the laser efficiency and consequently the thermal management. Moreover, advances in PV efficiency would significantly reduce the required mass and volumes for launch to space. In addition, assembly in space has never been done on the scale needed for Heliodromus and it opens a totally new field of research and development. The largest contribution in the cost for Heliodromus are the launches.

With improvements in launcher technology to reduce launch costs, solar cell efficiency to reduce loss and system size, laser efficiency to reduce losses and heat sink weight, combined with rising prices caused by fossil fuel reserve depletion: Heliodromus will become a realistic option for renewable energy in the future.

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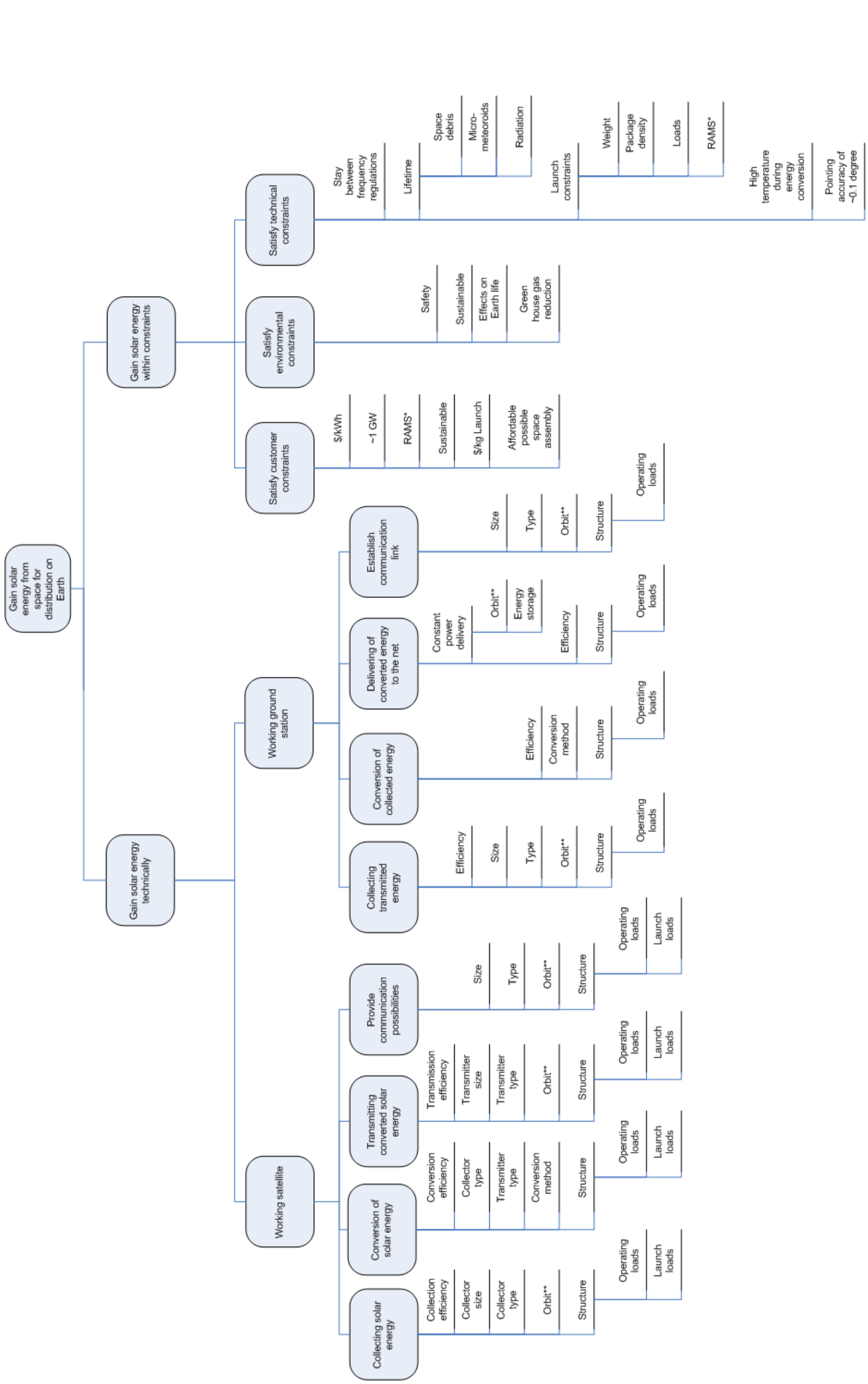
Appendix A

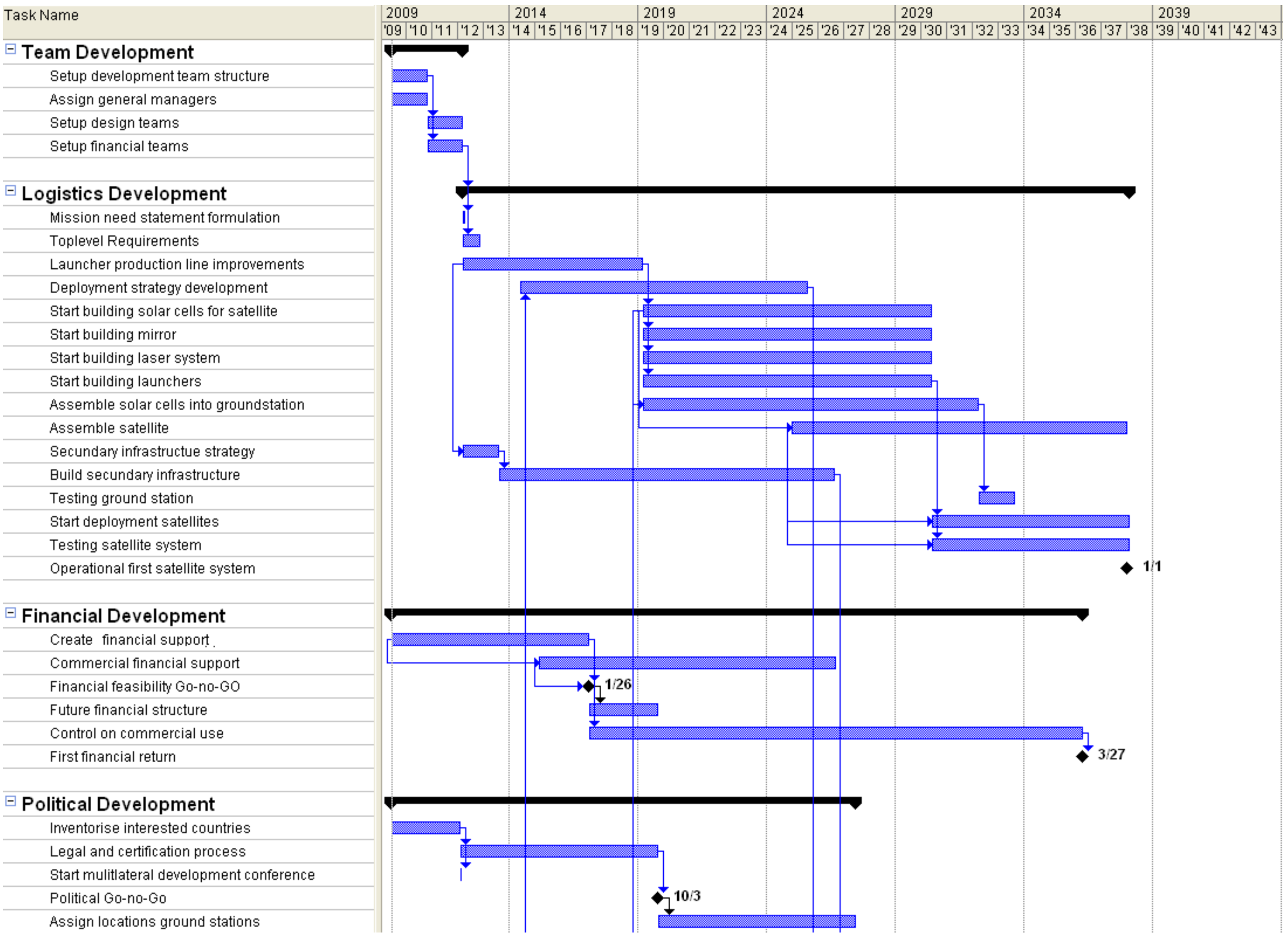
The Requirement Discovery Tree

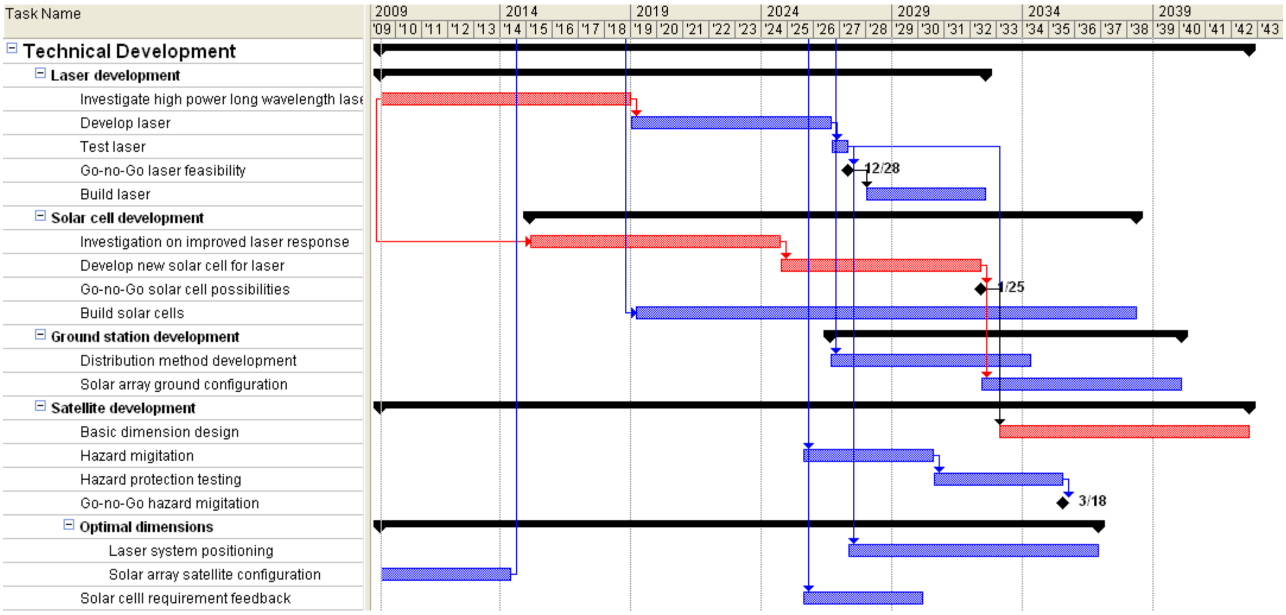


* Reliability, Availability, Maintainability and Safety
** Orbit control, Orbit determination, position and orientation

Appendix B

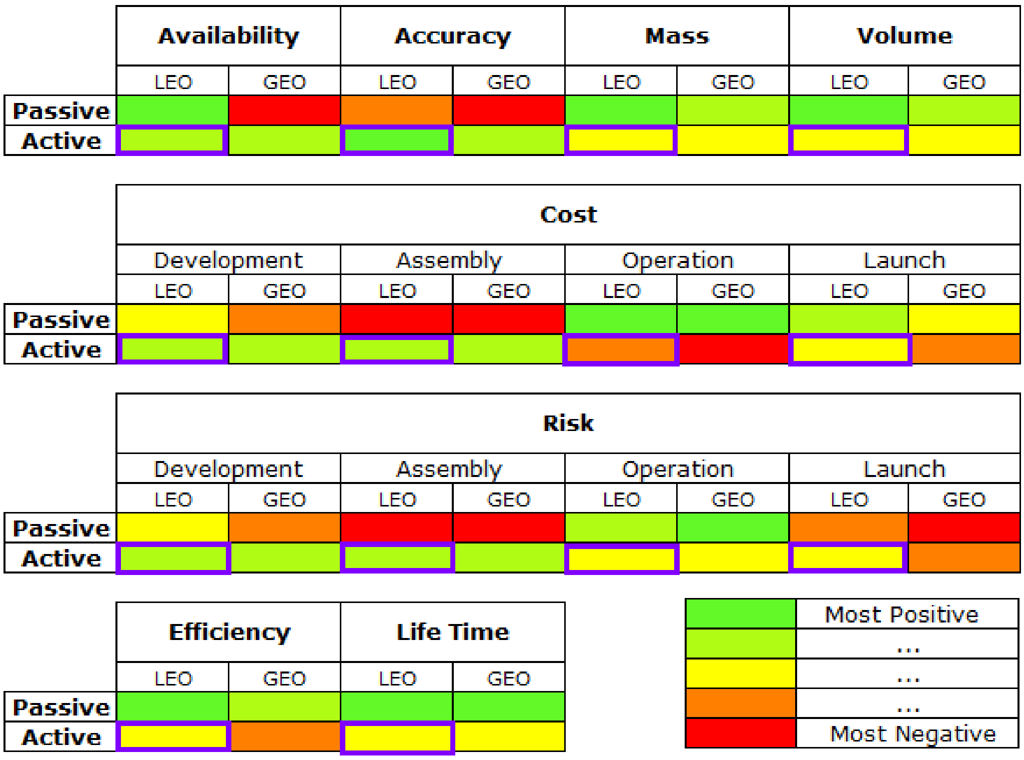
The Gantt Chart




Task Name
2009
2014
2019
2024
2029
2034
2039
'09 '10 '11 '12 '13 '14 '15 '16 '17 '18 '19 '20 '21 '22 '23 '24 '25 '26 '27 '28 '29 '30 '31 '32 '33 '34 '35 '36 '37 '38 '39 '40 '41 '42 '43
Technical Development
Laser development
Investigate high power long wavelength lase
Develop laser
Test laser
Go-no-Go laser feasibility
Build laser
Solar cell development
Investigation on improved laser response
Develop new solar cell for laser
Go-no-Go solar cell possibilities
Build solar cells
Ground station development
Distribution method development
Solar array ground configuration
Satellite development
Basic dimension design
Hazard migitation
Hazard protection testing
Go-no-Go hazard migitation
Optimal dimensions
Laser system positioning
Solar array satellite configuration
Solar celll requirement feedback
12/28
1/25
3/18

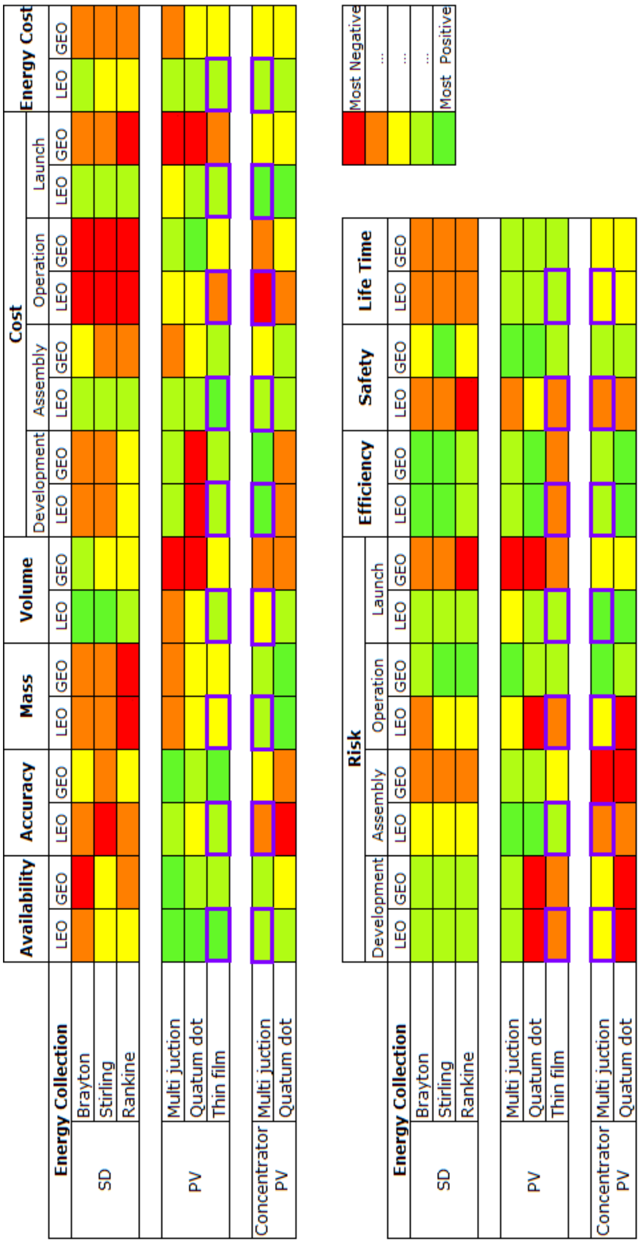
Appendix C

The Stabilization Trade-Off Matrix



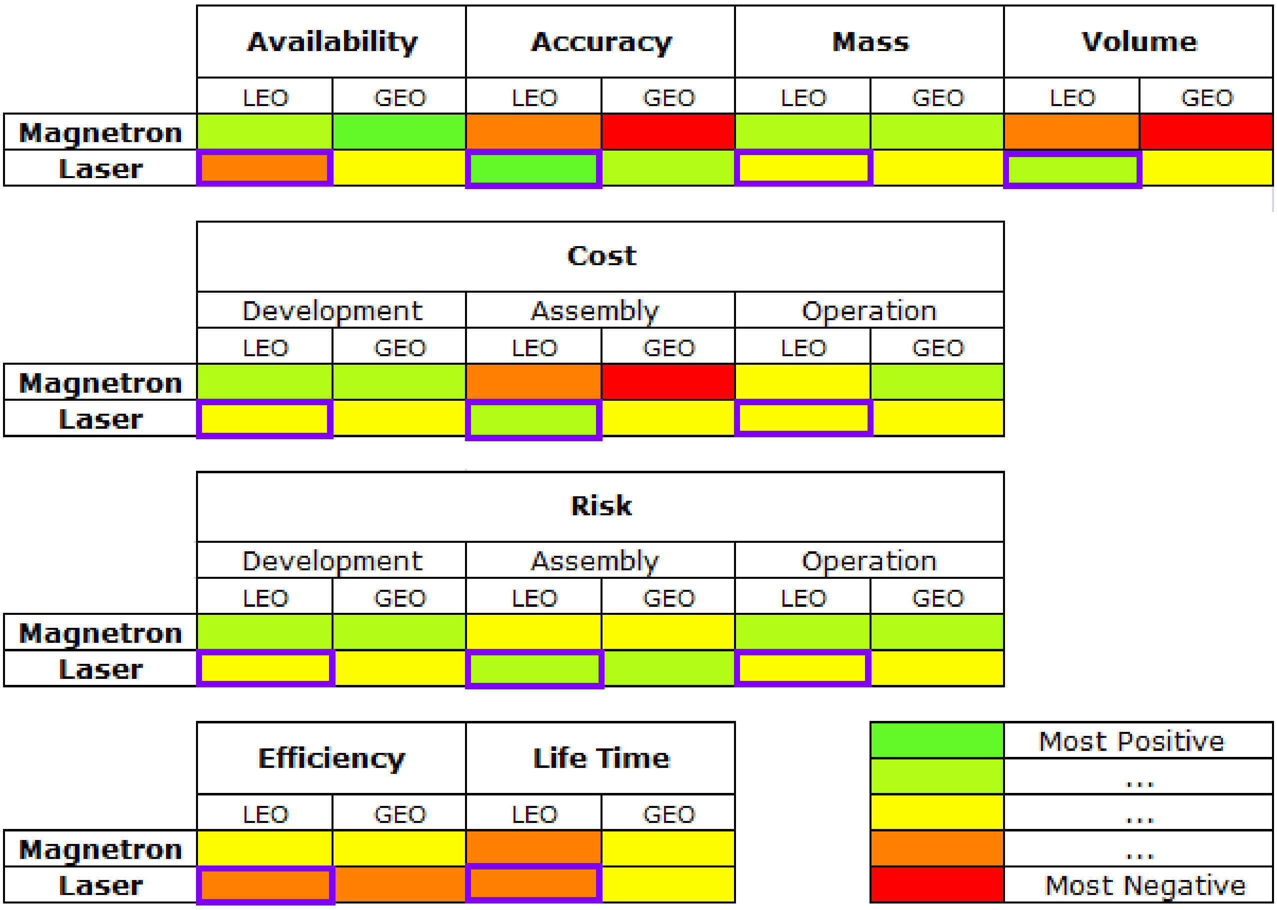
Appendix D

The Conversion Trade-Off Matrix



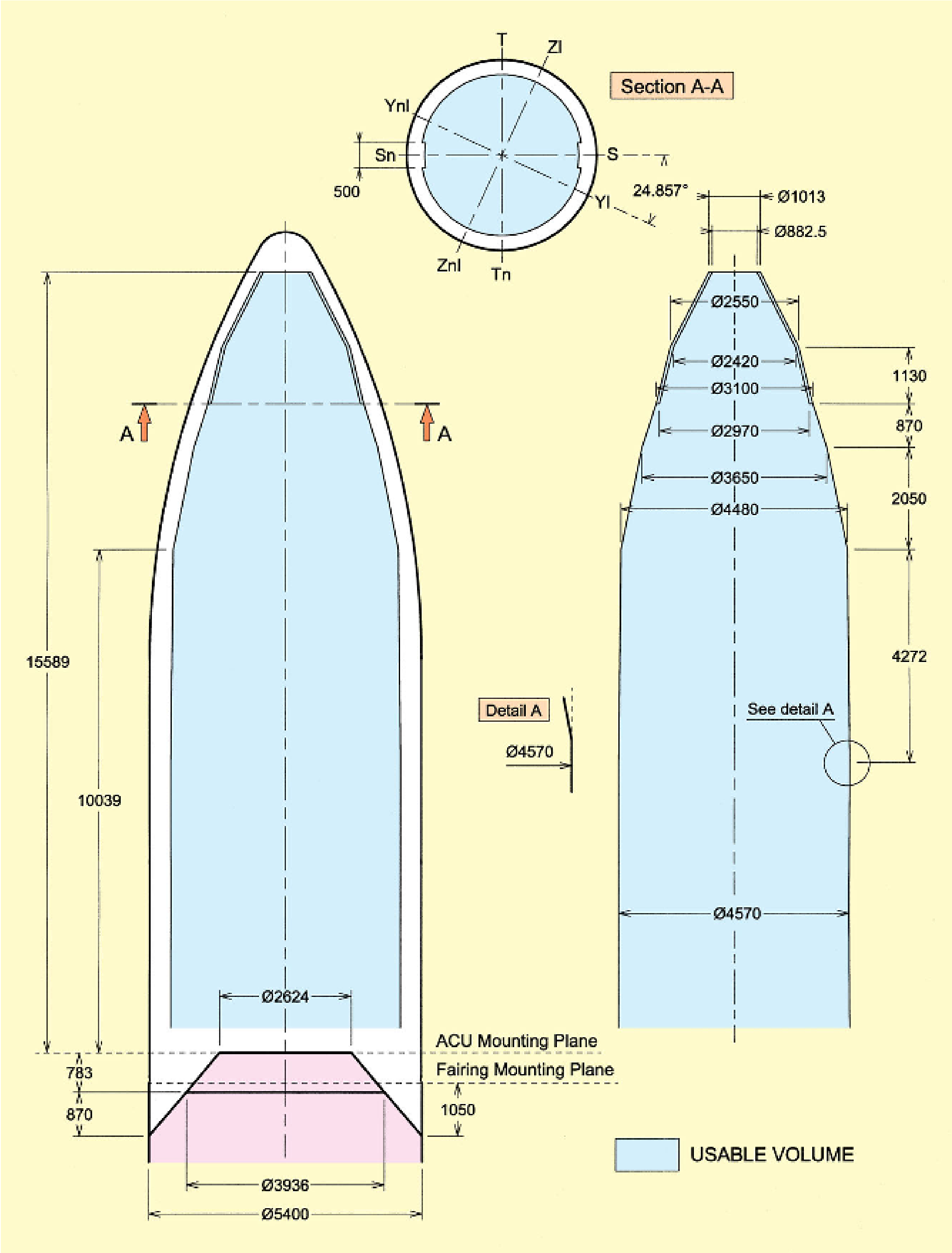
Appendix E

The Transmission Trade-Off Matrix



Appendix F

Ariane 5 Payload Volume



Usable volume beneath the payload fairing for the Ariane 5 launcher [41]

The payload space of the Ariane 5, known as the 'static volume', has been split up in six parts in order to calculate the total available payload volume. The lower, cylindrical part of the static volume has been split up in two parts with heights of 5428 mm and 4272 mm. Note that the lower dimension is not given and the 5428 mm height is an estimate. The top of the static volume has been split up in four parts with heights of 2050 mm, 870 mm, 1130 mm and 1500 mm, respectively from bottom to top.
The volumes of the first two parts are determined using the volumes of the corresponding cylinders. The volumes of the other four parts are determined using the volumes of the corresponding cone parts.

- Lower Cylinder:
$$5428\pi\left(\frac{4570}{2}\right)^2 \approx 8.90 \cdot 10^{10}[mm^3]$$
- Upper Cylinder:
$$4272\pi\left(\frac{4480}{2}\right)^2 \approx 6.73 \cdot 10^{10}[mm^3]$$
- Lower Cone:
$$\frac{1}{3}\pi\left(\left(\frac{4480}{2}\right)^2 \cdot 11065 - \left(\frac{3650}{2}\right)^2 \cdot 9015\right) \approx 2.67 \cdot 10^{10}[mm^3]$$
- 2^{nd} Cone:
$$\frac{1}{3}\pi\left(\left(\frac{3650}{2}\right)^2 \cdot 5774 - \left(\frac{3100}{2}\right)^2 \cdot 4904\right) \approx 7.80 \cdot 10^{9}[mm^3]$$
- 3^{rd} Cone:
$$\frac{1}{3}\pi\left(\left(\frac{2970}{2}\right)^2 \cdot 6102 - \left(\frac{2420}{2}\right)^2 \cdot 4972\right) \approx 6.47 \cdot 10^{9}[mm^3]$$
- Upper Cone:
$$\frac{1}{3}\pi\left(\left(\frac{2420}{2}\right)^2 \cdot 2361 - \left(\frac{882.5}{2}\right)^2 \cdot 861\right) \approx 3.44 \cdot 10^{9}[mm^3]$$

The total static volume is the sum of the volumes of all parts:

$$8.90 \cdot 10^{10} + 6.73 \cdot 10^{10} + 2.67 \cdot 10^{10} + 7.80 \cdot 10^{9} + 6.47 \cdot 10^{9} + 3.44 \cdot 10^{9} \approx 2.01 \cdot 10^{11}[mm^3] \approx \ 201[m^3]$$

The effective static volume will be smaller due to packaging, manufacturing tolerances, thermal protection installation and appendices. When assuming 20 % effective volume loss the available payload volume for an Ariane 5 launch is:

$$201 \cdot 80\% = 160[m^3]$$