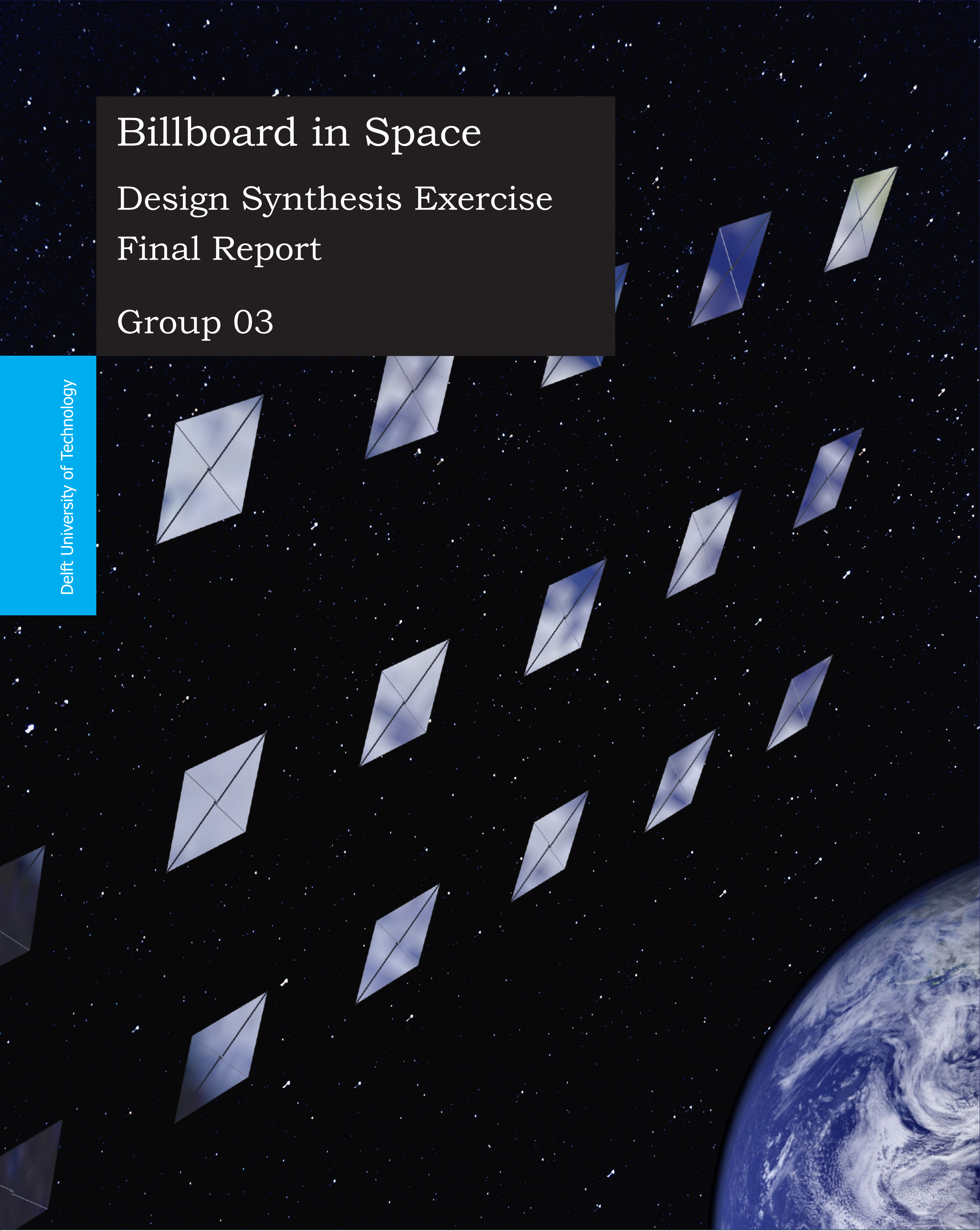
Billboard in Space

Design Synthesis Exercise
Final Report

Group 03



Delft University of Technology

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Challenge the future

Billboard in Space

Final report

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in partial fulfillment of the requirements for the degree of

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in Aerospace Engineering

at the Delft University of Technology,

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Delft University of Technology

Preface

A lot of money is invested in the advertisement industry to display advertisements through several different types of media such as television, posters and billboards. The goal of these advertisements is to reach as many people as possible. Placing a billboard in space (BBIS) provides an unique and optimal opportunity to reach a large target group at once.

A team of nine students took on the challenge to investigate the feasibility of such a billboard. This was done during the Design Synthesis Exercise (DSE), a project that is part of the curriculum of the Aerospace Engineering Bachelor at Delft University of Technology. A technical design of the spacecraft, as well as a mission proposal are part of the solution presented in this report.

This report is the last report from a series of four mandatory deliverables as part of the DSE. The first report was the Project Plan, which helped the team plan and execute the designing of the BBIS. Secondly, the requirements were ordered and several concepts were created, most of which were found instantly unfeasible. This process was documented in the Baseline Report. In the Midterm Report, the three best concepts were worked out in more detail and a trade-off between them was conducted. The winner of this trade-off was designed in detail in the final stage of the project, a process that is captured in this report. It contains the design of the spacecraft itself, as well as various other aspects that are part of the mission it is to perform.

Acknowledgements

The following people deserve to be thanked for their help during the process that is concluded in this report. Firstly, a huge thank you to Dr. ir. O. K. Bergsma for his support, guidance and input throughout the DSE. Team would also like to thank Ir. V. Stuber and ir. M. Rovira Navarro who helped out as coaches during the project. Intensive guidance for the Project Management and System Engineering aspect of the project was provided by Ms. S. Singh, who patiently answered every question the team had. Several professionals helped out during the technical design of the BBIS. Dr. ir. L. C. G. Rossi clarified a lot about observing light that is coming from space, for which the team would like to thank him. Furthermore, we would like to express our gratitude to Dr. ir. M. J. Heiligers, expert on the field of solar sails, for thinking along on the use of a solar sail in the design. Thanks should also go to ir. J. Breuer, who advised us on the design of our payload. Next, the astrodynamics and attitude determination and control department had a lot of valuable advice from Dr. S. Engelen, which was very much appreciated. Mr. M. S. Uludağ is thanked for his help on the communications and command and data handling subsystems. Last, but not least, the team would like to express its appreciation to R. Florea and N. von Storch, for their insights during the cost analysis. Without the help of all these people, this report would not have been possible.

Sincerely,

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Executive Overview

Different kinds of advertisements are seen all over the world. Traditional advertisement methods are well developed, but advertisers are keen to explore new options. The possibility of a billboard in space is investigated for this project. The billboard should have a comparable visibility to a full moon and should target the United States of America (USA). The following statements drive the design for the project.

Mission Need Statement

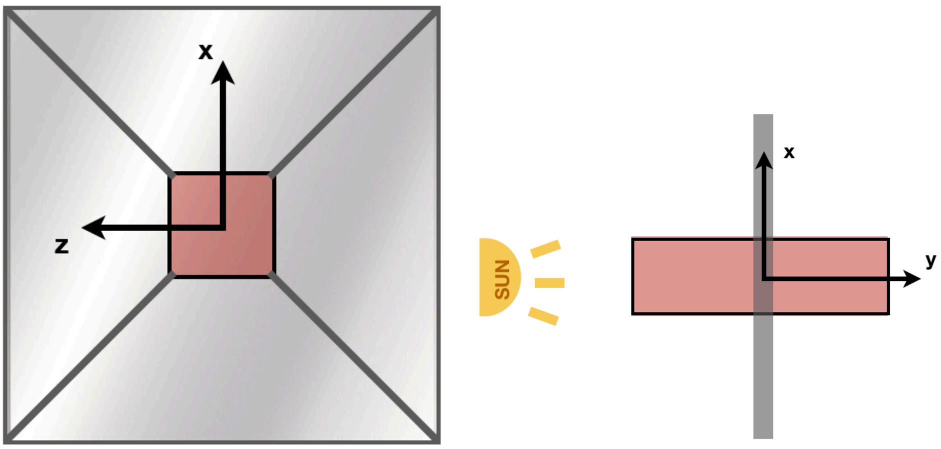
Explore new advertisement options by designing a billboard in space that has a visibility comparable to a full moon.

Project Objective Statement

The team consisting of 9 students has the objective to design, in 11 weeks, a billboard that orbits around the Earth.

Previously, a trade-off between the three main concepts was performed. These three concepts were a non-rigid structure with lights, a swarm with lights and a swarm with reflective surfaces. The most important criteria during this trade-off were the mass of the spacecraft, the visibility quality, the performance and the life span. During the trade-off it was determined that the swarm with reflective surfaces was the best option; mainly due to its low mass and low power compared to the two other concepts. This report contains the detailed design of the swarm with reflective surfaces.

In order to determine the location and distribution of the components placed inside and around the spacecraft, it is necessary to determine the coordinate system used in the upcoming explanations. The coordinate system in Figure 1 is right-handed and has its origin in the centre of gravity of the spacecraft. For a view of the open spacecraft see Figure 2. The spacecraft consists of a spacecraft bus ($0.34 \times 0.34 \times 0.66$ $meters$) and a solar sail of 404 m^2. The reflective side of the solar sail faces the Sun. In addition, solar cells are placed on the solar sail. Both of those components are elaborated on later.



(a) Front View, not to Scale. (b) Side View.

Figure 1: Body-Fixed Reference Frame, not to Scale.

The BBIS mission is divided in seven general phases: mission analysis and identification, feasibility analysis, design, development, qualification and production, operation and disposal. In the first phase, mission analysis and identification, the different tasks to be performed are identified and divided between the team members. In order to establish guidelines for the different design tasks, requirements

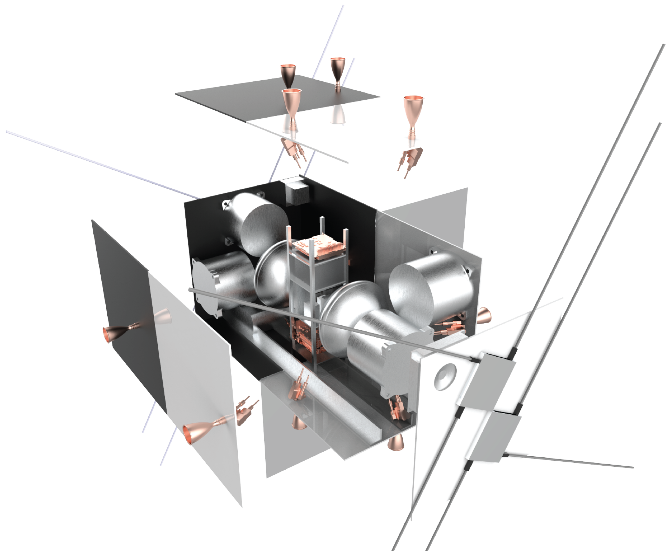


Figure 2: Exploded View of the Spacecraft.

are established, some of which are considered to be driving or killing. The killing requirement states that the spacecraft shall not light pollute other spacecraft. This requirement is reflected upon later.

The driving requirement when determining the orbit for 900 spacecraft is the flyover time. The requirement states that BBIS shall be visible for at least 14 seconds per flyover. The visibility is obtained by reflecting light from the Sun towards the Earth. The general orbit is designed in such a way that the visibility time for the USA is optimised. The contributions from disturbance forces are also taken into account in the orbit design. The main disturbance forces exerted on the BBIS are the solar radiation pressure, which is very high due to the enormous area of solar sail, the attraction force of Earth and of the Moon. These considerations lead to the decision of placing the swarm in a geosynchronous orbit.

In addition to the design of a general orbit, the main challenge in the astrodynamic analysis of the BBIS deals with the fact that the 900 spacecraft should follow different paths, in such a way that they do not collide with each other. This problem is solved by designing two types of formation flying. In the first formation, the spacecraft follow orbits with the same inclination but different longitude of ascending node. This formation is suitable for displaying logos. The second formation is based on the use of orbits with different inclination but similar ascending node (the ascending nodes cannot be coincident to prevent collision). This formation is suitable for displaying text. The design accounts for 6 different formations during the operational life time of the billboard. By adding these changes in formation to the effect of the disturbance forces, the ΔV budget in Table 1 is obtained.

	Required ΔV [m/s]	
	Per Year	**Total**
Non-Spherical Mass Distribution Earth	0.29	5.72
Third Body Interactions	10.2	204
Orbit Insertion Correction	-	3.99
End-of-Life	-	34
Changing formation	-	39.4
Total	10.5	287

Table 1: ΔV Overview.

The BBIS is visible as a circular area from the ground with a radius of $193\ km$. All spacecraft point their mirrors in such a way that they scan the entire USA. A rough estimation of the potential number of views can be observed in Figure 3.

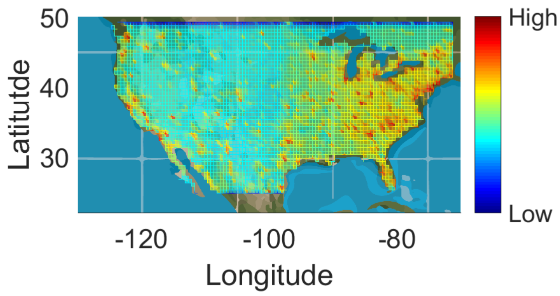


Figure 3: Potential View Distribution.

A solar sail is used to reflect the sunlight to the Earth. This sail consists of two layers of material, a $7.6\ \mu m$ layer of kapton and a $0.1\ \mu m$ layer of aluminium. The kapton ensures the rigidity of the sail during folding and deployment. The reflective aluminium film ensures the visibility of the spacecraft. Four booms are used to put the sail in tension. These booms are made of carbon-fibre reinforced struts. Load cases that apply to the booms are analysed and designed and it is made sure that the booms can withstand these. This results in booms with a length of $14\ m$ and a mass of $0.1\ kg/m$. There are three rings on each boom. A rope is connected to the spacecraft bus and to the tip of each boom. The rope is looped through three rings and the loops are connected to the solar sail. When the boom is deployed, both the rope and the solar sail are in tension. A deployed boom is seen in Figure 4. The booms are deployed from the spacecraft by an engine that rotates a base, around which the boom is folded.

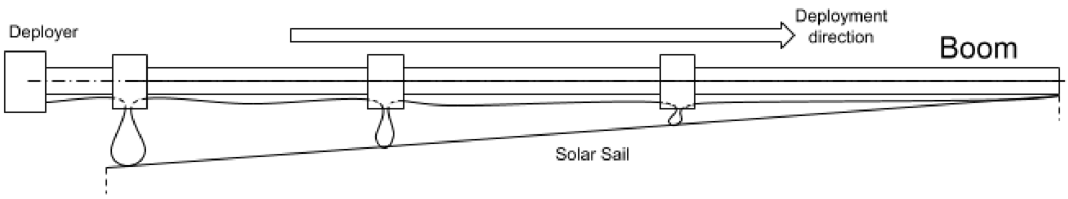


Figure 4: Demonstration of the Solar Sail Attachment on the Rope.

The reflective surface has a curvature of $0.0004m^{-1}$ to enable visibility on Earth. This curvature is forced into the sail by connecting the sail on different locations on the ring, see Figure 5. Starting on the top part at the rings closest to the base of the boom, and gradually connecting them lower on the rings at the end of the boom. The sail is then divided in four sections, all under a different angle, to form a curved surface.

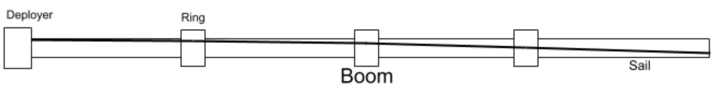


Figure 5: Method of Forcing Curvature.

To be able to switch the pixels off and on, reflectivity control devices are present on the sail. Those devices can control the reflectivity of part of the sail by switching from specular reflectivity to diffuse reflectivity. This concept has been first introduced by the IKAROS mission, which proved the feasibility of solar sails. The reflectivity control devices make the solar sail less reflective and ensure that it is less visible from Earth. Calculations are started by taking the illuminance of a full moon and dividing it by the number of spacecraft in the swarm. Resulting from this is the illuminance each solar sail should generate. A switched of pixel is not going to be completely invisible, the RCDs are not able to provide this decreased illuminance. The pixels that are switched off are significantly less bright than the pixels that are switched on. The reflectivity control devices are capable of decreasing the illuminance so that the pixel is as bright as Vega, the 5^{th} brightest star in the sky. The two visibilities are compared and due to the reflectivity control devices, the intensity of the light decreases to 0.3 of its original value. It was investigated how much area of these reflectivity control devices was needed to get the illuminance of the solar sail to be lower than the illuminance of Vega. Calculations on the reflectivity show that roughly $14\ m^2$ of the solar sail should be covered with the devices to permit this. This means the pixels that are switched off are still slightly visible, but are significantly less bright than the pixels that are switched on. Solar cells are also present on the sail to generate power.

To navigate the spacecraft and to keep the correct orientation, the attitude and orbit determination and control subsystem is present. This subsystem minimises the influence of disturbance torques on the spacecraft, such as gravity, solar pressure, the magnetic field and aerodynamic drag. Sensors determine the spacecraft's attitude and orbit. Each spacecraft has a Sun sensor, a star tracker, a global positioning system (GPS) receiver, a GPS-enhanced navigation system, an extended Kalmann filter and an inertial measurement unit on board. The signals of the sensors are merged to determine the attitude and orbit.

In case the attitude or orbit is not optimal, changes are applied. Actuators are on board of the spacecraft to make the desired changes in attitude or orbit possible. Four large and two small reaction wheels enable a change in attitude around every axis. Thrusters are used to desaturate these reaction wheels. The thrusters are used for both attitude and orbit change. In total, there are 12 thrusters present on the spacecraft bus. The propellants used to fire these engines are monomethyl hydrazine and dinitrogen tetroxide. The actuators of the spacecraft provide 3-axis stabilisation.

All data flows of the spacecraft should be regulated, which is done by the communications and the command and data handling subsystems. Two types of communication need to be supported by the communication subsystem. Uplink and downlink communication is between the spacecraft and the ground station and crosslink communication is between the spacecraft in the swarm. The latter communication is started at one spacecraft, sending a signal that all other spacecraft receive. After that, the next spacecraft does the same and this action is repeated sequentially by all the spacecraft in the swarm. A $10\ s$ delay is encoded in the communication protocol. This means the next spacecraft starts communicating even if the previous spacecraft malfunctions. Communication between individual spacecraft is crucial in formation flying. It is imperative for the spacecraft to know the precise position of each other in order to prevent collision.

The command and data handling subsystem consists of on-board computers and data busses. The former handles all commands that need to be processed internally. For redundancy, three on-board computers are used, which are all equipped with a data storage possibility. The data busses connect all the subsystems to the on-board computers, and are therefore present throughout the complete spacecraft bus.

To provide the other subsystems with power, the electrical power system is tasked with the generation of sufficient power. This is done by thin-film solar cells that are present on the solar sail, as mentioned earlier. The required area of the solar cells is obtained by taking degradation of the components and an extra safety factor into account. This analysis results in a solar cell area of $1.69\ m^2$.

The spacecraft encounters an eclipse during its orbit, therefore, batteries are needed to provide the spacecraft with power when the solar cells cannot generate any. These batteries are sized taking peak power into account, as they have enough time to fully recharge during every orbit. A total of $42.76\ Whr$ needs to be provided, and this is done by two batteries, one of $30\ Whr$ and one of $20\ Whr$. As usual in spacecraft, a backup battery is present. In case of the BBIS, this battery enables the spacecraft to function for half an hour after power generation stopped. The back-up battery has a capacity of $20\ Whr$.

The distribution and regulation of the electrical power is done by a board that is connected to the batteries. Peak power trackers enable maximum power extraction from the solar cells. To make sure the voltage and current coming from the data bus going to the subsystems is correct, distributors are present between all the components.

In order to ensure correct functioning of all the subsystems, the environment in which the spacecraft operates is analysed. Radiation presents a problem for all electronic components. Therefore, aluminium boxes are placed around all those components in order to keep them operational for $20\ years$.

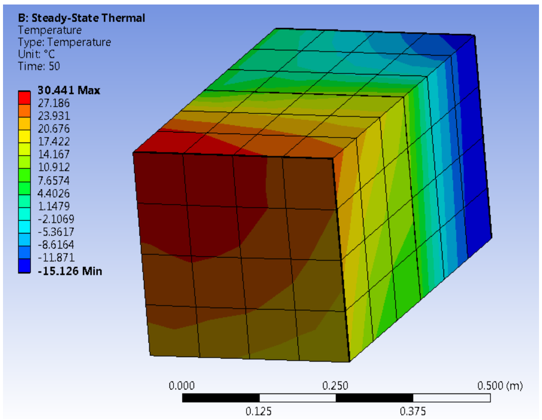


Figure 6: Thermal Model of the Bus with Sail Hidden.

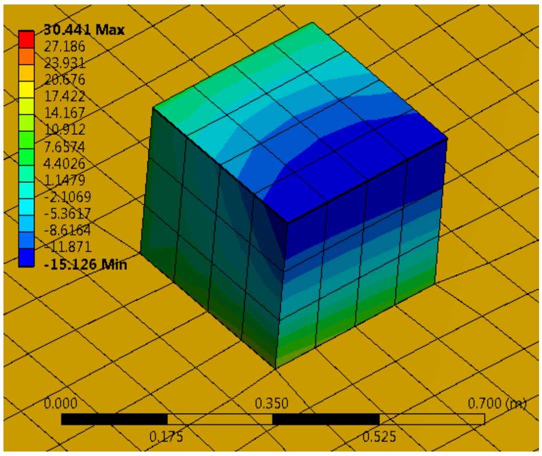


Figure 7: Thermal Model of the Bus with Sail.

Moreover, the spacecraft is covered with highly conductive paint in order to protect it from charging. In addition, the paint changes the emissivity and absorptivity of the bus in order to achieve favourable temperatures throughout the bus. See Figure 6 and 7 for the thermal models of the spacecraft.

The structures and mechanisms are formed by the primary and the secondary structure. The primary is the main load-carrying member during the critical flight conditions, while the secondary deals with the deployment and attachment of all the spacecraft components. The secondary structure of each BBIS spacecraft is formed by a 3 unit ($1\ unit\ =\ 10\ cm$) CubeSat structure and the deployment mechanisms for the solar sail. The CubeSat structure is used to make placing of the components easy and efficient. In addition, a 10% increase in structural weight is assumed to account for bolts and other joints.

The spacecraft is designed considering the worst loading conditions. The BBIS experiences those conditions during the first stage of the launch. The loads exerted on the structure during this phase are $6g$ in x-direction and $1g$ in y- and z-direction. The material chosen for the spacecraft bus and reinforcements is Aluminium 7075-T73, mainly due to its high specific strength. In addition to the aforementioned CubeSat structure, the bottom plate of the bus is reinforced with three stiffeners of $4\ mm$ in thickness, along its entire length.

The stresses created by the bending moment and the compressive forces are maximum at the points at which the structure is attached to the launcher adapter, which means that the bottom and top plate are critical for this spacecraft. The bottom plate is subjected to compression with buckling as main failure mode. The top plate is subjected to tension, and the maximum stress on it is therfore compared to the material's ultimate and yield strength. A more detailed analysis shows that the structure resists the buckling with a safety margin of 2.3, and the yield and failure with margins of 1.23 and 1.45, respectively. Adding the bus, stiffeners, CubeSat structure and bolts weight, the structures and mechanisms have a total mass of $14.13\ kg$.

In Table 2, the mass and power budget of one BBIS spacecraft can be seen. The used power during all calculations is $15\ \%$ higher, namely $37.32\ W$, to make sure enough power is available during operations.

The launcher chosen for this project is the Ariane V. Multiple options were considered, but a cost and reliability optimisation led to the conclusion that the Ariane V is the most suitable choice for the BBIS. In order to launch the 900 BBIS spacecraft, 9 different launches are needed, leading to a total cost of $1.6\ bln\ USD$. The chosen launch side is the Guiana Space Centre, in Kourou, French Guiana. This location is optimal due to its proximity to the equator, which gives access to numerous target orbits, and its proximity to the coast, which reduces the possibility of launch debris hitting populated areas.

The 100 spacecraft placed inside of each launch vehicle are attached to the adapter using a beam structure. When the final stage of the launcher is attained, each spacecraft deploys its antenna to enable crosslink communication and prevent collision during deployment. Each spacecraft separates from the beam structure with the help of explosive charges. When a safe distance is obtained, validated by comparing GPS signals, the propulsion system of each spacecraft is activated and the spacecraft

Subsystem	Mass [kg]	Power [W]
Payload	17.56	0
Attitude & Orbit Determination Subsystem	18.41	23.45
Communications	0.54	6
Command & Data Handling	4.96	3
Electric Power Subsystem	1.35	0
Structures	13.71	0
Total	56.49	32.45

Table 2: Mass and Power Budget of BBIS.

are brought to their final orbit. The total time to get all the spacecraft in the correct formation in geosynchronous orbit is approximately $90\ hours$.

Planned disposal is performed at the end-of-life of the BBIS. Due to the high altitude of the orbit, re-entry is not a feasible option for the disposal of the spacecraft. Therefore, a manoeuvre to a graveyard orbit should be performed, which is started by the propulsion subsystem, burning the remaining fuel on board, and then finished by the solar sail. This ensures that the disposal method used is as sustainable as possible. The target graveyard orbit for this mission is located $12\ 280\ km$ above geosynchronous orbit. The solar pressure acting on the solar sail is periodical, therefore, the BBIS remains in the graveyard orbit and does not sail away.

Unplanned disposal is performed when a subsystem failure requires one of the spacecraft to abandon the formation. Different failure modes and their corresponding disposal methods are studied during the design process. Some subsystems and failure modes are more critical than others, and the disposal procedure therefore varies per event.

Once the first phase of the project is over, the design and development stages start. This strategy elaborates on the feasibility of the project and the design and specifies the qualification needed for the production of the BBIS. Lastly, it elaborates on the launch, operation and disposal of the spacecraft. It is expected that the first four phases will take $25\ years$ and that the BBIS will be launched in 2043. A manufacturing, assembly and integration plan is generated to provide an overview of the activities needed to construct the spacecraft. The manufacturing phase covers acquisition of all the COTS components, the manufacture of unique BBIS components and the launcher. During this phase, the components are tested and qualified. The assembly phase contains the assembling of the components into the subsystems. These subsystems are then tested and integrated into the spacecraft. In the integration phase, the spacecraft are integrated into the launcher after which final checks are performed.

A collision protection system is proposed as a way to make sure the BBIS is not critically damaged by space debris. It uses the catalogue of the Space Surveillance Network to predict the orbits of space debris. Extra propellant is taken on board to perform space debris avoidance manoeuvres.

Risk assessment for the BBIS is required in order to identify the possible threats for the mission and develop corresponding mitigation strategies. The identified risks can be divided into launch, deployment and disposal, subsystem failure, external factors and unknown risk. The risks are organised according to their consequence and likelihood. The risks have been mitigated to such an extent that there are no high-risk threats to the mission left after mitigation.

The sustainability strategy aims to optimise results, while minimising the negative effect the project might have on society, economy and environment. The life-cycle analysis proposes certain requirements to be followed, mainly during the design and selection process for the launcher, electric power subsystem and propulsion subsystem. Moreover, when dealing with disposal of the BBIS, sustainability is a great concern. As was previously mentioned, the disposal of the BBIS is performed in the most sustainable way possible, ensuring no fuel is left on board at the end-of-life. Green fuels were considered, but not chosen for the propulsion subsystem. Green propulsion would result in a mass increase, which leads to the use of more launchers to deliver the swarm, which is, in fact, less sustainable than using non-green propellants. Social sustainability is crucial when analysing the ethics of the project. Despite the fact that the ultimate goal of the advertisement industry is to introduce an idea in the mind of the viewers, BBIS aims to have the smallest intrusive effect possible. This is done by by moving the projected light every $14\ seconds$ on a different part of the USA. In addition, it is possible to switch the reflective surface on and off when necessary.

In order to analyse the BBIS's position on the market, two driving requirements were established. The first requirement states that the project shall have a return on investment of at least 0%. The return on investment depends on both the mission cost and the revenues. A cost analysis is performed, breaking down the total budget into costs for the launch, spacecraft design, development, test and evaluation and mission operational phase. For this analysis, a combination of commercial off the shelf component costs and typical system costs obtained for similar spacecraft missions are used. Also, the influence of mass production and inflation are is taken into account for the spacecraft fleet configuration costs. Ultimately, the total mission cost is estimated to be $15.0\ bln\ USD$. To estimate the mission revenues, the advertisement market is analysed. In total, it is estimated that BBIS will have an income equal to $19.4\ bln\ USD$. Thus, the total return on investment is 29%.

The second driving requirement states that the billboard shall be visible from the USA for 905 hours per year under ideal weather conditions. This requirement is related to the the minimal viewing time such that a substantial return on investment is yielded. Both requirements are met and it can be concluded that the BBIS is financially feasible.

The verification of the system is mainly done by verifying the requirements per subsystem. Validation for the complete mission is impossible, as there has never been a comparable mission. For this reason, a plan is constructed to perform validation. This plan consists of several steps. First, a pioneer spacecraft is launched to space in order to validate all the subsystems. Second, an experimental spacecraft with sensors is launched in an orbit between the pioneer spacecraft and Earth. This spacecraft can determine whether the BBIS spacecraft causes light pollution to other spacecraft.The next step in the validation would be formation flying. Three more spacecraft are launched into the orbit and the visibility and formation flying of the spacecraft is checked.

Finally, recommendations to improve the design are given. These recommendations are all related to specific validation of certain concepts. Firstly, it is advised to reconsider the total mission operation time equal to $20\ years$. Here, the predominant constraining factor is the limited lifetime of many commercial off the shell components. Few components have performed for such a long lifetime, and due to the damaging environment, the risk of failure before end-of-life is very high. An extensive validation and testing of the components is advised before launching BBIS. Secondly, it is recommended to further research the concept of formation flying. Previously, formation flying has never been performed with such a large fleet. Realising the mission requires more research and experiments. Aspects such as the launch and deployment of the configuration, the spacecraft inter-communication and the precision of attitude and orbit determination and control are critical formation flying characteristics that need to be further analysed. Finally, one last mission characteristic need to be reconsidered. The mission requirement stating that the BBIS shall not light pollute other spacecraft orbiting around Earth is not verified. Initially, this requirement was categorised to be 'killing', implying that BBIS is not feasible if it does not meet this prerequisite. However, it is unknown what the consequences of this violation are. Validation of the light polluting effects on other spacecraft orbiting Earth is required before determining whether this requirement is indeed killing.

Nomenclature

Abbreviations

$A\&ODCS$	Attitude and Orbit Determination and Control System
Ads	Advertisers
AU	Astronomical Unit
$BBIS$	BillBoard in Space
BER	Bit Error Rate
$C\&DH$	Command and Data Handling
CDR	Critical Design Review
$CFRP$	Carbon-Fibre Reinforced Polymer
$CIGS$	Copper Indium Gallium Selenide
$COTS$	Commercial Off-The-Shelf
DSE	Design Synthesis Exercise
ECC	Error-Correcting Code
EoL	End of Life
EPS	Electrical Power System
FEM	Finite Element Model
FY	Financial Year
GBL	γ-Butyrolactone
GEO	Geosynchronous Equatorial Orbit
$GNSS$	Global Navigation Satellites System
Gov	Government
GPS	Global Positioning Systems
GSO	Geosynchronous Orbit
H/W	Hardware
$IA\&T$	Integration Assembly & Testing
IMU	Inertial Measurement Units
ISS	International Space Station
$ITAR$	International Traffic in Arms Regulations
Lan	Launcher Company
LEO	Low Earth Orbit
MAI	Manufacturing, Assembly and Integration
Man	Manufacturers
MMH	Monomethyl hydrazine
$MPPT$	Maximum Peak Power Tracker
$NASA$	National Aeronautics and Space Administration
OBC	On-Board Computer
$PD\&D$	Product Design and Development
PDR	Preliminary Design Review
PM/SE	Project Management & Systems Engineering
Pub	Public
$QPSK$	Quadrature Phase-Shift Keying
RCD	Reflectivity Control Device
RTG	Radioisotope Thermoelectric Generator
s/c	Spacecraft
S/W	Software
SA	Space Agencies
SPC	Specialty Polymer Coatings
SRB	Solid Rocket Booster
SUp	Suppliers
$SWRS$	Swarm with Reflective Surfaces
$TFSC$	Thin-Film Solar Cells
$TU\ Delft$	Delft University of Technology
Tud	Delft University of Technology
UHF	Ultra High Frequency
USA	United States of America
USD	United States Dollar
UV	Ultraviolet
VHF	Very High Frequency

Greek symbols

δ_{max}	Maximum deflection at the tip of the boom $[m]$
η	Efficiency [-]
μ	Gravitational parameter of Earth $[m^3s^2]$
Ω	Longitude of Ascending Node $[deg]$
ω	Argument of Periapsis $[deg]$
ρ	Density $[kg/m^3]$
σ	Stress $[N/m^2]$
τ	Torque $[Nm]$

Roman symbols

F	Force vector in x-y-z $[N]$
P	Position vector [-]
Sun	Position vector of the Sun in x-y-z $[m]$
$\mathbf{S_p}$	Vector perpendicular to solar sail [-]
A	Surface area $[m^2]$
a	Semi-major axis $[km]$
A_{cross}	Cross-sectional area of the solar sail $[m^2]$
ac	Angle change influence [-]
B	Magnetic field strength $[nT]$
b	Separation between stiffeners $[m]$
c	Speed of light $[m/s]$
C_d	Drag coefficient $[-]$
C_n	n^{th} spacecraft cost $[USD]$
C_R	Solar radiation coefficient $[-]$
C_{pa}	Centre of aerodynamics pressure $[m]$
$cp_{s_{ss}}$	Solar pressure centre $[mm]$
C_{tot_n}	Total cost of n spacecraft $[USD]$
cg	Position of centre of gravity $[mm]$
D	Residual dipole $[Am^2]$
E	Modulus of elasticity of the material $[GPa]$
e	Eccentricity [%]
E_{sp}	Specific energy $[J/kg]$
F	Force $[N]$
f	Frequency $[Hz]$
G	Gain factor
g	Gravitational constant $[m/s^2]$
h	Altitude $[m]$
I	Area moment of inertia $[m^4]$
i	Inclination $[deg]$
I_f	Solar irradiance $[W/m^2]$
I_{sp}	Specific impulse [s]
J_2	Scaling coefficients of the gravity field [-]
l	Length $[m]$
L_d	Solar panel degradation [-]

l_D Desired longitude $[deg]$
l_s Nearest stable longitude $[deg]$
M Bending moment $[N \cdot m^2]$
m Mass $[kg]$
N Number of orbits per day [-]
n Mean motion $[deg/day]$
N_0 Number of bits $[-]$
O/F Oxidiser-fuel ratio
P Power $[W]$
q_{ss} Reflectant factor of solar sail [-]
R Data rate $[bit/s]$
r Radius [m]
r_{cov} Coverage radius $[km]$
R_{Earth} Earth radius $[km]$
R_{Moon} Moon radius $[km]$
T Temperature $[K]$
t Time $[s]$
T_a Aerodynamic drag $[N]$
T_g Gravity-gradient torque $[Nm]$
t_h Thickness [m]
T_m Magnetic torque $[Nm]$
T_s Solar torque $[Nm]$
V Velocity $[m/s]$
v Poisson's ratio [-]
V_{fuel} Fuel volume $[m^3]$
$V_{oxidiser}$ Oxidiser volume $[m^3]$
V_{sphere} Volume of a sphere $[m^3]$
w Width $[m]$
x Distance $[m]$

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Contents

1. Introduction

Different kinds of advertisements are seen all over the world. Traditional advertisement methods are well-developed, but advertisers are also keen to explore new options. This project explores the feasibility of a Billboard in Space (BBIS) that has a comparable visibility to a full moon. The main target for the BBIS is the United Stated of America (USA). A mission need statement and a project objective statement were specified by the team in [2], these statements read as follows.

Mission Need Statement

Explore new advertisement options by designing a billboard in space that has a visibility comparable to a full moon.

Project Objective Statement

The team consisting of 9 students has the objective to design, in 11 weeks, a billboard that orbits around the Earth.

After the project familiarisation phase of the Design Synthesis Exercise, three concepts were analysed in [3]; a non-rigid structure with lights, a swarm with lights, and a swarm with reflective surfaces. A trade-off was made with criteria in five themes: mass budget, customer's interest, technical aspects, risk and sustainability. The four most important trade-off criteria were the spacecraft's mass, the visibility quality, the performance and the life span. The trade-off concluded that the swarm with reflective surfaces was the most promising concept, mainly because of the low mass and the low required power compared to the other designs. This report will further analyse the swarm with reflective surface and answer the question if the project is technically and economically feasible. In addition, it explains to the orbit of the swarm, the physical design of the spacecraft, the structural concept for deployment and the end-of-life procedure.

In this report, the general layout of the spacecraft and the reference frame are described in Chapter 2 and all requirements following from [2] are discussed in Chapter 4. Then, the organisation of the project is discussed in Chapter 3. Next, the formation flying is analysed in Chapter 5. All subsystems - payload, position and attitude control, data management, electrical power system, environment, and structures and mechanisms - are analysed in Chapter 6, 7, 8, 9, 10, 11, respectively. Then, the launch and disposal plans are discussed in Chapter 12. Next, operations and logistics are discussed in Chapter 13. Furthermore, the risk assessment and sustainability analysis are elaborated on in Chapter 14 and 15, respectively. Then, the revenue and cost of the project are analysed in Chapter 16. Next, verification and validation is performed in Chapter 17 and 18, respectively. Finally, the report is concluded in Chapter 19.

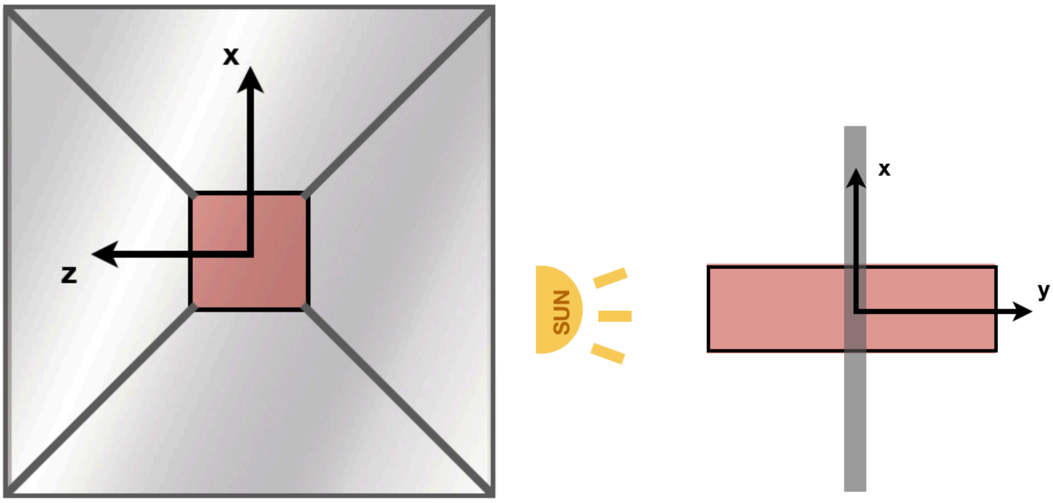
I
Mission Description

2. Spacecraft Description

This chapter elaborates on configuration of the spacecraft. Section 2.1 introduces the reference frame used throughout this report. This is followed by Section 2.2, which describes the spacecraft and swarm configuration. The last section of this chapter, Section 2.3 gives an overview of the mass and power budget.

2.1. Reference Frame

A right-handed body-fixed reference frame is the main reference frame used throughout the report. If another reference frame is used, it is clearly stated. The positive y-axis points to the side where the reflective surface does not reflect the sunlight, which is the dark side of the spacecraft. The reference frame is shown in Figure 2.1.



(a) Front View, not to Scale.

(b) Top View, not to Scale.

Figure 2.1: Body-Fixed Reference Frame.

2.2. Configuration

BBIS consists of 900 satellites which are orbiting the Earth in a swarm. The spacecraft are not connected to each other, but communicate to make sure they are correctly positioned. The spacecraft all have a solar sail attached to the bus. The solar sail functions as a pixel in a big screen and can be pointed by rotating the spacecraft itself. Also attached to the sail are the solar cells. The antennas are attached to the spacecraft bus. In Figure 2.2, a closed render of the spacecraft can be seen, while Figure 2.4 shows the exploded view. For a view of the total spacecraft with the solar sail, see Figure 2.3.

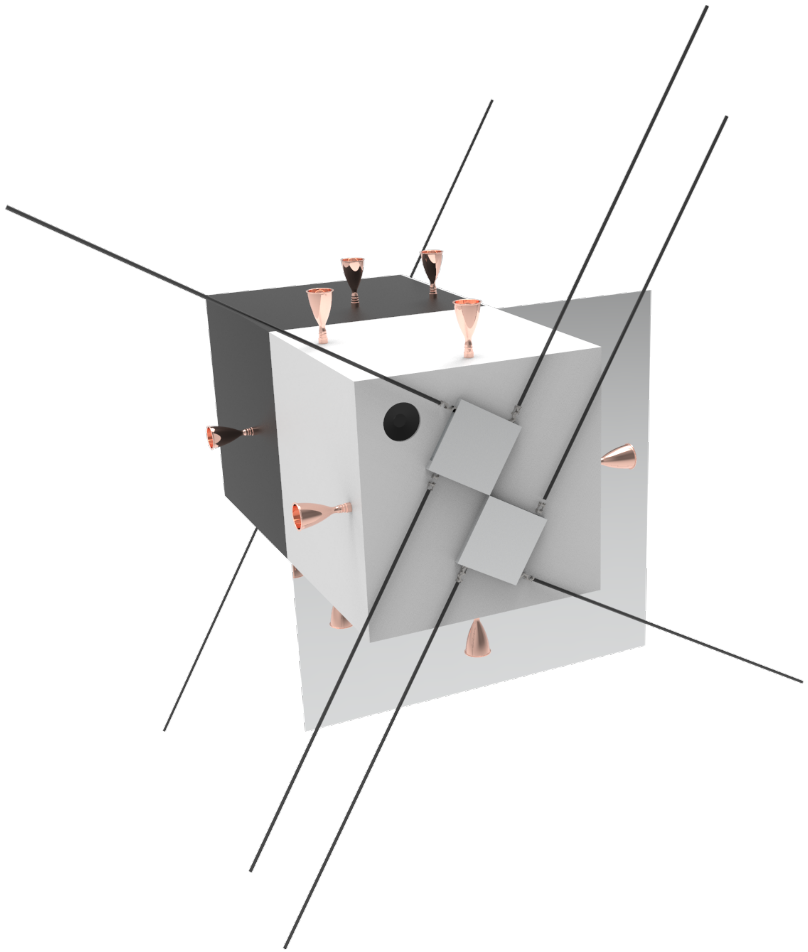


Figure 2.2: External View of One Spacecraft With a Partly Hidden Solar Sail.

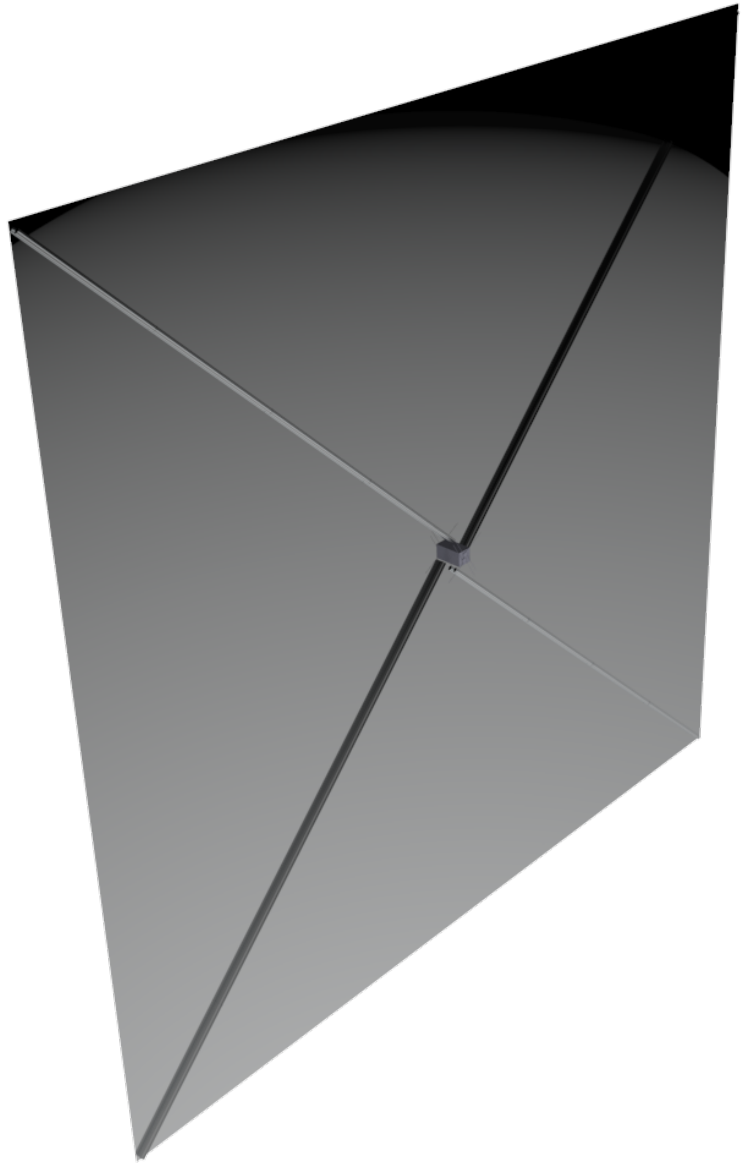


Figure 2.3: External View of One Spacecraft.

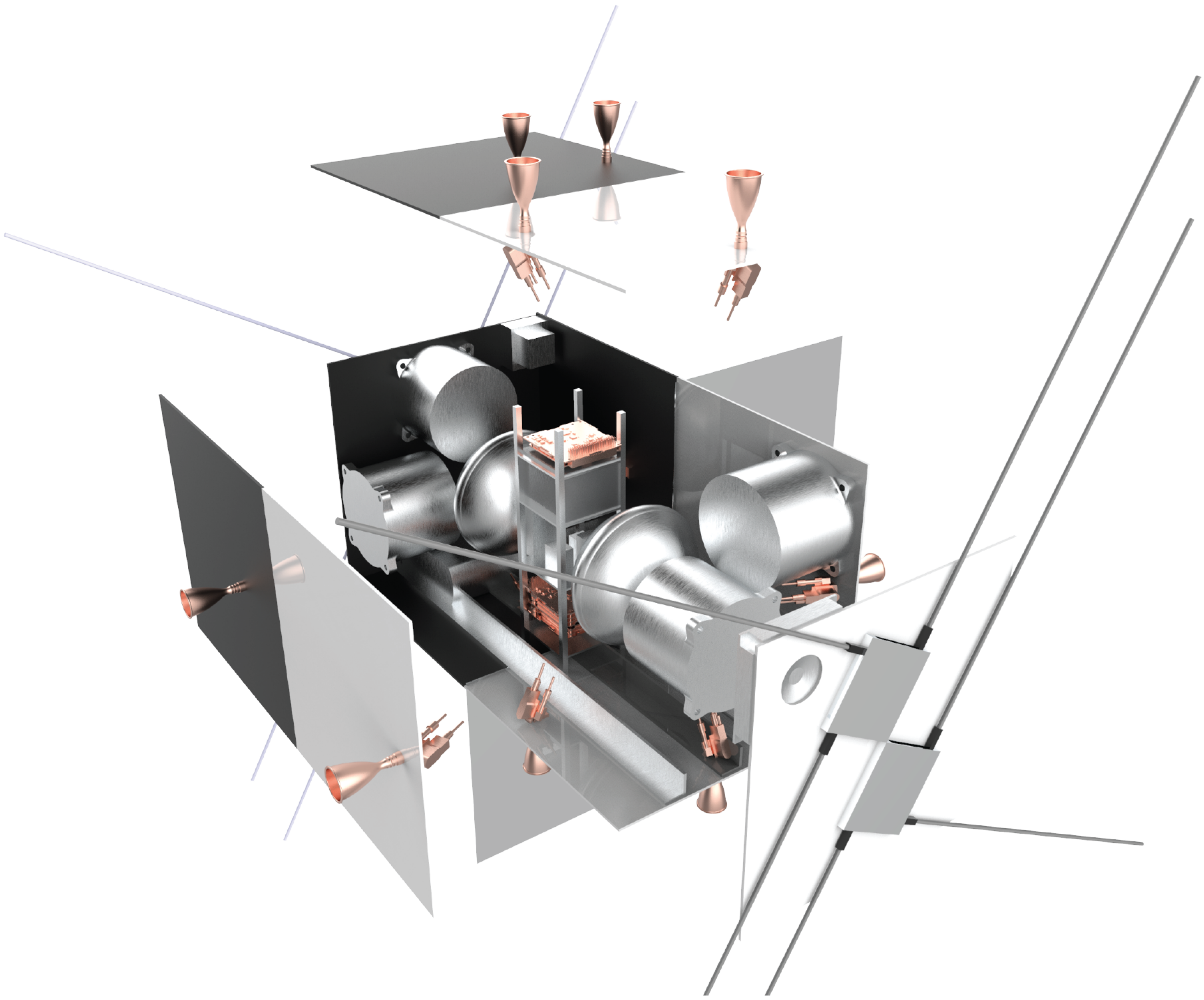


Figure 2.4: Exploded Render of the Spacecraft.

Technical drawings of the inner arrangement of the spacecraft bus are shown in appendix A (Figure A.1, A.2). The overall view of the whole spacecraft is shown in Figure A.3 also found in appendix A.

The spacecraft bus dimensions and different component sizes and geometry can be found in the corresponding subsystem chapters. For the iteration of each subsystem, an initial wet mass of $35\ kg$, and a dry mass of $30\ kg$ are used.

2.3. Resource Allocation

In this section, the mass and power budget of the spacecraft are described. The former is detailed in Table 2.1 and considers all subsystems and components. The row *Protections* refers to the total mass of the aluminium boxes that are placed around all electronic components, to protect these components from radiation. The latter, the power budget, is displayed in Table 2.2. An elaboration on the power budget and the 15% margin is given in Section 9.2.

Subsystem	Component	Mass per unit $[kg]$	Quantity $[-]$	Total Mass $[kg]$
Structures	3U CubeSat Structure	0.30	1	0.30
	Bus	-	-	11.06
	Stiffeners	0.41	3	1.24
	Bolts and Other Joints	-	-	1.11
Propulsion	Thrusters	0.35	12	4.20
	Oxidiser	-	-	2.45
	Fuel	-	-	1.48
	Tank	0.75	2	1.50
	Valves			0.37
Payload	Solar Sail	-	-	6.64
	Booms	1.44	4	5.74
	Deployment Mechanism	1.44	4	5.17
A&ODCS	GPS	0.72	1	0.72
	Star Sensor	0.33	1	0.33
	Sun Sensor	0.28	1	0.28
	Inertial Measurement Unit	0.02	1	0.02
	Reaction Wheel (big)	1.75	4	7.00
	Reaction Wheel (small)	0.03	2	0.05
C&DH	Processing Unit	0.01	3	0.02
	Wiring	-	-	0.002
Communications	Antenna	0.18	3	0.54
EPS	Back-up Battery	0.34	1	0.34
	Storage Battery	0.78	1	0.78
	Wiring	-	-	0.12
	Solar Cells	-	-	0.11
Protections				4.91
Total Mass				56.49

Table 2.1: Mass Budget.

Subsystem	Component	Quantity $[-]$	Average Power Required $[W]$
Communication	Antenna	3	6.00
A&ODCS	Reaction Wheels	6	12.00
	Sun Sensor	1	1.00
	Star Sensor	1	2.45
	GPS	1	7.00
	IMU	1	1.00
C&DH	On-Board Computer	3	3.00
	Data Bus	1	≈0.00
Total			32.45
Total Power With 15% Margin			37.32

Table 2.2: Power Budget of On-Board Devices.

3. Organisation

This chapter gives insight into the organisational processes of the team over the past weeks. The functional flow diagram is given in Section 3.1, the functional breakdown structure is presented in Section 3.2 and a Gantt chart is given in Section 3.3.

3.1. Functional Flow Diagram

In Figure 3.1, 3.2 and 3.3, the functional flow diagram of the BBIS mission is presented. It indicates the order of functions that the spacecraft and the ground station are to perform. It contains the five general mission phases; construct spacecraft, launch spacecraft, deploy spacecraft, operate spacecraft and perform EoL procedure.

3.2. Functional Breakdown Structure

In Figure 3.4, the functional breakdown structure is presented. It contains the functions the spacecraft and the ground station are to perform, ordered hierarchically.

3.3. Gantt Chart

Figure 3.5 visualises the Gantt Chart of the last two phases of the DSE. It indicates the planned schedule of these phases and monitors the progress. All the deadlines and deliverables are added, next to the activities that need to be performed to meet these deadlines.

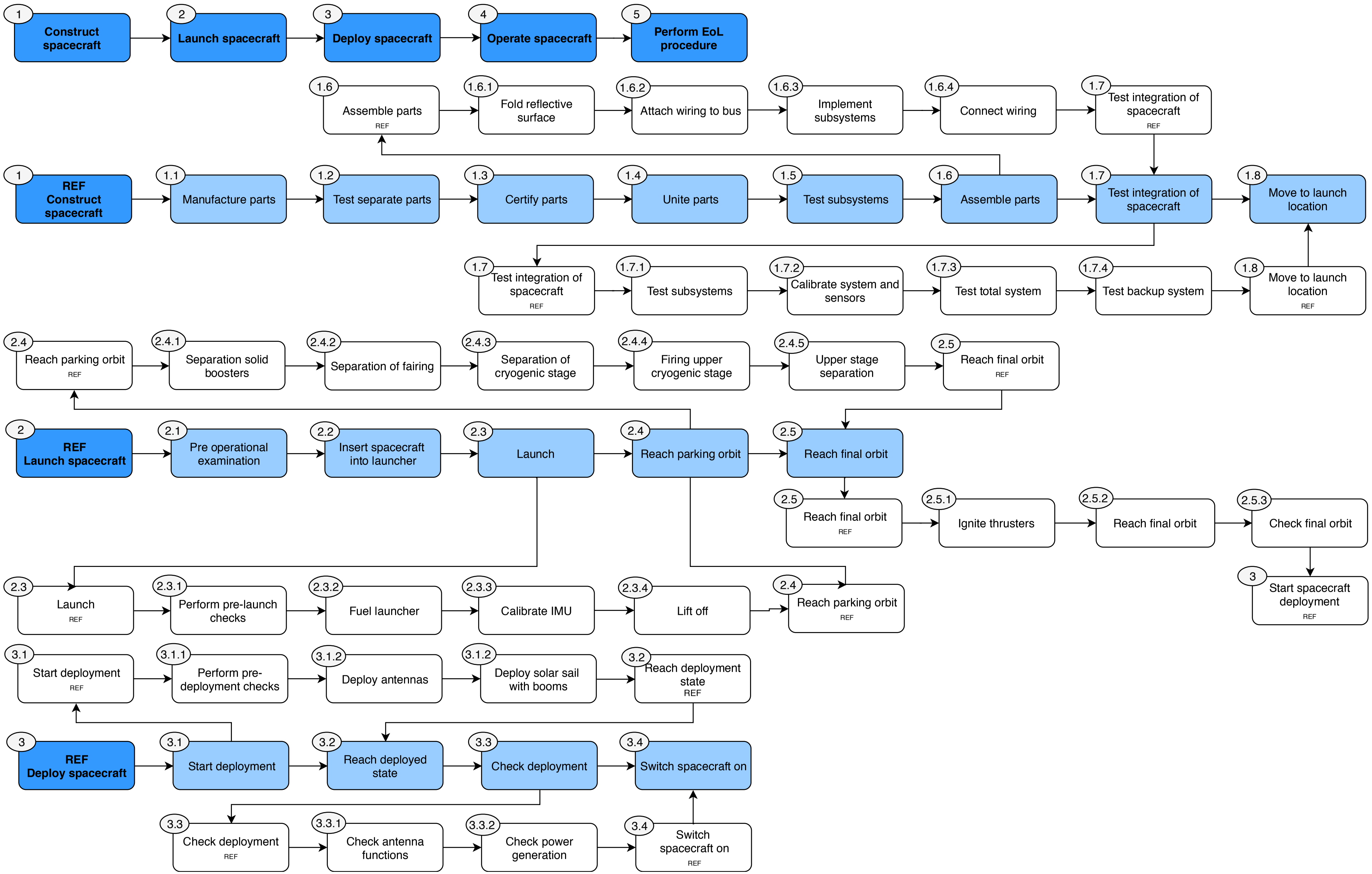


Figure 3.1: Functional Flow Diagram Part 1.

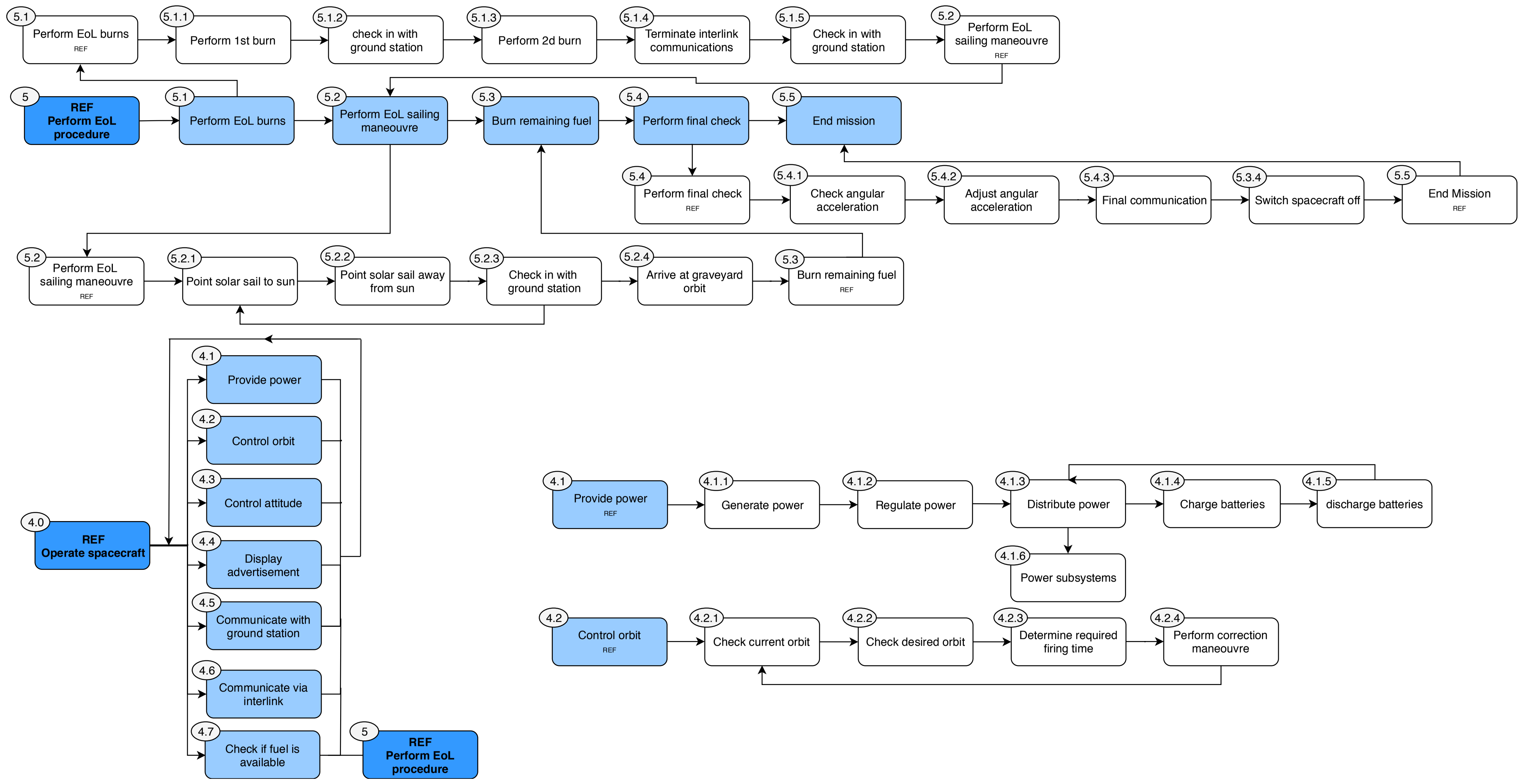


Figure 3.2: Functional Flow Diagram Part 2.

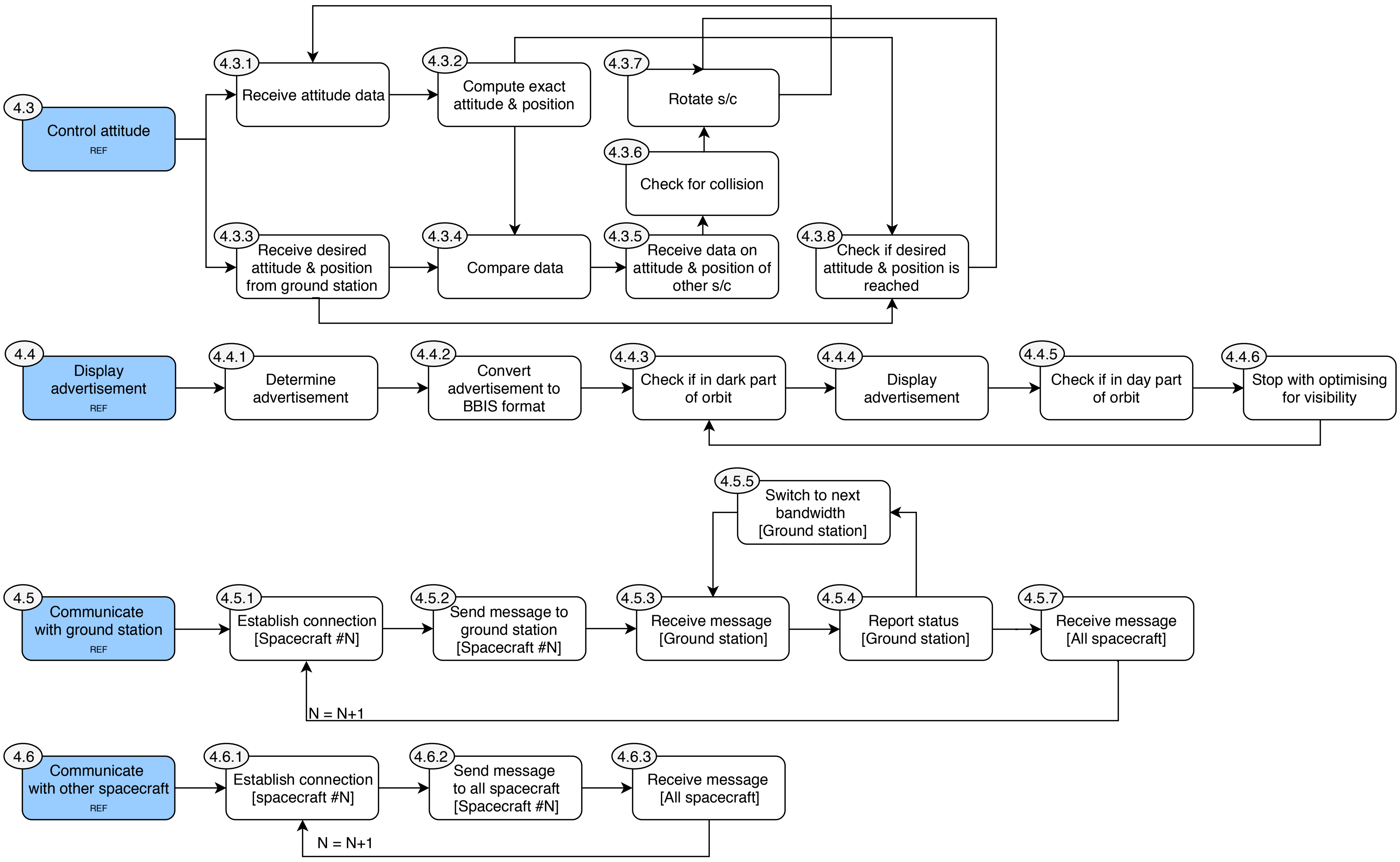


Figure 3.3: Functional Flow Diagram Part 3.

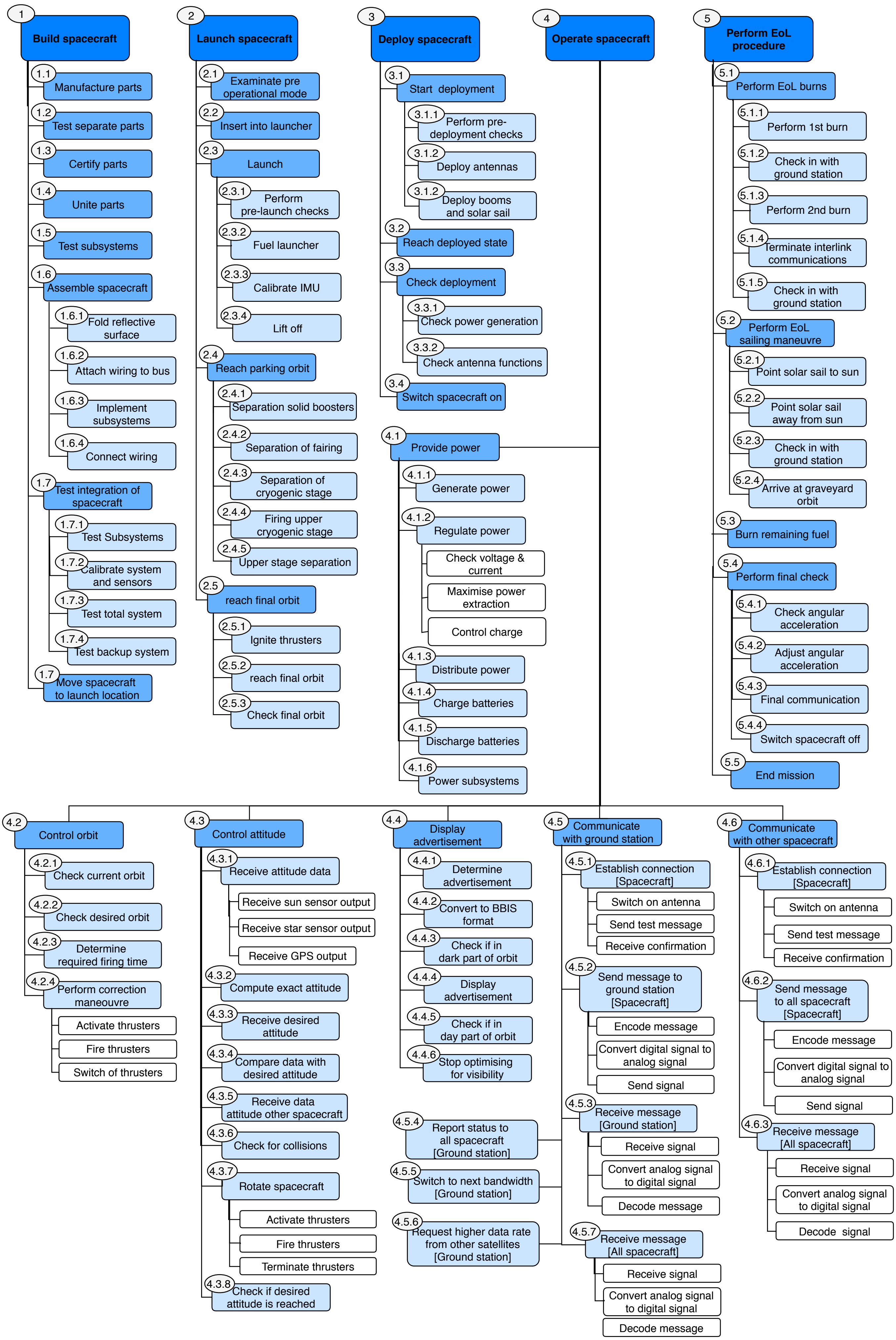


Figure 3.4: Functional Breakdown Structure.

	WBS	Name	Duration	Start	Person Respon...	Predecessors	Start Slack
2	5	Execute Detailed Design Analysis	16.625 days?	6/4/18 9...			0.25 days
3	5.1	Work Out Concept	16.625 days?	6/4/18 9...			0.25 days
4	5.1.3	Calculate Astrodynamics	10 days?	6/4/18 9...	Wierikx		4.5 days
5		Genral Orbit	2 days	6/4/18 9...			12.5 days
6		Orientation	2 days	6/4/18 9...	Bahnam		7.75 days
7		Disturbance Forces	2 days?	6/6/18 9...		6	7.75 days
8		Formation Flying	6 days	6/4/18 9...	Stutvoet		4.5 days
9		Lit. Study	2 days	6/4/18 9...			4.5 days
10		Orbit determination	2 days	6/6/18 9...		9	4.5 days
11		Visual Aid	2 days	6/8/18 9...		10	8.5 days
12		Visibility from Earth	2 days	6/8/18 9...		10	4.5 days
13		Orbit Perturbation	2 days	6/8/18 9...	Wierikx	10	4.5 days
14		Delta V budget	2 days	6/12/18 ...	Wierikx	13	4.5 days
15		Visibility analysis	2 days	6/12/18 ...	Wierikx	12	4.5 days
16		Verification	1 day	6/14/18 ...	Bahnam	15;14	4.5 days
17		Sensitivity Analysis	1 day	6/15/18 ...	Bahnam	16	4.5 days
18	5.1.1	Design Spacecraft	16.25 days?	6/4/18 9...	Yang		0.25 days
19		Subsystems	15.125 days?	6/4/18 9...			0.25 days
20		Payload	12.25 days	6/4/18 9...	van Beek		0.25 days
21		Lit. Study	1.5 days	6/4/18 9...			0.25 days
22		Material	2 days	6/5/18 2...		21	0.25 days
23		Booms	2 days	6/7/18 2...		22	0.25 days
24		Area	2 days	6/11/18 ...		23	0.25 days
25		Depolyment Mechanism	2 days	6/13/18 ...		24	0.25 days
26		Design Choices	2 days	6/15/18 ...		25	0.25 days
27		Verification and Validation	0.5 days	6/19/18 ...		26	0.25 days
28		Sensitivity Analysis	0.25 days	6/20/18 ...		27	0.25 days
29		Power	4.25 days?	6/19/18 ...	Yang		0 days
30		Power Budget	2 days	6/19/18 ...			0 days
31		Solar Pannels Selection	0.9 days?	6/21/18 ...		30	0 days
32		Battery Selection	0.5 days	6/21/18 ...		31	0 days
33		Distribution and Regulation	0.1 days	6/22/18 ...		32	0 days
34		Verification and Validation	0.5 days	6/22/18 ...		33	0 days
35		Sensitivity Analysis	0.25 days	6/25/18 ...		34	0 days
36		A&ODCS	8.75 days	6/4/18 9...	Vossen		7.75 days
37		Lit. Study	2 days	6/4/18 9...	Hallak, Vossen		7.75 days
38		Disturbances	2 days	6/6/18 9...		37	7.75 days
39		Sensors selection	2 days	6/8/18 9...		7;38	7.75 days
40		Actuators selection	2 days	6/12/18 ...		39	7.75 days
41		Lit. study for propulsion	1 day	6/4/18 9...			12.75 days
42		Propulsion Selection	2 days	6/5/18 9...	De Jong	41	12.75 days
43		Verification and Validation	0.5 days	6/14/18 ...		40;42	7.75 days
44		Sensitivity Analysis	0.25 days	6/14/18 ...		43	7.75 days
45		Data Management	4 days?	6/18/18 ...	Bahnam, Hallak		2.625 days
46		Communications	3 days	6/18/18 ...	Bahnam		2.625 days
47		Link Budget	2 days	6/18/18 ...			2.625 days
48		Component Selection	1 day	6/20/18 ...		47	2.625 days
49		C&DH	1.1 days	6/20/18 ...	Hallak		2.775 days
50		LIt. Study	0.5 days	6/20/18 ...			2.775 days
51		Component selection	0.5 days	6/20/18 ...		50	2.775 days
52		Block Diagram	0.1 days	6/21/18 ...		51	2.775 days
53		H/W & S/W Diagram	1 day?	6/21/18 ...	van Beek, Stutv...	46	2.625 days
54		Verification & Validation	0.5 days	6/21/18 ...		46;49	2.775 days
55		Sensitivity Analysis	0.25 days	6/21/18 ...		54	2.775 days
56		General Design	9.375 days?	6/13/18 ...			1.875 days
57		Structures	6 days?	6/13/18 ...	Seoane Alvarez		1.875 days
58		Loading Conditions	0.5 days?	6/13/18 ...			1.875 days
59		Force Diagrams	1.5 days	6/13/18 ...		58	1.875 days
60		CG location	0.25 days	6/15/18 ...		59	1.875 days
61		Section Properties	0.5 days	6/15/18 ...		60	1.875 days
62		Stress Distribution	1.5 days	6/15/18 ...		61	1.875 days
63		Materials	1 day	6/19/18 ...		62	1.875 days
64		Frequency Requirement	0.25 days	6/20/18 ...		63	3.625 days
65		Verification and Validation	0.25 days	6/20/18 ...		64	3.625 days
66		Sensitivity Analysis	0.25 days	6/20/18 ...		65	3.625 days
67		Weights	0.5 days	6/20/18 ...		63	1.875 days
68		Thermal Control	4 days?	6/20/18 ...	Hallak	20	0.25 days
69		Thermal Protection	1 day?	6/20/18 ...			0.25 days
70		FEM	2 days	6/21/18 ...		69	0.25 days
71		Model Set-Up	2 days	6/21/18 ...			0.25 days
72		Final simulation	1 day	6/21/18 ...		63	1.25 days
73		Analysis of the results	1 day	6/25/18 ...		70	0.25 days
74		Verification & Validation	2.25 days	6/20/18 ...			3.25 days
75		Model Set-up	1 day	6/20/18 ...			3.25 days
76		Verification of Results	0.25 days	6/22/18 ...		72	1.5 days
77		Sensitivity Analysis	0.5 days	6/22/18 ...		76	1.5 days
78		Environment	2.25 days	6/20/18 ...	Hallak		2.375 days
79		Pre-Orbit Environment	0.25 days	6/20/18 ...			2.375 days
80		In Orbit Environment	2 days	6/20/18 ...		79	2.375 days
81	5.1.2	Establish Launch and Retrieval	4 days?	6/20/18 ...	De Jong		1.875 days
82		Disposal	1.5 days	6/25/18 ...		29;5	-0.125 d...
83		Select a launcher	0.5 days	6/20/18 ...		67;4	1.875 days
84		Orbit Insertion	1 day?	6/21/18 ...		83	1.875 days
85		Deployment of Satellite	0.5 days	6/22/18 ...		84	1.875 days
86		Prepare Report	15 days	6/4/18 9...			1.5 days
87	5.2	Verification	3 days?	6/18/18 ...	Hallak, Stutvo...		3 days
88		Compliance Matrix Set-up	1 day?	6/18/18 ...			3 days
89		Compliance Matrix check	2 days	6/19/18 ...		88	3 days
90	5.5	Risk Managment	2 days	6/18/18 ...	De Jong	4	4.5 days
91	5.4	Check for Sustainability	1 day	6/21/18 ...	Seoane Alvarez	83	2.375 days
92	5.3	System Requirements check	0.9 days?	6/20/18 ...			0.725 days
93	5.6	System Integration	0.5 days	6/20/18 ...	Hallak		0.725 days
94	5.6.1	Interface Management	0.5 days	6/20/18 ...		92	0.725 days
95	5.6.2	System Design Integration	0.5 days	6/20/18 ...	Hallak	92	0.725 days
96	5.7	Integration Requirements Check	0.5 days	6/21/18 ...		95;94	0.725 days
97		PD&D	1 day?	6/18/18 ...	Hallak		2.675 days
98	5.8	Write Business Plan	2.75 days	6/21/18 ...	Vossen	97	0.675 days
99	5.8.2	Market Analysis	0.5 days	6/21/18 ...			0.675 days
100	5.8.1	Cost Analysis	2 days	6/21/18 ...		99	0.675 days
101		Return Investment	0.25 days	6/25/18 ...		100	0.675 days
102	5.9	Validation	1 day	6/21/18 ...	Yang	96	0.725 days
103		Sensitivity Analysis	0.5 days	6/25/18 ...	Bahnam	87	1.125 days
104		Functional Breakdown Structure	1.5 days?	6/20/18 ...	van Beek		1.625 days
105		Functional Flow Diagram	1.5 days	6/21/18 ...	van Beek	104	1.625 days
106		MAI	0.2 days?	6/25/18 ...		98	0.675 days
107	5.13	Final Review Preparation	1 day	6/22/18 ...		102	0.725 days
108	5.10	Finalise Design	1 day	6/21/18 ...		93	2.225 days
109		Status Report	0.125 days	6/18/18 ...			0 days
110	5.12	Jury Summary	0.1 days?	6/21/18 ...	van Beek		0 days
111	5.11	Holy Draft	0.1 days?	6/26/18 ...		3;86;87;98;108	0 days
112		Status Report	0.125 days	6/25/18 ...			0 days
113	5.14	Final Review	0.375 days?	6/28/18 ...		107	0 days
114		Final Report Preparation	1 day?	6/28/18 ...		113	1.475 days
115		Final Report	0.011 days?	7/2/18 4...		114	0 days
116	6	Check-out	11.963 days?	6/20/18 ...			2.862 days
117	6.1	Finalise Report	2 days	6/25/18 ...		108;87;86	3.875 days
118	6.3	Create Poster	2 days	6/20/18 ...	Wierikx		2.862 days
119		Deliver Poster	0.1 days	6/26/18 ...	Wierikx	118	0 days
120	6.2	Symposium Preparation	2.875 days	7/2/18 9...			0.125 days
121		Status Report	0.125 days	7/2/18 2...			0 days
122		Final Report	1 day?	7/3/18 8...		117	0 days
123	6.4	Red Booklet	0.1 days?	7/4/18 4...	van Beek	122	0 days
124	6.5	Symposium	0.875 days	7/5/18 9...		120	0 days

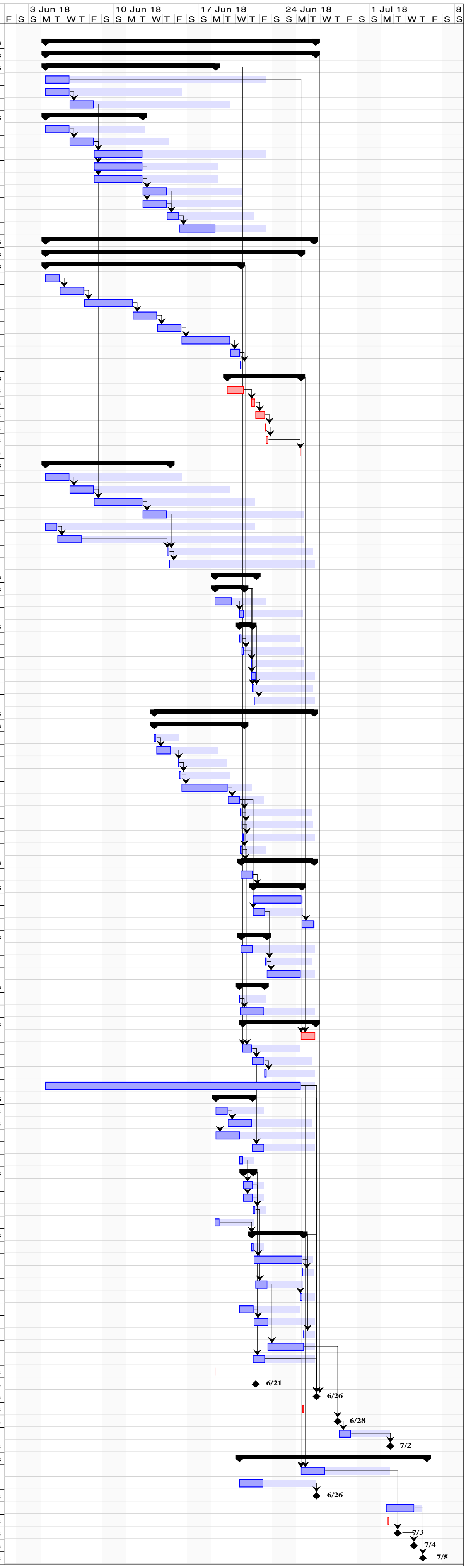


Figure 3.5: Gantt Chart of Final Phases of DSE.

4. Requirements

This chapter discusses the the user requirements established in [2], where a more detailed analysis of the requirements can be found. First, the user requirements are discussed in Section 4.1, followed by a discussion on the feasibility of those requirements in Section 4.2. The system and functional requirements are discussed in the chapters of the subsystems itself.

4.1. User Requirements

The user requirements of Delft University of Technology (Tud), the advertisers (Ads), the public (Pub), the governments (Gov), the launcher company (Lan), other space agencies (SA), the manufacturers (Man) and the suppliers (SUp) are stated in this section. Meeting these requirements ensures a positive reaction from all parties listed above. However, some requirements do not have specific values yet and are noted with to be determined ([TBD]). The requirements related to each stakeholder are listed below.

4.1.1. Delft University of Technology

Firstly, the TU Delft is the contractor who wants to create a cost effective project. The requirements of the TU Delft are listed below.

- **BBIS-Tud-01** The project shall be cost effective.
- **BBIS-Tud-02** The spacecraft shall have a minimal lifetime of 50 years.[1]
- **BBIS-Tud-03** The spacecraft shall be launched before 1-1-2028.[1]
- **BBIS-Tud-04** The advertisement shall be optimised for the United States of America.
- **BBIS-Tud-05** The spacecraft shall be able to withstand all of the conditions exerted on it during its life.
- **BBIS-Tud-06** The spacecraft shall fly in space.
- **BBIS-Tud-07** The spacecraft shall be sustainable.

Furthermore, TU Delft has two more wishes that need looking into. The first wish is to have more than one advertisement on the billboard. The second wish is to extend the time in space.

4.1.2. Advertisers

Secondly, the revenue of the billboard will be generated by the advertisements payed by advertiser. This is reflected in the requirements of the advertiser listed below.

- **BBIS-Ads-01** The perceived billboard size shall at least be equal to the size of a full moon.
- **BBIS-Ads-02** The perceived billboard light intensity shall at least be equal to the intensity of a full moon.
- **BBIS-Ads-03** The advertisement shall be easy to see by public.
- **BBIS-Ads-04** The advertisement shall be recognisable from Earth.

4.1.3. Public

Thirdly, the public needs to be considered, the viewers do not want their ordinary life to be interrupted by the billboard. Moreover, sustainability should be considered in order to add value to the public's perception of the device. The requirements for the general public are listed below.

- **BBIS-Pub-01** The spacecraft shall not be disturb the day to day life of the observers.
- **BBIS-Pub-02** The spacecraft shall not destroy life on Earth.
- **BBIS-Pub-03** Advertisements shall not be provoking.

[1]This requirement is further elaborated on in Section 4.2.

4.1.4. Governments

Next, the different governments try to protect the public with legislation. The user requirements of the governments are listed below.

- **BBIS-Gov-01** The project shall follow the legislation of the countries involved in the process.
- **BBIS-Gov-02** The billboard shall be able to deliver important announcements during emergencies.
- **BBIS-Gov-03** The billboard shall not bring people in danger by distracting them.
- **BBIS-Gov-04** The spacecraft shall comply with all Inter-Agency Space Debris Coordination Committee (IADC) regulations.

BBIS-Gov-02 requires that the spacecraft shall be able to send this emergency announcement. This has a big influence on the design while the government is not the main user. Therefore, this requirement is considered as a wish.

4.1.5. Other Users

The remaining users which have not been discussed are the launcher company (Lan), other space agencies (SA), the manufacturers (Man) and the suppliers (SUp). Their user requirements are listed below.

- **BBIS-Lan-01** The spacecraft shall be able to be launched by the launcher.
- **BBIS-Lan-04** The spacecraft shall not damage the launcher.
- **BBIS-SA-02** The spacecraft shall be disposed after service so that the risk of collision is less than 0.1% in the next 10,000 years.
- **BBIS-Man-01** The parts from the suppliers shall be delivered at least [TBD] days before the launch.
- **BBIS-Man-02** The assembly of the spacecraft shall be producible.
- **BBIS-SUp-01** The order shall be placed [TBD] days before the delivery.
- **BBIS-SUp-02** The spacecraft shall only be designed with commercial off the shelf (COTS) components.

Furthermore, it has been determined that the spacecraft can not be launched from the USA due to legislation, this has been elaborated on in [3].

4.2. Feasibility

During the initial analysing and approaching phase, the requirements have been reviewed and discussed. This section discusses the feasibility of the requirements briefly, a more elaborated discussion can be found in [2]. First, Section 4.2.1 elaborates on the set launch date. Then, in Section 4.2.2 the life span of the spacecraft is discussed.

4.2.1. Launch Date

The requirements state that the spacecraft shall be launched before 1-1-2028. However, this deadline is not considered feasible, because it is a new concepts. Taking this into account, the launch date proposal is rather going to be considered as a wish instead of as a requirement. This is further elaborated on in Section 13.2.

4.2.2. Life Span

Requirement **BBIS-Tud-02** states that the life span should be at least 50 years. However, after careful analysis, it turned out that the design would be driven too far by the lifespan. Mainly because, it is a new concept, mostly only commercial off the shelf components are to be used, a lack of space on the designated orbit, limited fuel and the spacecraft might not be relevant after 50 years. Taking these reasons into account, a lifetime of 50 years does not seem achievable. This is further elobareted on in Chapter 9.

4.3. Driving and Killing Requirements

In order to optimise the outcome of the design phase, the driving and killing requirements for the design were selected from the complete requirement list in [2] and [3]. The driving requirements, listed below, drive the design more than average.

- **BBIS-Sys-T01-1** The project shall have a return on investment of at least 0%.
- **BBIS-Sys-T05-5** The spacecraft shall be able to withstand a collision with space debris smaller than [TBD] m at an collision speed of [TBD] m/s such that it will not lead to total failure of the mission.
- **BBIS-Sys-A01-1** The spacecraft shall have a minimum radius of $0.00436 \times$ altitude of orbit m when it passes over the USA.
- **BBIS-Sys-A02-1** The spacecraft shall provide an illumination at Earth of at least $0.108\ lx$.
- **BBIS-Sys-A03-2** The spacecraft shall be visible for at least $14\ s$ per flyover.

The killing requirement is as follows.

- **BBIS-Sys-G01-3** The spacecraft shall not light pollute other spacecraft.

5. Astrodynamics

This chapter describes the different calculations and assumptions made to determine the orbit and its effects. Firstly, the requirements for astrodynamics are described in Section 5.1. Secondly, the used reference frame is presented in Section 5.2. Then, the general orbit is set in Section 5.3, followed by Section 5.4, where the chosen formation is explained. Subsequently, the ΔV-budget is discussed in 5.5, respectively. Visibility analysis is discussed in Section 5.6. Finally, the chapter is concluded with verification and validation in Section 5.7 and Section 5.8, respectively.

5.1. Requirements

The system and functional requirements were established in [3]. It was chosen to change **BBIS-Sys-A01-1**, the requirement based on the radius of the billboard [3]. The requirement has been changed to the total area of the billboard, instead of the radius; the new identifier is **BBIS-Sys-A01-3**. With this new requirement, it is possible to use different formations, which will be further explained in Section 5.4. The relevant requirements for astrodynamics are listed below. The full requirement list can be found in Section 17.2.

- **BBIS-Sys-A01-3** The spacecraft shall have an area of at least $\pi \cdot (0.00436 h_{sat})^2\ m$ when it passes over the USA.[1]
- **BBIS-Sys-A03-2** The spacecraft shall be visible for at least $14\ s$ per flyover.
- **BBIS-Sys-A04-1** The light source shall be focused on Earth's surface.

5.2. Reference Frame

In this chapter, an Earth-centred inertial reference frame is used, which is shown in Figure 5.1. For the visibility analysis, the Earth is rotated with a full revolution every day around the z-axis. At $t = 0$, the Sun in positioned in x-direction. All nomenclature used is shown in Figure 5.2.

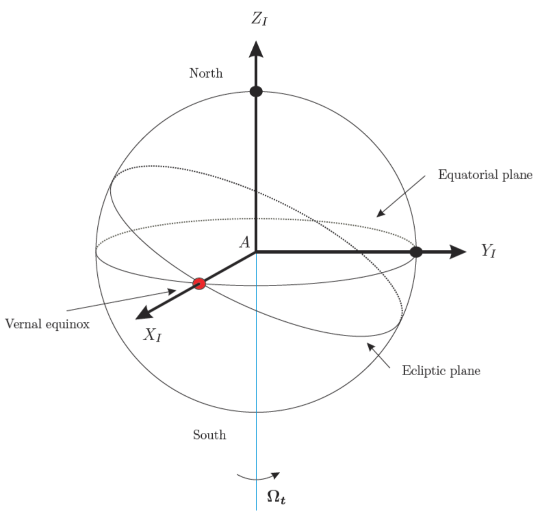


Figure 5.1: Earth-Centered Inertial Reference Frame [4].

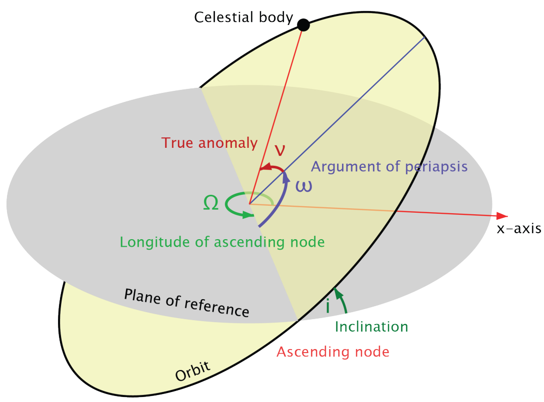


Figure 5.2: Nomenclature for Orbital Mechanics.[2]

5.3. General Orbit

This section explains how the general orbit of the spacecraft is determined. First, the general method of astrodynamics is explained, followed by the chosen orbit.

[1] $R_{sat} = h_{sat} \frac{R_{Moon}}{d_{Moon} - R_{Earth}} = \frac{1737}{405400-6371} h_{sat} = 0.00436 h_{sat}$ [2], $A_{sat} = \pi R_{sat}^2 = \pi (0.00436 h_{sat})^2$.

[2] URL https://www.hackster.io/30506/calculation-of-right-ascension-and-declination-402218 [cited 15 June 2018].

5.3.1. Method

The structure of the program used is shown in Figure 5.3.

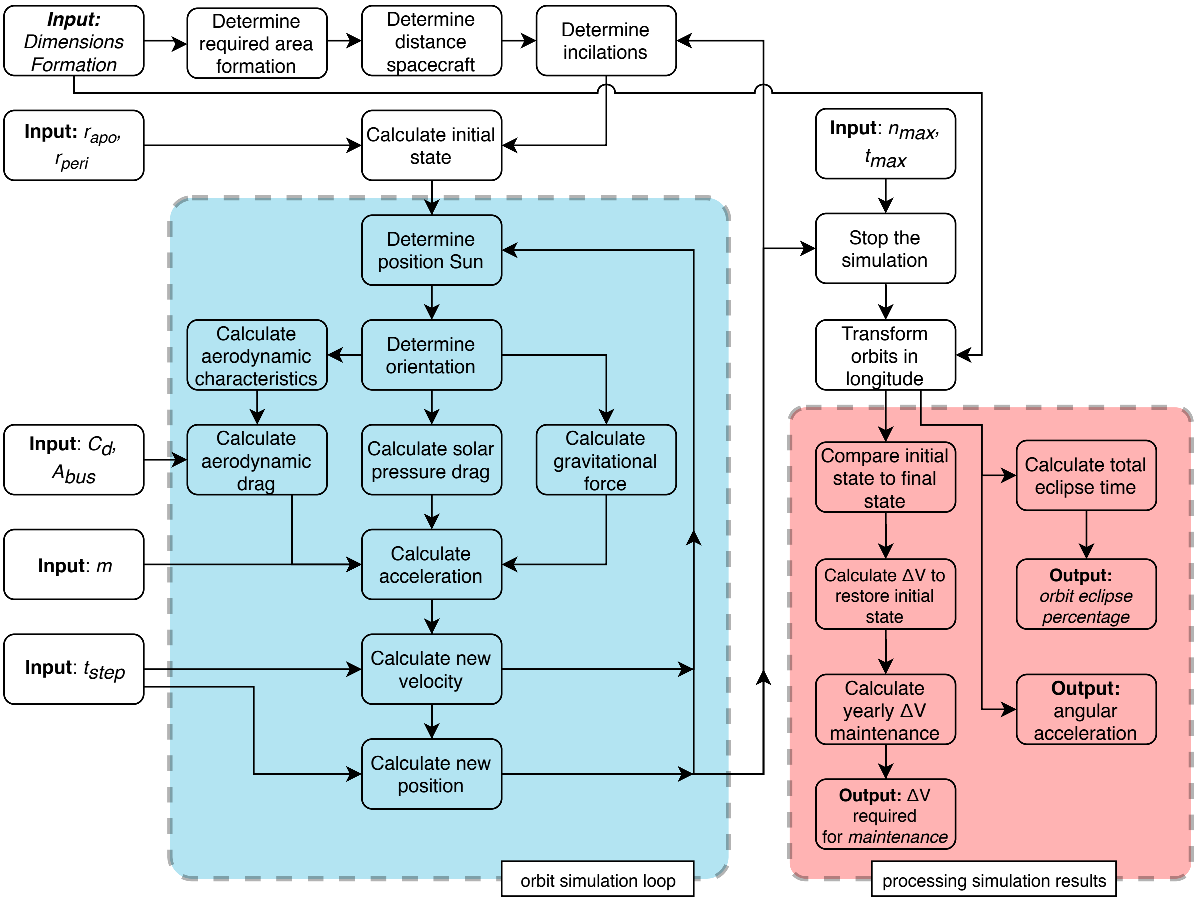


Figure 5.3: Orbit Program Structure.

Orientation

The main goal of the spacecraft is to display the advertisement on Earth, as requirement **BBIS-Sys-A04-1** states. Therefore, the payload needs to have the correct orientation. Since the payload is fixed to the spacecraft, the spacecraft orientation is determined by the payload. The required orientation of the spacecraft can be determined with Equation 5.1.

$$\mathbf{\hat{S}_p} = \frac{-(\mathbf{\hat{P}_{sun}} - \mathbf{\hat{P}_{SC}})}{|\mathbf{\hat{P}_{sun}} - \mathbf{\hat{P}_{SC}}|} \tag{5.1}$$

Where $\mathbf{\hat{S}_p}$ is the unit vector perpendicular to solar sail (pointing away from the Earth), $\mathbf{\hat{P}_{sun}}$ is the unit vector of the position of the sun, and $\mathbf{\hat{P}_{SC}}$ is the unit vector of the position of the spacecraft, both measured from the Earth's centre.

Gravitational Force

The gravitational force is computed using Equation 5.2. The gravitational force is constant for a circular orbit. The direction of the force is always pointing to the centre of the Earth.

$$\mathbf{F_g} = -\mathbf{\hat{P}_{SC}} \cdot \frac{\mu m_{SC}}{(h + R_{Earth})^2} \tag{5.2}$$

Where F_g is the gravitational force in N, μ is the gravitational parameter, which equals $3.986 \cdot 10^{14}\ m^3 s^{-2}$, m_{SC} is the mass of the spacecraft in kg, h is the altitude of the orbit in m and R_{Earth} is the radius of the Earth in m.

Solar Radiation Force

Solar pressure force is especially important to consider for spacecraft with a low mass compared to the size of its reflective area. The spacecraft uses a solar sail to reflect the sunlight to Earth. The solar pressure force can be computed using Equation 5.3.

$$\mathbf{F_S} = \mathbf{S_p} \cdot (1+\eta)\frac{I_f}{c}A_{cross} \tag{5.3}$$

Where F_S is the solar radiation pressure in N, η is the reflecting efficiency of the material, I_f is the solar irradiance, which is approximately $1361\ W/m^2$ close to Earth. c is the speed of light, which is $3.0 \cdot 10^8\ m$. A_{cross} is the cross-sectional area of the solar sail in m^2.

Aerodynamics Drag

Aerodynamic drag is important when considering low Earth orbits. For higher orbits, like a geosynchronous orbit, the aerodynamic drag is small, because of the low air density at that altitude. The aerodynamic drag vector can be calculated with Equation 5.4, where the drag is always in the opposite direction of the velocity.

$$\mathbf{F_D} = -0.5\rho C_d A_{cross}\mathbf{V^T V} \tag{5.4}$$

Where F_D is the aerodynamic drag in N, ρ is the density in kg/m^3, which varies with altitude and C_d is the drag coefficient. A drag coefficient of $8/3$ is used for the bus and $4cos(\alpha)^2$ is used for the solar sail [5], where α represents the angle between the solar sail and the velocity vector. A_{cross} is the cross-sectional area perpendicular to the velocity vector in m^2 and V is the velocity in m/s.

5.3.2. Results

Different orbits are compared. One of the main arguments to choose for a certain orbit is the amount of time spent above the USA during the BBIS's lifetime. Therefore, the orientation of the orbit with respect to the Sun over a year is of importance. There exist two types of orbits that have a the right orientation throughout the entire year, the two possibilities are a geosynchronous orbit (GSO) and a Sun-synchronous orbit.

A Sun-synchronous orbit is not used with an altitude higher than a $1\ 000\ km$ altitude, because the Van Allen Belts start at this height, the amount of radiation increases drastically above that altitude.[3] Above the Van Allen Belts no Sun-synchronous orbit is possible. A low Earth orbit result in a low amount of potential views and therefore a low revenue, such an orbit will result in launch cost higher than the revenue making the project financially unfeasible. For GSO, the potential views and therefore the revenue is much higher, therefore the average orbit has been set to be GSO. **BBIS-Tud-04** is met, as the formation is able to point at the USA continuously. Requirement **BBIS-Tud-06** and **BBIS-Sys-T06-1** are also met, because GSO is higher than $100\ km$ altitude, which is generally assumed to be the start of space.

The orientation of the spacecraft during one orbit is shown in Figure 5.4. In order to keep displaying the advertisement, the spacecraft need to rotate about the z-axis (a right-handed coordinate system is used). When the spacecraft is not displaying the advertisement, the spacecraft needs to either rotate 270° about the positive z-axis or 90° about the negative z-axis. Due to the angular velocity, both rotations requires the same magnitude of acceleration. It is chosen to rotate the spacecraft 90° about the negative z-axis, because it is not preferred rotate the spacecraft 360° every orbit, mainly because the reaction wheel would have to be desaturated more often. Also, more power needs to be generated by the solar cells. Herewith, requirement **BBIS-Sys-A04-1** is satisfied.

The maximum solar radiation force is $3.7 \cdot 10^{-3}\ N$. For the aerodynamic drag, this is $3.1 \cdot 10^{-9}\ N$. As expected, the aerodynamic drag is significantly lower than the solar radiation force.

5.4. Formation Flying

The formation of the swarm determines the size and the placement of the pixels in the advertisement as seen from Earth. This sections start with the method of how this was determined in Section 5.4.1, followed by the results in Section 5.4.2.

[3] URL https://image.gsfc.nasa.gov/poetry/tour/AAvan.html [cited 24 June 2018].

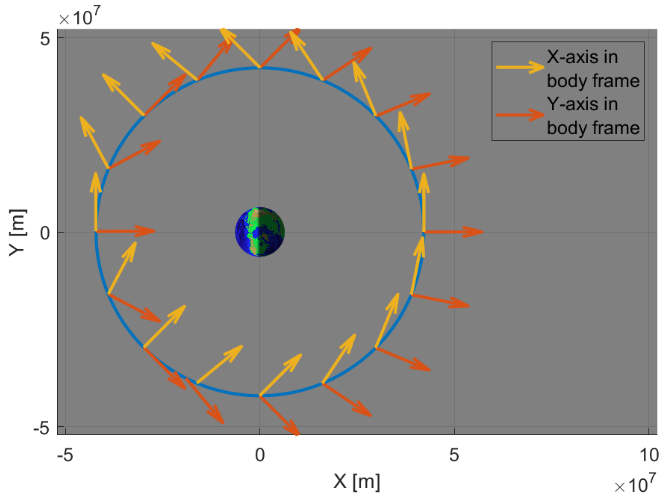


Figure 5.4: Orientation of the Solar Sail.

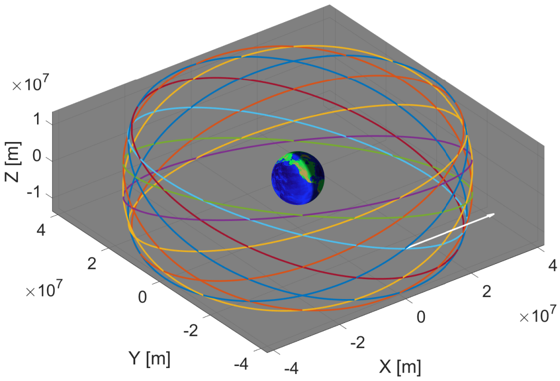
5.4.1. Method

All spacecraft in a formation have to orbit around the centre of the Earth. Therefore, a formation cannot stay the same throughout the entire orbit without orbit manoeuvres. One could only keep a formation the same throughout the entire orbit if continuous thrust is applied to correct for not flying around the centre of the Earth. However, such manoeuvres are not feasible for a long lifetime. Different orbits are therefore chosen for the spacecraft.

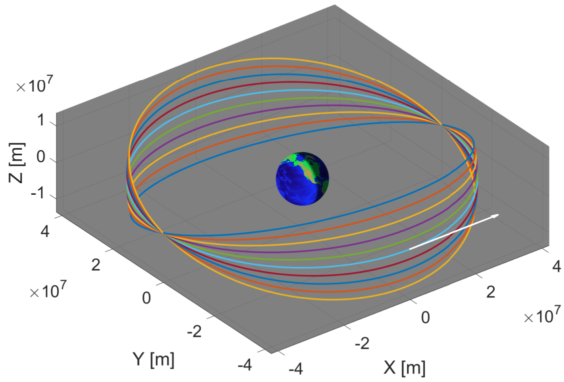
Because all orbits centre around the Earth, the previously explained program in Section 5.3 can be used. The structure of the program is shown in Figure 5.3. The whole program is placed in a loop for different inclinations. After the loop, the orbits are transformed for different argument of periapsis and different ascending nodes.

5.4.2. Results

It was decided to use 900 spacecraft. 900 pixels is ought to be the minimum amount of pixels to display an advertisement. Two different types of orbits are shown in Figure 5.5. The inclination in both of the graphs is enlarged by a factor 300. The white arrow depicts the direction of the velocity. In Figure 5.5a, all orbits have the same inclination, but their ascending nodes are distributed equally over the whole plane of reference. In Figure 5.5b, a formation is shown where all orbits have different inclination. The difference in ascending nodes of the orbits are small, such that the spacecraft do not collide when the inclination is $0°$. Both figures show 10 different orbits, for the final formation dedicated to logos is decided to use 20 different orbits, and for the formation dedicated to text 5 different orbits are used.



(a) Same Inclination, Different Longitude of Ascending Node.

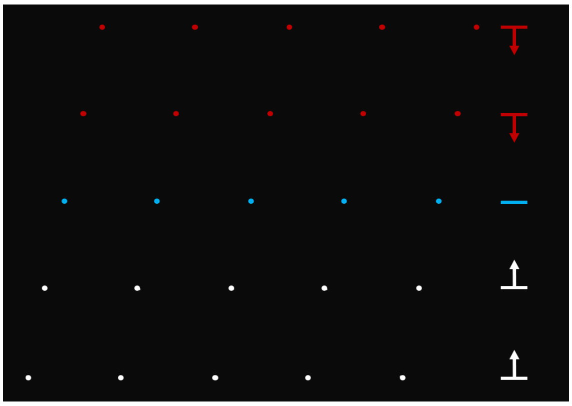


(b) Different Inclination, Similar Ascending Node.

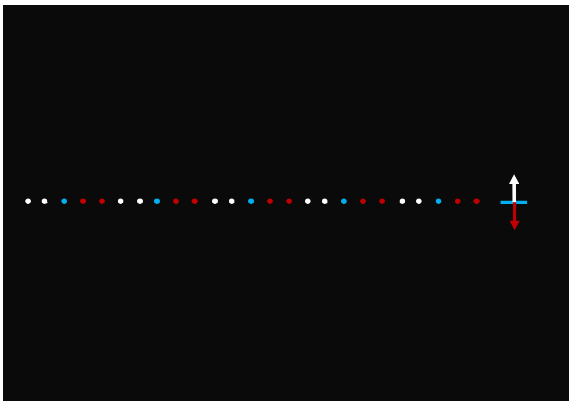
Figure 5.5: Orbits With Inclination 300 Times Enlarged.

Figure 5.5 shows that all orbits cross each other at least twice per orbit. Therefore, the shape of the formation influences the orbits. **BBIS-Sys-A01-3** first stated that the formation had to be circular, but a circular shape requires the spacecraft to fly relatively close to each other. Therefore, the requirement has been changed to make its area based as further explained in Section 5.1.

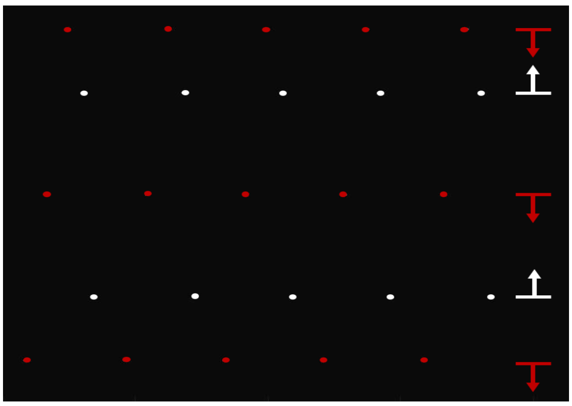
Figure 5.6 shows a 5×5 formations of the spacecraft from a point of view at the equator. Each dot represent a spacecraft, where the red dots are moving downwards, the white dots moving upwards and the blue dots are stationary. Each figure indicates the best or worst formation during one orbit.



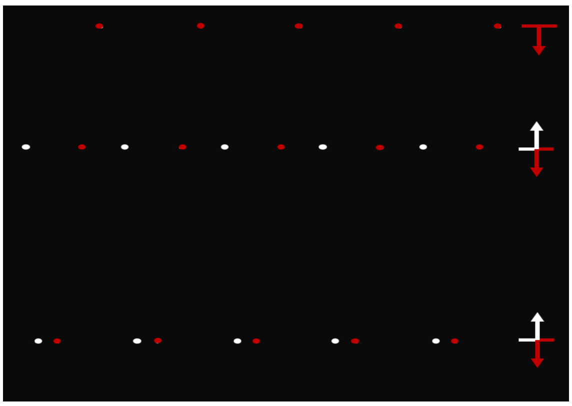
(a) Different Inclinations, Best Pattern.



(b) Different Inclinations, Worst Pattern.



(c) Different Ascending Nodes, Best Pattern.



(d) Different Ascending Nodes, Worst Pattern.

Figure 5.6: Formation of Spacecraft as Seen From the Equator.[4]

The formation with different inclinations has, at every single moment in time, a constant pitch between the spacecraft. This makes the formation favourable for displaying text, because it is always possible to make a symmetrical letter. The formation with the different ascending nodes has a changing latitude pitch.

Changing the formation does not require much ΔV relatively to the total ΔV budget, as is explained in Section 5.5.2. Therefore, it has been chosen to account for 6 different formations in the ΔV-budget. Multiple formations make the BBIS more appealing for advertisers.

The size of the formation is partly determined by the space available in GSO. In 2017 531 operational spacecraft were positioned in GSO. The longitude of the USA approximately ranges from 75 to $125°W$. The widest place available above the USA is between 107.3 and $109.9°W$[5], so the formation cannot be wider than $2.6°$ including drifting and margins. On both sides, a $0.1°$ margin is taken for safety and a margin of $0.15°$ is taken for the monthly drift of all spacecraft as explained in Section 5.5. This result in a total width of $2.25°$.

The goal of the formation with different inclinations is to display text. The formation should be as long as possible, to be able to display as many words as possible. Next, there should be at least 5 spacecraft in the latitude direction to display all symbols and letters. Therefore, a $5x180$ grid of spacecraft is chosen. This formation has 5 different orbits, with 180 spacecraft in each orbit with a different argument of periapsis. At the best situation, the formation has a length of $1\ 406\ 045\ m$ and a height of $54\ 220\ m$ at 6am and 6pm (Greenwich Mean Time -6), which means requirements **BBIS-Sys-A01-3** and **BBIS-Ads-01** are met.

For the formation with different ascending nodes, a $20x45$ grid of spacecraft is selected with the goal to display logos. This formation has 20 different orbits, with 45 spacecraft in each orbit with a different argument of periapsis. Mainly because most logos are rectangular, less ΔV is required to change the formation, and the orbital perturbations are less compared to a square shape. The length of the formation is $420\ 178m$ and the height is $181\ 441\ m$ long, which means requirements **BBIS-Sys-A01-3** and **BBIS-Ads-01** are met.

Due to the large distances between the spacecraft, it can be concluded that a non-rigid or rigid structure is not feasible for this particular design. The structure would be too heavy. Next, it should also be noted that formation flying is a cutting edge technology which has not yet been explored on a large scale. Therefore, the high amount of spacecraft in formation increases the risk of the project significantly.

As explained in Section 6.8, the advertisement is ought to be readable and recognisable by having a sufficient resolution and brightness level. Therewith, requirement **BBIS-Ads-03** and **BBIS-Ads-04** are satisfied. Furthermore, requirement **BBIS-Sys-A01-2** from [3] states which altitude should be kept with which precision, but no values were given at that time. The altitude of the spacecraft should be kept at $35\ 786\ km$ with a precision of $100\ m$ to avoid collisions.

5.5. ΔV-Budget

Several forces act on the spacecraft, slowly perturbing its orbit. To account for them, a certain ΔV budget is needed. Section 5.5.1 discusses the method to compute those forces and their ΔV. Subsequently, Section 5.5.2 gives an overview of all perturbations and a total budget.

5.5.1. Method

In this section, manoeuvres and orbit perturbations that require ΔV are addressed.

Non-Spherical Mass Distribution

The Earth is not an exact sphere, but is shaped more like an oblate spheroid. This is one of the reasons that the orbit of the spacecraft is not a perfect Kepler orbit. The biggest effect is an East-West shift for most orbits, which is caused by the equatorial bulge and flattening at the poles. This is also known as the so-called J_2 effect, referring to the second coefficient for zonal harmonics. However, for a GSO this effect is already accounted for by increasing the semi-major axis by $2\ km$, compared to the theoretical geosynchronous semi-major axis [6].

[4] An animation of both formations is made and can be found on Youtube with https://youtu.be/MbyNJExcuVo.

[5] URL https://www.ucsusa.org/nuclear-weapons/space-weapons/satellite-database#.Wyd3zi17E_U [cited 18 June 2018].

Although the J_2 effect does not affect the orbit, it has a slightly different effect on orbits with a small difference in inclination. Therefore, the spacecraft within the swarm will experience a movement relative to one another as a result of the J_2 effect. The right ascension of ascending node Ω and argument of perigee change ω at a certain rate can be calculated using Equation 5.5 and Equation 5.6 [7]. In these formulas, n is the mean motion in $°/day$, R_E is the Earth's equatorial radius, a is the semi-major axis in km, e is the eccentricity, i is the inclination, and $\dot{\Omega}_{J_2}$ and $\dot{\omega}_{J_2}$ are in deg/day.

$$\dot{\Omega}_{J_2} = -1.5\ n\ J_2\ (R_E/a)^2 (cos(i))(1-e^2)^{-2} \tag{5.5}$$

$$\dot{\omega}_{J_2} = 0.75\ n\ J_2\ (R_E/a)^2 (4\ -\ 5sin^2(i))(1-e^2)^{-2} \tag{5.6}$$

For some spacecraft in the swarm, the inclination is slightly different. By using orbital parameters and subtracting the rates of change of the spacecraft with the biggest differences in inclination can be computed. Equation 5.7 and Equation 5.8 determine differences in the rate of change. The difference in rate of change is given in $arcsec/year$.

$$\Delta\dot{\Omega}_{J_2} = -4.81(cos(i_2) - cos(i_1)) \tag{5.7}$$

$$\Delta\dot{\omega}_{J_2} = -9.61\ + 12.0(sin^2(i_2) - sin^2(i_1)) \tag{5.8}$$

Filling in these equations leads to a total relative movement of $40\ arcsec/year$. This means that the spacecraft will move approximately $22\ m$ relative to each other every day in the most extreme case. Since the distance between these spacecraft is $1.91\ km$, this should be accounted for by performing small orbit correction manoeuvres.

Next to the well-known J_2 effect, the non-spherical mass distribution causes another significant perturbation to spacecraft in GSO. The out-of-roundness of the Earth's equator causes an East-West drift. This effect can be computed using the J_{22} sectoral term. If this effect is not corrected, it will result in a sinusoidal motion about either of two stable longitudes at $75°E$ and $105°W$. The approximate amount of ΔV in m/s can be computed by using Equation 5.9, where l_D is the desired longitude and l_s the nearest stable longitude. [5]

$$\Delta V = 1.715\ sin(2|l_D - l_s|) \tag{5.9}$$

Third Body Interactions

The gravitational forces of the Sun and the Moon have an influence on the orbit of the spacecraft. As a result, the longitude of ascending node Ω changes, as well as the argument of periapsis ω and the inclination i. These changes are undesirable, but the change in inclination is negligible for the spacecraft's performance. The inclination will vary periodically between 0 and 15 over a period of approximately 15 years [5]. Although this is visible from Earth, it does not decrease the visibility of the spacecraft. However, changes in Ω and ω have to be counteracted to keep the spacecraft at the longitude of the USA and to prevent collisions with other spacecraft in GSO. The rates of change of Ω and ω can be found in Equation 5.10, 5.11, 5.12 and 5.13 [5][7].

$$\dot{\Omega}_{MOON} = -0.00338(cos(i))/N \tag{5.10}$$

$$\dot{\Omega}_{SUN} = -0.00154(cos(i))/N \tag{5.11}$$

$$\dot{\omega}_{MOON} = -0.00169(4 - 5sin^2(i))/N \tag{5.12}$$

$$\dot{\omega}_{SUN} = -0.00077(4 - 5sin^2(i))/N \tag{5.13}$$

In these equations, the rates of change are computed in $°/day$ and N is the number of orbits per day. Since the inclination is not constant during the lifetime, the worst rate of change is determined, which for all formulas is at an inclination of $0°$. Furthermore, it should be noted that not all spacecraft have the same inclination in one of the two formations. This causes the spacecraft to move relative to each other. However, this effect is only $37\ mm$ per day, which is negligibly small compared to the relative motion resulting from the J_2 effect ($22\ m$ per day).

Solar Radiation Pressure and Atmospheric Drag
The solar radiation pressure and atmospheric drag can be computed using Equation 5.3 and Equation 5.4, respectively [5] [7]. The perturbations as a result of these two forces are computed using the simulation. The ΔV needed to counteract the atmospheric drag is relatively low compared to other elements in the ΔV budget; 0.00051 m/s per year.

Due to the exceptionally high area over weight ratio, the solar radiation pressure acceleration is considerably higher for this spacecraft than for other spacecraft orbiting in GSO. However, the average force over the period of one year is exactly zero as a result of the Sun rotating 360 degrees with respect to an Earth-centred frame. Therefore, the perturbation is a periodical change of apocentre and eccentricity and does not have to be countered [5]. All spacecraft are accelerated in the same manner since this perturbation is not dependent on inclination or longitude of ascending node, so the formation is not in danger.

Other ΔV-Requiring Manoeuvres
All spacecraft are inserted into GSO by the launcher. However, the launcher has a certain accuracy and it is therefore possible that a correcting manoeuvre has to be performed to reach the exact required orbit. Section 12.1.3 specifies that the worst-case deviation is an apocentre difference of $40\ km$ and an inclination difference of $0.020\ deg$. The Hohmann-transfer and inclination change that are used to take the spacecraft to the desired orbit need approximately $2.9\ m/s$ and $1.1\ m/s$ of ΔV respectively.

The spacecraft have to be transferred to a graveyard orbit at the end-of-life. It is decided to use $34\ m/s$ of ΔV for the first part of the disposal. More information about the disposal can be found in Section 12.3.2.

Two possible formations are possible with the 900 spacecraft. In order to optimise the versatility of the advertisement, changes between those two formations should be possible during the mission. Assuming that all spacecraft start the mission in one straight line, it takes an average of $0.8/m/s$ of ΔV per spacecraft to create the formation with different inclinations. Additionally, it takes an average of $7.0/m/s$ of ΔV per spacecraft to create the formation with different longitudes of ascending nodes. To change formation, all spacecraft first need to go back to one straight line, which also required ΔV. If both formations are used 3 times, this results in a total required ΔV of $39.4m/s$.

5.5.2. Results
By combining all formulas and values from Section 5.5 and 5.5.1, a total required ΔV is obtained. An overview is presented in Table 5.1. As can be seen, the total required ΔV during the lifetime of the spacecraft is $289\ m/s$. Herewith, requirement **BBIS-Sys-SA02-01.1** is met.

	Required ΔV [m/s]	
	Per Year	**Total**
Non-Spherical Mass Distribution Earth	0.29	5.72
Third Body Interactions	10.20	204.00
Orbit Insertion Correction	-	3.99
End-of-Life	-	34.00
Changing formation	-	39.40
Total	**10.50**	**287.00**

Table 5.1: ΔV Overview per Year and Total Lifetime of a Spacecraft.

5.6. Visibility Analysis
To get an estimate for the number of views the BBIS will have, a simulation was created. The method is explained in Section 5.6.1 and the results follow in Section 5.6.2

5.6.1. Method
For analysis and optimisation of the visibility, another simulation was created. In Section 12.1.3 it is determined that the solar sail has a beam angle of $0.618°$. This means that the spacecraft is visible from

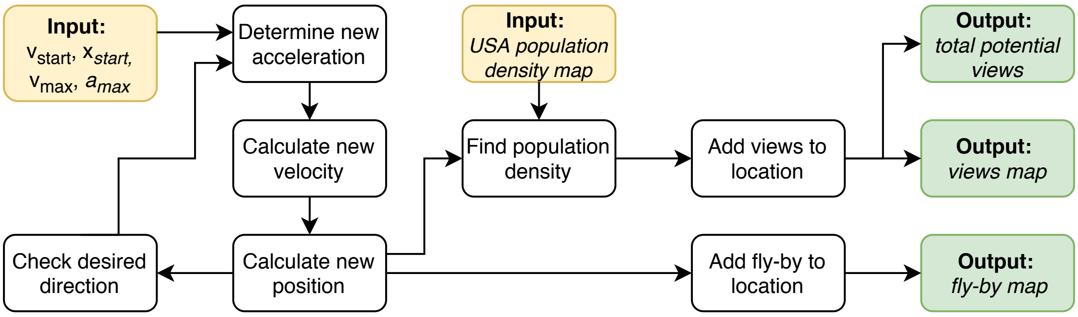


Figure 5.7: Visibility Analysis Program Structure.

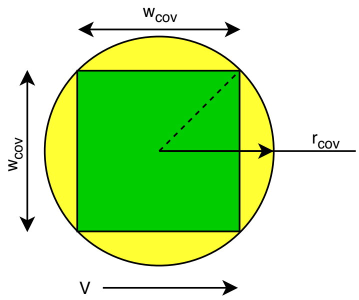


Figure 5.8: Covered Area Dimensions.

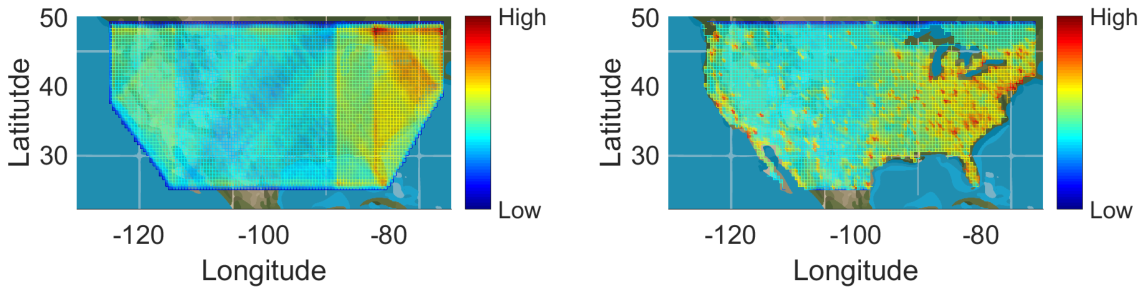
a circular area on the ground with a radius of $193\ km$ (the coverage radius r_{cov}). To meet requirement **BBIS-Sys-A03-2**, "the spacecraft shall be visible for at least $14\ s$ per flyover", the rotational rate of the spacecraft is limited. The speed of the moving covered area should be such, that every person is in the covered area for at least $14\ s$.

The covered area is a circle, but that means that people located at the top or bottom part of the circle will see the spacecraft for a shorter time than people located in the middle part, as Figure 5.8 illustrates. It is therefore decided that only a square area within the circle is considered as the covered area. Geometry shows that the width of the covered area, $w_{cov} = 2 \cdot 1/\sqrt{2} \cdot r_{cov} = \sqrt{2} \cdot r_{cov} = 273\ km$. This yields that the maximum ground speed of the covered area can be $273/14 = 19.49\ km/s$.

All spacecraft point their mirrors in such a way that they are scanning the USA; the groundpath of the covered area eventually covers the entire USA. To minimise the required power for these rotations, the groundpath goes from one border to the other border and then turns around. It is decided that this turn-around manoeuvre should take a maximum of $20\ s$, since this is an ineffective time use due to the lower ground speed. To optimise the number of views, more focus is put on the coastal (more densely populated) areas of the USA. This is also visible in Figure 5.9a. The current optimisation for groundpath is just a preliminary program and can be more detailed in later stages of the design. The current program structure can be found in Figure 5.7.

5.6.2. Results

By combining the ground path location with a population density map of the USA, the number of potential views can also be computed. This is the number of views in an idealised case; people can see the spacecraft 24 hours per day and everyone in the covered area sees the spacecraft. Although these assumptions are not valid, the obtained number of potential views can be used to compute a more realistic value for the number of views. This is done in Section 16.4. Figure 5.9b shows the locations with a relative high and low number of potential views.



(a) The Number of Fly-Bys.

(b) The Number of Potential Views.

Figure 5.9: Visibility Analysis Results.

5.7. Verification

In order to use this program, it is verified. First, the orbit propagator is verified in Section 5.7.1. This is mainly a recap of the verification of [8]. Secondly, the formation flying code is verified in Section 5.7.2. Finally, the visibility analysis is verified in Section 5.7.3.

5.7.1. Orbit Propagator

First the discretaion error is investigated, the discretisation error is $\pm 37.136\ km$ and $\pm 0.367\ km$ for a time step of $10\ s$ and $0.1\ s$ at an altitude of $1000\ km$. In [8] the method is described how the discretisation error is determined.

Next, the simulated orbit time is compared to the analytical orbit time using the formula for Kepler's orbit. This results in an orbit time error of $0.01\ s$ and $1.03\ s$ for a time step of $0.1\ s$ and $10s$ at an altitude of $1000\ km$, respectively. Again, in [8] the methodology is explained.

Furthermore, the direction of the forces are verified by vector calculations. The dot product between two vectors is mainly used. This ensures that a vector is either tangent or perpendicular to another vector. The directions of the forces are listed below.

- **Radiation Pressure Drag** Perpendicular to the solar sail.
- **Aerodynamic Drag** Opposite to the velocity vector.
- **Gravity Force** Pointing to Earth's centre

A more elaborated explanation about the verification method can be found in [8].

The new input of the rotation of the Sun is verified by running the program for a whole year. The spacecraft should end up in more or less the same orbit as it started, as explained in Section 5.5. Furthermore, the position of the Sun is plotted throughout a whole year. This results in a circular plot with a radius of $1\ AU$.

5.7.2. Formation Flying

Formation flying is verified by comparing different inputs for both formations. First, it os checked if the spacecraft fly at the right spacing. The total area the spacecraft occupied have to equal the requirement set for their apparent size.

Second, the transformation for the different ascending nodes is checked. The orbits are transformed with $-360, -180, 0, 180$, and $360°$. A transformation with $-360, 0$, or 360, results in the same orbit, as expected. The transformation with -180 and $180°$ result in a orbit with seemingly an opposite inclination, which is also as expected.

5.7.3. Visibility Analysis

The visibility analysis determines the amount of views based on the groundpath location and covered area of the advertisement, and the population density in the USA. The covered ground area is checked to be zero when the beam angle is equal to zero degrees. The beam angle is also checked to be a value close to half of the Earth's surface area when the angle equals $180°$.

In order to verify that the population density is correctly implemented to calculate the total views, two unit tests are done. Firstly, the population density is put to zero everywhere. The result must be that the total number of views equals zero. Secondly, the population density is set to be uniformly distributed everywhere. This results that the number of fly-bys plot is exactly the same as the number of potential views plot. This means that Figure 5.9a is equal to Figure 5.9.

Another unit test is to run the visibility analysis for the whole USA without focusing on the coastal areas, which would mean that a random ground path is followed. Equation 5.14 should be true and the difference should be small.

$$Pop_{avg} \geq \frac{Total\ views}{t_{sim}/t_{view}} \tag{5.14}$$

Where Pop_{avg} is the average population density of the USA, $Total\ views$ is the total views determined by the program and t_{im} is the simulated time in s and t_{view} represent the time required per view, which is $14\ s$. Increasing the simulation time and/or the angular acceleration to scan, should decrease the difference.

5.8. Validation

The main program consist of the orbit propagator, formation flying and the visibility analysis. It is not possible to validate all three parts at once, because there has not been a similar mission. Therefore, the validation has to be done for each part of the code separately. The code for the visibility analysis cannot be validated, because there are no spacecraft with a similar mission.

To validate the orbit propagator, the simulation needs to be compared with different spacecraft where data is available of their position over time. The current orbit propagator takes aerodynamic drag, radiation pressure and the Earth's gravitational force into account.

To ensure that the aerodynamic drag is implemented correctly, the program result needs to be compared with a spacecraft orbiting in a low Earth orbit. To ensure that the radiation pressure is implemented correctly, the simulation can be compared with a solar sail mission, such as IKAROS.

The Earth's gravitational force does not have to be validated, as verification with a Kepler model is sufficient. However, one more validation process is needed to validate the orbit propagator. The simulation needs to be compared with a spacecraft in a GSO. This will show what the effect of other forces, which are not taken into account, influencing the orbit. An example is the gravitational force of the moon.

It is harder to validate formation flying, because there have only been a couple of missions with spacecraft in a formation, for example, the PROBA and Magnetospheric Multiscale mission. The simulation can be reduced to two spacecraft which are flying in formation. There are two main things to consider when comparing the simulated data with the mission data. Firstly, the results of one orbit have to be looked into. If they are similar, it validates that the current code is correct. Secondly, the long-term data has to be considered. In the simulation, it is assumed that the formation does not change over time. However, the mission data can show a different result. This might be because not all the forces are taken into account.

6. Payload

In this chapter the design of the payload is discussed. In Section 6.1 the requirements applicable to the payload are discussed. Next, in Section 6.2, the material choice is elaborated on. The booms are highlighted in Section 6.3, after which the area of the payload is described in Section 6.4. Following, in Section 6.5, the deployment mechanism is elaborated on. The design of the curved surface is given in Section 6.6, after which additional design choices are stated in Section 6.7. A summary of all the payload design characteristics is given in Section 6.8. The chapter is concluded with a sensitivity analysis in Section 6.9 and verification and validation in Section 6.10.

6.1. Requirements

The following requirements all have an influence on the design of the payload.

- **BBIS-Ads-02** The perceived billboard light intensity shall at least be equal to the intensity of a full moon.
- **BBIS-Ads-04** The advertisement shall be recognisable from Earth.
- **BBIS-Sys-A02-1** The spacecraft shall provide an illumination at Earth of at least $0.13\ lx$.
- **BBIS-Sys-A04-1** The light source shall be focused on Earth's surface.

Requirement **BBIS-Sys-A02-1** follows from requirement **BBIS-Ads-02**, as explained in [8]. After more elaborate calculations it turned out the minimal illumination was $0.13\ lx$. instead of the $0.108\ lx$ used up till now. Thus, the value in this requirement is modified.

6.2. Material

The material of the solar sail must highly reflectively to ensure visibility on Earth. With current techniques, it is not possible to manufacture a solar sail consisting of only one reflective material. That is why, in former missions, coatings of reflective film are applied to strong, lightweight materials, called substrates.

6.2.1. Substrates

Substrates are the main contribution to the rigidity and mass of the solar sail. They are solely present to support the folding and deployment of the reflective film. Polymers are used as substrates due to the fact that they can be formed into difficult shapes and are very tough. They are divided into two groups, aliphatic and aromatic polymers. Aliphatic polymers contain carbon chains and can be cyclic, but they do not contain a benzeen ring. They are sensitive to ultra violet (UV) and infrared (IR) radiation and are therefore not suitable for the substrate of the reflective material. The second group are the aromatic polymers, which contain a carbon chain with benzeen rings present. Due to this benzeen ring, the polymer can store electric charges generated by radiation hitting the substrate and is therefore more resistant to radiation. Based on these properties, an aromatic polymer is chosen to be the substrate of the BBIS solar sail. [9]

The substrate needs to provide tensile strength such that the sail does not fail when it is in its deployed state and under tension. The glass transition temperature should be as high as possible, to ensure a wide range in temperature in which the substrate can function. If the temperature of the substrate is above the glass transition temperature, the substrate properties decrease, meaning that it is not able to withstand the tension in the sail. A higher glass transition temperature is therefore preferred. Three different aromatic polymers used in space are Mylar, Lexan and Kapton. Mylar has a high tensile strength, but a very low glass transition temperature and its UV life is not good when considering long term missions. Lexan has a low density, but a very poor UV life and low tensile strength. That is why Kapton from DuPont USA is chosen. It has a good resistance to radiation and high tensile strength. It has a density of $1.42\ g/cm^3$ and a high glass transition temperature of $360\ ^\circ C$, which is useful when the solar sail is in sunlight. The thickness of the substrate is limited by the fabrication, it is preferably as thin as possible. Currently, Kapton is fabricated in thicknesses of $7.6\ \mu m$. [10],[9],[11].

6.2.2. Reflective Film

For the substrate to become reflective, it must be coated with a reflective film. This film is usually a very thin layer of metal with a high melting point and low density. Furthermore, it should reflect in the visible part of the radiation spectrum. In addition to this, it is preferably also reflective in the other wavelengths as well, to prevent degradation of the substrate as much as possible. Suitable films are made of lithium, silver and aluminium. Lithium has a very low density compared to the other two, but also a relatively low melting point. Silver has a very high melting point, but also a high density. It reflects the visible wavelengths very well, but is almost completely transparent in the UV part of the spectrum. This leaves aluminium density of $2.70\ g/cm^3$ and a melting point of $933\ K$ [11] to be the most suitable materia. A very thin layer of $0.1\ \mu m$ is required to reach a high enough reflectivity. Aluminium has a reflectivity of 0.91[1] for the visible part of the spectrum and is also reflective in the UV part of the spectrum.

6.3. Booms

Due to the fact that the solar sail is a relatively new technology and that there is no previous application of the use of a solar sail as a reflective surface, limited information is known. Therefore, several assumptions are made.

- The dimension of the boom and the solar sail are all designed based on the GSO environment, implying that the thermal influence has been considered.
- The booms are designed in such a way that the solar sail is fully extended and the tension on the boom is negligible.

It is chosen to use the ultra-light Carbon-Fibre Reinforced struts (CFRP booms) developed by the German Aerospace Center (DLR) [1] for the boom structure due to its light weight per meter (approximately $0.1\ kg/m$) and the fact that it is rollable. The maximum tested boom length equals $14\ m$, which is therefore the maximum length constraint for this design. Moreover, it has been determined from the design report of the boom that the structure has a critical bending moment of $M_{x,cr} = 81.8\ Nm$ and a $EI_x = 5298\ Nm^2$ [1]. The analysis results derived by DLR are shown in Figure 6.1.

The solar radiation pressure at the Earth's distance from the sun $(4.5\mu\ Pa)$ [12], which generates quite little force to the solar sail and the tension of the solar sail on the boom structure is negligible. Therefore, the only aspect needed to consider is the thrust force on the boom. From Section 7.5, it can be derived that each thruster generates $10\ N$ of thrust. It is analysed that the critical deformation of the boom by thrust force is caused by the $Th5$ and the critical bending moment on the boom is caused by $Th3$ or $Th4$. The position of each thruster can be found in Table 7.3.

When the thrusters on the spacecraft are switched on, the bus and payload are accelerated. As a result, specific loads act on the centre of gravity of each boom. The loading situation on the boom can be simplified to a cantilever beam, whose maximum deflection function at any point of the beam is shown in Equation 6.1 [13].

$$\delta_{max} = \frac{Fl^2}{6EI}(3l - a) \tag{6.1}$$

In Equation 6.1, with the force calculated from the solar radiation pressure of $4.5\mu\ Pa$, the solar sail area of $406\ m^2$ and a boom length of $14\ m$, there is a maximum deflection $\delta_{max} = 0.315\ mm$ at the tip of the boom, which is small enough to be neglected. F is the load in N caused by the acceleration, which is acting at the centre of gravity of the boom. l is the length of the beam in m, I is the moment of inertia of the cross-section in $kg \cdot m^2$ and E is the elastic modulus of the material in GPa. Due to the fact that the mass of the solar sail sheet is significantly lower than the boom, the influence of the solar sail is relatively low and could be neglected.

In Section 7.2, all possible torques the spacecraft undergoes are listed and analysed. By comparing with the experimental data $M_{x,cr}$ in Figure 6.1 with the coordinate system in Figure 6.2, the booms are stiff enough to encounter the loads. Moreover, due to the different elasticity of the materials and the fact that only one side of the solar sail is covered with aluminium, there is a compressing load at

[1] URL https://laserbeamproducts.wordpress.com/2014/06/19/reflectivity-of-aluminium-uv-visible-and-infrared/ [cited 6 June 2018].

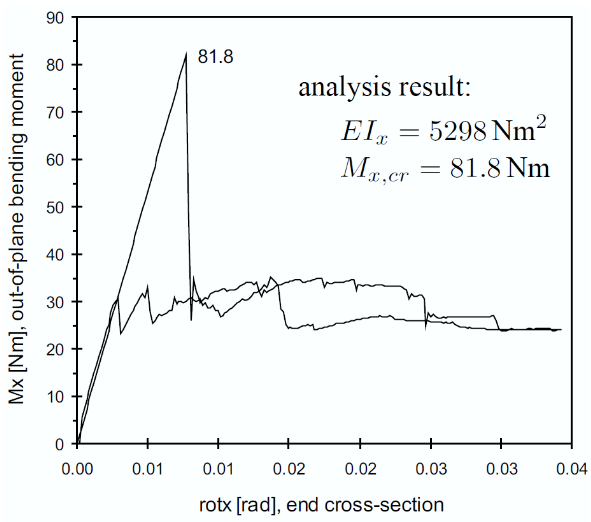


Figure 6.1: Explicit Finite Element Analysis of the Buckling Behaviour According to Uniaxial Bending M_x [1].

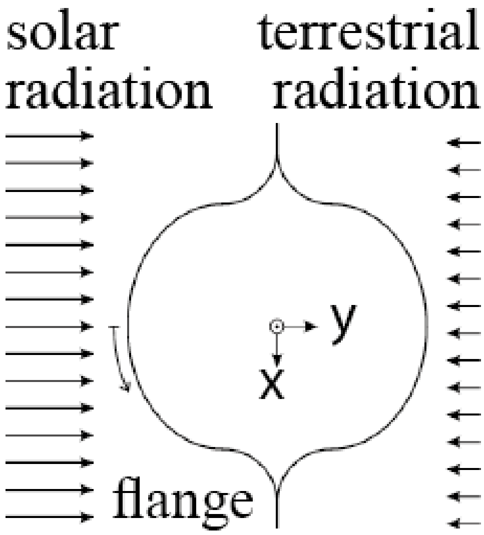


Figure 6.2: Coordinate System Used during Experiment of the Boom [1].

the root and the tip of the boom. Based on the research report of DLR, the critical load in longitudinal direction of the boom is $2000\ N$[14].

There is no extra pointing mechanism for the solar sail, and the direction changing is done by the AODCS, which is explained in detail in Section 7.4.

6.4. Area

Now that the material is chosen, the solar sail area only depends on the altitude and the beam angle. The former is chosen in Chapter 5, so that only the beam angle is variable. The beam angle is the angle over which the light spreads when it hits a surface. In Figure 6.8 the effects of a beam angle are displayed at zero beam angle, a narrow and a wide beam angle, respectively. It can be seen that the intensity of the light decreases as the beam angle increases. This has been taken into account while computing the area of the solar sail.

It is preferable to have the biggest area possible in order to have better visibility performance as well as larger beam angle because this is easier to manufacture. From the structures and payload perspective, due to the low forces on the sail, the only area constraints are the booms that cannot be longer than $14\ m$, as indicated before. Using these booms, the biggest possible area that can be constructed is $406\ m^2$. This the most optimal payload surface area possible, and is therefore chosen for BBIS. This area has a beam angle of $0.618°$. This angle is formed by the payload, it is calculated by a programme developed to meet the $0.13\ lx$ visibility requirement, which makes it possible for people on Earth to see and read the BBIS. Therefore, requirement **BBIS-Ads-02**, **BBIS-Sys-A02-1**, **BBIS-Ads-03** and **BBIS-Ads-04** are met. The mass of the sail itself is calculated to be $6.637\ kg$.

6.5. Deployment Mechanism

During launch, the solar sail is folded and rolled with the method shown in Figure 6.6, the booms are folded flat and rolled in the deployers. The main concept of deployment mechanism consists of an actuator that spins out the boom and the solar sail, during which the booms swell to the shape shown in Figure 6.2 when they are extracted out of the deployers. The boom attached to the solar sail corner at the tip is extracted from the deployer located at each corner of the bus to extend the whole solar sail.

The boom deployment mechanism is chosen to have the same CFRP boom deployer as the one developed by University of Surrey[2] for the DEPLOYTECH Project[3], which have a mass of $408\ g$ each.

[2]URL https://www.surrey.ac.uk/ssc/research/space_vehicle_control/deorbitsail/files/gsf_keynote_2012.pdf [cited 19 June 2018].
[3]URL https://cordis.europa.eu/project/rcn/101853_en.html [cited 19 June 2018].

From the DLR technical report of the CFRP booms [1], although the Earth orbit deployment demonstration has not been done yet, the boom deployment mechanism has been successfully tested on ground. Moreover, there have been many successful missions of solar sail deployment with solar sail, therefore it is assumed that the deployment of the CFRP boom with solar sail is applicable.

Each corner of the solar sail is attached at the tip of the boom. Similar to the way a curtain is open and closed, the edge of the solar sail is attached onto rings that move along the boom. Each ring has the same shape of the boom cross section. The size of each ring is slightly bigger than the boom in order to slide freely on the boom. There are three rings on each boom, each connected by strings when fully deployed with a distance of $3.5\ m$ in between each ring. The furthest one is $3.5\ m$ away from the tip, enabling the solar sail to stay in a certain shape after deployment.

There are three ropes attached to each boom; one is connected to the three rings and the tip of the boom, the other two wires go through the holes on the ring and are attached to the solar sail. As shown in Figure 6.3, at the beginning of the deployment, solar sail is fully folded next to the deployer and the rings are all locating at the exit of the deployer. In Figure 6.4, when the boom spins out the deployer, the rings are pushed out by the rope attached at the tip of the boom and the folded solar sail is extended by the rope going through the holes on the boom and attached at the tip of the boom. The fully extended solar sail is shown in Figure 6.5.

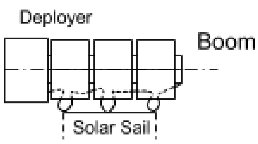


Figure 6.3: Beginning of the Solar Sail Deployment.

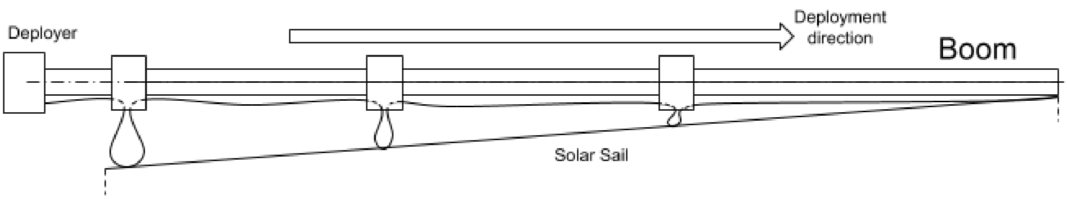


Figure 6.4: During Solar Sail Deployment.

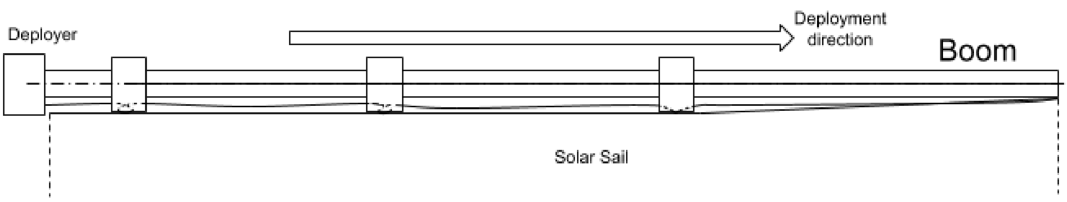


Figure 6.5: Fully Extended Solar Sail.

A square solar sail is chosen in order to have maximum solar sail area as well as desired shape. Four triangular solar sails with a specific curvature constitute the whole sail, as shown in Figure 6.9. The modified spiral folding pattern is chosen, depicted in Figure 6.6 [15]. The sail is cut into four pieces as shown in Figure 6.7, in which the blue squares indicate the distribution of solar cells, which are elaborated on in Section 6.7.2. Each solar sail is folded 72 times evenly with the same folding method as in Figure 6.6.

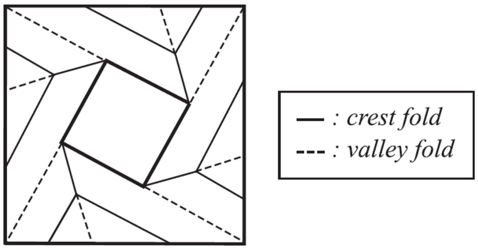


Figure 6.6: Folding Pattern of the Solar Sail.

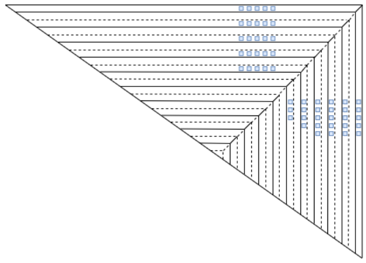


Figure 6.7: Configuration of Each Piece of the Solar Sail with Solar Cells.

As investigated in Section 6.3, the load from solar radiation pressure is negligible. Furthermore, it is assumed that the booms are deployed steadily with constant low speed. Therefore the stresses and forces acting on each of the ring are neglected. Each ring is chosen to be made of Aluminium 2024-T3 due to its low density and relatively high strength, which has an density of $2.78\ g/cm^3$[4]. The resulting ring weight can thus be derived to equal $0.12\ kg$.

For the material of the rope, nylon is chosen due to its relatively high specific strength[5] and low density. The nylon wire has a minimum breaking strength of $6610\ kN$ and a linear density of $0.023\ kg/m$[6] such that it is able to withstand the tension of the solar sail as well as the tension between the root and the tip of the boom. After calculation, a total wire length of $154\ m$ of is needed, resulting in $3.542\ kg$ of wires.

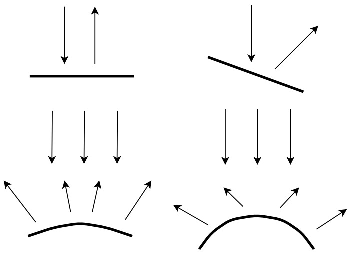


Figure 6.8: Reflection Behaviour of Light.

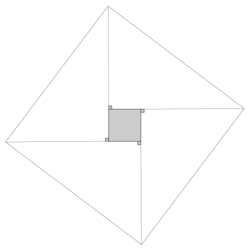


Figure 6.9: Solar Sail Configuration.

6.6. Design of Curved Surface

In the previous section it was indicated that a very small beam angle of the reflective surface is not preferred. Up until now, the solar sail was assumed to be a flat plate, with a small beam angle that could be changed into something that suited the design. In reality, a solar sail is harder to adjust due to the structure needed for a curved reflective surface and the compact storage of this structure. Wrinkles are formed in the structure due to the folding of the sail, or by the absence of tension in the sail. When radiation hits the wrinkle, the light is not scattered back in the direction the sail is pointed in, but it is diffused. This causes the intensity of the radiation to be less. That is why a curved, smooth surface of the solar sail is preferred. An easy way to construct a curved surface is to divide a flat surface into five pieces, each piece is a fully extended flat sail part and there is angle difference between them in order to simulate a curved surface. These angle difference are achieved by attaching the solar sail at different locations on the rings that slide over the boom.

6.7. Additional Design Choices

This section elaborates on two additional design choices that are made concerning the payload. Firstly, the use of reflectivity control devices is discussed in Section 6.7.1, after which the application of solar

[4]URL http://asm.matweb.com/search/SpecificMaterial.asp?bassnum=ma2024t3 [cited 19 June 2018].
[5]URL http://www-materials.eng.cam.ac.uk/mpsite/interactive_charts/spec-spec/NS6Chart.html [cited 19 June 2018].
[6]URL https://www.engineeringtoolbox.com/nylon-rope-strength-d_1513.html [19 June 2018].

cells on the solar sail is elaborated on in Section 6.7.2.

6.7.1. Reflectivity Control Devices

To be able to control the displayed advertisement and switch pixels off, not all the spacecraft reflect sunlight to the USA at the same time. In order to make a pixel appear 'black', the pixel must not reflect sunlight towards the USA. This can be controlled by thin-film reflectivity control devices (RCD), which can influence the reflectivity of the solar sail. They have been successfully used on the IKAROS (Interplanetary Kite-craft Accelerated by Radiation Of the Sun) mission from The Japan Aerospace Exploration Agency (JAXA) and they consist of electrodes in a sandwiched structure with crystals in between. The orientation of the reflection can be changed by applying voltage to the electrodes, such that the reflectivity of the RCD is changed from unidirectional to omnidirectional.This results in a lower intensity in light and thus the particular solar sail is not visible from Earth anymore. [7]

The scattering of light when using RCDs behaves differently, as indicated before. To compute the needed area of RCDs, assumptions are made. If the RCD is switched on, the sunlight diffuses over $180°$, while normally it is only scattered over an angle of $0.618°$ as indicated before. This means that only 0.3% of the light is scattered in the right direction when the RCD is switched on. The intensity of the light thus also decreases to 0.3% of its original value.[16] To make the solar sail invisible, it does not have to decrease down to this percentage. The total formation has the same visibility of the moon, namely $0.13\ lux$, as indicated before. That means one spacecraft is $0.000144\ lux$. To switch a pixel off, it is not going to be completely invisible, the RCDs are not able to provide this decreased illuminance. They are, however, capable of decreasing the illuminance so that the pixel is as bright as Vega, the 5^{th} brightest star in the sky. Vega has an illuminance of $2.13 \times 10^{-6}\ lux$.[17] Lux is equal to lumen per square meters, so together with the percentage of light that is scattered in the right direction, this gives the required RCD area. This area is found to be $400\ m^2$, so almost the complete solar sail. These RCDs are specifically manufactured per mission and therefore, no mass is known. It is assumed that the mass of the RDCs is negligible with respect to the mass of the solar sail itself. The RCDs have a length of $1.0\ m$, a width of $0.25\ m$ and a thickness of just $50\mu m$[16].

6.7.2. Solar Cells

To generate as much energy as possible, thin-film solar cells (TFCS) are mounted onto the solar sail. This configuration is preferred over separate solar arrays with their own deployment mechanism because these add the need for mechanisms that are prone to failure and also add mass. TFSC are used due to their flexibility and low mass. The TFSCs are connected to the solar sail via silicone. The actual selection of the solar cells is highlighted in Chapter 9.

6.8. Final Design

The final design of the payload of BBIS consists of a solar sail that reflects light to Earth. The square solar sail is made of $7.6\ \mu m$ Kapton and $0.1\ \mu m$ aluminium. Four booms divide the solar sail into four triangular parts. On each boom, three rings are present. These rings function as an attachment during the deployment of the solar sail and also force a required curvature into the sail. Wire is connected to the tip of the boom, looped through each ring, and connected to a motor on the spacecraft bus. To provide tension in the sail, the wire is put in tension as well. On the sail, RCDs and solar panels are present, to influence the reflectivity and provide the spacecraft power.

6.9. Sensitivity Analysis

If the length of each boom decreases, the solar sail area decreases as well as the beam angle in order to still meet the visibility requirement. If the solar sail area decreases 10%, the beam angle decreases approximately 5.14%.

If the required illumination at Earth increases 10%, either the solar sail area needs to increase 10.03% or the beam angle of the solar sail needs to decrease 4.67% to meet the requirement.

Moreover, smaller solar sail also decreases the mass moment of inertia of the whole structure, which makes it easier for orientation and orbit control. However, smaller solar sail requires even smaller beam angle, which is more difficult to achieve and not favourable.

[7]URL https://directory.eoportal.org/web/eoportal/satellite-missions/i/ikaros [cited 19 June 2018].

6.10. Verification & Validation

The solar sail is designed typically to meet the visibility requirement. In Section 6.4, the beam angle is calculated, which determines the curvature of the solar sail. The verification of the solar sail is done by verifying the programme used for the calculation. First the errors in the programme file is checked, then the relation between beam angle and the solar sail area is plotted. From the graph, the beam angle is zero when the solar sail area is zero and the beam angle do not exceed $180°$ when the area goes to infinity. Both of them are reasonable, therefore the solar sail is verified.

The validation of the payload contains the successful deployment of the booms and the solar sails from fully folded condition, which are relatively straightforward and easy to validate with a deployment test on ground. Similar deployment tests have been done by NASA Marshall Space Flight Center for NanoSail-D [18] and LightSail-A [19].

The boom chosen is developed by DLR, which has been verified and validated by DLR. Moreover, the validation of the spacecraft performance in space is explained in Chapter 18.

Verification and validation must also be performed on the use of RCDs. Since it is not a technique that has been used a lot, detailed investigation needs to be performed. Validation can be done with the pioneer spacecraft of BBIS, as is explained in Chapter 18. As this turns out to be unfeasible, the attitude determination and control system can be used to turn the spacecraft to switch the pixels off and on. This, however, will alter the current design and is therefore not preferred.

II
Subsystem Design

7. Position and Attitude Control

The attitude and orbit determination and control system (A&ODCS) is crucial for the stabilisation, orientation and navigation of the spacecraft during its mission. It enables the spacecraft to maintain the desired position and direction despite external disturbance torques acting on it. Furthermore, it allows the spacecraft to manoeuvre in order to obtain the correct orientation. In this case, it is important that the BBIS's payload maintains a stable orientation such that the billboard is directed towards Earth. Furthermore, the correct orbit must be maintained. First, the subsystem requirements are listed in Section 7.1. Then, the disturbances are discussed in Section 7.2. Then, the subsystem's sensors are elaborated on in Section 7.3, followed by the actuators in Section 7.4. The the verification and validation process of the the position and attitude control system is described in Section 7.7.Finally, sensitivity analysis of the subsystem is discussed in Section 7.6.

7.1. Requirements

The requirements with respect to the position and orientation of the spacecraft were established in [3]. The relevant requirements for the design of BBIS are listed below.

- **BBIS-Sys-A01-2** The spacecraft shall be able to maintain an altitude of $34\ 786\ km$ with a maximum deviation of $100\ m$.
- **BBIS-Sys-A03-1** The orientation of the spacecraft shall be controlled with a precision of $0.0618°$.
  - **BBIS-Sys-A03-1.1** The orientation of the spacecraft shall be determined with a precision of $0.0618°$.
  - **BBIS-Sys-A03-1.2** The spacecraft shall be able to adjust its orientation with a precision of $0.0618°$.
- **BBIS-Func-Att-02** The spacecraft shall be able to determine the relative position of the sun relative to the spacecraft with a precision of $0.0618°$ in 3-axis directions.
- **BBIS-Func-Att-03** The spacecraft shall be able to determine the relative position of the ground station relative to spacecraft with a precision of $0.0618°$ in 3-axis directions.
- **BBIS-Func-Att-04** The spacecraft shall be able to determine the direction of the Earth centre relative to spacecraft with a precision of $0.0618°$ in 3-axis directions.
- **BBIS-Func-Att-07** The spacecraft shall be able to accelerate around its 3 axes individually.
  - **BBIS-Func-Att-07.1** The spacecraft shall be able to accelerate around the body's x-axis with $5.42 \cdot 10^{-5}\ rad/s^2$.
  - **BBIS-Func-Att-07.2** The spacecraft shall be able to accelerate around the body's y-axis with $3.98 \cdot 10^{-9}\ rad/s^2$.
  - **BBIS-Func-Att-07.3** The spacecraft shall be able to accelerate around the body's z-axis with $5.48 \cdot 10^{-5}\ rad/s^2$.

The requirements **BBIS-Sys-A01-2**, **BBIS-Sys-A03-1**, **BBIS-Func-Att-02**, **BBIS-Func-Att-03** and **BBIS-Func-Att-04** are equal to those stated in [3]. The accuracy of the determination systems is based on the payload's pointing requirement. The reflective surface's beam angle is equal to $0.618°$, as previously determined in Section 6.4. For the attitude and orbit determination system, this accuracy requirement is set to be 10% of this value, resulting in an accuracy requirement equal to $0.0618°$. The pointing accuracy has a direct influence on the potential views, thus the total revenue. This is further elaborated in Section 16.4. Note that accuracy of the attitude determination and control of the spacecraft is combined for these requirements, the subsystem is not individually designed for the precise determination of the Sun, ground station and Earth's center separately. Requirement **BBIS-Func-Att-06** concerning the spacecraft's acceleration around its axes has been discarded as not all the rates of acceleration are equal for the spacecraft's 3-axis stabilisation. **BBIS-Func-Att-07** and its respective sub-requirements have therefore been added to the list. The slew rates per axis are dependent on the experience disturbance torques and required manoeuvres. These calculations are further discussed in Section 7.2, 7.3 and 7.4.

7.2. Disturbances

During orbit, the spacecraft's orientation are affected by disturbances. The external forces acting on the spacecraft create a torque that destabilise the body. These forces are individually analysed as they have specific magnitudes and direction, as well as a variation throughout the orbit. The external disturbances and their respective equations to calculate them are listed below. Here, specific assumptions are also stated. An overview with the parameters, values and units is given in Table 7.1. [20]

- **Gravity-Gradient** This gradient produces a torque which is primarily determined by the spacecraft's moment of inertia's and orbit altitude. The gravity-gradient torque, τ_g in Nm, is calculated using Equation 7.1 [20]. Here, μ stands for the Earth's gravity constant in m^3/s^2, r for the orbit radius in m, I_{yy} and I_{zz} represent the spacecraft's mass moment of inertia in $kg \cdot m^2$ around its y- and z-axis, respectively, and θ for the maximum deviation of the z-axis from the local vertical in rad.
$$T_g = \frac{3\mu}{2r^3}|I_{zz} - I_{yy}|sin(2\theta) \tag{7.1}$$
- **Solar Pressure** The radiation creates a disturbance depending on the spacecraft's geometry and reflectivity. For BBIS, it is assumed that the spacecraft's bus contribution is negligible due to the low reflectivity and relatively small area compared to the payload. Therefore, the solar sail is assumed to produce the solar torque. Thus, the surface area A and reflectant factor q_{ss} are based on the solar sail characteristics. Furthermore, the solar pressure centre $c_{ps_{ss}}$ is assumed to coincide with the solar sails centre of gravity. Equation 7.2 [20] is used to determine the solar torque τ_s in Nm. The solar irradiance I_r is assumed not to vary due to the relatively large distance to the sun. Finally, the spacecraft's centre of gravity cg_{sc} is assumed to remain constant during operation.
$$T_s = \frac{I_r}{c}A_{ss}(1 + q_{ss})cos(i)(c_{ps_{ss}} - cg_{sc}) \tag{7.2}$$
- **Magnetic Field** The Earth's magnetic field creates a torque depending on the spacecraft's orbit altitude and inclination, as well as the body's residual magnetic dipole moment. The magnetic torque, τ_m in Nm, is determined using Equation 7.3 [20]. It is assumed that the residual dipole D is equal to $1.0\ Am^2$, which is a typical value for small uncompensated spacecraft [20]. The strength of the magnetic field , B in nT differs in value along the x-, y- and z-axis which can be calculated with Equation 7.4, 7.5 and 7.4.[1] In these equations, x, y and z represent the coordinates of a point in space in multiples of the radius of Earth. R_{Earth} is the radius of Earth in m. Furthermore, M is a constant equal to $7.96 \cdot 10^{15}\ T$.
$$T_m = DB \tag{7.3}$$
$$B_x = \frac{3xzM}{R^5_{Earth}} \tag{7.4}$$
$$B_y = \frac{3yzM}{R^5_{Earth}} \tag{7.5}$$
$$B_z = \frac{(3z^2 - R^2_{Earth})M}{R^5_{Earth}} \tag{7.6}$$
- **Aerodynamic** The last disturbance is caused by the aerodynamic drag, T_a in Nm, that the spacecraft experiences. This disturbance depends on the spacecraft's altitude and geometry, as can be seen in Equation 7.7 [20]. For the aerodynamic torques, both the spacecraft bus and the solar sail are considered, where the solar sail is the main contributing element (i). For the elements, the centre of aerodynamic pressure c_{pa_i} is assumed to coincide with the centre of the spacecraft bus. Again, the spacecraft's centre of gravity cg_{sc} is assumed to remain in the same position during operation.
$$T_a = \frac{1}{2}\rho C_{d_i}A_iV^2(cg_{pa_i} - cg_{sc}) \tag{7.7}$$

The spacecraft's mass moment of inertia expressed in $kg \cdot m^2$ can be written in the form of a matrix, shown in Equation 7.8. The spacecraft is assumed to be symmetric along its x-, y- and x-axis. Thus, all parameters except I_{xx}, I_{yy} and I_{zz} in the mass moment of inertia matrix go to zero. To calculate

[1] URL https://www.nasa.gov/pdf/417438main_Magnetic_Math.pdf [cited 22 June 2018].

Symbol	Description	Value	Unit
μ	Earth's gravity constant	$3.986 \cdot 10^{14}$	m^3/s^2
R	Orbit radius (Earth's radius and the orbit altitude)	$6.371 \cdot 10^6 + h$ (variable)	m
I_z	Mass moment of inertia about spacecraft's z-axis	612.16	$kg \cdot m^2$
I_y	Mass moment of inertia about spacecraft's y-axis	1222.6	$kg \cdot m^2$
θ	Maximum deviation of z-axis from local vertical	θ (variable)	rad
I_f	Solar Irradiance	1361	W/m^2
c	Speed of light	$3 \cdot 10^8$	m/s
A_{ss}	Solar sail surface area	406	m^2
q_{ss}	Solar sail reflectant factor	0.91	[-]
$c_{ps_{ss}}$	Solar sail centre of solar pressure	$[0.009, -0.0035, -0.023]$	$[m, m, m]$
cg_{sc}	Spacecraft centre of gravity	$[0, 0, 0]$	$[m, m, m]$
D	Residual Dipole	1.0	Am^2
M	Magnetic moment of the Earth	$7.96 \cdot 10^{15}$	$T \cdot m^3$
ρ	Atmospheric density	$4.040 \cdot 10^{-19}$	kg/m^3
$C_{d_{bus}}$	Drag coefficient spacecraft bus	$8/3$	[-]
$C_{d_{ss}}$	Drag coefficient solar sail	$4sin(\alpha)^2$	[-]
A_{bus}	Spacecraft bus surface area	0.1156	m^2
V	Undisturbed flow velocity	3074.9	m/s
$c_{pa_{bus}}$	Spacecraft bus centre of aerodynamic pressure	$[0.009, -0.0035, -0.023]$	$[m, m, m]$
$c_{pa_{ss}}$	Solar sail centre of aerodynamic pressure	$[0.009, -0.0035, -0.023]$	$[m, m, m]$

Table 7.1: Disturbance Torque Parameters [20].

I_{xx}, I_{yy} and I_{zz}, the spacecraft is divided into two parts, the spacecraft bus and the payload, due to the difference in the mass moment of inertia contribution. Equation 7.9, 7.10 and 7.11 show the calculations for I_{xx}, I_{yy} and I_{zz}. Here, m_{bus} represents the spacecraft's bus mass equal to the first iteration mass 16.93 kg, $m_{payload}$ represents the payload's mass equal to 18.07 kg, w the width of the bus equal to 0.34 m, h the height of the bus also equal to 0.34 m, l the length of the bus equal to 0.66 m and p the length of the payload's sides equal to 20.15 m. The payload thickness, t_h goes to zero due to the small magnitude of the parameter. Note that the values to determine the mass moments of inertia for the attitude control originate from the first iteration. In Section 7.6 the consequences of having a heavier spacecraft are elaborated on.

$$I = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} \tag{7.8}$$

$$I_{xx} = \frac{1}{12}(m_{bus}(w^2 + l^2) + m_{payload}(p^2 + t_h^2)) \tag{7.9}$$

$$I_{yy} = \frac{1}{12}(m_{payload}((p^2 + p^2) - (w^2 + h^2)) + m_{bus}(w^2 + h^2)) \tag{7.10}$$

$$I_{zz} = \frac{1}{12}(m_{bus}(h^2 + l^2) + m_{payload}(p^2 + t_h^2)) \tag{7.11}$$

When filling in Equation 7.9, 7.10 and 7.11, the following mass moment of inertia values are obtained: $I_{xx} = 612.18\ kg \cdot m^2$, $I_{yy} = 1222.8\ kg \cdot m^2$, $I_{zz} = 612.18\ kg \cdot m^2$.

Finally, when calculating the disturbance torques expressed in Equation 7.1, 7.2, 7.3 and 7.7, the disturbance torque about the x-axis, y-axis and z-axis during one orbit can be seen in Figure 7.1, 7.2 and 7.3, respectively. The spacecraft's minimum and maximum experienced disturbances are:

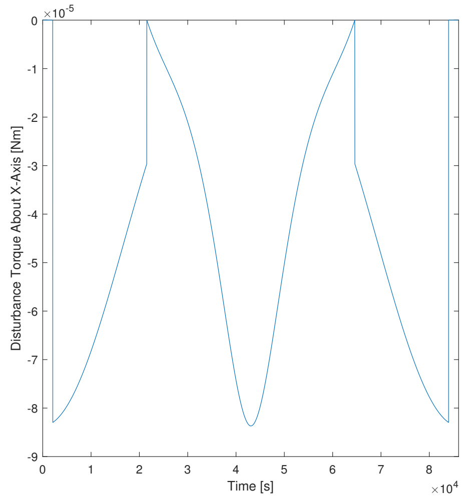


Figure 7.1: Torque About the X-Axis due Solar Pressure Over One Orbital Period.

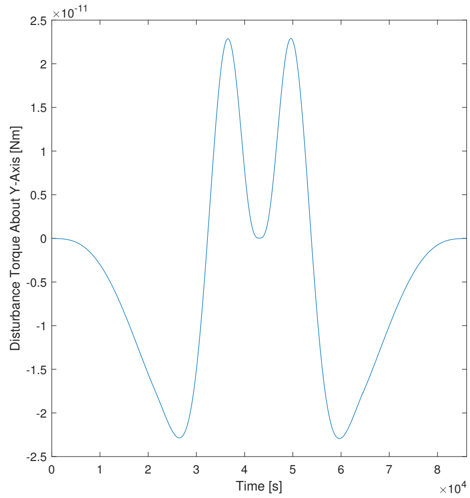


Figure 7.2: Torque About the Y-Axis due Aerodynamic Drag over One Orbital Period.

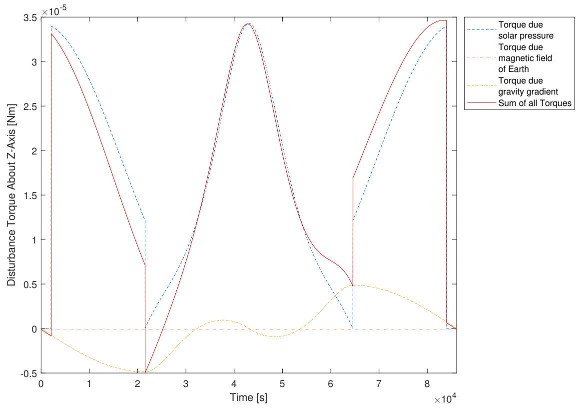


Figure 7.3: Torque About the Z-Axis over one Orbital Period.

- $\tau_{g,z}\ \pm 4.87 \cdot 10^{-6}\ Nm$ about the y-axis,
- $\tau_{s,x} = -8.37 \cdot 10^{-5}\ Nm$ about the x-axis and $\tau_{s,z} = 3.43 \cdot 10^{-5}\ Nm$ about the z-axis,
- $\tau_{m,z} = -1.06 \cdot 10^{-7}\ Nm$ about the z-axis, and
- $T_{a,y} = \pm - 2.29 \cdot 10^{-11}\ Nm$ about the y-axis

The torque due to aerodynamic drag around the z-axis is negligibly small compared to the magnetic torque and gravity gradient torque, since the maximum drag is $3.1 \cdot 10^{-9}\ N$ as was shown in Section 5.4.2. And the maximum distance from the cg in the y-direction is $3.5 \cdot 10^{-3}\ m$ in the y-direction.

The spacecraft is stabilised by these torques by activating actuators to produce a torque in the opposing direction of the disturbances. In order to ensure that the spacecraft does not fail, a 10% safety margin is included. Due to the coinciding force and moment arm of the solar radiation force acting on the solar sail and the spacecraft's centre of gravity, it can be concluded that in perfect conditions this disturbance does not generate a torque. However, due to manufacturing imperfections and to include a safety margin, a small moment arm between the solar sail's centre of solar pressure is accounted for. To correct the spacecraft's orientation and positioning for this disturbance, either a countering force acting through the same line of action or torque around the same body axis in the opposite direction must be produced.

7.3. Sensors

In order to determine the spacecraft's attitude and orbit, specific A&ODCS devices are included in the subsystem design. External references, such as the Sun, the Earth's IR horizon, the local magnetic field direction and the stars, are used to determine the spacecraft's absolute attitude and position. Measurements with respect to external references represent the body-centred angular distances to a vector. As these vector measurements only provide a two-dimensional orientation, multiple sensors are required on board to obtain an accurate orientation determination for all three independent axes. Additionally, it can be chosen to include inertial measurement units (IMU), which includes gyroscopes acting as internal sensors for short-term attitude updates. These provide smoother data at a higher frequency than external references. Furthermore, the Global Navigation Satellite System (GNSS) can be used in order to provide the spacecraft's position.

For attitude determination, the following devices are included. In order to provide 3-axis stabilisation, a minimum of two sensors is required. Furthermore, for orbit determination, an additional device is required. For the design of BBIS, a total of 4 A&ODCS sensors are included in order to achieve a higher accuracy and to add a safety factor. As listed in **BBIS-Sys-A01-2**, **BBIS-Sys-A03-1**, **BBIS-Func-Att-02**, **BBIS-Func-Att-03** and **BBIS-Func-Att-04**, a precise spacecraft determination of $0.0618°$ is required in order to accurately point the payload to its target location on Earth.

Firstly, a Sun sensor is chosen to determine the spacecraft's attitude with respect to the position of the Sun. Due to the required minimum operational time of spacecraft, many devices with a lifetime $< 15\ years$ were eliminated. In order to provide an accurate determination with a frequent refreshing rate and large field of view, the S3 Smart Sensor[2] was chosen. The device has an accuracy of $< 0.02\ °$ and resolution of $< 0.005\ °$. The spacecraft's face has a maximum inclination angle equal to $45\ °$ with respect to the Sun, which falls within the sensor's measurement field. The sensor's field of view is $128 \times 128\ °$, implying that only one Sun sensor is required to provide accurate attitude determination. The sensor is mounted on the spacecraft's side that always faces the Sun, as can be seen in Figure 2.4.

Additionally, a star sensor is included to accurately determine the spacecraft's attitude. To ensure a high determination accuracy is obtained, a star sensor, ST400[3] developed by Hyperion, commonly used for CubeSats, is chosen. This star sensor has optimal specific performance characteristics with a precision of $0.0028\ °$ in the pitch and yaw plane and $0.033\ °$ in the roll plane. This star sensor requires sufficient protection in order to obtain a life time of $> 15\ years$. The star sensors are placed on the deep space facing side of the spacecraft such that minimal light source interference occurs. Despite the innovation of modern techniques, the possibility that a Sun or star sensor confuses the light emitted or reflected by a neighbouring spacecraft with a Sun or star remains a critical risk to consider. The spacecraft are continuously communicating between each other, exchanging a broad variety of data. The spacecraft data handling system are programmed in such a way that the Sun and star sensor information observed by the spacecraft orbiting in the outer edges of the fleet are leading for the entire spacecraft configuration. The spacecraft orbiting at the edge have the lowest chance of spacecraft light interference, resulting in a low error Sun and star sensor determination system.

Thirdly, an inertial measurement unit is included. The IMU acts as a temporary determination substitute section by filling in missing data gaps. For this sensor, the priority is given to its continuity in order to have precise attitude determination. In [8], a trade-off was performed in order to determine the most suitable device. This trade-off was based on typical COTS components with low costs [21]. The most cost effective IMU resulted to be the Microstrain 3DM-GX3-45. This device is integrated into the spacecraft's design in order to improve the determination accuracy with its high update rate equal to $100\ Hz$.

Finally, a Global Positioning System (GPS) receiver is chosen to provide the spacecraft's position for orbit control. The Global Positioning System (GPS) is a GNSS technology that has developed over the past years and is now widely used for the orbit control of spacecraft due to its accuracy and evolved method of application. Alternatives to the GPS system include Russia's GLONASS, Europe's Galileo and China's COMPASS. However, GPS is currently the only system that has been tested as a navigation determination system above LEO, making it the most favourable GNSS type [22]. The Galileo constellation is expected

[2]URL http://www.leonardocompany.com/documents/63265270/65745274/S3_Smart_Sun_Sensor_LQ_mm07948_.pdf [cited 14 June 2018].

[3]URL https://hyperiontechnologies.nl/wp-content/uploads/2015/08/HTBST-ST400-V1.01_Flyer.pdf [cited 22 June 2018].

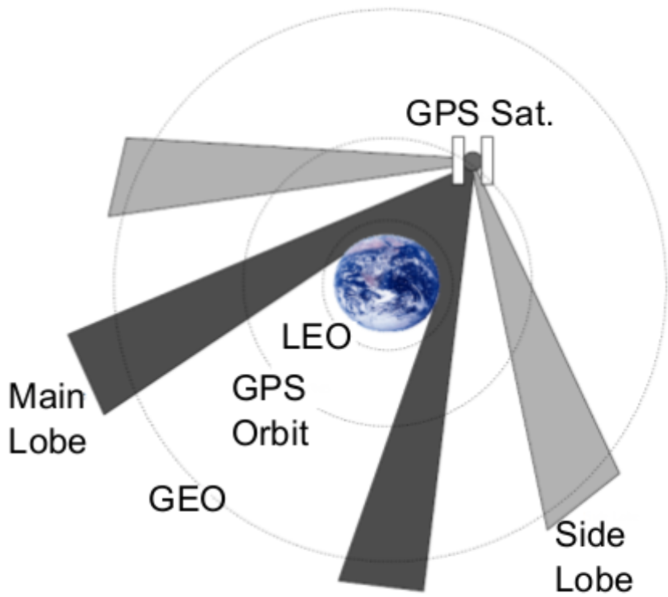


Figure 7.4: GPS Signal Path to a Spacecraft in a GSO.[4]

to have higher accuracy than GPS. Therefore, once the Galileo GNSS is fully functioning it is possible the BBIS changes its GPS receiver. However, at the moment GPS receiver is the only option.
In Figure 7.4 the GPS signal path to a spacecraft in GSO is shown.
The path loss is significantly larger for a GSO spacecraft compared to a spacecraft in LEO. Therefore, the signal is significantly lower, which results in a lower precision. In order to have sufficient precision, a GPS-Enhanced Onboard Navigation System (GEONS) is required [23]. GEONS processes data from the GPS receiver to increase the reduce position error with a factor of 15 and velocity error with a factor of 50. Furthermore, it includes on-board maneuver control and relative navigation for formation flying[5]. Using GPS in GSO is done before, a military satellite from the USA has been using GPS for several years [24]. Therefore, it is concluded that GPS in GSO is possible. For navigation in GSO, GPS has an accuracy approximately equal to $1\ m$ using the increased acquisition and tracking sensitivity of Navigator.[4]

Based on the trade-off made in [8], the GPS reciever PolaRx2[6] is chosen. This GPS receiver has not been tested in GSO before, but is used as a reference GPS receiver due to the limited amount of data found on receivers functioning in GSO. PolaRx2 has multiple antenna inputs, flexible inherent architecture[7], and is capable of communicating with GSO based satellites[8] This GPS has an update frequency of $10\ Hz$ and a standalone vertical accuracy equal to $1.9\ m$ and horizontal accuracy equal to $1.1\ m$ that can be further improved through augmentation. This determination accuracy meets the requirements for orbit control stated in **BBIS-Sys-A01-2**.

[4] URL https://www.navcen.uscg.gov/pdf/gps/news/Apr2010/Navigator_Space_Receiver_info_04212010_rev4.pdf [cited 29 July 2018].
[5] URL https://partnerships.gsfc.nasa.gov/downloads/featured_technologies/aerospace_aeronautics/gsc_14687_1_geons.pdf [cited 29 June].
[6] URL http://www.ppmgmbh.com/pdf_d/GPS%20Hardware/OEM-Boards/High%20End%20GPS%20OEM%20Boards/PolaRx2_Board.pdf [cited 8 June 2018].
[7] URL http://gpsworld.com/professional-oemcontinuous-product-improvements-10954/ [cited 8 June 2018].
[8] URL https://www.esa.int/Our_Activities/Navigation/Satellite_navigation_receiver_uses_EGNOS_signals_delivered_via_Internet/(print) [cited 8 June 2018].

7.4. Actuators

After it has been determined what the deviation in attitude and/or position is, a manoeuvre is performed to correct the spacecraft's orientation and positioning. In order to ensure that the payload is accurately pointed towards Earth with the desired incidence angle, an active control system is required. In the preliminary BBIS design phase [8], an attitude correcting actuator system was proposed consisting of magnetic torquers and thrusters as an additional safe mode system. However, due to the design's altitude change to a GSO orbit, magnetic torquers cannot be used because the magnetic field is too weak and unpredictable [25]. This results in the magnetic torquers not being capable of delivering sufficient torque [26].

Instead, in GSO orbit, reaction wheels, thrusters, control momentum gyros and solar flaps are widely used as actuators [27]. Control momentum gyros are discarded for this design due to their complexity and common application on spacecraft $> 1000\ kg$. Furthermore, solar flaps are eliminated as they perform slow manoeuvres which is not beneficial for the accurate formation flying and controlled payload orientation. The remaining two actuators are further reviewed for the BBIS design. First, the reaction wheels are elaborated on in Section 7.4.1 after which the propulsion system including the thrusters are discussed in Section 7.5.

7.4.1. Reaction Wheels

Reaction wheels allow the spacecraft to rotate the spacecraft to the desired orientation around it's x-, y- and z-axis. Reaction wheels have the advantage that they are a linear control system that are able to ensure simply controlled manoeuvres. However, the disadvantage is that reaction wheels are only able to create torques and not forces. Furthermore, reaction wheels commonly suffer mechanical wear out when operated continuously over the years. Therefore, to guarantee a specific life time, the reaction wheels are chosen such that the EOL is not threatened. The required torque to correct for the spacecraft's attitude rotation around its axes are dependent on the subjected disturbances and required manoeuvres. Per rotation plane, two thrusters are placed. One thruster is capable of acquiring the required to be delivered torque. The other thruster is added for redundancy and can furthermore function as a counter device to produce an opposing torque to decelerate the spacecraft's rotation. The disturbance torques are discussed in Section 7.2, the additional manoeuvres will be discussed per axis below.

- **Torque around x-axis** The payload is required to scan over and point at the USA by rotating around its x-axis. The angular acceleration of this manoeuvre is equal to $0.0031\ deg/s^2$, which equals $5.4 \cdot 10^{-5}$. Using Equation 7.12, it is determined that a total required torque of $0.0331\ Nm$ is needed to rotate the spacecraft around its x-axis.
- **Torque around y-axis** For this axis, no additional manoeuvres are required. The spacecraft must remain in the same orientation throughout the entire orbit.
- **Torque around z-axis** Similar to the required torque around the x-axis, the payload is also required to scan over and point at the USA by rotating around the z-axis. However, an additional rotation about the z-axis. The reflective surface's orientation depends on the spacecraft's orbit position as can be seen in Figure 5.4. However, this is relative small compared to the acceleration required for scanning the USA. This resulted a total torque required of $0.0335\ Nm$ about the z-axis.

$$\tau_x = I_{xx}\alpha_x \quad (7.12) \qquad \tau_y = I_{yy}\alpha_y \quad (7.13) \qquad \tau_z = I_{zz}\alpha_z \quad (7.14)$$

Combining the required torque for the manoeuvres with the torque needed to withstand the disturbances, the following required torque for each axis rotation are yielded: $\tau_x = 0.0335\ Nm$, $\tau_y = 4.87 \cdot 10^{-6}\ Nm$ and $\tau_z = 0.0331\ Nm$. These torques are in line with the required slew rates listed in requirements **BBIS-Func-Att-07.1**, **BBIS-Func-Att-07.2** and **BBIS-Func-Att-07.3**.

Based on these constraints, the reaction wheel RSI 02-33/30A by Rockwell Collins[9] is chosen to rotate the spacecraft around its x- and z-axis. This reaction wheel is able to deliver a total torque of $0.033\ Nm$.

[9] URL http://www.electronicnote.com/RCG/RSI%2002_A4.pdf [cited 22 June 2018].

To provide sufficient torque, both reaction wheels lying in the same plane can be activated simultaneously. For the required rotation around the y-axis, a smaller reaction wheel is chosen to minimise the weight and power consumption of the actuators. For this determination control, the RW210[10] reaction wheel designed by Hyperion is chosen. This light weight actuator delivers a torque equal to $0.001\ Nm$. The placement of these reaction wheels is specified in Chapter 11.

Orbit correcting procedures require a point force input which is carried out by thrusters. To ensure that the spacecraft has 3-axis stabilisation, multiple thrusters are included on different faces of the spacecraft such that the desired correcting reaction forces and torques can be obtained. Moreover, thrusters can be activated to desaturate the spacecraft's reaction wheels when moment dumping is required. Here, the reaction wheel has reached its maximum speed and is thus 'saturated'. The wheel cannot rotate faster, and a small deviation resulting in a slower rotation would cause a torque in the opposite direction. Therefore, the reaction wheel must be forced to standstill by thrusters. This command can be given automatically or by the ground station. For the choice of the reaction wheels, the storable angular momentum is not a constraint because the actuator is only operated a short period of time to provide an initial impulse like torque. The large reaction wheels have a storage momentum equal to $0.2\ Nm$.,For the small ones, this value is $0.0015\ Nm$.

7.5. Propulsion

This section discusses the propulsion subsystem, that is, the thruster, the propellant and the propellant tank. First, the constraints for the propulsion subsystem are listed. Then, it is explained how the sizing of this subsystem was executed, after which the final design choices are explained.

7.5.1. Method

This subsection first discusses the constraints that are imposed on the propulsion subsystem. Next, a determination method for the different components of the propulsion subsystem will be given. These components include the thrusters themselves, the propellant tank and the supporting structures.

Propulsion System Constraints

When selecting components for the propulsion system, several aspects have to be taken into account. These aspects are listed below.

- **The thruster** The following aspects of the thruster are relevant for the spacecraft design.
  - **Type of thruster** For the BBIS, it is decided to use a liquid bi-propellant engine. Other options, and the reason why these options are not suitable for the BBIS's design, are listed in [8].
  - **Force generated by the thruster** When the thrusters are activated, they exert force on the rest of the spacecraft's structure. If this force is too strong for the spacecraft to withstand, the thrusters could cause damage to the structure.
  - **Size of the thruster** The thruster will have to be launched along with the rest of the spacecraft, so it needs to fit inside the launcher. Also, its size should be proportional to the rest of the spacecraft.
  - **Mass of the thruster** In order to simplify the launching process, the spacecraft's mass should be limited.
- **The propellant** As the used thruster type requires propellant to function, the propellant also has to be considered. The following aspects were considered when selecting the propellant.
  - **Compatibility with the thruster** Most thrusters can only function with certain types of propellant; the chosen thruster and propellant should be compatible.
  - **Specific impulse** If the specific impulse is high, the spacecraft will need less fuel in order to obtain the desired ΔV.
  - **Sustainability** The BBIS team wishes to create a sustainable design, so safe and sustainable propellant options should be considered for the propulsion system.
- **The propellant tank** This is used to store the propellant.

[10] URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-RW210_V1.01_Flyer.pdf [cited 22 June 2018].

- **Storage capacity** The propellant tank should be able to store enough propellant for the duration of the entire mission.
- **Size of the propellant tank** As the propellant tank will be stored inside the spacecraft bus, minimising the propellant tank's size will also decrease the size of the spacecraft bus. A small propellant tank, and thus a small spacecraft bus, is beneficial for the launching of the spacecraft.
- **Mass of the propellant tank** A lightweight propellant tank is preferred in order to minimise the total spacecraft mass.

Thrusters

Several thrusters are placed on the spacecraft. The considered constraints when placing these thrusters are listed below.

- **Operating temperature of the thrusters** The thrusters, using monomethyl hydrazine (MMH) and dinitrogen tetroxide (N_2O_4), can operate at a temperature range of $10\ °C$ to $50\ °C$.[11]
- **Other spacecraft subsystems** The thrusters should be located such that they do not interfere with other subsystems, such as the solar sail or the spacecraft's antennas.
- **Orbit control** The spacecraft should be able to perform orbit control in three dimensions. When performing orbit control, the attitude of the spacecraft should remain constant.
- **Attitude control** The spacecraft should be able to perform attitude control around its three body axes. When performing attitude control control, the orbit location of the spacecraft should remain constant. Even though the spacecraft's reaction wheels are able to perform attitude control, the thrusters should be able to perform this as well, as a safe mode for the reaction wheels.

Propellant Tanks

The needed amount of propellant is determined using the calculated ΔV for orbit maintenance. Also, it is known that the launcher, the Ariane 5, has an inaccuracy in the orbit where it can deliver the spacecraft. It is thus assumed that the spacecraft will be delivered in an orbit that is not the desired orbit, so the spacecraft will require additional ΔV to be inserted into the right orbit. The ΔV required for orbit insertion is added to the determined ΔV for orbit maintenance. The method used to calculate the ΔV for insertion and maintenance is explained in Section 5.5.2.

After an estimation is made for the amount of ΔV required for the entire mission, Tsiolkovsky's equation (see Equation 7.15) can be used to determine the expected amount of propellant for orbit maintenance [20].

$$\Delta V = g_0 \cdot I_{sp} \cdot ln\left(\frac{m_{dry} + m_{prop_{nom}}}{m_{dry}}\right) \tag{7.15}$$

Here, ΔV is the total ΔV for orbit insertion and maintenance during the lifetime in m/s, per spacecraft. g_0 is the Earth's gravitational constant in m/s^2, I_{sp} is the specific impulse in s, m_{dry} is the spacecraft's dry mass in kg and $m_{prop_{nom}}$ is the required propellant mass for a nominal mission in kg, without attitude control. Using [20] as a reference, is it estimated that 6.5% of the spacecraft's propellant is used for attitude control, 10% is present as a safety margin for unexpected events and 1.5% is a residual, meaning this propellant is unavailable for use during the mission. As a result, the orbit's ΔV budget only accounts for 82% of the propellant mass, so the $m_{prop_{nom}}$ determined in Equation 7.15 is divided by $\frac{82}{100}$ to determine the total amount of required propellant: $m_{prop_{total}}$.

As the chosen engine is bipropellant, two different propellants (an oxidiser and a fuel) have to be stored separately. This means that two propellant tanks are required. It is usual in such cases to operate the thruster with an oxidiser-fuel ratio that allows the two propellant tanks to be of the same size [20]. This allows the same tank model to be used twice, allowing for an easier manufacturing process. The oxidiser-fuel ratio required to have an equal volume for the two propellants is determined in Equation 7.16.

$$\frac{O}{F} = \frac{m_{oxidiser}}{m_{fuel}} = \frac{\rho_{oxidiser}V_{oxidiser}}{\rho_{fuel}V_{fuel}} = \frac{\rho_{oxidiser}}{\rho_{fuel}} \tag{7.16}$$

[11] URL http://ecaps.space/hpgp-characteristics.php [cited 14 June 2018].

Supporting Structures

Thrusters and propellant tanks are the main components of the propulsion subsystem, but the subsystem also needs other components to function properly. These supporting components include propellant management devices, such as valves and lining, and mounting hardware. According to statistics, these components will have a mass equal to approximately 25% of the overall tank mass [20].

7.5.2. Results

Each of the BBIS spacecraft should be able to perform attitude and orbit control in three dimensions. In order to do so, multiple thrusters have to be located on the spacecraft. The used thrusters will be the S10-13 10N bipropellant thruster by Ariane Group.[12] Relevant specifications of this thruster are listed in Table 7.2.

Thrust	$10\ N$
Specific Impulse	$292\ s$
Thruster Mass	$350\ g$
Qualified Accumulated Burn Life	$69\ hrs$
Qualified Cycle Life	$1\ 000\ 000\ cycles$
Oxidiser	Dinitrogen tetroxide (N_2O_4)
Fuel	Monomethyl hydrazine (MMH)
Required Power	$0\ W$

Table 7.2: Specifications of Ariane Group's S10-13 10N Thruster.

The chosen thrusters use a combination of MMH and N_2O_4 to propel the spacecraft. These propellants have good storage and performance capacities [20], but MMH is known to be damaging to the environment when not handled properly.[13]

The option of using green propellants was considered in order to increase the sustainability of the BBIS design. However, the most suitable green engine, the 1N HPGP Thruster by Bradford Ecaps, has a significantly lower specific impulse than the non-green alternatives.[14] A lower specific impulse requires the spacecraft to bring more propellant in order to obtain the desired ΔV for the mission.[15] The mass of this additional propellant requires the launcher to be launched more times in order to get all the BBIS spacecraft into their orbit. An additional launch of the launch vehicle is considered to be less sustainable than changing the BBIS satellite's propellant from green to non-green, meaning the BBIS spacecraft use non-green propellants in order to limit the needed amount of launches. Another advantage of using MMH and N_2O_4 is that this combination of propellants is hypergolic, meaning they will ignite when they get into contact with one another.[16] An electric spark to ignite the engine is thus not necessary.

The oxidiser to fuel ratio for N_2O_4 ad MMH follows from the density ratio between these substances. The density of N_2O_4 is $1440\ kg/m^3$[17] and the density of MMH is $880\ kg/m^3$.[18] This gives an oxidiser to fuel ratio of 1.64.

With the oxidiser to fuel ratio known, it can be determined what fraction of the total propellant mass ($m_{prop_{total}}$) is fuel and what part is oxidiser. The BBIS spacecraft will bring 1.64 times more oxidiser (N_2O_4) than fuel (MMH). This means that N_2O_4 will contribute to $\left(\frac{1.64}{1.64+1}\right) \cdot 100\% \approx 62.1\%$ of the total propellant mass, $m_{prop_{total}}$. Consequently, MMH will be 37.9% of $m_{prop_{total}}$.

Taking the above mentioned constraints into account, it was found that the spacecraft need a total of 12 thrusters to perform all necessary functions for attitude and orbit control. The configuration of the

[12] URL http://www.space-propulsion.com/brochures/bipropellant-thrusters/bipropellant-thrusters.pdf [cited 14 June 2018].
[13] URL https://onlinelibrary.wiley.com/doi/pdf/10.1002/asia.201500711 [cited 14 June 2018].
[14] URL http://ecaps.space/products-1n.php [cited 14 June 2018].
[15] URL https://spaceflightsystems.grc.nasa.gov/education/rocket/rktpow.html [cited 3 July 2018].
[16] URL http://www.daviddarling.info/encyclopedia/H/hypergolic_propellant.html [cited 3 July 2018].
[17] URL https://pubchem.ncbi.nlm.nih.gov/compound/Dinitrogen_tetroxide [cited 3 July 2018].
[18] URL https://pubchem.ncbi.nlm.nih.gov/compound/methylhydrazine [cited 3 July 2018].

		Thrusters											
		1	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**	**11**	**12**
Moment arm [*mm*]	**x**	0	0	0	0	292	110	292	110	0	0	0	0
	y	0	0	0	0	0	0	0	0	132	132	132	132
	z	292	110	292	110	0	0	0	0	110	110	110	110

Table 7.3: Each Thruster's Moment Arm with Respect to the Spacecraft's X-, Y- and Z-Axes.

thrusters is shown in Figure 7.5. The thrusters are numbered $1-12$. For the thruster in this sketch, the Table 7.3 shows the moment arm of each of the thrusters, with respect to the spacecraft's main axes, x, y and z.

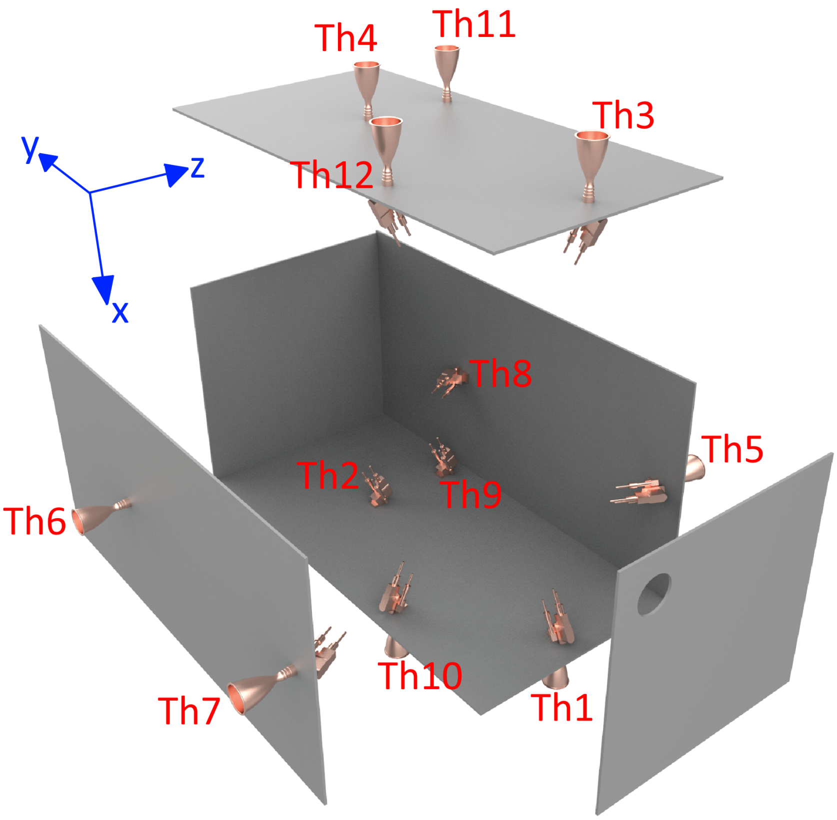


Figure 7.5: Thruster Configuration on a Single BBIS Spacecraft.

Each of the thrusters has a separate function in the attitude and orbit control of the spacecraft. The thrusters used for each of the needed control manoeuvres are explained below.

- **Attitude control around the x-axis**
  A positive rotation around this axis is obtained by simultaneously activating thrusters 5 and 6.
  A negative rotation around this axis is obtained by simultaneously activating thrusters 7 and 8.
- **Attitude control around the y-axis**
  A positive rotation around this axis is obtained by simultaneously activating thrusters 2 and 3.
  A negative rotation around this axis is obtained by simultaneously activating thrusters 1 and 4.
- **Attitude control around the z-axis**
  A positive rotation around this axis is obtained by simultaneously activating thrusters 9 and 12.
  A negative rotation around this axis is obtained by simultaneously activating thrusters 10 and 11.
- **Orbit control in x-direction**
  An orbit change in positive x-direction is obtained by activating thrusters 3 and 4. Thruster 4 should be activated longer, to avoid creating a moment around the z-axis.

An orbit change in negative x-direction is obtained by activating thrusters 1 and 2. Thruster 2 should be activated longer, to avoid creating a moment around the z-axis.

- **Orbit control in y-direction**
  It is not possible to directly perform an orbit change in y-direction. However, the spacecraft can easily be rotated around its y-axis in order to use the x-direction control's thrusters for the desired orbit manoeuvre in z-direction.
  An orbit change in positive y-direction can be obtained by performing a sequence of actions. First, thrusters 2 and 3 are activated to rotate the spacecraft around its z-axis. Then, thrusters 3 and 4 are activated to perform the desired orbit change. Finally, thrusters 1 and 4 are activated to return the spacecraft to its original attitude.
  An orbit change in negative y-direction can be obtained by performing a sequence of actions. First, thrusters 1 and 4 are activated to rotate the spacecraft around its z-axis. Then, thrusters 3 and 4 are activated to perform the desired orbit change. Finally, thrusters 1 and 4 are activated to return the spacecraft to its original attitude.
- **Orbit control in z-direction**
  An orbit change in positive z-direction is obtained by activating thrusters 6 and 7. Because of the different moment arms of these thrusters, thruster 6 should be activated longer, to avoid creating a moment around the x-axis.
  An orbit change in negative z-direction is obtained by activating thrusters 1 and 2. Because of the different moment arms of these thrusters, thruster 2 should be activated longer, to avoid creating a moment around the z-axis.

A summary of the usage of the thrusters is given in Table 7.4. It is shown which thrusters are used for a positive (+), and which for a negative (+), change in what attitude or orbit direction.

		Thrusters											
		1	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**	**11**	**12**
Attitude	**x**					+	+	-	-				
	y									+	-	-	+
	z	-	+	+	-								
Orbit	**x**									-	-	+	+
	y	+,-	+,-	+,-	+,-								
	z					-	+	+	-				

Table 7.4: Overview of Thrusters Used for Positive (+) and Negative (−) Attitude and Orbit Changes.

The propellant tanks used are product number 4593, produced by Ardé Inc. The tanks have a volume of $1.82\ L$ and weigh $0.748\ kg$.[19]

A summary of the propulsion subsystem components and masses is given in Table 7.5.

Mass of 12 Thrusters	$4.20\ kg$
Mass of Two Propellant Tanks	$1.50\ kg$
Fuel Mass (MMH)	$1.48\ kg$
Oxidiser Mass (N_2O_4)	$2.45\ kg$
Supporting Components	$0.37\ kg$
Total Subsystem Mass	$10.00\ kg$

Table 7.5: Summary of Propulsion Subsystem Components and Masses.

7.6. Sensitivity Analysis

In this section, the sensitivity of the spacecraft's positioning and orientation design is reviewed. The design sensitivity for the determination and control system will be reviewed individually in Section 7.6.1,

[19]URL http://www.ardeinc.com/sketches/propellant/4593.pdf [cited 25 June 2018].

m_{bus} $[kg]$	I_{xx} $[kg \cdot m^2]$	τ_x $[Nm]$	I_{yy} $[kg \cdot m^2]$	τ_y $[Nm]$	I_{zz} $[kg \cdot m^2]$	τ_z $[Nm]$
16.93 (0%)	612.18	0.033122	1222.78	0	612.18	0.033122
16.08 (-5%)	612.14	0.033120	1222.77	0	612.14	0.033120
15.24 (-10%)	612.10	0.033118	1222.62	0	612.10	0.033118
8.47 (-50%)	611.79	0.033101	1222.75	0	611.79	0.033101
17.78 (+5%)	612.22	0.033124	1222.80	0	612.22	0.033124
18.62 (+10%)	612.26	0.033126	1222.82	0	612.26	0.033126
25.93 (+50%)	612.57	0.033143	1222.95	0	612.57	0.033143

Table 7.6: Sensitivity Analysis Torque Around X-Axis, Y-Axis and Z-Axis.

7.6.2 and 7.6.3, respectively.

7.6.1. Sensors

The sensors are primarily chosen based on their accuracy constraint. This constraint originates from the payload's pointing requirement. In order for BBIS to be visible from Earth, a minimum pointing accuracy is required. This accuracy is influenced by the reflective surface's beam angle, the precision required in order for the billboard to be visible from Earth and the chosen orbit. These characteristics are fixed for the material and project design. Other criteria such as the spacecraft's mass or size do not influence the sensor choice, thus, no sensitivity analysis can be done. However, recommendations for finalising the design would include the analysis of noise levels. A high required accuracy was chosen such that a large safety factor was accounted for. However, no additional research has been done regarding the effects of noise.

7.6.2. Reaction Wheels

The reaction wheels' design depends on the following characteristics: the disturbance torques, the required manoeuvres and the spacecraft's design. Comparing the disturbance and manoeuvre torque magnitudes, it can be observed that the disturbance torques are typically a factor 10^{-3} smaller. Thus, these torques are not considered to be driving for the reaction wheel design. The required manoeuvre torques, however, are of great influence. Assuming that the spacecraft orbit and required manoeuvring rates do not change, the magnitude of the to be delivered torques are primarily dependent on the spacecraft's mass moment of inertia. Table 7.6 shows the change in required torque around its respective axis, dependent on the spacecraft's mass moment of inertia which is determined according to Equation 7.9, 7.10 and 7.11. A sensitivity analysis could be done for each variable independently. However, because the mass of the spacecraft is considered the most critical design characteristic, the reaction wheel sensitivity analysis will only be done for this parameter. For the analysis, the payload mass and required manoeuvring angular acceleration are kept constant and the spacecraft's bus mass varies.

Analysing Table 7.6, it can be observed that the required manoeuvre torque is minimally affected by a change in the spacecraft's bus mass. Thus, the conclusion can be drawn that the reaction wheels are not sensitive to a change in mass. It is recommended to perform further analysis on the the degradation of the reaction wheels as a decrease in performance could lead to catastrophic consequences for the attitude control of the spacecraft.

7.6.3. Thrusters

The sensitivity analysis of the propulsion subsystem includes an analysis of the consequences of thruster failure. The designed thruster configuration contains some redundancies in the thrusters, meaning the spacecraft will still be able to function if a thruster fails. As is summarised in Table 7.4, at least two thrusters are activated at the same time to perform an attitude or orbit manoeuvre. The six main manoeuvres of the spacecraft can still be performed if one thruster were to fail.

For attitude control, the thrusters are divided into three groups. Of each of the groups, one thruster is allowed to fail before the spacecraft loses its functionality for attitude control. It should be noted that the absence of one of the thrusters will reduce the efficiency at which attitude control manoeuvres are performed. Of thrusters 5, 6, 7 and 8, one can fail and the spacecraft can still perform attitude control around the x-axis. If two thrusters of this group would fail, the spacecraft would not anymore be able to perform attitude manoeuvres without changing the orbit. The same holds for thrusters 9, 10, 11 and 12 and attitude control around the y-axis and for thrusters 1, 2, 3 and 4 and attitude control around the z-axis.

For orbit control in x-direction, the task of thrusters 1 and 2 can be replaced by thrusters 9 and 10, in case either thruster 1 or 2 fails. The same holds for thrusters 11 and 12 replacing 3 and 4. If one of the thrusters responsible for orbit control in z-direction fails, the spacecraft can be rotated around its y-axis to achieve the desired orbit change. For orbit control in y-direction, the alternative method to rotate the spacecraft about its z-axis can be used.

It can thus be concluded that attitude and orbit control of the spacecraft can still be performed if a thruster fails. Further analysis could be performed to determine the thruster's and the propellant tank's sensitivity to radiation and their degradation after use.

7.7. Verification & Validation

Here, the verification and validation for the attitude and orbit determination and control system is presented. For the sensors, no verification will be done because the only design constraint is determined by the pointing accuracy of the payload. Furthermore, the devices are COTS which have been chosen based on their validated specifications and successful performance in previous missions. For the actuators, the verification and validation process for the reaction wheels are discussed in Section 7.7.1 and for the thrusters in Section 7.7.2.

7.7.1. Reaction Wheels

The manoeuvring equations used throughout this chapter, Chapter 7, are applied to the specific design case in order to determine the design constraints. In order to analyse the change in the required torque depending on the spacecraft's mass and dimensions, a MATLAB programme is set up to iterate calculations depending on the updates of specific variables. This code is verified by observing that required to be delivered torque goes to zero when one or more of the following variables is set to zero: bus mass, payload mass, angular acceleration per axis. Furthermore, it is verified that the torque is specifically calculated for one plane depending on the angular acceleration rotating axis.

Validation of the reaction wheels is done by testing the actuators. Here, the delivered torque is reviewed per reaction wheel. These test are non-destructive, as the reaction wheel will not be subjected to any forces. However, it should be noted that the lifetime of the reaction decreases over operation time. Therefore, it is recommended to use new reaction wheels for the BBIS final design. Additionally, it is important to test the reaction wheels' resistance to radiation as severe degradation could be catastrophic for BBIS. This testing is destructive and must be iterated with multiple reaction wheels with varying protection layers/types. Finally, additional specification such as the reaction wheel reaction accuracy, reaction time and severeness vibration are also important to test.

7.7.2. Propulsion

As of 2017, the chosen thrusters have been used on 97 successful missions in space.[20] These thrusters are therefore considered to be reliable COTS components. The scheme used to calculate the amount of propellant can be checked by setting the spacecraft's mass or the required ΔV budget to zero: if this is done, the required amount of propellant is also found to be zero.

The used propellant tanks can be tested for their functioning in a vacuum chamber. It should be tested whether the tanks can function in these conditions when filled, and when empty. Also, the amount of residual left in the tank should be tested. For the thrusters, it should be tested how much thrust and specific impulse they are able to produce. If the used components are suitable for the mission, these tests should not be destructive. Finally, the thruster's burn time and cycle life should be tested, after which the thrusters used for testing have reached the end of their lifetime.

[20] URL http://www.space-propulsion.com/spacecraft-propulsion/bipropellant-thrusters/10-bipropellant-thrusters.html [cited 26 June 2018].

8. Data Management

This chapter elaborates on BBIS's data handling system. First, the system requirements are presented in Section 8.1.Secondly, the communication subsystem is presented in Section 8.2, followed by command and data handling in Section 8.3. Then in Section 8.4 the hardware and software diagrams are elaborated on. Followed by the verification and validation of the subsystem in Section 8.5. Finally, a sensitivity analysis is performed in Section 8.6.

8.1. Requirements

In [3], the system requirement with respect to the BBIS project were established. Below, the requirements for the communication subsystem are listed.

- **BBIS-Func-Com-02** The spacecraft shall be able to receive a command from Earth.
- **BBIS-Func-Com-2.1** The spacecraft shall decode the signal.
- **BBIS-Func-Com-04** The spacecraft shall be able to send information to the ground station.
- **BBIS-Func-Com-05** The communication system shall not use more than $15\ W$.
- **BBIS-Func-Att-05** The spacecraft shall be able to determine the desired attitude.

Requirement **BBIS-Func-Att-05.3** is discarded as the spacecraft computer determines the required range of attitudes for the advertisement display with precision of degrees.

8.2. Communication

This section determines the communication subsystem. First the communication for BBIS is explained in Section 8.2.1. Followed by, the characteristics of the selected components in Section 8.2.2. Then, the communication logistics are explained in Section 8.2.3. Finally, the link budgets are shown in Section 8.2.4.

8.2.1. Communication for BBIS

The spacecraft need to communicate with the ground station (uplink and downlink) and with each other (crosslink). The crosslink communication ensures that all spacecraft communicate together and prevent collision. The downlink communication contains information of every spacecraft position and attitude. Once a day the spacecraft has a downlink connection. The uplink communication allows to have an input from the ground station to all spacecraft. Uplink communication is needed when the spacecraft change formation, display another advertisement, in case of unexpected events or to ask any subsystem info for any other reason.

The complication of communication for the BBIS is the amount of spacecraft. There are 900 spacecraft which all need to communicate, while one receiver can only receive a message of one transmitter at a time. However, there are solutions to receive multiple signals from different transmitters with code division multiple access communication, but this decreases the quality of the signal and is not usually done for spacecraft.

For the crosslink communication each spacecraft needs to transmit the position and attitude of the spacecraft, position of the Sun and the location of the navigation stars. To estimate the amount of data to transfer, 21 digits are used for the x-y-z coordinates and 11 digits are used for angles. Based on this, it is estimated that one message is $300\ bytes$. $1\ byte$ consist of $8\ bits$, thus, $2\ 400\ bits$ are required to send the information needed.

There are 900 spacecraft which need to communicate with the ground station. When all spacecraft operate as predicted, the downlink data contains only the last measured position, attitude (x,y,z,α,β and γ), housekeeping data and state of the subsystems. The total size of this message is estimated to be $400\ bytes$, which means $3\ 200\ bits$. However, when the position or attitude seems to be off, the measured data from the sensors are sent to the ground station.

The available frequency is important to consider. However, it is not possible to get a definite bandwidth frequency. Therefore, the current available frequencies are being looked into. Due to the fact that the spacecraft have low data rates, a small bandwidth is required, thus low frequencies can be used. The lowest available frequency in the USA is $322 - 328.6\ MHz$[1] according the International Communication Union. The next available bandwidth frequency in the USA is $2\ 200 - 2\ 290\ MHz$[2]. Not the whole bandwidth is required, the bandwidth depends on the data rate and modulation code. **BBIS-Sys-GO1-2** is met, because none used frequency are used for ground communication.

8.2.2. Component Selection

The spacecraft is equipped with three antennas. Two antennas are used for crosslink and one antenna is used for ground communication. This means that the spacecraft is not able to transmit and receive at the same time. Half of the orbit the spacecraft is able to communicate with the ground station, which is sufficient. However, a fourth antenna could be placed to make communication with the ground station always possible.

All antennas are low gain antennas. The advantage of low antenna gain is that it increase communication time with ground station, because the spacecraft's orbit is close to a geostationary orbit. Furthermore, it allows to focus the spacecraft orientation for the payload. When the attitude control has failed, the spacecraft is able to communicate with Earth and/or other spacecraft. Therefore, BBIS-func-Att-5.1 can be discarded. Using a microstrip patch antennas with a beam width of $90\ degrees$ (horizontal and vertical) is most efficient, because the sail blocks the signal when the beam width angle is bigger. A rectangular microstrip antenna has those properties as can be seen in Figure 8.1.

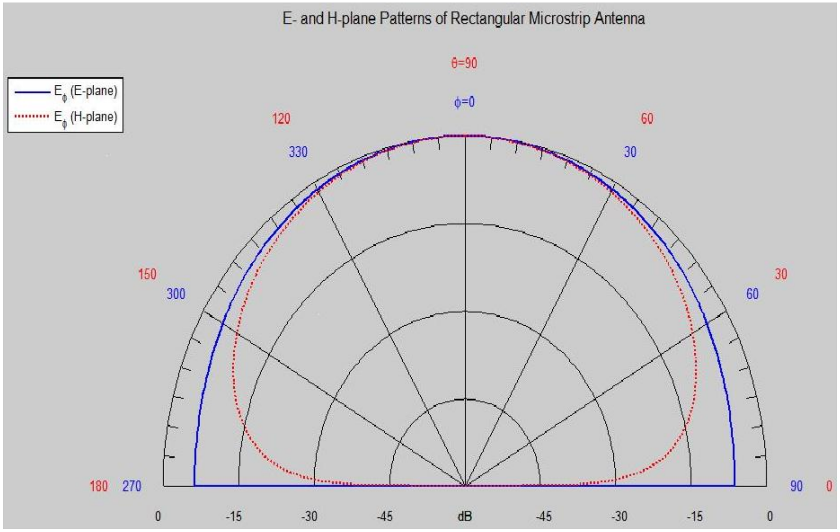


Figure 8.1: Radiation Pattern of a Rectangular Microstrip Antenna.[3]

Microstrip antenna do exist, however information (like price) is only available on request. Therefore, in this stage of the design omni-directional antennas are used for the link budgets and cost estimation. A fully omni-directional has a lower antenna gain ($0\ dB$) than a microstrip patch antenna ($> 0\ dB$).

To create a signal a transmitter is needed and to receive a signal a receiver is needed. Three different transceivers are used. An UHF (Ultra High Frequency) uplink/VHF (Very High Frequency) downlink transceiver and a VHF uplink/UHF downlink transceiver, both are used for the crosslink communication. Those transceivers modulate the data in Binary Phase Shift Keying (BPSK). One UHF uplink/S-band transceiver is used for the downlink and uplink. This transceiver is modulated in Offset Quadrature Phase-shift Keying. For the UHF uplink/S-band downlink transceiver it is assumed that the cost and weight are similar as the VHF/UHF transceivers. Since, two different receivers are on board, the

[1]URL https://www.ecfr.gov/graphics/pdfs/er07jy15.025.pdf [cited 25 June 2018].
[2]URL https://www.ecfr.gov/graphics/pdfs/er07jy15.036.pdf [cited 25 June 2018].
[3]URL http://pubs.sciepub.com/wmt/2/1/2/figure/3 [cited 24 June 2018].

spacecraft is able to decode two different signals, thus **BBIS-Func-Com-02.1** is met. The advantage of using a different frequency for transmitting and receiving is that it is possible to receive a message with a different antenna, while another antenna is transmitting. Also the ground communication and the crosslink communication happens on different frequencies, this is due to legal constrains [28].

For the ground station there are two options. The first option is to rent an already built ground stations. The advantage of this option is that those ground stations have good performances. The second option is to build a ground station, with slightly less performance. The advantage is that it can be build on the desired location. Also the ground station will always be available, so lower data rates are allowed. The second option is considered in this phase of the project. The main reason is the availability of the ground station. The properties of a reference ground station are listed below[4].

- **S-Band Receiver noise figure:** 0.9 dB,
- **S-Band Antenna:** $31.35\ dB$ gain,
- **UHF Antenna:** $15.5\ dB$ gain,
- **UHF Receiver Noise Figure:** $2.0\ dB$,
- **UHF RF Output Power:** $120\ W$, and
- **Price:** $64\ 500\ Euro$.

8.2.3. Communication Logistic

In order to have downlink communication with all spacecraft, the ground station sends a message to all spacecraft. In return, each spacecraft communicates its position to the ground station. This message contains the status of the spacecraft and information that the next spacecraft can start the downlink. This message is negligibly small, therefore, this transmitting time is neglected. However, when something appears to be wrong with one of the spacecraft, the ground station should be able to have a sufficient uplink data rate and connection time to solve the problem. For example, if the self-determination system does not work, the ground station shall be able to send the required information to de-orbit. All uplink data is received by all spacecraft, since they all receive on the same frequency, which reduces the required crosslink communication. In Figure 8.2 the ground communication logistic is shown. All spacecraft are defined as a number between 1 and 900, n. When one spacecraft has a problem, the ground station establish a high data rate connection with that spacecraft after it has finished normal downlink with all other spacecraft.

For the crosslink communication, one spacecraft transmits data and the other spacecraft receive the message. In this message it is also specifies which spacecraft is next to transmit his data. One spacecraft is transmitting for $\frac{2\ 400}{8\ 000}\ s$. Which means that every spacecraft has to transmit once every $270\ seconds$. If one spacecraft does not transmit the message, the next spacecraft will communicate $10\ seconds$ later, based on the cycle time of $270\ seconds$. The crosslink communication happens all the time, whereas downlink and uplink only is possible during half of the orbital time. The crosslink logistic diagram can be seen in Figure 8.3.

8.2.4. Link Budgets

In this section the link budgets for downlink, uplink and corsslink are presented. In order to have a sufficient communication link the signal to noise ratio (SNR) should be positive and have a margin of $3\ dB$. The energy per bit and the modulation code are used to determine the bit error rate (BER). The energy per bit can be calculated with Equation 8.1.

$$\frac{E_b}{N_0} = 10log_{10}(P) + L_l + G_t + L_a + G_r + L_s + L_{pr} + L_r - 10log_{10}(k) - 10log_{10}(R) - 10log_{10}(T_s) \quad (8.1)$$

Where k is the Boltzmann constant, $1.38 \cdot 10^{-23}\ m^2kgs^{-2}K^{-1}$. The other symbols with their units can be found in Table 8.1, which shows the link budgets of downlink, uplink and crosslink. Note that the downlink link budget is for the worst case scenario, where something is wrong and the spacecraft sends all sensor data. The elevation angle is assumed to be zero, because it is relatively small due to the fact that we can place the ground station on the desired location. Therefore, the altitude is used as maximum distance to calculate the path loss. Furthermore, the bit error rate (BER) of the

[4]URL https://www.isispace.nl/product/full-ground-station-kit-for-vhfuhfs-band/ [cited 25 June 2018].

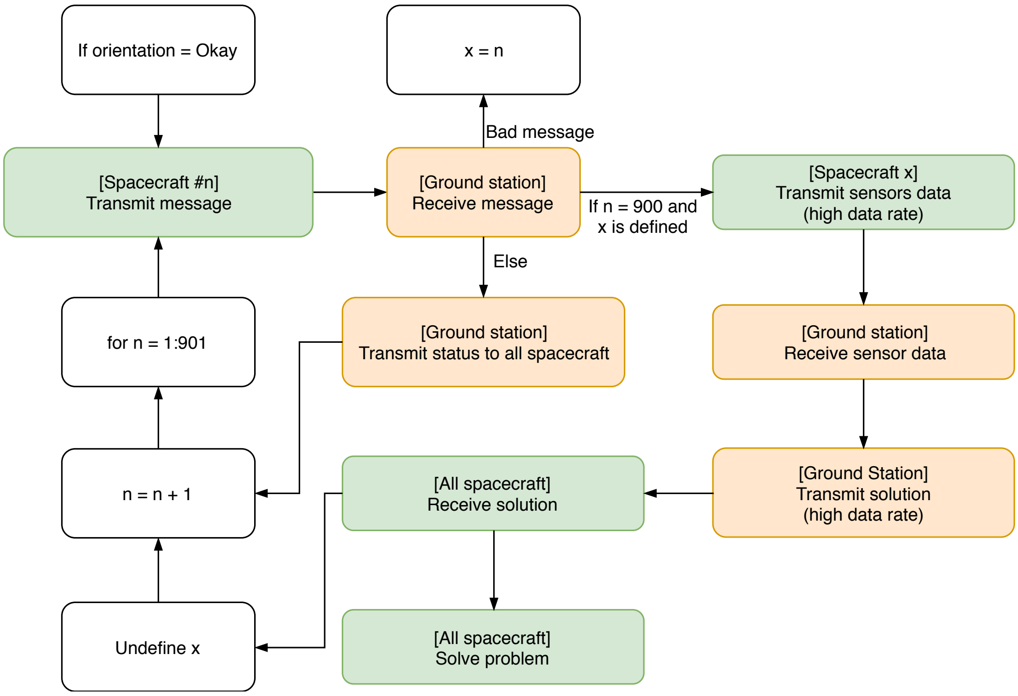


Figure 8.2: Ground Communication Logistic.

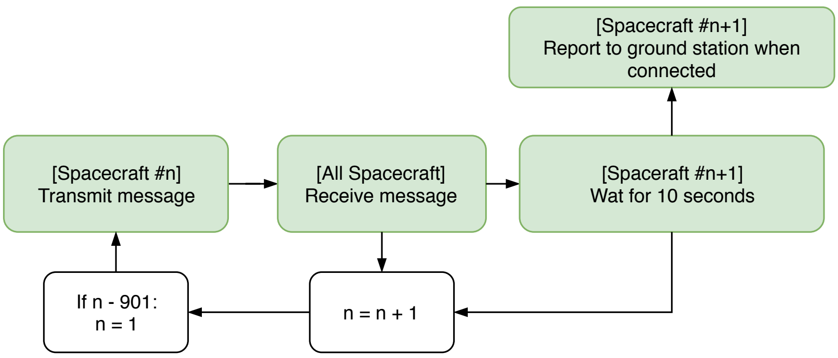


Figure 8.3: Crosslink Communication Logistic.

downlink is relatively high. However, due to the fact that all measurements of the sensor are sent to the ground station this error can be filtered. The normal downlink has a lower data rate of $1\ 600 bps$, which contains only the current position and attitude of the spacecraft. Therefore, the energy per bit is higher, thus a lower BER. For the crosslink communication the two furthest apart spacecraft are considered in the link budget. This means that other spacecraft could communicate with less power and still have the same data rate and BER.

All the link budgets are closed, thus ground communication and crosslink communication is possible. Therefore, **BBIS-Func-Com-02** and **BBIS-Func-Com-04** are met. In order to deliver an announcement during emergencies, the spacecraft should be able to receive a command from Earth and change advertisement. As was discussed in Section 6.7.1 the advertisement is controllable. Therefore, **BBIS-Gov-02** is met. And the maximum power consumption during the high data rate downlink is $15\ W$, thus **BBIS-Func-Com-05** is also met. Keep in mind that this is the power required for the high data rate of the downlink, which does not happen often. The average power required is mostly influenced by the crosslink communication. The average power for communication is $6\ W$.

	Downlink	Uplink	Crosslink
Antenna Type	Omni-directional	Omni-directional	Omni-directional
Antenna mass	$0.1\ [kg]$	$0.1\ [kg]$	$0.1\ [kg]$
Spacecraft Antenna Gain (G_t/G_r)	$0\ [dB]$	$0\ [dB]$	$0\ [dB]$
Spacecraft Transmitter/ Receiver Power	$9.2/0\ [W]$	$0/0.2\ [W]$	$5.7/0.68\ [W]$
Radio Frequency Power (P)	$5.8\ [W]$	$120\ [W]$	$2.0\ [W]$
Elevation Angle	$0\ [°]$	$0\ [°]$	$0\ [°]$
Maximum Data Rate (R)	$25\ 200\ [bit/s]$	$100\ 000\ [bit/s]$	$8\ 000[bit/s]$
BER	$6 \cdot 10^{-3}$	10^{-6} cite baseline	10^{-6}
Frequency	$2\ 290\ [MHz]$	$328.6\ [MHz]$	$438/146\ [MHz]$
Bandwidth	$36\ [kHz]$ (per spacecraft)	$114\ [kHz]$	$1.1\ [kHz]$ (2 times)
System Temperature (T_s)	$135\ [K]$[5]	$614\ [K]$[5]	$682\ [K]$[5]
Line Loss (L_l/)	$-0.5\ [dB]$[5]	$-0.5\ [dB]$[5]	$-0.5/-3.0\ [dB]$[5]
Antenna Pointing Loss (L_pr)	$0\ [dB]$	$0\ [dB]$	$0\ [dB]$
Path loss (L_s)	$-192.1\ [dB]$	$-175.3\ [dB]$	$-\ [dB]$
Atmospheric Loss (L_a)	$-0.03\ [dB]$[6]	$-0.03[dB]$[6]	$0\ [dB]$[6]
Ground Station Antenna Gain (G_r/G_t)	$31.35\ [dB]$[7]	$15.5\ [dB]$[7]	$0\ [dB]$
Receiver Line Loss (L_r/L_l)	$-0.9\ [dB]$[7]	$-3.0\ [dB]$[7]	$-5.0\ [dB]$[75]
SNR	$6.4\ [dB]$	$8.6[dB]$	$9.7/10.6\ [dB]$
Energy per Bit ($\frac{E_b}{N_0}$)	$8.0\ [dBJ]$	$9.2\ [dBJ]$	$8.3/9.1\ [dBJ]$

Table 8.1: Downlink, Uplink and Crosslink Link Budgets.

8.3. Command and Data Handling

The command and data handling (C&DH) subsystem has two primary functions. First, it receives commands from the ground station which it needs to decode, validate and distribute throughout the spacecraft. Second function, is to gather, process and format the data from other subsystems for downlink. In addition, to those primary functions C&DH monitors the health of the computer (watchdog) and does the timekeeping. [20]

BBIS's C&DH subsystem has to process the data from the ground station and other spacecraft and give input to the A&ODCS in order to avoid collision. Furthermore, the C&DH needs to process the data and send it to the ground station once a day. As a consequence, it needs to be able to store the data for up to $24\ hrs$. Therefore, as mentioned in Section 8.2, it is necessary to store $\approx 35\ Mb$[8] every day.

8.3.1. Components

For C&DH it is necessary to choose: a processing platform (i.e.: on-board computer - OBC), a data bus, a watchdog timer, and error-correcting code (ECC) memory. The whole C&DH subsystem has an over-current protection, therefore, no overheating is present, and all the components are protected. This is already elaborated on in Chapter 9 and is not addressed in this section.

[5] Typical value found in Table 13-10 in [20].
[6] Determined with Figure 13-10 in [20].
[7] URL https://www.isispace.nl/product/full-ground-station-kit-for-vhfuhfs-band/ [cited 25 June 2018].
[8] $400\ bytes \cdot 86\ 400\ s \approx 35\ Mb$.

On-Board Computer

Based on requirement **BBIS-Func-Att-05**, and based on (Section 8.2), a sufficient choice for the processing platform is the *CP400.85*[9]. *CP400.85* has a Linux based operating system, therefore, it is easier to adapt the software as it is open source. Most importantly, due to the operational system it can reset itself when errors appear. In addition to that, internal watchdog is present. A watchdog is a timer used to detect and recover malfunctions of the main processing platform. It is basically a reset button of the processor. If necessary a companion board with up to $7.5Gb$ of radiation tolerant storage and over $64\ GB$ of bulk data storage can be added to the primary base.[9] Moreover, *CP400.85* is able to protect itself from single event latch-up and has an ECC memory.[9] ECC memory corrects for flipped bits, those can be caused by single event effect.[10] ECC memory maintains data which are single bit error free.

In order to have a redundant system three processing platforms are used, which is further explained in Section 8.6. The three *CP400.85* are on a carrier board next to each other with the possibility of a companion board being stacked above it containing the radiation tolerant storage and bulk data storage. However, unless the scope of the mission is enlarged and more data than mentioned in Section 8.2 needs to be stored the primary data storage of $512Mb$ is enough. In the event that additional software packages need to be uploaded to the OBC, there is a sufficient memory reserve.

Data Bus

The primary job of the data bus is to control the data transfer between different spacecraft components (i.e. reaction wheel to OBC). [11] In [8], a *MIL-STD-1553* data bus is selected. Since *MIL-STD-1553* only defines properties of boards and components it is necessary to select a specific data bus which complies with those properties. *MIL-STD-1553* has a high reliability as it has been used on several missions like GAIA, Vega, Small Geo, etc.[12] However, it is necessary to realise that each spacecraft component (i.e. IMU, reaction wheels) might need different communication line. Some components might need *MIL-STD-1553* others might prefer I^2C others use USB or Ethernet protocols.

To give an example PolaRx2, GPS receiver chosen in Section 7.3, can operate on Linux and comes in a standard Euro-card sized board. However, it can be integrated in several other ways, for example via Ethernet.[13] The battery chosen in Section 9.5 has an I^2C data bus. That means one wire is used for outgoing communication and one wire is used for incoming communication. It is decided that choosing a data bus for each component is too detailed for the level of DSE and if components were to change the data bus would have to changed too. As the mission objective is to investigate the feasibility of the project, it is known that data bus is something that can be done. COTS components exist, and therefore it is not a deciding factor for BBIS.

However, in order to be able to continue with the design, a specific data bus component is decided upon. A *MIL-STD-1553* data bus is chosen for BBIS: *BU-67521* [29].

Data Handling Block Diagram

Figure 8.4 shows the command and data transfer between the C&DH components and other subsystems. The figure indicates the storage memories and processing speed.

8.3.2. Cables

Cables are the veins of the spacecraft. Cables connect all the components and distribute the energy and commands from and to the OBC. This section addresses all the cables in the spacecraft in general, regardless of their usage.

Cables for power distribution need to have low resistance. Furthermore, they need to have good operating temperatures and need to be protected against radiation. Qualified space cable *SPC* for GEO

[9] URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-CP400.85-V1.01_Flyer.pdf [cited 19 June 2018].
[10] URL https://community.arm.com/processors/b/blog/posts/beware-all-error-correcting-code-memory-systems-are-not-created-equally [cited 20 Jue 2018].
[11] URl https://www.milstd1553.com/resources-2/desginers-guide/designers-notes/mil-std-1553-overview/ [cited 20 June 2018].
[12] URL https://www.esa.int/Our_Activities/Space_Engineering_Technology/Onboard_Computer_and_Data_Handling/Mil-STD-1553 [cited 19 June 2018].
[13] URL http://www.ppmgmbh.com/pdf_d/GPS%20Hardware/OEM-Boards/High%20End%20GPS%20OEM%20Boards/PolaRx2_Board.pdf [cited 20 June 2018].

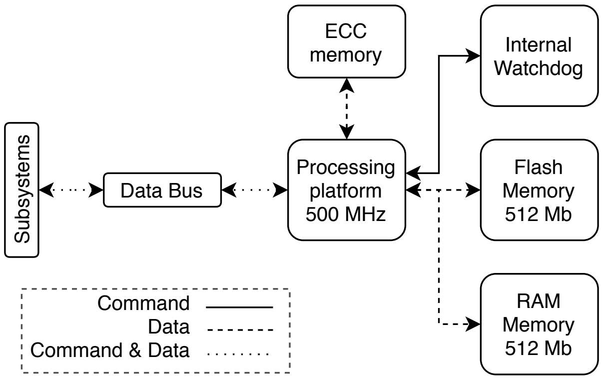


Figure 8.4: Command and Data Handling Block Diagram.

application from GORE Space Cables are selected. The conductor used is copper alloy with a double layer of expanded PTFE and Polyimide.[14]

Data cables have an impact on the speed of transmission, and more importantly on the quality of the signal. *GBL* cable from GORE Space has a signal transmission up to $1\ GHz$ [14]. In addition to that, it is also space qualified for geostationary orbit. The conductor is silver-plated, high-strength copper and copper alloy with an outer jacket of Perfluoroalkoxy alkane (PFA).[14]

8.4. Hardware and Software Diagrams

During operation, the spacecraft's subsystems' hardware and software are continuously communicating with each other. Figure 8.5 shows the interaction of hardware between the subsystems. Note that thermal control is excluded because this system is passive, as discussed in Section 10.4. The hardware diagram is not time based, all processes are continuous. The diagram includes different levels of depth; some systems within a subsystem are linked specifically, whereas other subsystems are linked as a whole. For example, the power distributor in EPS provides power to all subsystems except for the ground station. While the on-board data storage unit only communicates with the data handling function within the on-board computer in Command & Data Handling subsystem.

Additionally, Figure 8.6 shows the software design of the spacecraft. The coloured blocks indicate the hardware, already highlighted in the hardware diagram. The arrows contain the specification of data that is given in that connection. Special attention should be paid to the power check, in between the EPS board and the attitude & position comparison. This power check looks at the available power and the power usage of the subsystems. If there is not enough power available, the power check can overrule a desired attitude & position command from the ground station to optimise for power generation instead. This function results in requirement **BBIS-Func-Att-05.2** being discarded, because the exact required range is not yet specified in this function.

8.5. Verification and Validation

For now omni-directional antennas are used, however this is not optimal as it is elaborated on in Section 8.2.2. The antenna gain of the microstrip antenna could be verified by modelling the radiation pattern. Furthermore, the model of the radiation pattern of the antenna will also indicate half-power beam angle. The antenna gain can be calculated with Equation 8.2.

$$Gain[dB] = 10log_{10}\left(\frac{V_{sphere}}{V_{radiation}}\right) \tag{8.2}$$

Where V_{sphere} is the volume of a sphere in m^3 and $V_{radiation}$ is the volume of the radiation pattern in m^3. It is possible to validate the communication link budgets by comparing them to other spacecraft missions operating in GEO. The main difference between BBIS and other missions is the data rate.

[14] URL https://www.gore.com/sites/g/files/ypyipe116/files/2016-04/GORE%20Space%20Cables%20-%20Product%20Portfolio_02-10-2016.pdf [cited 20 June 2018].

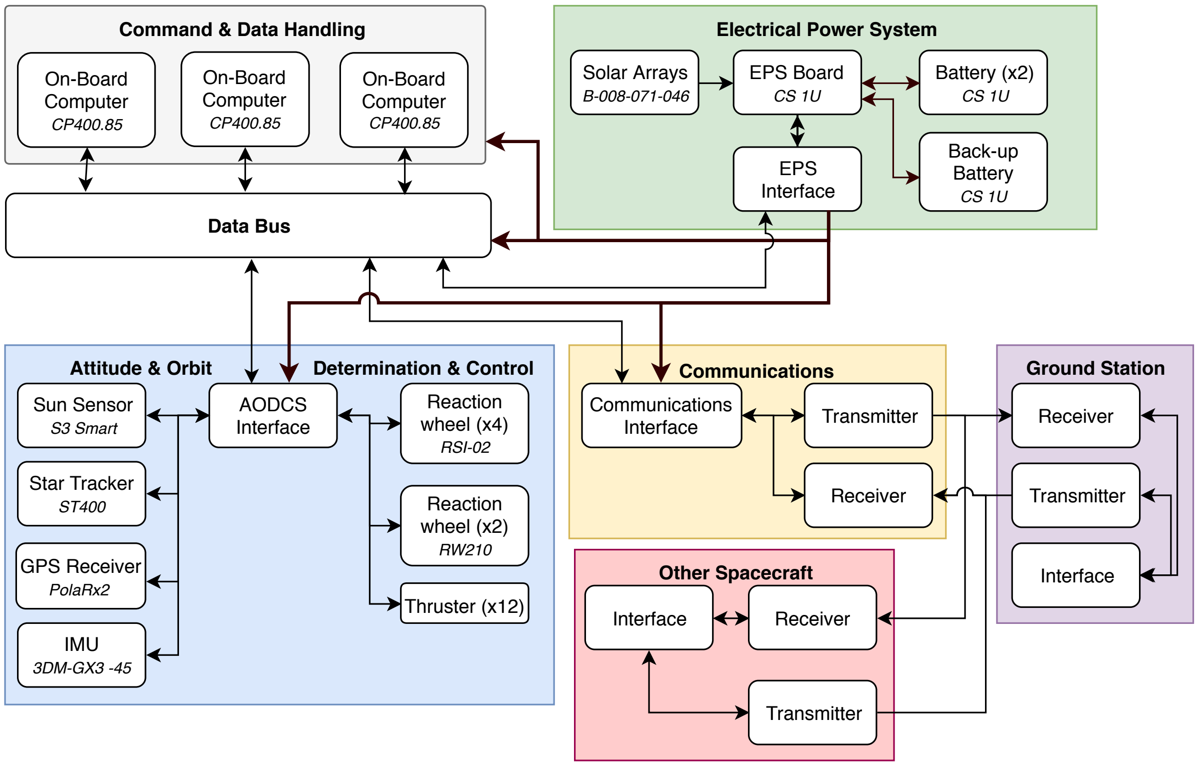


Figure 8.5: Hardware Diagram.

The link budget can be validated by comparing it to other spacecraft in GEO. The main differences is be the data rate, this results in a difference in bandwidth frequency. Also a relative lower frequency is used, which will result in smaller space loss. Therefore, the required radio frequency power of BBIS will be lower.

The C&DH subsystem component selection has to be verified. No calculations are present in Section 8.3, therefore, no calculations need to be verified. Only one component is selected, on-board computer. The selection is based on the requirement that BBIS needs to be built with COTS components. In addition, it has enough storage memory with the possibility to expand the memory storage. The cables selected are used on missions such as XMM (GBL cable) and Alphabus (SPC cable). [15] Those spacecraft are still operational and orbiting in GEO. Functional testing is done while the BBIS is still on Earth in order the validate the function of C&DH.

8.6. Sensitivity Analysis

In case a rectangular microstrip antenna is used, instead of an omni-directional antenna the gain of the antenna increase with $3\ dB$. This is due to the radiation patternm which is roughly half a sphere as can be seen in Figure 8.1. This allows to have a lower RF power, while maintaining same data rate and BER. For the crosslink, the RF power can be reduced to $0.5\ W$. For uplink the RF power can be reduced to $2.9\ W$ and for downlink the RF power can be reduced to $60\ W$. When comparing those values to the values in Table 8.1, the uplink and downlink RF power are halfed, and the crosslink RF power is only a quarter. Therefore, it is worth it to look into this type of antenna.

Three OBC's are used in each spacecraft. However, a processing power and storage memory of only one OBC is needed. In addition, to having two extra OBC's extra storage is accounted for as well. Even though, at this time no extra storage is necessary, during the design (sizing, mass) an extra board with three storage extensions, one for each processing platform, is accounted for. This part of C&DH is extremely over designed. This is due to the fact that without a functioning OBC it would be impossible to control the spacecraft which could cause a chain effect of individual BBIS spacecraft crashing into

[15]URL https://www.gore.com/sites/g/files/ypyipe116/files/2016-04/GORE%20Space%20Cables%20-%20Product%20Portfolio_02-10-2016.pdf [cited 24 June 2018].

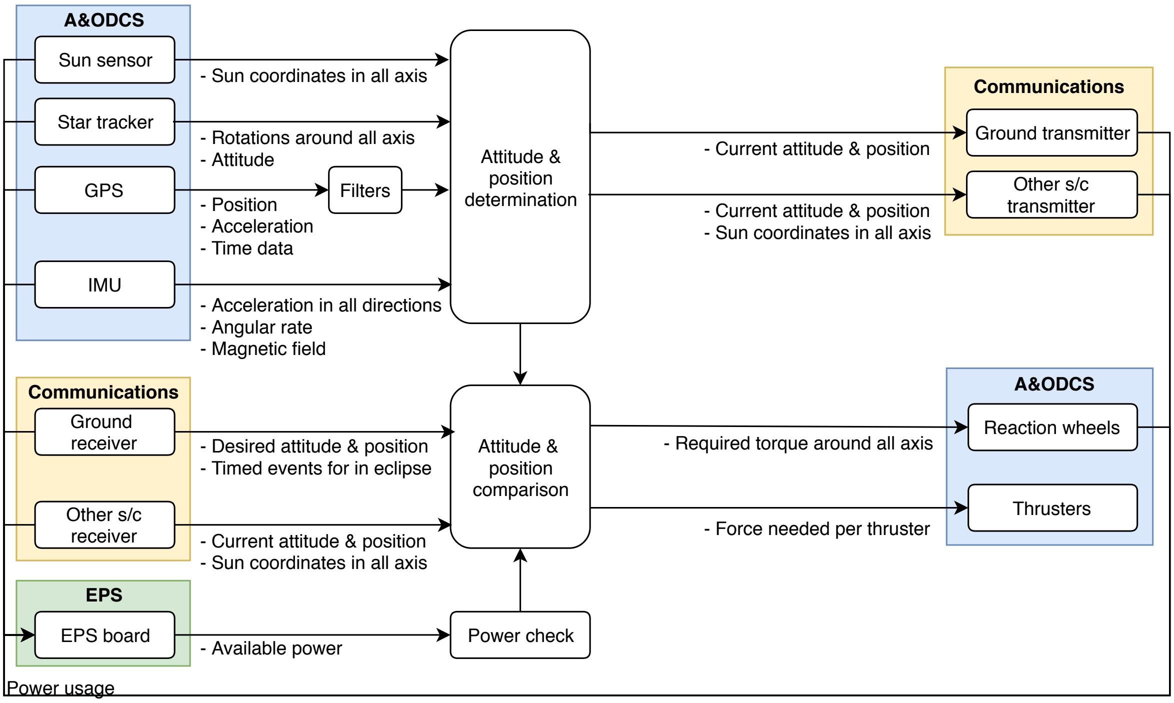


Figure 8.6: Software Diagram.

each other. That is unacceptable as it could affect other spacecraft orbiting in the vicinity. In addition, it increase the space debris which is unacceptable.

The OBC consumes $1\ W$ and has a mass of $7\ g$. The total power budget is $37.32\ W$ which is $\approx 8\%$ of the total power budget. Therefore, increasing or lowering the number of OBC's would change the total power consumption with less than 3%. The total mass budget is $57.06\ kg$, therefore, increase or decrease of one or two OBC's is negligible. Despite the extra power consumption, it is necessary to have three OBC's. The main reason, is that the spacecraft is not controllable without a well-functioning OBC. Therefore, when only one OBC is functional the spacecraft needs to de-orbit.

9. Electrical Power System

In this chapter the design of the Electrical Power System (EPS) is explained. Firstly, the requirements applicable to this subsystem are stated in Section 9.1. The power budget is highlighted in Section 9.2. In Section 9.3 the methodology of the sizing of the EPS is explained. The selection of solar cells and batteries is elaborated in Section 9.4 and 9.5, respectively. In Section 9.6, the distribution and regulation of power flows is highlighted in an electric block diagram. And the sensitivity of the design choices is discussed in Section 9.7, Lastly, the verification and validation methods used are stated in Section 9.8.

9.1. Requirements

The requirements for the BBIS project have been established in [3]. Listed below are the requirements that are relevant for the EPS.

- **BBIS-Func-Eps-01** The electrical power system shall distribute the required power to all subsystem.
- **BBIS-Func-Eps-02** The spacecraft shall base the generated power on the mode it is in.
- **BBIS-Func-Eps-03** The spacecraft shall generate a peak power of $66.45\ W$.
- **BBIS-Func-Eps-04** The spacecraft shall generate a average power of $37.32\ W$.
- **BBIS-Func-Eps-05** The spacecraft shall be able to store $61.42\ Whr$.

The EPS consists of solar panels, batteries, a distribution unit and a regulation unit. In the following sections these components are elaborated on, designed and chosen.

9.2. Power Budget

The power budget of all instruments on board is shown in Table 2.2, in Section 2.3. The total power is estimated to be 37.32 W with a 15% margin for unexpected situations. This margin is included because some subsystems have a peak power that is used during emergencies. An example of such a subsystem is communications, which needs a power of $15\ W$ when emergency messages need to be transmitted. Due to the importance of the functioning of the communications subsystem during emergencies, extra power was reserved for this. Another mode of the spacecraft is needed when the attitude is to be changed and the large reaction wheels are used. These consume $40\ W$ in total when they are spinning, and this is not accounted for during the sizing of the batteries. The orbit control is only done when the spacecraft is out of the eclipse and at the night side of the Earth. The additional power needed is provided by solar cells, which are sized for this power mode, among other design choices.

Including different modes for average power and peak power helps to make sure that requirements **BBIS-Func-Eps-02** is met.

9.3. Method

The main changes implemented after [8] are listed below.

- The spacecraft is orbiting at a geosynchronous orbit (GSO).
- The solar panel is replaced by copper indium gallium selenide (CIGS) thin-film solar cell (TFSC), which typically have a conversion efficiency $\eta = 18\%$. The selection of these solar cells is explained in Section 9.4.
- The new solar sail orientation is taken into account during calculations.
- It is assumed that the spacecraft shall always be able to deliver at least the average power required.

Because the peak power is only for a short duration and is provided directly by the solar cells, the average power $P_{average} = 37.32\ W$ has to be delivered all the time for operation.

The first aspect that needs to be considered is the incident angle. In Chapter 5, the worst-case incident angle is indicated to be 45°, when the spacecraft is above the terminator of the Earth. The solar cells are rotating w.r.t. sun when orbiting, as shown in Figure 5.4. to make sure that the solar cell always be able to deliver the required additional power, the worst case incident angle is used. Therefore, the influence of the incident angle is $ac = cos(45^{\circ}) = 0.707$.

The method of solar panel calculation is almost the same as the solar panel sizing in [8]. The required area of a planar solar array A_{sa} is mainly related to the power required during the daylight P_{sa}, using the solar constant ($1367\ W/m^2$).

$$P_{sa} = \frac{1}{t_{charge}} \frac{P_{average} t_d}{0.85} \tag{9.1}$$

Equation 9.1 is the result of several substitutions of equations from [20], and Equation 9.2 has been derived in [8].

$$A_{sa} = \frac{P_{sa}}{P_0 I_d cos\theta L_d} = \frac{\left(\frac{P_{average} t_d}{0.85}\right)}{t_{charge} \cdot \eta \times 1367\ W/m^2 \cdot 0.77 \cdot ac \cdot L_d}; \quad L_d = (1 - 3.75\%)^{20} = 0.4656 \tag{9.2}$$

Until now, the solar panel is sized in order to ensure operation of the spacecraft. However, the A&ODCS subsystem need approximately $28\ W$ more to change the attitude for scanning the USA. As it is determined that the spacecraft is only doing orientation when the spacecraft is exposed to the sun, the power needed is directly supplied by the solar cells. Therefore, by taking the degradation of solar panel L_d from Equation 9.2 into account, an additional $60.14\ W$ needs to be generated by the solar cells at the beginning of life.

After calculation, the area of solar cells is determined to be $1.69\ m^2$, which supplies $161.88\ W$ of electric power at the beginning of life and $75.37\ W$ at the end of life. During the complete lifetime, the solar cells can provide enough power to have functional subsystems. Therefore **BBIS-Func-Eps-03** and **BBIS-Func-Eps-04** are met.

9.4. Solar Cells

Due to the location of the solar cells on the solar sail, TFSC are used. There are three big groups of TFSC, copper indium gallium selenide (CIGS), cadmium telluride (CdTe) and amorphous silicon (a-Si) cells. CdTe celss are not sustainable due to the presence of cadmium. This heavy toxic material is indicated in the top ten chemicals of major public health concern[1], therefore this material is not chosen. The availability of commercial a-Si cells is decreasing due to the fact that the prospects of the increase of its efficiency are not as good as the other two groups of TFSC. CIGS cells are an interesting type of TFSC, due to their relatively high efficiency of $18\ \%$ and their low manufacturing costs and they are therefore used on BBIS.

The solar cell model B-008-071-046 by Ascent SOLAR[2] is chosen for this mission. It has a length and width of $86\ mm$ and generates $0.71\ W$. It is the most effective as well as the cheapest model among several CIGS TFSC. Based on the results in Section 9.3, a total amount of 226 solar cells are needed to meet the power requirement. Every quarter of the solar sail contain the same amount of solar cells, results in 57 cells per quarter sail. The distribution of the cells is shown in Figure 9.1.

The placement of the solar cells is driven by the folding pattern. The cells are flexible to a certain extent, but it is not possible to wrap the solar cells around the corner of the spacecraft. When the sail is folded, the solar panels can also not touch each other with their surface, otherwise scratches are formed and performance is reduced. That is why the position of the solar panels is interchanged between the quadrants, as can be seen in Figure 9.2. The distance between the solar cells along one fold is $10\ mm$. When the solar sail is expanded, the solar cells are immediately ready to generate electrical power.

[1] URL http://www.who.int/ipcs/assessment/public_health/chemicals_phc/en/ [cited 8 June 2018].
[2] URL http://www.ascentsolar.com/bare-modules.html [cited 22 June 2018].

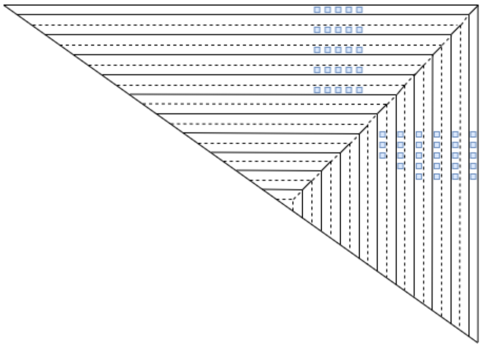


Figure 9.1: Location of Solar Cells.

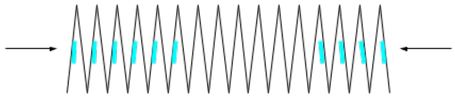


Figure 9.2: Folding with Solar Panels.

9.5. Battery

Initially, **BBIS-Tud-02** indicates that the spacecraft shall have a minimum lifetime of 50 years. However, the lifetime of the batteries is only available for 20 years [8], therefore **BBIS-Tud-02** is discarded and a new requirement **BBIS-Tud-08** is created, which states that the minimum lifetime of the spacecraft shall be at least 20 years. The whole project is designed for a 20-year mission, therefore **BBIS-Tud-08** is met. Due to the orbit being in GSO, the spacecraft is experiencing a eclipse of $4125\ s$ ($\approx 1.15\ hrs$). By multiplying the eclipse time with the operation power, approximately $42.76\ Whr$ of electrical energy is needed per eclipse. Therefore storage battery is needed to guarantee operation during the eclipse. Due to the fact that the battery supplier only offer $20\ Whr$, $30\ Whr$ and $40\ Whr$ batteries, therefore one $20\ Whr$ and one $30\ Whr$ batteries are needed for electric energy storage.

In [8], the NiH_2 battery was chosen for the power storage. However, in this final design, the lithium-ion polymer battery from Clyde Space[3] is chosen due to its light-weight and relatively high power density compared with the NiH_2 battery.

Moreover, the batteries chosen contain an EPS board with all the other electric components required, as explained in Section 9.6. There are two sizes of battery needed for this mission: CS 1U Power Bundle B: EPS + 20Whr Battery[4]and CS 3U Power Bundle A: EPS + 30Whr Battery[5].

Same as the requirement from the midterm report, a back-up battery should ensure that the spacecraft is able to operate for half an hour, which is enough to handle the emergency situation in case of solar panel failure and send the spacecraft to the graveyard orbit. Based on the estimated power budget in Section 9.2, an additional battery which has a minimum storage of $18.66\ Whr$ is needed. A battery with a storage of $20\ Whr$ is chosen. By including this battery, a total $70\ Whr$ of electrical energy can be stored, thus the requirement **BBIS-Func-Eps-05** is met. So, in a total the spacecraft has two batteries for operation and one battery as a back-up.

9.6. Distribution and Regulation

Next to the solar panels and the batteries, control units are part of the EPS. These control units contain different systems that distribute and regulate the power from the solar panels.

The regulation of the power is done by a maximum peak power tracker (MPPT) together with the voltage and current sensors, to extract as much power from the solar cells as possible. The MPPT enables the presence of a voltage difference between the solar cells, the data bus and the components that consume power.

The distribution of power is done by a distributor, which makes sure the voltage from the data bus is the same as the voltage to other subsystems require. The data bus on BBIS operates at $3.3\ V$, so the power from the solar panels or batteries needs to be converted to this value. The flow from the data bus has to be regulated by the MPPT and converted to the right voltage, such that it matches the voltage of the components. The distributor also protects against overcharging and overvoltage

[3]URL https://www.clyde.space/ [cited 20 June 2018].
[4]URL https://www.clyde.space/products/16-cs-1u-power-bundle-b-eps-20whr-battery [cited 21 June 2018].
[5]URL https://www.clyde.space/products/39-cs-3u-power-bundle-a-eps-30whr-battery [cited 21 June 2018].

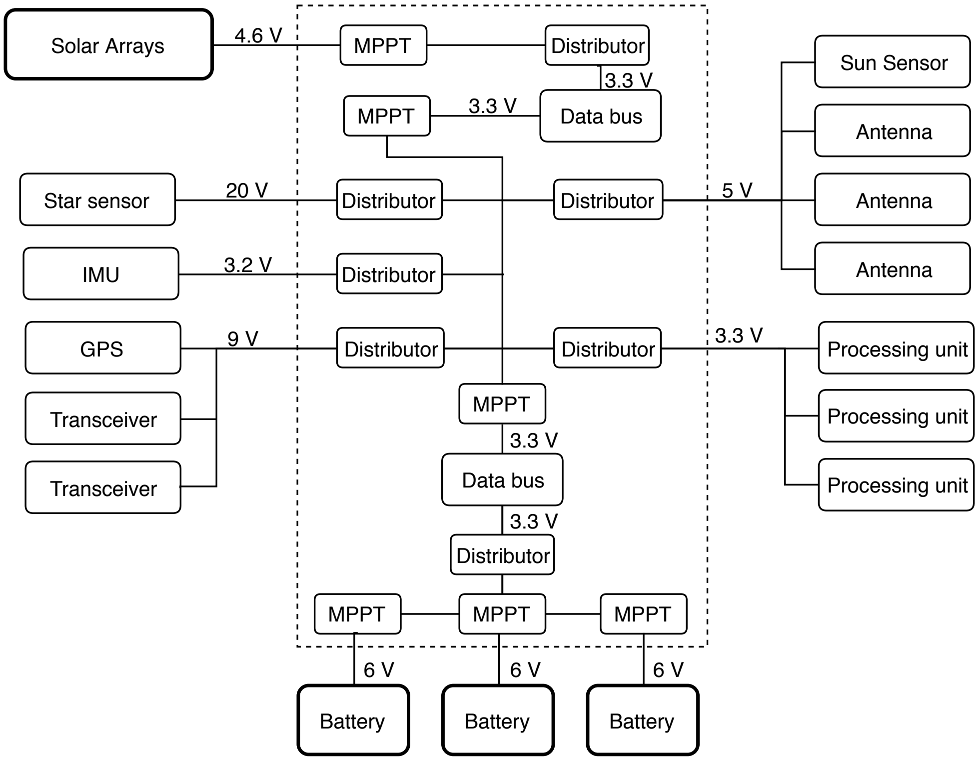


Figure 9.3: Electrical Block Diagram.

of batteries and components [20]. The complete process of the EPS is visualised in an Electric Block Diagram in Figure 9.3.

Both components mentioned above are integrated in the EPS board which comes with the batteries that were selected before. The wires used to connect all the components take roughly $10\ \%$ of the total EPS mass. The wire selection is also elaborated on in Section 8.3.2. This results in the requirement **BBIS-Func-Eps-01**, **BBIS-Func-Eps-01.1** and **BBIS-Func-Eps-01.2** being met.

9.7. Sensitivity Analysis

In this section the sensitivity of the EPS is investigated and elaborated on. A test is conducted by increasing the power consumption with 10%, the results of this test are displayed in Table 9.1. After the increase of power consumption, the mass of the EPS increases with 8.5%. This is mainly due to the fact that a bigger battery is needed to store the required energy, the total battery capacity is $80\ Wh$, instead of the original $70/Wh$. The increased solar cell area only attributes to 8.4% of the increase in EPS mass, the rest is due to the bigger battery. To actually implement the extra solar cells is not difficult, they can just be added to the already present solar cells. The reflectivity of the sail has to be calculated again, because the presence of solar cells decreases the reflectivity by a small amount.

Input Parameter	**Power Required** $[W]$	**Solar Cell Area** $[m^2]$	**Battery Capacity** $[Wh]$	**Total Mass** $[kg]$
Design Condition	37.32	1.69	61.41	1.29
10% **Power Increase**	41.05	1.78	67.56	1.40 (8.5% increase)

Table 9.1: Sensitivity Analysis of EPS.

9.8. Verification & Validation

To make sure the EPS actually generates the power required when it is in operation, verification and validation are performed. Several equations from [20] are used during the computations of the power budget and they are assumed to be verified already. These equations are implemented in a program and together with an assumed typical characteristic values of CIGS TFSC and batteries. Firstly, the solar cell area and batter power is estimated. The result of the program gives a solar cell area, mass and required battery capacity. These are in the range of the first estimation and are therefore verified.

The characteristics of the components as well as the program are then validated by comparing the results with reference missions from [20]. During calculations, the specifications of all the EPS components are computed taking into account the degradation of the components as well as a safety margin. To make sure the EPS is not underdesigned, these specifications are based on EoL power generation. All selected EPS components are verified and certified for use in space and have proven their functionality in space numerous times. They therefore, do not need to be tested on itself before they van be implemented on BBIS. Once BBIS is in space, a check needs to be performed to see if they are functional.

10. Environment

This chapter analyses the pre-orbiting and in-orbit environment in Section 10.1 and Section 10.2 respectively. Those sections are followed by requirements of the thermal subsystem in Section 10.3. General information about thermal subsystem is presented in Section 10.4. The method used to design the subsystem is presented in Section 10.5 and the results are presented in Section 10.6. The verification and validation of the design are presented in Section 10.7. The last section, Section 10.8, discusses the sensitivity of the design.

10.1. Pre-Orbiting Environment

Before the spacecraft is launched it is subjected to various environments on Earth. The terrestrial atmosphere has dust, water, oxygen, etc. Water and oxygen can cause corrosion. Therefore, a careful transportation of parts and assembled spacecraft is crucial. In order to avoid dust collection, the spacecraft are assembled in clean rooms, nevertheless, the spacecraft still experience particle contamination. Particulate contamination is unavoidable as small pieces of material deposit on the spacecraft during manufacturing, integration, testing, transportation and launching. Particulate contamination is especially undesirable for optical instruments such as star trackers and Sun sensors.

During launch the spacecraft experiences extreme stresses (lateral loads due to wind gusts and axial loads due to launcher acceleration), furthermore, BBIS needs to withstand mechanical vibrations and large dose of acoustic energy. During launcher separation the spacecraft experiences shock mechanical loads. Those issues are addressed in Chapter 11.

The atmospheric pressure varies during all the various launch phases. Depressurisation loads need to be taken into account while designing the spacecraft. In addition, to various load and pressure changes BBIS experiences thermal loads caused by aerodynamic heating. The average heat load variation during launch is approximately $250\ W/m^2$ [30]. Compared to what the spacecraft experiences during its orbit this variation is negligible. Therefore, the requirement **BBIS-Sys-T05-3.5** is met while requirement **BBIS-Sys-T05-3.4** is discarded.

10.2. In-Orbit Environment

The effect of space medium is grouped into five categories: micrometeoroids and orbital debris, vacuum, plasma, radiation and electrically neutral particles [30]. The first category is addressed in section Section 13.4. In vacuum, solar ultraviolet radiation has a negative effect on surface degradation. This issue is addressed in Section 10.5.1 as is plasma, which charges the spacecraft which shifts the electrical potential. The electronic degradation and single event effect which is caused by radiation is discussed in Section 10.2.1. Finally, the electrically neutral particles which have mechanical and chemical effect on the BBIS are addressed in Section 10.5.1. Section 10.2.2 discusses the heat flows spacecraft experiences while in orbit.

10.2.1. Radiation

When designing a long term mission, radiation is a big problem for all components. Electronics are largely affected when only COTS components can be used. There are three main radiation effects: charging, ionisation and single-event effects [31].

Charging, can create a voltage potential which can lead to sparking if the voltage level becomes high and the insulator is unable to withstand the charge [31]. In order to protect the spacecraft from this phenomenon and to ensure the components of the spacecraft remain in operational temperature, the spacecraft is coated with a conductive layer of paint. The paint has a high conductivity in order to protect the spacecraft from build-up of of harmful potential gradients, which result from charging [32]. The specific type of paint used is elaborated on in Section 10.6.

Ionisation is caused by an impact of radiation particles. Ionisation is only a problem over long period of time as it can lead to high leakage currents. The total ionisation dose is expressed in radiation absorbed dose (rad). COTS integrated circuits can typically sustain less than $10\ krad$, after that, they start to malfunction [31]. However, some of them can withstand more than $10\ krad$ as can be seen in Table 10.1. Radiation tests are expensive, and therefore some companies do not do them. Radiation hardened electronics can withstand $10 - 100$ more ionisation than COTS components, however, their prices are between 100 and $100\ 000$ times higher [31].

Single event effects, just like ionisation, occur after a radiation impact particle. The particle can release heat, that can locally change the property of the material and a short circuit can occur. This is called *single event latch-up (SEL)*. In case this event persists, it can cause a cascading effect on other parts of the circuit and result in permanent failure. [31]

In order to protect the electronic components from ionisation it is necessary to shield them. Figure 10.1 shows the necessary thickness of the aluminium shield vs. the rad dose over the lifetime of the spacecraft in various orbits.

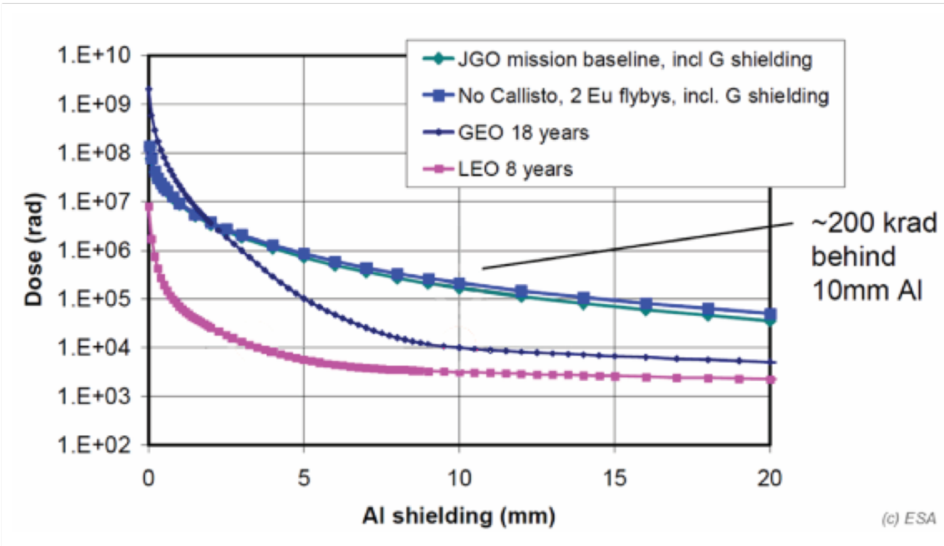


Figure 10.1: Necessary Shield Thickness [31].

When Figure 10.1 is consulted, a conclusion can be drawn about the protection of each component. First, it is necessary to realise Figure 10.1 only shows a trend line for spacecraft orbiting for $18\ years$ in GSO and various other orbits. Therefore, a safety factor of 1.1 is assumed in order to compensate for a lifetime of $20\ years$. For example, a Sun sensor can withstand $100\ krad$ during its lifetime which yields necessary protection of $\approx 5\ mm$. With a safety factor of 1.1 the aluminium thickness necessary needs to be $\approx 5.5\ mm$. The spacecraft bus has a thickness of $3.5\ mm$ (see Chapter 11), so an additional aluminium shielding of $\approx 2\ mm$ is required. All the components that need to be protected are listed in Table 10.1. As mentioned above, some of the COTS components that have been selected might have higher radiation tolerance then typical $10\ krad$. However, since it is necessary to always design for worst case scenario, it is assumed COTS components (unless specified otherwise) can only experience $10\ krad$ during their lifetime. With all the components protected the requirements **BBIS-Sys-T05-4**, **BBIS-Sys-T05-4.4** and **BBIS-Sys-T05-4.5** are met. Requirement **BBIS-Tud-05** cannot be verified at this stage.

10.2.2. Heat Flows

The BBIS is flying at geosynchronous orbit. It experiences an eclipse of $4\ 113\ seconds$. Furthermore, there are seasonal differences that affect the environment the spacecraft is flying in. Those deviations can be found in Table 10.2. This analysis is necessary to accurately predict the environment BBIS experiences during its lifetime.

	Dose $[krad]$	Extra Aluminium Shielding $[mm]$
Back-up Battery [1]	10	7.5
Storage Battery [1]	10	7.5
Sun Sensor[2]	100	2
Star Tracker [3]	11	5.3
IMU [1]	10	7.5
OBC [4]	25	4.8
Small Reaction Wheels [5]	45	3.7
Big Reaction Wheels [6]	100	2
GPS Receiver [1]	10	7.5
Data Bus	300	1.45

Table 10.1: Aluminium Shielding [29].

	Solar Constant $[W/m^2]$		Albedo Coefficient $[-]$
Summer	1318 [7]	**Cold Case**	0.35
Winter	1422 [7]	**Hot Case**	0.25
Average	1361	**Nominal**	0.3

Table 10.2: Thermal Environment [33], [34].

When in space, BBIS experiences three heat flows: solar radiation (solar intensity), solar energy reflected from the surface of Earth (albedo) and Earth flux (infrared radiation). Due to the large solar sail that is located in the middle of each spacecraft it is assumed the back of the BBIS does not experience any external heat flows.

10.3. Requirements

In [3] functional requirements are listed. Functional requirements are based on the system requirements. The system requirements are stakeholder requirements written in technical terms. The functional requirements relevant for the thermal control subsystem are listed bellow.

- **BBIS-Func-Temp-02** The spacecraft shall make a prediction of every subsystems temperature per orbit.
- **BBIS-Func-Temp-03** The spacecraft shall check if every subsystem temperature complies with indicated mode.
- **BBIS-Func-Temp-04** The spacecraft shall be able to regulate the temperature within the requirements of the subsystems.

Several requirements listed in [3] are omitted from the aforementioned list due to the thermal control subsystem being passive. Therefore, no active temperature regulation is present which renders requirements **BBIS-Func-Temp-01**, **BBIS-Func-Temp-04.1** and **BBIS-Func-Temp-04.2** unnecessary and they are not elaborated on. Requirement **BBIS-Func-Temp-02** is not exactly met as the spacecraft does not predict every subsystem temperature per orbit as it is not necessary. The placement of each subsystem component is based on the temperature distribution in the bus. Therefore, it is unnecessary to predict each components temperature and the requirement is also discarded.

[1] Radiation tolerance is not mentioned, therefore, $10\ krad$ is assumed.
[2] URL http://www.leonardocompany.com/documents/63265270/65745274/S3_Smart_Sun_Sensor_LQ_mm07948_.pdf [cited 22 June 2018].
[3] URL https://hyperiontechnologies.nl/wp-content/uploads/2015/08/HTBST-ST400-V1.01_Flyer.pdf [cited 22 June 2018].
[4] URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-CP400.85-V1.01_Flyer.pdf [cited 22 June 2018].
[5] URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-RW210_V1.01_Flyer.pdf [cited 22 June 2018].
[6] URL http://www.electronicnote.com/RCG/RSI%2002_A4.pdf [cited 23 June 2018].
[7] URL https://oce.jpl.nasa.gov/practices/2301.pdf [cited 8 June 2018].

10.4. Thermal Control

The thermal control subsystem is essential to ensure the operation of all spacecraft payload components and the subsystems by maintaining the temperature within the required limits. Two different types of limits are frequently defined: operational limits, which the component must remain within while operating, and survival limits, which the component must remain within at all time, even when out of power, during launch, etc. The operational temperature values, of all the components affected by a temperature, of BBIS can be found in Table 10.3. Those values are leading when designing the spacecraft.

	T_{min} [$^{\circ}C$]	T_{max} [$^{\circ}C$]
Back-up Battery	0	40
Storage Battery	0	40
Sun Sensor [8]	-30	60
Star Tracker [9]	-20	40
OBC [10]	-45	85
Small Reaction Wheels [11]	-40	60
Big Reaction Wheels [12]	-15	45
Bi-propellant Tank	-5	50
IMU [13]	-40	65
Antenna [14]	-20	60
Data Bus	-55	125
Cables [15]	-200	180
GPS [16]	-30	70

Table 10.3: Operational Temperatures of BBIS Components [20], [29].

The temperature of the spacecraft is affected by internal heat, which is produced by electronic equipment. Furthermore, it is affected by external solar radiation, the albedo of Earth and Earth flux, as discussed in Section 10.2.2. The coldest temperature the spacecraft encounters is $-267^{\circ}C$ (in shade behind the sail) and the maximum is $300^{\circ}C$ (direct sunlight). Those extremes are accounted for during the design of BBIS.

10.5. Method

The environment the BBIS is orbiting in, is discussed above. The various methods used to keep the spacecraft in the right temperature are discussed in Section 10.5.1. In order to be able to analyse more precisely how the heat is distributed in the spacecraft a finite element model set-up is discussed in Section 10.5.2.

10.5.1. Thermal Protection

During the lifetime of the spacecraft the solar absorbance of the paint changes. It is estimated that white coating present on ISS degrades with time. In its lifetime of $30\ years$ the absorbance of the

[8] URL http://www.leonardocompany.com/documents/63265270/65745274/S3_Smart_Sun_Sensor_LQ_mm07948_.pdf [cited 22 June 2018].
[9] URL https://hyperiontechnologies.nl/wp-content/uploads/2015/08/HTBST-ST400-V1.01_Flyer.pdf [cited 22 June 2018].
[10] URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-CP400.85-V1.01_Flyer.pdf [cited 22 June 2018].
[11] URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-RW210_V1.01_Flyer.pdf [cited 22 June 2018].
[12] URL http://www.electronicnote.com/RCG/RSI%2002_A4.pdf [cited 22 June 2018].
[13] URL http://files.microstrain.com/3DM-GX3-25-Attitude-Heading-Reference-System-Data-Sheet.pdf [cited 8 June 2018].
[14] URL https://www.isispace.nl/product/dipole-antenna/ [cited 23 June 2018].
[15] URL https://www.gore.com/sites/g/files/ypyipe116/files/2016-04/GORE%20Space%20Cables%20-%20Product%20Portfolio_02-10-2016.pdf [cited 24 June 2018].
[16] URL http://www.ppmgmbh.com/pdf_d/GPS%20Hardware/OEM-Boards/High%20End%20GPS%20OEM%20Boards/PolaRx2_Board.pdf [cited 24 June 2018].

white pain is expected to almost double ($\alpha_{BoL} \cong 0.18$, $\alpha_{EoL} \cong 0.33$) [30]. The probable cause of solar absorbance degradation on ISS is the ultraviolet (UV) radiation, outgassing contamination, bi-propellant deposition [30] and the presence of atomic oxygen [32]. However, it is necessary to realise BBIS is orbiting Earth at geosynchronous orbit. Therefore, no erosion of the paint due to atomic oxygen occurs. Furthermore, presence of a coating defect can lead to erosion as well, and therefore it can affect the lifespan of the coating. The emissivity degradation over time is negligible [30].

10.5.2. Finite Element Analysis

In order to analyse the heat flows the BBIS experiences during its lifetime a finite element analysis is developed in ANSYS. The thermal model analyses the worse case scenarios the spacecraft experiences during its lifetime. The model has dimensions $0.34 \times 0.34 \times 0.66\ m$ with thickness of $3.5\ mm$ for the bus. The thickness of the spacecraft is based on preliminary thermal finite element method (FEM) simulations. With a thickness of $0.45\ mm$ (preliminary estimated thickness from Chapter 11) the weight distribution was unfavourable. Therefore, a thickness of $3\ mm$ is assumed at first. After several iterations the final thickness of $3.5\ mm$ resulted in the best thermal distribution while still not making the spacecraft too heavy. The sail is a square with each side being $20.15\ m$ long and with thickness of $0.1\ mm$. The materials used in the simulation is aluminium $7075 - T73$. The size of the mesh is $0.1\ m$ as it is the most the academic licence from ANSYS supports on such a large structure. Steady-State Thermal model is used to simulate the external and internal heat flows of the BBIS.

10.6. Results

Due to the size of the spacecraft bus it is not necessary to use active thermal control to regulate its temperature. The bus dimensions, mentioned above, are relatively small and as a consequence, enough heat can be emitted through the bus itself. Therefore, it is not necessary to have a radiator. As the BBIS is not too close to the Sun the sunshield is also not necessary. The only passive thermal control component needed is paint.

10.6.1. Paint

Various paints for different parts of the BBIS are considered. This section is divided into two main parts. Spacecraft bus which elaborates on the paint used on the bus and payload which elaborates on the paint used on the solar sail.

Spacecraft Bus

Two different types of paint are needed. For the side of the bus facing the Earth which is also illuminated by the Sun a paint with high emissivity and low absorbtivity is needed. Therefore, the paint used is *GSFC White NS43C* with an emissivity 0.92 and absorbance of 0.2 [35]. The part of the bus behind the sail is painted with *GSFC Green NS53-B*, the emissivity is 0.87 and absorbtivity is 0.52 [35]. The high emissivity and a moderate absorbtivity is chosen in such a way to achieve ideal internal heat distribution. Both paints have a thickness of $0.1\ mm$ when applied on the surface of the spacecraft [17]. The thickness of the paint is based on a data sheet from *AZ Technology*, and the typical thickness of a conductive layer of a paint that is applied on the outer surface of the spacecraft. No lifetime estimation is present, therefore, it is possible more than one layer of the paint is necessary followed by a protective layer of film discussed bellow.

In order to protect the paint from the UV radiation a protective film of silicon nitride (SiN) is applied. The film is $0.7\ \mu m$ thick. This protective film does not alter the optical properties of the paint, it is thin, lightweight and has high conductivity in order to prevent the build-up of harmful potential gradients. In addition, the protective film must be free of defects. This protective film is applied on the whole bus. [32]

Payload

The solar sail has a very thin substrate and film with thermal properties which are not ideal. The solar sail has a reflective aluminium film with reflectivity of 0.91 on the side facing Earth. On the rear side of the sail a layer of kapton has an emissivity of 0.34, which is not enough to provide thermal control

[17] URL http://www.aztechnology.com/pdfs/materials-catalog.pdf [cited 13th June 2018].

for the payload. Therefore, a chromium layer on the back of the solar sail, with emissivity of 0.64, provides the thermal control needed. [11]

In order to protect the sail from pre-launch oxidation a protective film of SiN with thickness of 0.7 μm is used. In addition, to the qualities mentioned above, a layer of SiN also makes the spacecraft resistant to atomic oxygen attack. Therefore, the paint does not loose its reflectivity while still on Earth. [32]

10.6.2. Thermal Model

When all the absorbtivity and emissivity of different sides of the spacecraft are imputed and modeled in ANSYS the maximum and minimum temperature of the spacecraft is found. Figure 10.2 shows the spacecraft from the side it faces Earth. The solar sail is hidden in the image in order to be able to observe the thermal variation in the spacecraft. As expected the side behind the sail is colder as it is not exposed to any external heat flows.

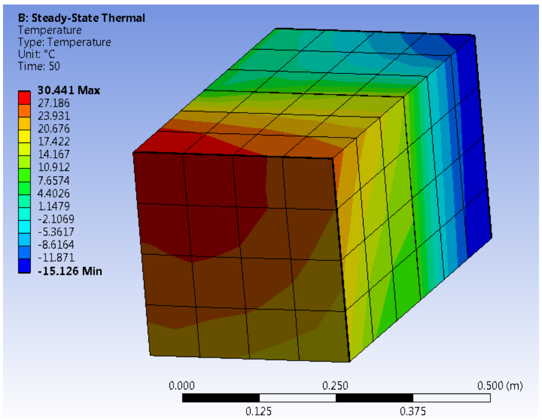


Figure 10.2: Thermal Model of the Bus with Sail Hidden.

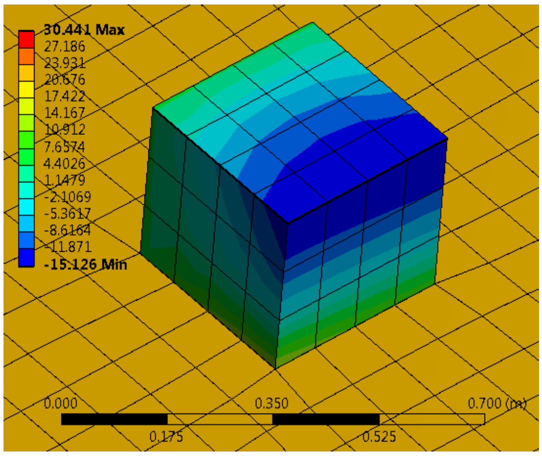


Figure 10.3: Thermal Model of the Bus with Sail.

Figure 10.3 shows the bottom (negative x-direction) and side view of the BBIS and the sail. Internal heat produced by all the electronic equipment is accounted for. However, it is necessary to realise the heat those electronics dissipate does not only heat up the environment but also the electronic equipment itself. Even when in idle those electronics still produce heat and warm up the environment surrounding them.

It can be seen the bi-propellant tank that is located in the cold part of the spacecraft is not in its ideal operational temperatures. However, it is assumed that since the thermal model is only modelled as a simple empty box with sail surrounding it, once the components are imputed in they are in a closer proximity to the tank. As a consequence the small and large reaction wheels, battery, start tracker, GPS receiver, OBC and IMU heat up their surroundings enough to keep the tank in its operational temperatures.

Figure 10.3 shows part of the sail. Since it is significantly larger than the bus it seams unnecessary to show the whole sail. Worse case conditions are when the whole front part of the sail is illuminated by the Sun, experiences albedo, Earth flux and the back side does not experience any external heat flow.

The maximum temperature, $30.44\ ^{\circ}C$, is on the front face. Therefore, all the components located in this environment have a minimum of $15^{\circ}C$ spare to heat up. The components located on the opposite side of BBIS experiences the lowest temperatures, $-\ 15.13^{\circ}C$. As those components heat up when in use there is no danger of achieving the minimum operational temperature. The average temperature of BBIS is $21.95^{\circ}C$. When the requirements from Section 10.3 are reflected upon it can be seen that **BBIS-Sys-T05-4.6**, **BBIS-Func-Temp-03** and **BBIS-Func-Temp-04** are met.

10.7. Verification & Validation

The MATLAB program used in [8] has been modified in order to fit the aforementioned situations and the same assumptions introduced in this chapter are used. The program is based on equations from

[20] which were introduced in [8]. The average temperature of the spacecraft calculated is $10.36°C$. The temperatures throughout this chapter are listed in degrees Celsius as it is easier to relate to those numbers. However, if the values are converted to Kelvin the average temperature obtained by FEM is $295.1\ K$. Temperature obtained from MATLAB program is $283.51\ K$. When those two numbers are compared the difference is 4%. That is an acceptable difference, and therefore the model is considered to be correct. In addition, the sail has a temperature of $\approx 22°C$ which influences the value of total average temperature from the FEM. However, the MATLAB model only verifies the bus temperature distribution. It was deemed unnecessary to verify the temperature distribution of the solar sail as the sail is design to withstand large range of temperatures when it sails through space.

In order to validate the paint used the spacecraft bus shall be subjected to thermal testing. Thermal testing consists of thermal cycling, thermal balance and thermal vacuum testing [30]. The thermal protection of solar sail is based on [11], therefore, it is a proven working design.

10.8. Sensitivity Analysis

In order to see how sensitive the design is based on the thickness of the spacecraft bus two new models are developed. First model in Figure 10.4, lowers the thickness of the bus by 20%. Therefore, the new thickness is $2.8\ mm$. Second model in Figure 10.5, increases the thickness of the bus by 20% which results in a new thickness of $4.2\ mm$. The new temperatures can be seen on legends in each figure.

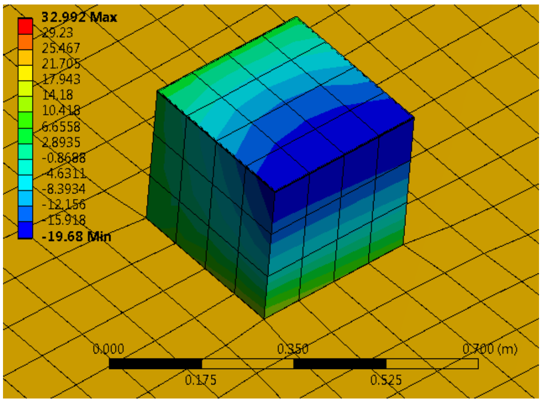


Figure 10.4: Thermal Model with 2.8 mm Bus Thickness.

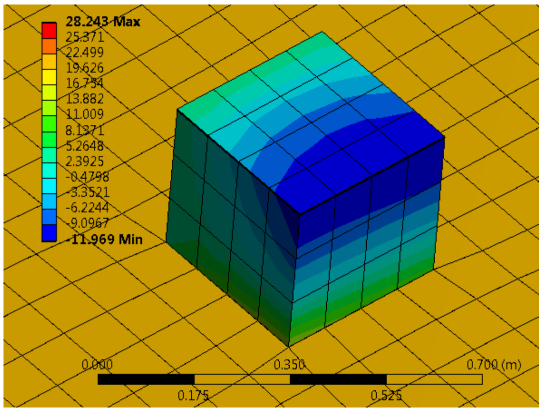


Figure 10.5: Thermal Model with 4.5 mm Bus Thickness.

Table 10.4 shows the maximum and minimum temperature of all three different bus thicknesses investigated. When the thickness is decreased by 20% the minimum temperature changes by 30% and the maximum temperature differs by 8%. Similar trend can be observed when the thickness is increased by 20%. The minimum temperature differs by 26% and the maximum temperature changes by 7.8%. As expected the larger the thickness of the bus more favourable temperatures are achieved. Thickness of $2.8\ mm$ gives unacceptably low temperatures.

Bus Thickness	**Decreased**	**Original**	**Increased**
	$2.8\ mm$	$3.5mm$	$4.2\ mm$
Min Temperature [$°C$]	-19.68	-15.13	-11.97
Max Temperature [$°C$]	32.99	30.44	28.24

Table 10.4: Effects of Different Thicknesses on Thermal Distribution.

11. Structures and Mechanisms

This chapter presents the structures and mechanisms subsystem. The requirements in which the structural design of the spacecraft is based are introduced in Section 11.1, then the subsystem is defined in Section 11.2. The results of the structural analysis are presented in Section 11.3, followed by verification and validation for the structural model, and its sensitivity analysis in Section 11.5 and 11.6, respectively.

11.1. Requirements

Since the loads in the spacecraft are only analysed at the critical conditions, requirements **BBIS-Tud-05**, **BBIS-Sys-T05-3.3**, **BBIS-Sys-T05-4.1**, **BBIS-Sys-T05-4.2**,and **BBIS-Sys-T05-4.3**, have been considered, but not yet investigated for the BBIS design. The requirements driving the design of the structure are the following.

- **BBIS-Sys-T05-3** The spacecraft shall be able to withstand all the exposed conditions during launch.
  - **BBIS-Sys-T05-3.1** The spacecraft shall be able to withstand a g-load range of 1 to 6 in launch mode.
  - **BBIS-Sys-T05-3.2** The spacecraft shall be able to withstand the vibrations ranging from $8\ Hz$ to $27\ Hz$ during launch in launch mode.

The specific numbers in this requirements are determined by the launcher vehicle, the loading conditions and the specific design goal for the strength of the structure. Therefore, the origin of these values is specified in the section developing the critical loading conditions, Section 11.2.1.

11.2. Structural Model Definition

The ultimate goal of the spacecraft structure is to withstand the loads it experiences during its lifetime. In addition, the structures, in combination with the mechanisms provides support and attachment to the other subsystems.

The bus is considered to be the primary structure of the spacecraft, it carries the loads during launch and protects the more delicate subsystems from the hostile environment. The secondary structure design is assessed separately and chosen from OTS components. In order to arrange the small components a 3-Unit CubeSat structure from Inovative Solutions in Space is chosen.[1] The weight of this structure is added to the bus and the remaining subsystems, to determine the mass distribution of the spacecraft.

11.2.1. Loading Conditions

The method presented in [8] established the need to determine the critical regions for this structure during the different flight conditions. It is not feasible to design a structure that uses different attachment and support mechanisms during its different operational phases, and therefore the structure needs to be designed to survive the worse loading conditions, which occur during launch. However, this process is not uniform, and therefore different stages are analysed based on this configuration. The different loads the spacecraft is subjected to during launch can be observed in Table 11.1.

[1]URL https://www.isispace.nl/product/3-unit-cubesat-structure/ [cited 24 June 2018].

Loading case	Longitudinal Loads [g]	Lateral Loads [g]
Lift-off	3.3	2.0
Aerodynamic Phase	3.2	2.0
Pressure oscillations / SRB end of flight	6.0	1.0
SRB Jettisoning	3.9	0.9

Table 11.1: Loading Conditions During Launch [36].

In addition, the structure shall have a natural frequency greater than $8\ Hz$ in the lateral direction (y-direction in the coordinate system in Figure 11.1) and $27\ Hz$ in the axial direction (x-direction).

11.2.2. Material Properties

The spacecraft bus and the bottom plate stiffeners are made of Aluminium 7075-T73. Its main properties can be observed in Table 11.2.

Elastic Modulus (E)	$71 \cdot 10^9\ N/m^2$
Poissons ratio (v)	0.33
Ultimate Strength	$4.8 \cdot 10^8\ N/m^2$
Yield Strength	$3.9 \cdot 10^8\ N/m^2$
Density (ρ)	$2\ 800\ kg/m^3$

Table 11.2: Aluminium 7075-T73 Properties [20].

11.2.3. Force Diagrams

As previously stated, the maximum loading cases occur during launch. Therefore, the spacecraft configuration chosen for the structural analysis is modelled as a beam with a square cross-section (the solar sail is folded around the bus centre in the y-direction), supported by the launcher adapter placed on the reflective side of the spacecraft. The final bus size, the coordinate system and a basic beam model can be observed in Figure 11.1. The thickness of $3.5\ mm$ comes as a minimum requirement from the thermal control (see Section 10.4).

The different components placed inside of the spacecraft contribute to the beam free body diagram creating a non uniformly distributed line load on the x-, y- and z-directions. Moreover, the beam experiences reaction forces and moments at the points it is attached to the launcher adapter as can be seen in Figure 11.2.

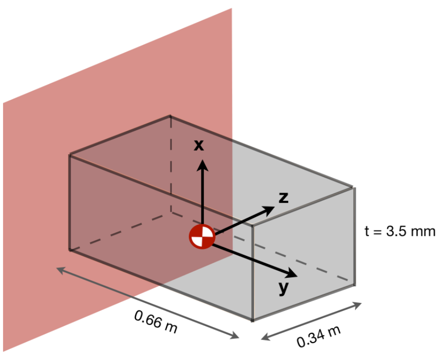


Figure 11.1: Spacecraft Model During Launch.

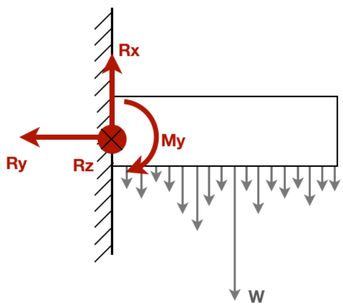


Figure 11.2: Launch Configuration Free Body Diagram.

The shear and normal force diagrams resulting from this configuration can be observed in Figure 11.3 and 11.4 , respectively.

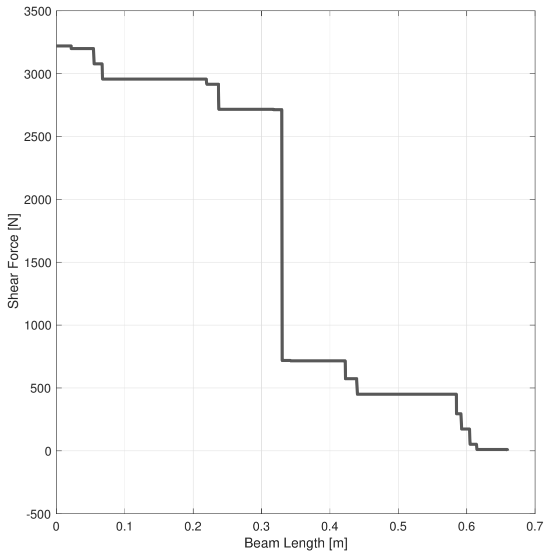


Figure 11.3: Shear Force Diagram.

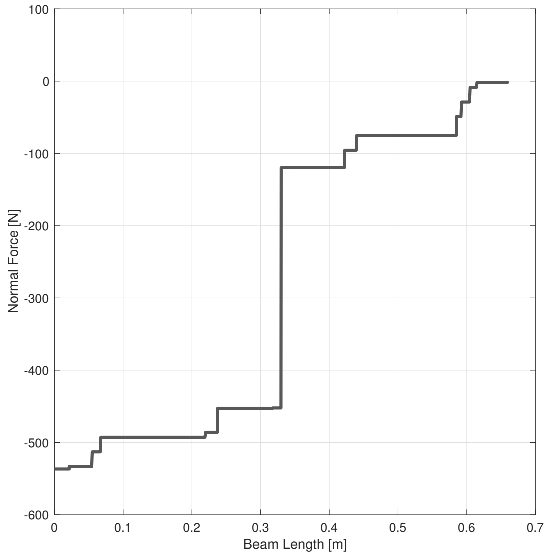


Figure 11.4: Normal Force Diagram.

The bending behaviour observed in Figure 11.5, corresponds to the bottom plate of the spacecraft bus, chosen due to the fact that it is a critical region for buckling. This plate has 3 stiffeners attached, dividing the plate surface in four equal sections. The stiffeners have a side length of $30\ mm$ and a thickness of $4\ mm$.

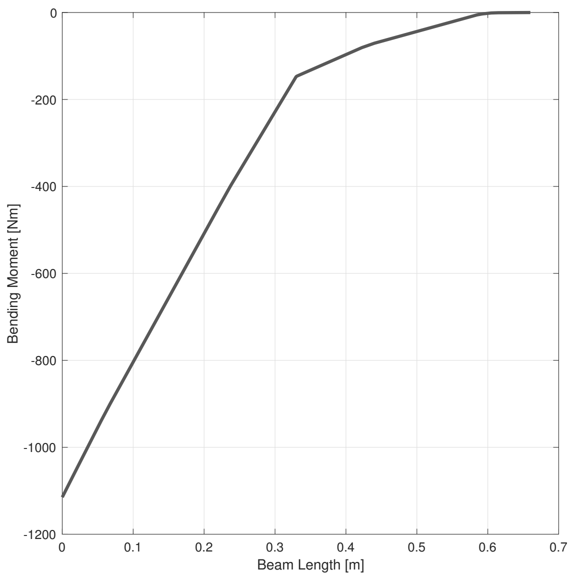


Figure 11.5: Bending Moment Diagram.

11.2.4. Centre of Gravity Location

The different components placed inside of the spacecraft bus are arranged in such a way that the centre of gravity location is optimised, this means that it is almost coincident with the geometric centre of the bus. In order to achieve this goal, similar components are placed symmetric with respect to the centre line. From this mass distribution it is obtained that the center of gravity is located $0.1739\ m$ above the bottom plate, $0.3265\ m$ from the attached face along the y-axis, and its deviated $4\ mm$ from the bus centre line in the z-direction.

11.2.5. Section Properties

Due to the fact that critical stresses occur at the bottom and top plates (they are subjected to bending, compression and shear), it is necessary to determine the moment of inertia of the entire cross section. It equals to $5.34 \cdot 10^4\ m^4$ for the configuration of bus plus stiffeners. The other property that should be studied is buckling of the bottom plate. From [37], it is obtained that the buckling of thin plates follows Equation 11.1.

$$\sigma_{cr} = 3.6\frac{E}{1-v^2} \cdot (\frac{t_h}{b})^2 \tag{11.1}$$

Where σ_{cr} corresponds to the critical buckling stress, E is the modulus of elasticity of the material, v is the Poisson's ratio, t_h is the thickness of the plate and b is the distance between stiffeners. Plugging in the values leads to a buckling stress of $4.8633 \cdot 10^2 MPa$. The upcoming calculations show if the applied compression in the bottom plate is smaller than this value, and therefore if the structure fails.

11.3. Results

The results of the structural analysis of the spacecraft inside of the launcher are presented in the following subsections.

11.3.1. Stress distribution

The two main failure modes that can occur are, as was previously stated, buckling of the bottom plate and yield of the top plate. In order to determine if those failure modes occur it is necessary to determine the axial stresses on the bottom plate. In addition, the total Von Mises stress on the top plate have to be determined as well. The axial stress caused by the compressive forces (axial loading) and the bending is obtained according to Equation 11.2.

$$\sigma_a = \frac{F_{eq}}{A} \cdot 1.25 \tag{11.2}$$

Where σ_a stands for axial stress, A is the cross-sectional area of the analysed plate, 1.25 is the margin of safety (substituted by 1.1 for yield assessment) and F_{eq} is the combined load from compression and bending as shown in Equation 11.3.

$$F_{eq} = F + \frac{M \cdot x}{I} \tag{11.3}$$

Here F is the compressive load, that changes along the beam according to the normal force diagram. M is the correspondent internal moment (from the bending diagram), x is the distance between the bottom plate and the centre of gravity in the x-direction and I is the area moment of inertia of the cross section. From the above-mentioned equations results the axial stress distribution for the bottom and top plate in Figure 11.6 and 11.7, respectively. The minimal difference between both graphs is due to the fact that the centre of gravity is not located exactly in the middle of the bus, and due to the difference in plate cross-sections (the top plate does not have any stiffeners).

From Figure 11.6 it is observed that the maximum compressive stress in the bottom plate equals $2.074 \cdot 10^2\ MPa$. Comparing this value to the buckling stress, it is possible to affirm that the structure does not buckle under the applied loads. The safety margin for buckling is then 2.3.

In order to analyse failure of the top plate, it is necessary to combine the maximum stress in Figure 11.7 with the average shear caused by the lateral force in the direction of the z-axis. The maximum shear stress is experienced in the section close to the beam attachment, and it equals $0.4035\ MPa$. Combining this with a maximum compressive stress of $318\ MPa$, using the Von Mises equation, leads to a value of $317\ MPa$ for the maximum stress of the top plate. Considering the material yields at a value of $390\ MPa$ and fails at $460\ MPa$, the safety margin for yield and failure are 1.23 and 1.45 respectively.

As it was stated earlier, different launch phases were analysed in order to determine the critical stage. The numbers mentioned above correspond to the loading conditions of *Pressure oscillations / SRB end of flight*, which resulted in the higher structural stresses.

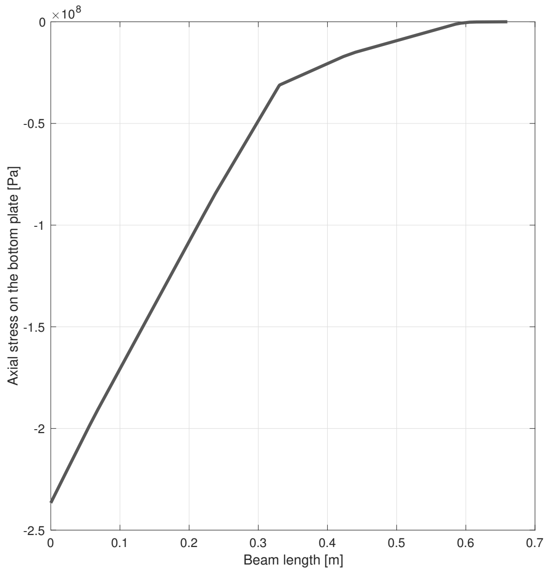


Figure 11.6: Axial Stress on the Bottom Plate.

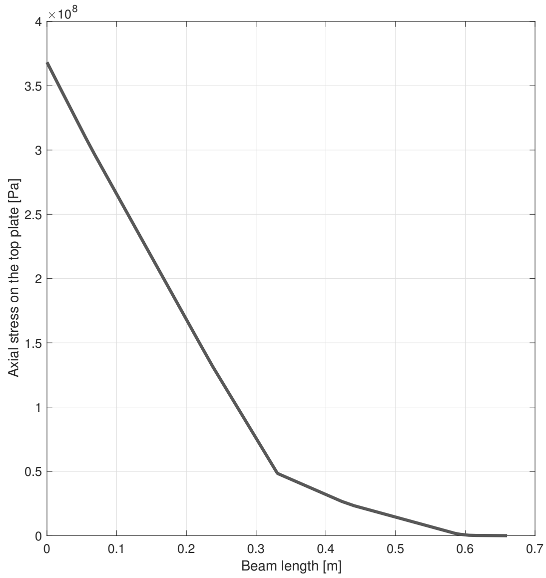


Figure 11.7: Axial Stress on the Top Plate.

11.3.2. Frequency requirement

The last failure mode to check in order to ensure that the structure is suitable for this mission, is the failure caused by the vibrations of the launcher. The rigidity of the beam-modelled spacecraft is determined by its natural frequency, that should be greater than the frequencies mentioned in Section 11.2.1. The beam natural frequencies in the axial (y-axis) and lateral (x-axis) directions are given by Equation 11.4 and 11.5.

$$f_{nat_{axial}} = 0.25\sqrt{\frac{AE}{ml}} \qquad (11.4)$$

$$f_{nat_{lat}} = 0.56\sqrt{\frac{EI}{ml^3}} \qquad (11.5)$$

In these equations, E corresponds to the modulus of elasticity of the material, I is the moment of inertia of the cross-section, A is the area of the cross-section, m is the total spacecraft mass, and l is the length of the bus. This leads to a value of $795\ Hz$ for the natural frequency in the y-direction, and $904\ Hz$ in the x-direction, significantly greater than the required, which means the spacecraft does not fail under vibration.

11.3.3. Conclusion

The main load carrying structure of the spacecraft is formed by the bus with a thickness of $3.5\ mm$, a length of $0.66\ m$ and a square cross-section with side $0.34\ m$. Three L-shaped stiffeners, with a thickness of $4\ mm$ and a side length of $30\ mm$ are added along the bottom plate, optimising the structure performance under buckling. It is determined that the structure does not fail under the known applied loads. Requirements **BBIS-Sys-T05-3**, **BBIS-Sys-T05-3.1** and **BBIS-Sys-T05-3.2** are therefore met.

The secondary structure of the spacecraft is formed by a CubeSat shelf system. This added to the bus and stiffeners mass, and accounting for a 10% structural weight increase due to bolts and other joints, leads to a total structural mass of $14.13\ kg$.

11.4. Last iteration

At a further stage of the project it was discovered that the payload required the use of $3.54\ kg$ of rope. This significant increase in mass lead to updated values for the maximum stresses in both plates. The results of a further iteration, including the effect of the ropes, are summarised, and compared to previous values, in Table 11.3.

Parameter	Initial values	Values after adding rope
Maximum Axial Stress (bottom plate) [MPa]	207	236
Buckling Safety Margin [$-$]	2.3	2.06
Maximum Axial Stress (top plate) [MPa]	318	365
Von Mises Stress on Top Plate [MPa]	318	368
Yield Safety Margin [$-$]	1.23	1.05
Failure Safety Margin [$-$]	1.45	1.25

Table 11.3: New Structural Characteristics.

The structure is still able to withstand the loads. However, the safety margin for yield on the top plate is now critical. Therefore, a new iteration is needed in order to optimise the design.

11.5. Verification and Validation

The formulas mentioned on the previous section are applied to the specific design case with the help of a MATLAB code. This code is verified observing that the moment and stresses at the non-supported end of the beam have a limit of zero. Further code verification is performed by setting the loads to zero and observing how the stresses become also zero.

Validation of the structures subsystem is done by subjecting the bus to certain tests. During which the structure undergoes tensile and compressive forces. These tests can be destructive, the structure is loaded until it fails, or non-destructive, in which the structure does not suffer any damage. Examples of non-destructive procedures are radioactive, ultrasonic or infrared testing.

11.6. Sensitivity Analysis

In order to proof the change in structural characteristics when a small variation is introduced, the program is ran one more time with two new thicknesses. Similar to Section 10.8, the thickness is reduced and increased by a factor or 20%. The results of this changes are compared to the original values in Table 11.4.

Parameter	2.8 mm	3.5 mm	4.2 mm
Maximum Axial Stress (bottom plate) [MPa]	321	236	183
Buckling Safety Margin [$-$]	0.96	2.059	3.8
Maximum Axial Stress (top plate) [MPa]	545	365	268
Von Mises Stress on Top Plate [MPa]	318	368	268
Yield Safety Margin [$-$]	0.71	1.05	1.45
Failure Safety Margin [$-$]	0.84	1.25	1.71
Structural Mass [kg]	11.28	14.13	16.14

Table 11.4: Structural Sensitivity Analysis.

III
Mission Development

12. Orbit Insertion and Disposal

This chapter deals with the techniques used to achieve the final orbit of the BBIS, and with the method chosen for the disposal of the spacecraft at the end-of-life. First, a suitable vehicle and deployment method is assessed in Section 12.1 and 12.2, respectively, followed by the explanation of planned and unplanned disposal in Section 12.3.

12.1. Launch

In order to introduce the spacecraft into the desired orbit, it is necessary to select a suitable launch vehicle. Different options are considered in this section, which implies that different configurations, flight envelopes and load cases are considered. The launcher needs to bring the spacecraft to GSO, which is circular orbit an altitude of $35,786\ km$ and $0°$ inclination. The goal of this section is to analyse the options and determine the most cost effective launcher.

12.1.1. Requirements

The requirements that the launcher company can ask for are identified in [3]. The relevant requirements are listed below. Due to changes in the launcher arrangement, some of the preliminary requirements are changed to meet the performance goal. **BBIS-Sys-L01-1** and **BBIS-Sys-L01-2** are therefore changed to **BBIS-Sys-L01-3** and **BBIS-Sys-L01-4**, respectively. The final requirements for the launcher are stated below.

- **BBIS-Sys-L01-3** The total amount of spacecraft per launcher shall occupy a maximum volume of $118.74m^3$.
- **BBIS-Sys-L01-4** The total weight of spacecraft per launcher shall not exceed $6000kg$.
- **BBIS-Sys-L04-1** The spacecraft shall have connector to connect the launcher and the spacecraft.
- **BBIS-Sys-L04-2** The spacecraft shall have no contact with the launcher except for the connection point(s).

In addition, requirements **BBIS-Lan-01** and **BBIS-Lan-04**, are met, since they are the ultimate goal of this system design process.

12.1.2. Launcher Selection Criteria

The main criteria are the cost and reliability of the launcher. In order to optimise total cost for the final design, a MATLAB program is created. The program uses the characteristic of each individual spacecraft to determine the amount that can be included in each launcher. The launcher capacity is determined based on the volume available inside the payload fairing and also based on the total payload mass as it is established in **BBIS-Sys-L01-3** and **BBIS-Sys-L01-4**. Different launch vehicles are studied for the cost determination, and therefore, included in the program. The program flow can be observed in Figure 12.1.

American launchers are not considered in the trade-off due to the U.S. Code Title 51 Subtitle V Chapter 509 § 50911[38]:

> **b. Launching.-** No holder of a license under this chapter may launch a payload containing any material to be used for purposes of obtrusive space advertising.

The definition of "obtrusive space advertising" is stated to be "advertising in outer space that is capable of being recognised by a human being on the surface of the Earth without the aid of a telescope or other technological device."[38], therefore, according to requirements **BBIS-Gov-01** and **BBIS-Sys-G01-01**, it is not possible to launch BBIS from the USA.

Combining the output of the program with statistical data on launch failure rate, it is possible to perform a trade-off for each of the possibilities and determine the most suitable option for this specific project.

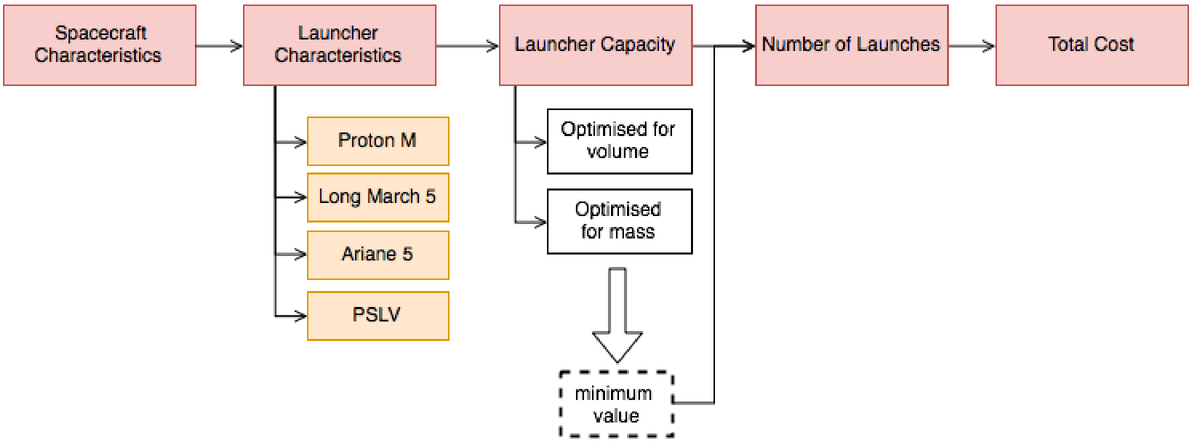


Figure 12.1: Cost Estimation Process.

12.1.3. Launcher Selection

Making use of the aforementioned criteria, it was determined that the most suitable launch vehicle for our mission is the Ariane V. This implies that the procedures and loads analysed during previous steps in [8] are valid for the current design phase.

The Ariane V is mainly propelled by a main core stage and two solid boosters. The main core is formed by a Vulcain 2 thruster loaded with 175 tonnes of liquid oxygen and liquid hydrogen. This stage is ignited for 6.05 seconds, during this time, the on-board computer checks the good functioning of the engine and ignites the solid boosters giving authorisation to lift-off. The main engine is shut-down when the intermediate target orbit is reached. Then this stage de-attaches and re-enters the Earth's atmosphere to crash in the Atlantic Ocean. [36]

This launcher has a typical insertion duration of 25-35 minutes. The possible inaccuracy of the target orbit is of $0.02°$ for the inclination, and a semi-major axis deviation of $40\ km$ [36]. Moreover, Ariane V has a reliability of 96.9%.

Due to the amount of spacecraft that need to be placed inside of each vehicle, new adapting methods are studied for this mission to ensure the compliance of **BBIS-Sys-L04-1** and **BBIS-Sys-L04-2**. In order to use the available adaptors, a rigid beam structure is placed inside of the launcher as shown in Figure 12.2. The different spacecraft forming the swarm are then attached at the side of each beam according to the distribution in Figure 12.3.

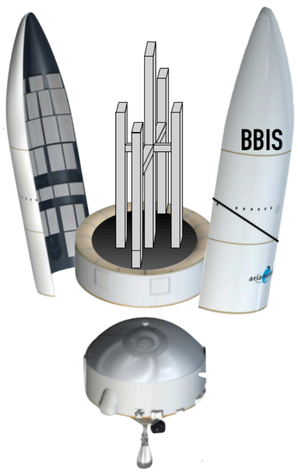


Figure 12.2: Adaptor Structure Inside of Launcher.

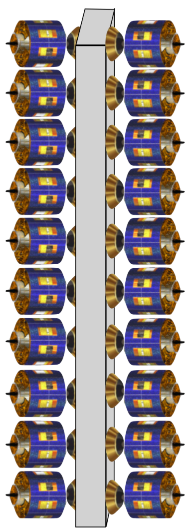


Figure 12.3: Spacecraft Arrangement.

From the aforementioned program, it is obtained that in order to insert the 900 spacecraft into the desired orbit, 9 launches of Ariane V are needed.

In addition, the team has decided that the space left available in the last launcher can be offered to other missions in order to increase the profitability of the project.

12.1.4. Launch Site

The designated spaceport for the Ariane V is the Guiana Space Centre (CSG), located North of Kourou in French Guiana. Its location is particularly convenient due to its proximity to the equator and to the coast, which means not only that the facilities are accessible via sea, but also that in case of launch failure, there is a smaller chance of debris hitting populated areas.

This spaceport has been operational since 1968, and has a launch distribution of up to 10 missions per year. CSG is also the location from which other Ariane rockets, Vega and Soyuz missions depart. Its optimal location represents an advantage from the sustainable point of view due to the fact that a smaller amount of fuel is required for each launch.

12.2. Deployment

Delivering 100 spacecraft per launch vehicle is one of the biggest challenges of this project. The reference mission used in this case is the PSLV (Polar Satellite Launch Vehicle) launch, which holds the record by launching 104 satellites with one vehicle [1]. The exact method used for this delivery is classified information. However, based on videos showing the deployment process, and analysing the techniques used with the Ariane V, it is possible to sketch a deployment procedure that should be further studied in next project stages.

As it was previously explained, the spacecraft are carried by the launcher with the help of a beam structure attached to the spacecraft adaptor. When the launcher reaches its target orbit, first, the fairing separates. Then all the spacecraft deploy one omni-directional antenna, so crosslink communication is possible. This is done to prevent a collision during separation. Small explosive charges similar to the ones used in re-entry vehicles are placed in the spacecraft connection with the beam structure. These charges are ignited causing a reaction between launch structure and spacecraft, when the safety distance between the spacecraft has been reached, the propulsion system of each individual spacecraft is started and the spacecraft are brought to the final orbit.

The target orbit of the spacecraft will be slightly higher or lower than GSO, which is favourable. This results in a orbital time difference. The further the orbit is away from GSO, the bigger the relative velocity is with the designated GSO. However, more ΔV is required.

For the 5×180 formation, the total deployment time can be estimated dependent on the orbital time of the launcher's orbit. The following calculation assumes a circular launcher orbit, that all spacecraft needs the same time to do manoeuvre, five spacecraft are separated and manoeuvre simultaneously and all 900 spacecraft are launched with one launcher. Equation 12.1 determines the distance between the spacecraft when they are deployed one second after each other.

$$\Delta s_{GSO}/t_{orbit} = |t_{launcher} - t_{GSO}| V_{GSO}/t_{launcher} \tag{12.1}$$

Where Δs_{GSO} is the distance between the spacecraft in km. t is the time in s, t_{orbit} is the orbital period in s and V_{GSO} is the velocity in a GSO. Equation 12.2 determines required time between the deployments, to get the desired distance between the spacecraft.

$$t = \frac{\Delta s_{formation}}{\Delta s_{GSO}/t_{orbit}} \tag{12.2}$$

Where $\Delta s_{formation}$ is 7.86 km for the 5 times 180 formation. Than the total time can be calculated get the spacecraft flying in formation with Equation 12.3, note the manoeuvre time of the first spacecraft is not accounted in this equation.

$$t_{total} = t \cdot 179 \tag{12.3}$$

[1]URL https://www.theverge.com/2017/2/14/14601938/india-pslv-rocket-launch-satellites-planet-doves [cited 24 June 2018].

For an altitude of $40\ km$ above GSO, the total time to get all the spacecraft in the correct formation in GSO is about $90\ hours$. Note that this is an estimation, due to the required assumptions. However, when the launcher gets exactly in GSO, the spacecraft first needs to go to a slightly higher or lower orbit to fly in the desired formation. Note that the that the spacecraft does not require to consume any power when it is attached to the adopter.

12.3. Disposal

This section describes the section starts with the requirements for disposal in Section 12.3.1. Followed by the plan for planned and unplanned disposal in Section 12.3.2 and Section 12.3.3, respectively.

12.3.1. Requirements

The requirements that are applicable to the disposal of the spacecraft were identified in [3]. Requirement **BBIS-Sys-SA02-01.1** has been determined to be $34\ m/s$ for a safe travel away from GSO. The relevant requirements are listed below.

- **BBIS-Gov-04** The spacecraft shall comply with all Inter-Agency Space Debris Coordination Committee regulations.
- **BBIS-SA-02** The spacecraft shall be disposed after service so that the risk of collision is less than 0.1% in the next $10\ 000\ years$.
- **BBIS-Sys-SA02-01** The spacecraft shall dispose itself in a controlled re-entry or in a suitable graveyard orbit.
- **BBIS-Sys-SA02-01.1** At the end of the mission the spacecraft shall have at least $34\ m/s$ velocity increment left.

12.3.2. Planned Disposal

The BBIS orbits around the Earth in GSO. In [8] it is stated that the BBIS will be burned in the atmosphere at end-of-life, however, with the new orbit determined in Chapter 5 it is decided to dispose the BBIS in a graveyard orbit above the initial orbit. De-orbiting the BBIS satellites back to Earth would require a disproportional amount of ΔV, meaning more propellant is needed, and also more launches. Leaving all spacecraft in a graveyard orbit is thus a more sustainable option. The recommended altitude increase imposed by the Inter-Agency Space Debris Coordination Committee is calculated with Equation 12.4 which immediately verifies requirement **BBIS-Gov-04** [39].

$$\Delta h = 235 + 1000 \cdot C_R \frac{A}{m} \tag{12.4}$$

In this equation, C_r is a solar radiation coefficient between 0 and 2 depending on the reflectivity, A is the cross-sectional area in m^2 and m is the spacecraft mass in kg. It has been chosen to use average values of the the before-named variables, because the attitude of the spacecraft cannot be controlled after end-of-life. Before the end-of-life procedure is finalised, a minimum angular velocity of $360°$ per season is enforced. In Chapter 6, it has been determined that the solar sail has an area of $406\ m^2$, due to the angular velocity the average frontal area is half of this: $203\ m^2$. The solar radiation coefficient of the front- and backside of the solar sail is 1.65 [40] and 1.91 (see Chapter 6), respectively, therefore an average of 1.78 is used. This results in a graveyard orbit $12\ 280\ km$ above GSO.

The manoeuvre to the graveyard orbit is split in three. Firstly, the propulsion system is used to get the spacecraft out of GSO and to increase the spacing between the spacecraft. Secondly, the solar sail is used to get the spacecraft in the final graveyard orbit, and thirdly, all remaining fuel is burned to avoid exploding danger. It is found that this is the most sustainable manner to put all spacecraft in a graveyard orbit, it will require the least amount of launches and least amount of fuel compared to removal with the propulsion system. The manoeuvre is shown in Figure 12.4, the blue line shows the original orbit in GSO, the white line illustrates the orbit after the burns, the red line shows the manoeuvre done by the mirror and the black line shows the graveyard orbit.

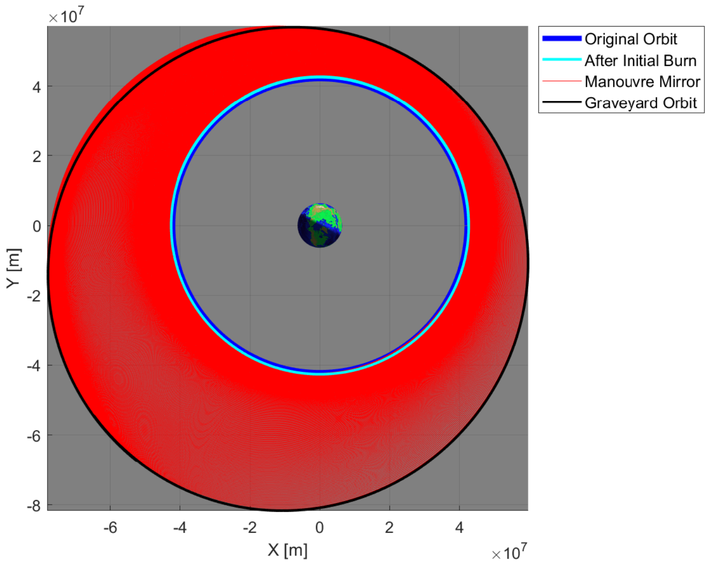


Figure 12.4: End-of-Life Manoeuvre.

For the initial burn to move away from GSO, two burns of $17\ m/s$ each are accounted for as explained in Section 5.5.2. This increases the altitude of the spacecraft with $431\ km$, while keeping a circular orbit. Herewith, new spacecraft can use that space in GSO immediately. Herewith, requirement **BBIS-Sys-SA02-01** and **BBIS-Sys-SA02-01.1** are met.

Afterwards, the mirror is used to move the spacecraft to the right altitude. Half of the orbit the mirror will accelerate the spacecraft, and the the other half the mirror will be pointed such that the solar pressure drag is minimised. This manoeuvre will take approximately $246 orbits$ and $362\ days$.

All fuel needs to be used when the graveyard orbit is reached to avoid the possibility that the spacecraft explode. Therefore, the manoeuvre is speed up by using all unused fuel at the end. It has been chosen to only use the remaining fuel at the end, because of the possibility that AODCS fails, the attitude of the spacecraft still needs to be controlled which can ultimately be done with the propulsion system. Furthermore, putting the spacecraft in a graveyard orbit is ought to be enough to verify requirement **BBIS-SA-02**.

12.3.3. Unplanned Disposal

A spacecraft should be removed from the formation when subsystems form critical danger for the operation of the spacecraft, otherwise, it cannot be guaranteed that the spacecraft do not collide. For different failures in the system the approach for disposal is different. All subsystems are listed below.

- **Communication**
  - *Interlink* If one of the interlink connections fail, the spacecraft can still be operated.
  - *Ground Link* If the ground link fails, the spacecraft is able to communicate to the ground via other spacecraft. Therefore, this does not necessarily have to be a problem. However, during the normal de-orbit procedure the spacecraft are moved away from each other which would mean that the spacecraft cannot be controlled from the ground anymore. Therefore, the manoeuvre to the graveyard orbit should be done in a close formation with at least one other spacecraft.
  - *More Than One Antenna* If more than one of the three antennas fails, the spacecraft should be moved to the graveyard orbit.
- **EPS**
  - *Batteries* If the emergency battery fails, a normal battery should be allocated as emergency battery. Only if one operational battery is left, the spacecraft should be moved to the graveyard orbit.

  - *Solar Cells* If the solar cells fail, the emergencies batteries are able to provide energy for at least half an hour with all of the subsystems on as explained in Section 9.5. During this period, the ground should be notified that the solar cells failed. The ground station gives clearance to move out of the formation. The spacecraft should increase its inclination and increase its eccentricity. Herewith, the spacecraft does not cross any GSO orbit.
  - *Electric Circuit* If the EPS fails, the spacecraft is uncontrollable. There is no clear solution for the removal of the spacecraft. The other spacecraft can only be positioned in such a direction that they avoid the spacecraft which failed.

- **AODCS**
  - *Attitude Determination* If a Sun sensor or star tracker fails, the spacecraft should be moved to the graveyard orbit, because orbit manoeuvres require attitude determination.
  - *Attitude Control* If the attitude control fails, the propulsion system can act as attitude control for a certain period of time. Therefore, the spacecraft is able to operate for a period of time, only not for its designed lifetime, because the spacecraft does not have enough fuel. The spacecraft should be moved to a graveyard orbit If there is only fuel left for this procedure.
  - *Reaction Wheel Failure*

- **Propulsion System**
  - *Thruster Failure* If more than one thruster fails, the probability is high that the spacecraft will still be able to have full attitude and orbit control. The thrusters are subdivided in three groups: thrusters 1, 2, 3 and 4 are one group, thrusters 5, 6, 7 and 8 are the second group and thrusters 9, 10, 11 and 12 are the third group. One thruster from each group can fail without causing problems in the spacecraft's functionality. If more than one thruster within a single group fails, the spacecraft loses part of the desired control and will thus not anymore be useful. This spacecraft should then be removed from the formation.
  - *Total Failure* If the total propulsion system fails, the spacecraft should be moved out of the formation with the use of solar pressure. The inclination should be increased to minimise the amount of interceptions with the initial orbit, and the size of the orbit should be increased.

- **Payload** If the payload fails, the spacecraft does not serve any purpose anymore and only forms a danger for the swarm. The spacecraft should be removed from the formation.

13. Operations and Logistics

This chapter discusses the operations and logistics related to BBIS. Section 13.1 discusses American legislation that should be considered. The project design and development strategy and the manufacturing, assembly and integration plan are discussed in Section 13.2 and Section 13.3, respectively. Then, in Section 13.4, the collision protection system is discussed.

13.1. International Traffic in Arms Regulations

International Traffic in Arms Regulations (ITAR) is an American regulatory regime that controls the import and export of technologies that could be used against the USA. In addition to those restrictions, ITAR includes space related technology and research. Therefore, it can be difficult to order COTS components from the USA as ITAR may be involved in the production process. Nevertheless, if ITAR has contributed to the component production, this is usually mentioned on the product's data sheet. Some of the components that have been selected, the reaction wheels and propellant tanks for example, are produced by American companies. Additionally, the solar sail used in the BBIS design is also an American product. It is not possible to launch from the USA, as previously determined in [3]. Assembling the components in the USA would furthermore lead to difficulties due to common delays caused by United States Customs and Border Protection. Therefore, it is favourable to have a European based company to order, integrate and assemble the COTS components for BBIS. Europe has a good relationship with USA, unlike other countries such as Russia, India, Turkey, Iran etc., to which is it difficult to get American products delivered. [41]

13.2. Project Design and Development Strategy

This section discusses the design and development phase of the project after the DSE. A time line of the whole project is furthermore presented. In Figure 13.2 it can be observed there are 7 different phases. Phase 0 is the pre-DSE and DSE phase of the project followed by phase A in which the feasibility of the project is determined. Phase B focuses on the design of the project and contains the preliminary design review (PDR). Development of the spacecraft and several test models are part of phase C which ends with critical design review (CDR). Phase D focuses on qualification and production of the spacecraft which is followed by launch and operation of the spacecraft which is part of phase E. Phase F show the disposal of the spacecraft.

As the whole concept of formation flying is more of an idea rather then a reality the timelines of other pioneering missions are compared. The idea of a habitable artificial satellite emerged in early $40's$.[1] However, only in 1986 it was proven that such a structure can orbit Earth when USSR launch the Mir station. The idea of an extra-terrestrial observatory is first introduced in 1946.[2] In 1979 the budget is assigned to developed $2.4\ m$ main mirror of a telescope and only in 1990 is the Hubble Telescope launched.[2] Therefore, the idea of orbiting station took $\approx 40\ years$ to become reality and the idea of an extra-terrestrial observatory took $46\ years$ to develop and launch. More technical resources are available now than $70\ years$ ago. Therefore, it is expected the design and development of BBIS is not going to take $40\ years$. The payload of the mission is not a COTS component and formation flying is not a working concept. Therefore, it is not expected that requirement **BBIS-Tud-03**, BBIS is launched by 1st of January 2028, is going to be met. It is expected the BBIS is going to launch in year 2043, therefore, the first five phases are expected to take approximately $25\ years$.

The expected timeline of the whole mission is listed bellow. The expected timeline in a Gantt chart format can be found in Figure 13.1.

[1] URL https://www.iss-casis.org/about/iss-timeline/ [cited 26 June 2018].
[2] URL http://www.spacetelescope.org/about/history/timeline/ [cited 26 June 2018].

- Phase 0: $11\ weeks$
- Phase A: $1.5\ years$
- Phase B: $\approx\ 10\ years$
- Phase C: $\approx\ 10\ years$
- Phase D: $3\ years$
- Phase E: $\approx\ 20\ years$
- Phase F: $1\ year$

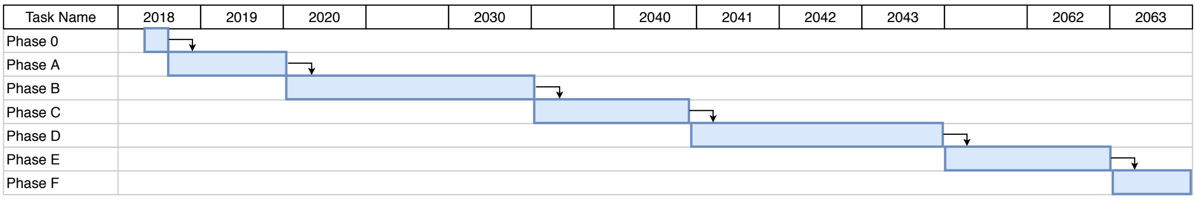


Figure 13.1: Seven Phases of BBIS.

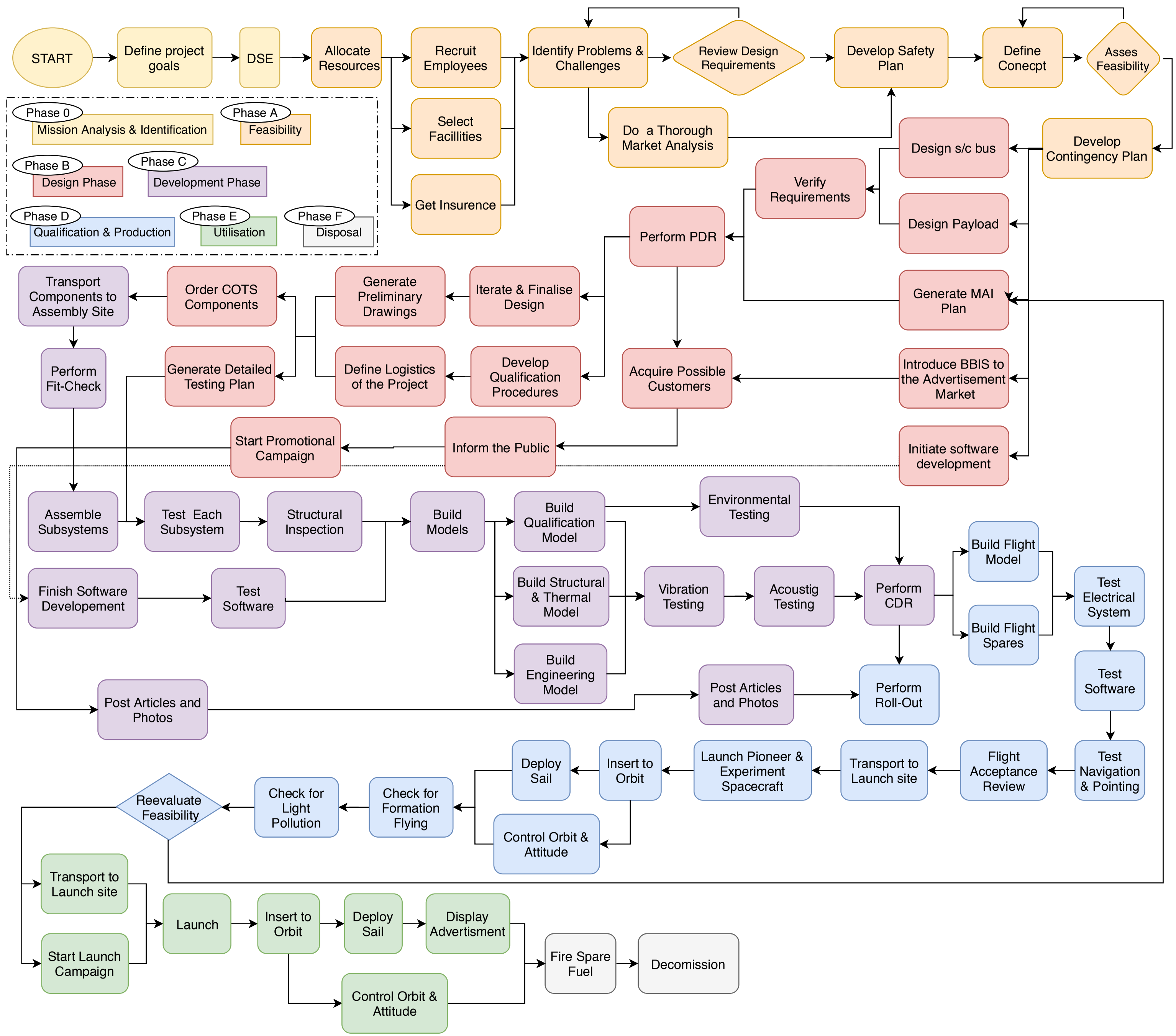


Figure 13.2: Project Design and Development.

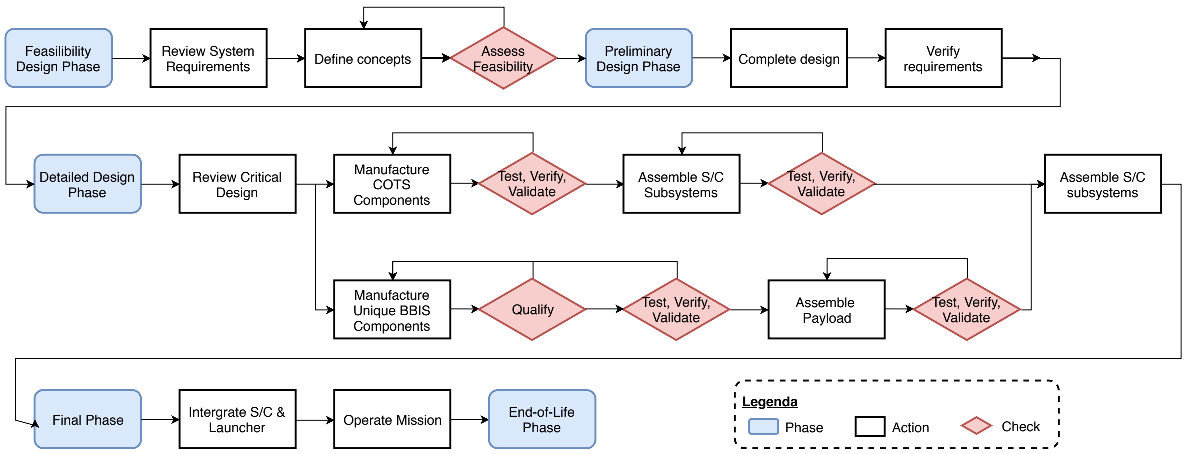


Figure 13.3: Manufacturing, Assembly and Integration Plan.

13.3. Manufacturing, Assembly and Integration Plan

The manufacturing, assembly and integration plan (MAI Plan) provides an overview of the activities required to construct the spacecraft and realise the formation in orbit. A production plan in included, which elaborates on the MAI plan, it can be seen in Figure 13.3. Several activities can be performed simultaneously, as is explained during the specification of the three phases. During testing and qualification the components or assemblies can be found inadequate, which takes the process a step back, and thus more time will be spend on manufacturing or assembling. Once the detailed design phase of BBIS is finished, the manufacturing phase of the MAI plan starts.

Manufacturing

After the design phase, all components need to be manufactured. Most of the components are part of COTS components or subsystems, such as the thrusters or the reaction wheels, those are already qualified for use on spacecraft. Some components, like the solar sail and the booms, are unique for every mission and therefore have a longer manufacturing time. These components need extensive qualification and testing before successful use is guaranteed. For the production of all components for the 900 spacecraft, assembly lines are efficient to use. After manufacturing, all components, both COTS and developed for this mission, are tested and certified. This manufacturing, testing and certificating can be done simultaneously to save time. Simultaneously with the manufacturing of the spacecraft, the launcher is manufactured and tested.

Assembly

After manufacturing and testing, the components are assembled to subsystems and tested again. This is done at several locations. All subsystems are then transported to a location as close to the launch site as possible and assembled in the spacecraft, meeting requirement **BBIS-Man-02**. Next, the complete system is tested as well. If this test is executed without any irregularities are observed, the next phase can start.

Integration

During the integration phase, the assembled and tested spacecraft can be implemented in the launcher. The details of the configuration used can be found in Section 12.1. Final checks are then performed before permission for launch is given. Not all spacecraft are launched at the same time, so tests can be performed while the formation in the orbit is not yet complete. This can result in modifications for better performance in next generation spacecraft that form the rest of the formation.

Because the exact manufacturing and assembling conditions are not known yet, requirements **BBIS-Sys-T05-1**, **BBIS-Sys-T05-1.1**, **BBIS-Sys-T05-1.2**, **BBIS-Sys-T05-1.3** and **BBIS-Sys-T05-1.4** cannot be verified at this moment. Transportation methods and thus conditions are also not specified, so requirements **BBIS-Sys-T05-2**, **BBIS-Sys-T05-2.1** and **BBIS-Sys-T05-2.2** cannot be verified yet either. Requirements **BBIS-Man-01** and **BBIS-SUp-01** are also not verified, because no specifications are given on the delivery times of the components unique for this mission. The verification of these requirements should be done in future investigation.

As specified in Figure 13.3, not all the components on BBIS are COTS, so requirement **BBIS-SUp-02** is not met. Requirement **BBIS-Sys-P02-1** is discarded, as production is not done in the USA, but in Europe.

13.4. Collision Protection System

In [8], it was mentioned that the BBIS needs to be protected against collisions. Now that the final stage of the BBIS design has been reached, the collision protection system has to be re-assessed.

A separate protection mechanism will be used for large (diameter $> 10\ cm$) and for small (diameter $< 10\ cm$) debris objects. A collision with a large debris object can be catastrophic, which is most often not the case when a collision with a small object occurs.[3] These two categories of space debris objects are thus considered separately.

To estimate how many collisions will occur over the lifetime of the BBIS, space debris statistics are used. First, the collision probability in the present is determined. Then, prediction models are used to estimate the amount of collisions in the future.

Currently, an arbitrary GEO satellite is expected to collide with a large debris object every $50\ years$. The collision frequency of a GEO satellite with a small debris object is estimated to be once every $4\ years$.[42] The expected collision probability for a spacecraft is thus estimated to be 0.02 for large debris objects and 0.25 for small objects.

Predictions are available for the amount of spacecraft and debris objects in LEO.[4] It is assumed that the amount of debris in LEO increases at the same rate as the amount of debris in GEO and that the amount of debris in space has a linear relation to the expected amount of collisions per spacecraft. It is estimated that the amount of debris in GEO at the end of the BBIS's expected lifetime will be 3.8 times higher than the current amount of debris.

Using the current collision likelihood and the expected increase in space debris, and thus, the expected increase in collisions, an estimation can be made of he amount of collisions that the BBIS spacecraft will be subjected to over their estimated lifetime from 2028 to 2048. The expected amount of collisions per spacecraft over the 20-year lifetime of the BBIS is:

- 0.76 for large debris, and
- 9.5 for small debris.

It is thus expected that each BBIS spacecraft will be hit by an average of 0.76 large debris object over the course of its lifetime. On a population of 900 spacecraft, this would mean that an expected 687 satellites are will be hit by debris large enough to cause catastrophic damage. This is not acceptable, which is why it is decided that the BBIS should be able to perform an avoidance manoeuvre in case debris is approaching. This is done by using the Space Surveillance Network, which contains a catalogue of all known objects orbiting the Earth, along with a prediction of their orbits.[5] The BBIS spacecraft will have a margin in the amount of on-board propellant, which is partly there to be able to perform the needed avoidance manoeuvres using the propulsion system. With this, requirement **BBIS-Sys-G04-1** is verified.

Each BBIS spacecraft is expected to be hit by 9.5 small debris objects over the course of its lifetime. It is assumed that a collision with a specific energy more than $40 \cdot 10^3\ J/kg$ is catastrophic.[6] The specific energy is obtained by dividing the kinetic energy of the impacting debris object by the spacecraft's mass [43]. See Equation 13.1.

$$E_{sp} = \frac{\frac{1}{2} m_{debris} V^2}{m_{spacecraft}} \tag{13.1}$$

[3]URL http://www.aerospace.org/cords/all-about-debris-and-reentry/debris-impacts-in-orbit/ [cited 19 June 2018].
[4]URL http://www.spacesafetymagazine.com/space-debris/impact-new-satellite-launch-trends-orbital-debris/ [cited 23 June 2018].
[5]URL http://www.au.af.mil/au/awc/awcgate/usspc-fs/space.htm [cited 24 June 2018].
[6]URL https://repository.tudelft.nl/islandora/object/uuid%3A9d609886-372d-466d-9cca-7e7b26b26a7f [cited 26 June 2018].

Here, E_{sp} is the specific energy of the collision in J/kg, m_{debris} is the mass of the impacting debris object in kg, V is the velocity of the impact in m/s and $m_{spacecraft}$ is the spacecraft's dry mass in kg. Assuming space debris is made of aluminium, which is a material used by many spacecraft, and that space debris has a spherical shape, the mass of space debris can be related to its radius using Equation 13.2 and Equation 13.3.

$$V_{sphere} = \frac{4}{3}\pi r^3 \tag{13.2}$$

Here, V_{sphere} is the volume of a sphere in m^3 and r is the sphere's radius in m.

$$\rho = \frac{m}{V_{sphere}} \tag{13.3}$$

Where ρ is the density of the material ($2720\ kg/m^3$ for aluminium),[7] v_{sphere} is the volume of the (spherical) object and m is the mass of the object.

Combining Equation 13.1, 13.2 and 13.3 will give the minimum debris radius needed to cause catastrophic damage to the BBIS spacecraft. The combined equation is shown in Equation 13.4.

$$r = \sqrt[3]{\frac{3}{2} \cdot E_{sp} \cdot \frac{m_{spacecraft}}{V^2 \cdot \rho_{Al} \cdot \pi}} \tag{13.4}$$

With $E_{sp} = 40 \cdot 10^3\ J/g$ (the maximum value before the collision becomes catastrophic), $m_{spacecraft} = 56.49\ kg$ and $\rho_{Al} = 2720\ kg/m^3$, the maximum radius r for a non-catastrophic collision can be determined. The collision is assumed to take place at a velocity of $1500\ m/s$, which is the expected maximum velocity of a collision in GEO, which occurs when an object in a GTO orbit collides in GEO.[8] The maximum radius of a non-fatal debris object is then found to be $5.47\ cm$, so the diameter can be $10.9\ cm$ without experiencing catastrophic damage. Over 99.9% of the space debris is smaller than this,[9] so the BBIS spacecraft are able to withstand an impact of almost all debris. Requirement **BBIS-Sys-T05-5** is thus met.

It should be noted that the calculations performed to predict the collision probability do not take the spacecraft's size and exact orbit (inclination, ascending node) into account. The used sources do not allow for such a detailed analysis. The results can therefore only be used as an indication.

[7] URL https://www.engineeringtoolbox.com/metal-alloys-densities-d_50.html [cited 26 June 2018].
[8] URL https://repository.tudelft.nl/islandora/object/uuid%3A9d609886-372d-466d-9cca-7e7b26b26a7f [cited 26 June 2018].
[9] URL http://www.aerospace.org/cords/all-about-debris-and-reentry/debris-impacts-in-orbit/ [cited 26 June 2018].

14. Risk Assessment

In both [3] and [8], risk assessments are performed. The risk assessment in the former is general and preliminary, while the latter is focused on the trade-off between the three main design concepts. The current stage of the project requires a risk assessment focused on the technical difficulties that could be encountered during the operation of the BBIS. In Section 14.1, the risks and their mitigating actions are identified. In Section 14.2, a summary of the risks is given in the form of two risk maps.

14.1. Risk Identification

Several types of risks have to be considered. To create a good overview, the risks all have their own identifier. The identifier includes a two-letter code and a number. Possible identifiers, and the types of risks that are part of that identifier, are listed below.

- **LDxx** Launch, deployment and disposal risks.
- **CPxx** Component or subsystem failure of the payload risks.
- **CExx** Component or subsystem failure of the EPS risks.
- **CAxx** Component or subsystem failure of the A&ODCS risks.
- **CCxx** Component or subsystem failure of the communication, C&DH and thermal subsystem risks.
- **EXxx** External factors in space risks.
- **UU00** Unknown risks.

A list of all identified risks and an explanation of these risks is given below. This explanation includes a short description of the mitigating actions that were taken to reduce these risks. An estimation of the likelihood (L) and consequence (C) of these risks after mitigation is also given. The likelihood and consequence are rated on a scale from very low to very high.

- **LD01** Catastrophic launcher failure. The launcher could explode or crash during launch, destroying the on-board BBIS spacecraft. The likelihood was reduced by choosing a launcher with a proven reliability. The consequence of this risk cannot be reduced.
  - L: very low. Of the past 98 launches of the Ariane V, 2 experienced a catastrophic launcher failure.[1] This gives the Ariane V a reliability of 98%, which means the likelihood of failure is very low.
  - C: very high. Crashing of the launcher will cause catastrophic damage to all on-board spacecraft.

- **LD02** Partial launcher failure. A partial launcher failure means that the launcher does not perform its intended task, but it also does not destroy the transported spacecraft. The result is that the spacecraft are delivered into an orbit that is significantly different from the desired orbit, which could severely limit the operation capability of BBIS. The likelihood of this event is reduced by choosing a launcher with a proven reliability. The consequence of this event is slightly reduced by having a propellant margin, so that propellant can be used to bring the spacecraft to its desired orbit.
  - L: very low. Of the past 98 launches of the Ariane V, 3 are considered to be partial failures.[1]. Statistically speaking, the probability of a partial launcher failure is thus 3%, rating the probability of this event very low.
  - C: high. If the spacecraft are delivered at an altitude lower than the desired altitude, there is still a possibility to increase the altitude by using the spacecraft's on-board thrusters. This, however, will significantly decrease the BBIS's lifetime, because the spacecraft will have less propellant left for the planned operations.

[1]URL http://www.esa.int/For_Media/Press_Releases/Ariane-502_-_results_of_detailed_data_analysis [cited 22 June 2018].

- **LD03** Launcher inaccuracy. The launcher could deliver the BBIS spacecraft into an orbit that is slightly different than the desired orbit, even though the launching sequence went well. The likelihood of this event cannot be mitigated, because the BBIS design does not influence the launcher accuracy. Propellant is reserved on board of the BBIS spacecraft to make sure that the spacecraft are able to reach the desired orbit.
  - L: very high. The Ariane V user manual mentions a standard deviation in the altitude at which the spacecraft are delivered[2]. Based on this, it can be assumed that the spacecraft will always be delivered at a slightly different location than desired.
  - C: very low. The BBIS spacecraft carry an additional amount of propellant specifically for the purpose of countering the inaccuracy of the launcher. This propellant is accounted for separate from the standard propellant margin.
- **LD04** Deployment failure of the solar sail. The solar sail has to be folded to fit inside the launcher. This means that the sail also has to be deployed at the beginning of BBIS's operational life. A failure could occur when deploying the solar sail. The likelihood of this event is reduced by choosing a deployment mechanism that has been proven. The consequence is reduced by the redundancy in the amount of spacecraft.
  - L: low. The deployment mechanism of the solar sail is based on a design that has been successfully tested, as explained in Section 6.5.
  - C: high. Partial failure of the deployment mechanism could reduce the visibility and reduce the power generation of the BBIS. Complete failure makes the considered spacecraft invisible from Earth. The solar sail also plays an important role during both planned and unplanned disposal, a function that is compromised if the solar sail fails to deploy (see Section 12.3).
- **LD05** Re-orbiting failure. After its lifetime, the spacecraft should be re-orbited to the desired graveyard orbit. A failure while re-orbiting could cause the spacecraft to not reach its graveyard orbit. The likelihood of this event is reduced by having multiple possible methods available for re-orbiting. The consequence is not mitigated.
  - L: very low. Both the spacecraft's propellant and the solar sail can be used for re-orbiting. The probability of both methods not being sufficiently available is very low.
  - C: medium. The BBIS mission is not be harmed if the BBIS spacecraft are not properly disposed of. However, not disposing of the spacecraft causes additional space debris in GEO, which means future missions 0.have to avoid the non-functional BBIS spacecraft.
- **CP01** Solar sail failure. The solar sail could reflect less light than desired, reducing the expected visibility of the BBIS. The likelihood of this event is reduced because much research was performed on the production and usage of solar sails. However, the BBIS will use solar sails for a purpose they were not designed for. The consequence is reduced by first testing a single spacecraft in space before sending all other.
  - L: high. The likelihood of this event is high, because solar sails have not been used before for the purpose of reflecting sunlight to the Earth.
  - C: low. Losing 1 of 900 satellites will not have a large influence on the visibility of the BBIS.
- **CE01** Solar cell failure. The solar cells could generate less power than expected. The likelihood of this event is reduced by choosing solar cells that successfully been used for other missions. The consequence is reduced by adding a safety factor to the expected amount of generated power.
  - L: low. The used solar cells have a proven reliability.
  - C: low. Generating less power than expected is only a problem if the difference between expectation and reality is very large. This difference will be small due to the safety factor.
- **CE02** Battery failure. The battery could store less power than desired, or have a lower discharge efficiency than desired. The likelihood of this event is reduced by choosing a battery that has been used on previous missions. The consequence is reduced by adding a safety factor to the known battery specifications and by having a backup battery on board.
  - L: low. The used battery was proven to work on other missions.

[2] URL http://www.arianespace.com/wp-content/uploads/2011/07/Ariane5_Users-Manual_October2016.pdf [cited 22 June 2018].

- C: medium. Reduced battery capacity could cause a power shortage in the spacecraft, which will affect all subsystems that require power. There is a backup battery on board that could take over if necessary.

- **CA01** Attitude and orbit determination sensor failure. Three attitude determination sensors are present per spacecraft: a Sun sensor, a star tracker and a GPS receiver. These work together for redundancy for attitude determination, which reduces the consequence of one of the three failing. For each of the three, a reliable component is chosen to reduce the likelihood of failure, as explained in Section 7.3.
  - L: low. Reliable components are chosen.
  - C: low. Due to the redundancy of the sensors, the consequence of a single sensor failing is low.
- **CA02** Inertial measurement unit failure. This component adds a higher accuracy and frequency to the attitude and orbit data. The risk of this component failing is not mitigated, as the spacecraft is able to function without it, the only difference being the accuracy and frequency of the measurements.
  - L: low. A reliable component is chosen.
  - C: low. Without this component, the attitude's measurement data is less accurate, but the spacecraft can still function.
- **CA03** Reaction wheel failure. This component is able to control the spacecraft's attitude. The likelihood of this component failing is reduced by choosing a component with a proven reliability. The consequence is reduced by designing the thrusters to be able to perform attitude control instead of the reaction wheels.
  - L: low. Reliable components are used.
  - C: very low. The spacecraft's propulsion system has enough propellant and thrusters on board to perform the task of the reaction wheels.
- **CA04** Thruster failure. Thruster malfunctioning could cause an undesired change in the spacecraft's attitude or orbit location. The likelihood of this event is reduced by choosing a thruster with a proven reliability. The consequence is reduced by having a redundancy in the thrusters' functioning.
  - L: very low. The manufacturer of the chosen thruster has provided thrusters since 1974. As of 2017, the thruster failed in only one of 176 missions.[3] Thus, according to statistics, this is an extremely reliable thruster.
  - C: low. As explained in Section 7.6.3, three of twelve thrusters could fail before the spacecraft loses part of its functionality. Also, the reaction wheels are able to perform attitude control instead of the thrusters.
- **CA05** Propellant tank failure. The tank could experience leakage, or more fuel could be left inside the tank as a residual than is expected. As a result, the spacecraft will have less fuel available to function. A safety factor is already added to account for residual fuel in the tank. The likelihood of this event is reduced by selecting reliable propellant tanks. A slight reduction in the consequence of this event is accounted for by having additional propellant on board.
  - L: low. The used propellant tanks have been successfully used on other missions.[4]
  - C: high. Depending on the severity of the failure, this event could cause the spacecraft to lose a minor to a significant portion of its propellant. The lifetime of the spacecraft is reduced accordingly with the amount of propellant lost.
- **CA06** Propellant shortage. The on-board propellant of the spacecraft could prove to be insufficient for the mission. The likelihood of this event is reduced by applying a safety margin for the amount of on-board propellant. The consequence is reduced by having a disposal manoeuvre ready that does not require any propellant.

[3] URL http://www.space-propulsion.com/spacecraft-propulsion/bipropellant-thrusters/10-bipropellant-thrusters.html [cited 25 June 2018].
[4] URL http://www.ardeinc.com/propellant.html [cited 25 June 2018].

  - L: low. The spacecraft carry more propellant on board than is expected to be necessary.
  - C: medium. When there is no more propellant left, the lifetime of the considered spacecraft should be ended, to make sure it does not collide with any of the other BBIS spacecraft. One of 900 spacecraft is then lost.

- **CC01** Antenna failure. There are three antennas on board of each spacecraft in order to be able to communicate with the ground station and with the other BBIS spacecraft. The likelihood of antenna failure is reduced by choosing a qualified component.[5] The consequence is reduced by having three antennas on board.
  - L: low. The used antennas are highly qualified.
  - C: medium. If one antenna fails, the other two can still be used to communicate. However, the communication will be less efficient, because, for example, the considered spacecraft might have to communicate via another spacecraft to send a message to the ground station.

- **CC02** On-board computer failure. The on-board computer gives commands to all subsystems. Without a working computer, the entire spacecraft will fail. To mitigate the risk of this event, three computers are taken on board, each of which can perform the task of on-board computer individually.[6]
  - L: low. A reliable manufacturer will deliver the on-board computers.
  - C: low. Three on-board omputers will be present, of which only one has to function.

- **CC03** Paint degradation. The spacecraft's thermal paint layer can be degraded by, for example, sunlight. A degraded layer could cause the spacecraft to absorb more sunlight, and thus the temperature of the spacecraft will increase. The likelihood of this event is reduced by adding a protective layer over the paint. The consequence of this event is not mitigated, as the likelihood is very low due to the protective layer.
  - L: very low. The protective layer will make sure the paint does not degrade.
  - C: high. The spacecraft's components have a temperature range within which they can function. An increase in temperature could render some subsystems useless. Also, radiation can harm the electrical components in the spacecraft.

- **EX01** Collision with large space debris object. A collision with large debris could cause catastrophic damage to a spacecraft. There are no mitigating actions taken to reduce the consequence of this risk. Instead, the BBIS will make use of the Space Surveillance Network, as explained in Section 13.4, to predict the orbit of incoming debris.
  - L: low. In Section 13.4, it was determined that the probability of an arbitrary BBIS spacecraft being hit by a large debris object over the course of its lifetime is 0.76. This is quite a large probability, but if the BBIS takes measures to avoid large, detectable debris objects, then the likelihood is significantly reduced.
  - C: very high. A collision with a large debris object is expected to be fatal for a single spacecraft. Also, the additional debris caused by breakup of the hit spacecraft could collide with other BBIS spacecraft, which further increases the consequence of this risk.

- **EX02** Collision with a small space debris object. Depending on the size of the debris, small debris could cause negligible to significant damage to the spacecraft. It was decided to not mitigate this risk, because it is not possible to detect small debris in GEO[7] and adding a debris shield causes a significant mass increase of the spacecraft.[8]
  - L: Very high. The BBIS spacecraft are expected to be hit by multiple small debris objects over the course of their lifetime, as explained in Section 13.4.
  - C: Low. As explained in Section 13.4, small debris objects in GEO are expected to not cause much damage to the BBIS spacecraft.

[5]URL https://www.isispace.nl/product/antennas/ [cited 25 June 2018].
[6]URL https://hyperiontechnologies.nl/wp-content/uploads/2016/08/HT-CP400.85-V1.01_Flyer.pdf [cited 25 June 2018].
[7]URL http://www.esa.int/Our_Activities/Operations/Space_Debris/Scanning_and_observing2 [cited 25 June 2018].
[8]URL https://ston.jsc.nasa.gov/collections/trs/_techrep/TP-2003-210788.pdf [cited 25 June 2018].

- **EX03** Collision with other functioning spacecraft. The spacecraft of the BBIS could collide with each other. The likelihood of this event is reduced by properly designing the spacecraft's subsystems and by having an emergency formation leaving method available, as explained in Section 12.3.3. The consequence of this event cannot be accounted for, as two BBIS spacecraft colliding will be catastrophic, even if the spacecraft would have a protective layer.
  - L: low. The spacecraft have sufficient spacing in between each other (Section 5.4) and are able to leave the formation in case of failure of one of the subsystems (Section 12.3.3).
  - C: very high. BBIS spacecraft colliding with each other will cause space debris within the BBIS formation, which could hurt other spacecraft as well.

- **EX04** Space radiation. Space radiation could negatively affect the spacecraft's functioning. The likelihood of encountering space radiation in GEO cannot be reduced. The consequence of space radiation is reduced by adding an aluminium shielding to the spacecraft's sensitive components.
  - L: Very high. The likelihood of encountering space radiation is 100%.
  - C: low. In Section 10.2.1, it is explained how the spacecraft is protected against radiation.

- **UU00** Unknown risks. Not all risks can be identified beforehand, but the likelihood of this event has been reduced by performing an extensive risk analysis of the known factors. The consequence of this event can be reduced by mitigating all known risks as much as possible.
  - L: very low. Proper risk assessment has already been performed. Possible failure modes of the subsystems have been analysed, as well as dangers from the environment and the launch and deployment phase of the mission.
  - C: high. The nature of the threat is unknown and could damage the spacecraft in any amount possible.

14.2. Risk Maps

The above mentioned risks can be summarised in a risk map. Two maps are provided: one before the mitigating actions are taken (Figure 14.1), and one after (Figure 14.2). It can be seen in Figure 14.2 that there are no critical risks after risk mitigation.

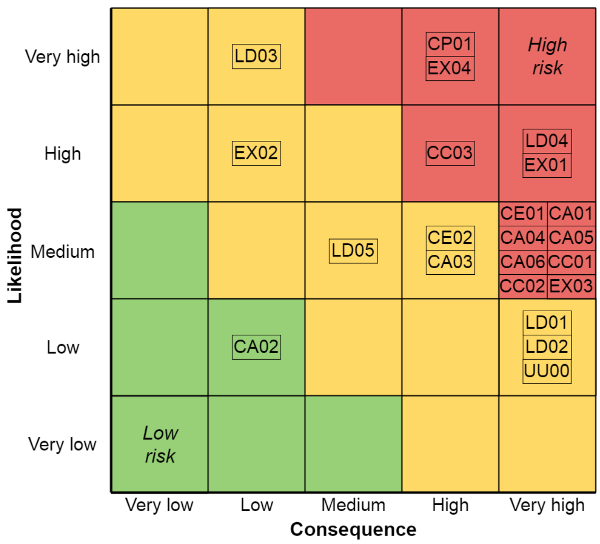


Figure 14.1: Risk Map Before Risk Mitigation.

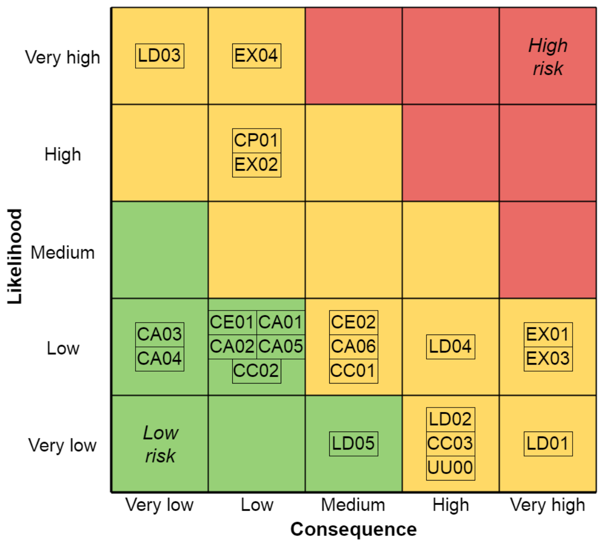


Figure 14.2: Risk Map After Risk Mitigation.

15. Sustainability

This chapter deals with the sustainability analysis of the project. Firstly, a summary of the strategy is presented in Section 15.1, followed by the life cycle analysis in Section 15.2. To conclude this chapter, the ethics of this project are discussed in Section 15.3.

15.1. Sustainable Development Strategy

The ultimate goal of any engineering project is to provide certain benefits to society. However, the methods used to obtain these benefits sometimes result in a compromise for the life quality of future generations and their available resources. Moving forward towards sustainable development means finding the equilibrium between optimising project results, while minimising the negative effect they have in society, economy and environment. These policies should be approved and followed by the totality of the team in order to fulfil requirement **BBIS-Tud-07**.

As it was previously mentioned [3], the sustainability engineer (SUE) develops a strategy that implements all these policies into the different phases of the process. During the trade-off, sustainability risks were considered as one of the main criteria. Their effects were minimised by choosing the most suitable concept. Some of these risks are, for example, emissions, intrusive effects or the generation of space debris. The mitigation of some of these aspects might result in a short term increase in cost, but is required to ensure long term benefits. These benefits include, for example, decrease in environmental and social impact while increasing profitability.

15.2. Life-Cycle Analysis

Even though the concept with the most optimal characteristics was chosen during the trade-off, sustainability should be implemented in all the upcoming project phases. Therefore, the SUE developed a strategy to follow, even after the design concept was determined. This strategy is the named life cycle analysis and during this analysis every step to be taken is analysed and given requirements. Only certain aspects of the spacecraft design have been treated yet and therefore the life-cycle analysis is adapted to establish the requirements that can be verified or analysed at this stage. See Figure 15.1 for a detailed version of the current steps of the sustainable development strategy.

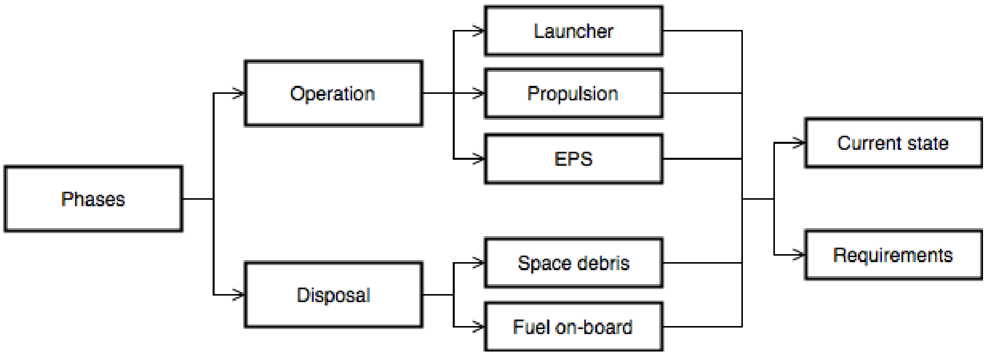


Figure 15.1: Life-Cycle Analysis.

The subsystems that are not indicated in the figure are also analysed but mainly follow the policy of not overdesigning. This policy results in the use of less material and resources, contributing to the waste reduction.

15.2.1. Ariane V

Amongst its many technical advantages, Ariane V is chosen due to its positive contribution to sustainable development. Each launcher is designed by Arianespace in such a way that there is a limited environmental impact. The number of launchers used to get BBIS in its orbit is minimised in order to reduce emissions and waste generation.

The spaceport in French Guiana, from which Ariane V is launched, follows a policy based on the continuous protection of the ecosystem around the spacecenter. Local air and water are checked after each launch and surrounding fauna and flora are analysed twice a year, in order to ensure that no chemicals or other damaging substances have been introduced [1]. Moreover, Arianespace contributes to the social growth of the communities around their facilities, providing jobs, collaborating with local schools, hiring young people to ensure knowledge is transmitted to new generations, and ensuring gender equality in the workplace.

In addition, eco-friendly practices are implemented into the launcher design and development process, reducing electricity and fuel consumption.

15.2.2. Propulsion

As is explained in Section 7.5.2, green propellant is not used, as this will lead to an undesired increase in the amount of launches needed, which is more harmful to the environment than the use of non-green propellants. Therefore requirement **BBIS-Sust-Op-01** is not met, which means **BBIS-Tud-07** is also not met.

15.2.3. EPS

The available power sources for space missions are limited. Currently the only available technology includes solar arrays or RTGs. The Radioisotope Thermoelectric Generator (RTG) produces electricity from the decay of a suitable radioactive material, which results in harmful material being stored on board and staying in the spacecraft at the end of its operational life. This technology provides a high efficiency, however, the amount of power required for this design of BBIS was significantly low compared to the other choices analysed in the trade-off, and therefore it is possible to use less efficient but more sustainable energy sources.

The BBIS uses solar arrays that are attached to the solar sail, staying in sunlight during most of its operational time. The solar cells produce electricity from the sun radiation, generating zero waste in the process. The choice for CIGS solar cells was elaborated in Section 9.4. These less efficient cells were chosen over the more efficient CdTe cells, because the latter contain cadmium, a toxic material, which is not in line with the sustainability strategy of BBIS.

15.2.4. Disposal

The end-of-life disposal represents one of the biggest sustainability threats for the space industry. Currently, the amount of non-operational spacecraft orbiting the earth outnumbers the amount of operational satellites. This is something to consider due to the high popularity of certain orbits that could be unavailable in the upcoming years due to debris accumulation.

In order to minimise the impact of the debris, graveyard orbits are established between the commonly used orbits. The 900 spacecraft forming BBIS will be sent to a graveyard orbit at its end-of-life, reducing its negative effect on future space missions. Moreover, to achieve this disposal orbit, each spacecraft will perform a solar sail followed by the burn-out of the fuel that still remains on board. This technique ensures not only that the minimum amount of fuel is used, but also that there will be no chemicals left on board at the end-of-life, which would be harmful in case of leakage .

15.2.5. Requirements

Certain sustainability requirements cannot be verified yet, due to the fact that certain phases of the project have not yet been reached, however it is necessary to consider them in upcoming steps. For requirements that hold for future phases see below.

- **BBIS-Sust-Mat-01** The extraction processes shall not cause irreversible damage in the surrounding areas.
- **BBIS-Sust-Mat-02** The machinery used in the extraction process shall use clean energy.

[1] http://www.arianespace.com/wp-content/uploads/2016/03/CSR_report_2014_2015_GB.pdf [cited 23 June 2018].

- **BBIS-Sust-Mat-03** The extraction sites shall not be inside endangered areas for the fauna and flora according to the *Convention on International Trade in Endangered Species of Wild Fauna and Flora (CITES)* [44].
- **BBIS-Sust-Man-01** Manufacturing processes shall have a reduction of 30% of their gas emissions before 2020[2].
- **BBIS-Sust-Man-02** The machinery used in the manufacturing process shall use clean energy.
- **BBIS-Sust-Trans-01** Electric trucks shall be used as the form of transportation of billboard components.
- **BBIS-Sust-Op-02** The ground station used for operation shall use clean energy.

The following requirements refer to the operation and disposal of the spacecraft. The final performance of the spacecraft cannot be predicted but, as it was explained in the previous sections, subsystems are designed to optimise the following requirements.

- **BBIS-Sust-Op-01** The spacecraft shall not emit harmful substances to the environment during its life cycle.
- **BBIS-Sust-EoL-01** The spacecraft shall comply with the *Space Debris Mitigation Guidelines of the Committee on the Peaceful Uses of Outer Space*.

In addition, requirements about social sustainability are added and discussed in Section 15.3.

- **BBIS-Pub-01** The spacecraft shall not disturb the day to day life of the observers.
- **BBIS-Pub-02** The spacecraft shall not destroy life on Earth.
- **BBIS-Pub-03** Advertisements shall not be provoking.
- **BBIS-Gov-03** The billboard shall not bring people in danger by distracting them.

15.3. Ethics

The implementation of sustainable techniques in engineering projects is highly related to the ethics in the industry. The ultimate goal of engineering is progress. However in the upcoming years, it will be no longer possible to access certain resources, such as the radioactive materials used in RTGs. Sustainability is therefore also growing, forced by these resource constraints. Numerous materials, overused in the past, are not available for current projects. The search for substitutes impulses the progress of a sustainable mentality as well.

From the ethical point of view, social sustainability is a crucial aspect of the mission as well. This is why the project chooses its partnerships to be with companies that impulse equality and progress of society. The best example being the previously mentioned Arianespace.

The ultimate goal of the advertisement industry is to introduce an idea in the public's mind. However, BBIS is designed in such a way that this effect is balanced with the social sustainability of the project. Therefore, in order to comply with requirements **BBIS-Pub-01**, **BBIS-Pub-02**, **BBIS-Pub-03** and **BBIS-Gov-03**, the billboard has a the ability to be switched on and off. In addition, the displayed advertisements are carefully reviewed before being displayed in order to avoid provocative or offensive content.

[2]URL https://eur-lex.europa.eu/legal-content/EN/TXT/?uri=LEGISSUM%3Al28012 [cited 4 May 2018].

16. Market and Cost Analysis

In this chapter, the market and cost analysis for BBIS is presented. For this mission, it is crucial to asses these aspects to determine the financial feasibility of the project. The current and future markets play an important role in this analysis. A low cost is one of the driving requirements. Throughout the mission design, costs are minimised by working with COTS components, as well as the goals to maintain a minimal spacecraft weight and size. A cost budget will be presented in this chapter. However, due to the fact that no mission exists that is similar to BBIS, the establishment of a detailed and accurate cost estimation is difficult.

First, the initial requirements set up in [3] are listed in Section 16.1. These requirements are related to the various stakeholders involved in the BBIS mission. Then, the mission's market is elaborated on in Section 16.2. Furthermore, the cost estimation method and budget breakdown is presented in Section 16.3. Finally, Section 16.4 discusses the return on investment.

16.1. Requirements

The requirements determined in [3] are listed below. For the market analysis, a distinguishment between various related systems and parties is made. The system requirements, relevant for the market analysis are listed below.

- **BBIS-Tud-01** The project shall be cost effective .
- **BBIS-Sys-T01-1** The project shall have a return on investment of at least 0%.
- **BBIS-Sys-T04-1** The billboard shall be visible from the USA for 905 hours per year under ideal weather conditions.

Here, requirement **BBIS-Sys-T01-1.1** is discarded because this project budget was based on the previous market and profitability analysis. The remaining requirements **BBIS-Sys-T01-1** and **BBIS-Sys-T04-1** will be reviewed in Section 16.2, 16.3 and 16.4.

16.2. Market Analysis

New advertisement methods are in high demand as companies are continuously competing and seeking new ways to outperform each other. Currently, digital media dominates the advertisement market.[1] BBIS adds an entire new dimension to advertising which has never been explored before. A big debate is currently taking place regarding the privacy issues related to digital advertising [45]. BBIS is not in conflict with this problem because the advertisement does not use any personal data. Instead it is a traditional passive type of advertising with an additional benefit of being unique in its field and able to reach a large target group at once.

In this section, first the strengths, weaknesses, opportunities and threats analysis is presented in Section 16.2.1. Then, Section 16.2.2 discusses BBIS's potential future markets. Finally, mission funding is elaborated on in Section 16.2.3.

16.2.1. SWOT Analysis

This subsection analyses the strengths, weaknesses, opportunities and threats of the market.

- **Strengths** The advertising market has a large growth potential and investors are continuously seeking for revolutionary methods to advertise their products. BBIS provides a innovative way to reach a large target group. Currently, there are no competitors on the market, making BBIS unique.
- **Weaknesses** Despite the innovating benefits of the advertising market, the space mission side of BBIS is still very undeveloped in this field. For example, no reference missions exist consisting of spacecraft flying in such a large formation. Software handling smooth communications and

[1] URL https://www.emarketer.com/topics/topic/ad-spending [cited 20 June 2018].

spacecraft collision protection operations will have to be specially developed and extensively tested. Furthermore, the use of solar sails as reflective surfaces to direct sunlight to Earth has never been done before. Thus, the uncertainty factor for this mission is very high, resulting in high costs.

- **Opportunities** Many opportunities arise from an investment in BBIS. Firstly, companies seeking for a new advertisement method will be able to use BBIS as a medium to broadcast. Moreover, scientific research, investigating the prospects of formation flying are beneficial for concepts related to for example space debris scanning and collecting and solar farming. Finally, BBIS could be used as an informative medium such as an alarm system for the government.
- **Threats** The main threat to the market is the high risk, cost and complexity of the mission. One system failure in one spacecraft could be catastrophic for the entire fleet. Additionally, a restricting legislation change of loss of costumer interest form large threats for the mission.

16.2.2. Future Market

The focus of this section is to give a prediction of the markets from which BBIS can profit in the future. The prospects for this project are positive due to its variety of application possibilities. For this project, the mission statement 'Explore new advertisement options by designing a billboard in space that has a comparable visibility to a full moon' suggests that the billboard is only used for commercial services. However, additional potential markets exist. These future markets, listed in the SWOT analysis in Section 16.2.1 as market opportunities, concern the following parties: advertising companies, scientific researchers and the government.

Companies

Because the spacecraft's pixels can be actively turned on and off, varying advertisements can be displayed. This enables multiple companies to be contracted, meaning that the market income is not constrained by the number of potential investors. Furthermore, in total, the billboard display configuration is able to change five times. Thus, the pixels will be obtain a new distribution, enabling new billboard lay-outs to be made. This also broadens the potential revenue.

BBIS is designed for an optimal view from the USA. However, for companies in that aim to target other countries, a similar mission would be able to take place with a different orbit and orientation. Furthermore, the if the companies are willing to invest, a more elaborate spacecraft payload can be developed such that the displayed billboard is refined. For example, experiments could be done on different pixel colours and intensities to extend the number of different advertisements that can be shown as well as making them more detailed. Furthermore, complete new spacecraft configurations and possible orbits can be reviewed to provide a more diverse range of display options.

Scientific Researchers

Researchers have been exploring space since the mid 1900s. Yearly, new milestones are reached regarding deep space discoveries and valuable mission prospects. A newly developing concept within the space industry is formation flying. BBIS is a unique mission, such a complex mission with 900 spacecraft flying in formation has never been previously executed. Investing in this project would be very beneficial for other potential space projects. The dimensions that are able to fit into a launcher are limited, restricting the size of spacecraft that are able to be launched into space. Currently, the largest spacecraft sent to space is the ISS (International Space Station), which is $357\ ft$ end-to-end [2] Launching multiple spacecraft that will work in coherence with each other once in orbit, will provide a solution to this sizing constraint.Two examples are elaborated on below.

- **Space Debris** The Department of Defence keeps track of objects orbitting Earth. Together with NASA, the Department of Defence is currently responsibility for a catalogue that tracks space debris.[3] A large formation flying fleet would enable debris scanning and collecting missions to cover a large area of space efficiently.
- **Solar Farming** Space-based solar farming is a concept that aims to collect solar energy in space and distribute it to Earth. This innovating technology has the following advantages: the collection

[2] URL https://www.nasa.gov/feature/facts-and-figures [cited 1 May 2018].
[3] URL https://www.nasa.gov/mission_pages/station/news/orbital_debris.html [cited 7 May 2018].

rate of solar energy in space is higher, the collection period is longer and the solar collector can orbit in a location which is continuously exposed to sun radiation. Combining this concept with a large fleet of spacecraft would provide beneficial prospects for the solar farming technology.

Government
In addition to BBIS's primary advertising function, the billboard could also be used for informative applications such as an alarm system and informative broadcasting system. For example, in the case of an extremely heavy storm a wide range of public can be informed instantly and warned about the upcoming event. A large target group is reached without the public having to invest in anything such as a television.

16.2.3. Mission Funding
The market revenue of this project has been roughly estimated in [3]. Few changes have been made after that. These modifications are listed below.

- The BBIS is flying at GEO, therefore the operation time is much longer and more viewing is possible.
- The total lifetime of the BBIS reduces to 20 years.

In 2020, the projected population in the USA is $334.5\ mln$[4] and the projected population in contiguous United States (subtracted projected population of $0.75\ mln$ in Alaska[5] and $1.39\ mln$ in Hawaii[6]) is equal to $332.4\ mln$.

As it is assumed in [3], data shows that in 2015, a total of $569.62\ USD$ was spent on advertising per person.[7] and the source projected that this total would increase to $725.75\ USD$ per person in 2020, which yields to a total value of $241.22\ bln\ USD$.

16.3. Financial Analysis
In this section, an estimation of the cost breakdown is given. First, the launching costs are addressed in Section 16.3.1. Then, the spacecraft production costs are discussed in Section 16.3.2. Then, the operational costs are elaborated on in Section 16.3.3. Finally, a total mission cost is given in Section 16.3.4

The calculations in this section include time value of money, because of the large timespan where money is spent and earned. It is chosen to correct all costs and revenues for inflation and to use the United States Dollars (USD) in 2018 as a base. This section uses the annual inflation rates of the USD.[8] As an example, this section abbreviates $1\ bln\ USD$ in the financial year 2018 as $1.6\ bln\ FY2018\ USD$.

16.3.1. Launching
The cost of launching depends on the number of launches and the price of one launch. The Ariane V is chosen for the launching. The total cost per launch using the Ariane V is set to equal $178\ mln\ FY2017\ USD$ [46]. In total, 9 launches are required to get the swarm of 900 spacecraft into the desired orbit. This results in a total launch cost, corrected for inflation, of $1.635\ bln\ FY2018\ USD$.

16.3.2. Spacecraft
Due to the absence of reference missions, it is difficult to accurately estimate the BBIS mission costs. A rough estimation is made using the following method. First, the readily retrieved prices for the COTS components used throughout most of the spacecraft are summed to retrieve the total system cost. For components or processes where no direct price is available, statistical data from [47] is used or a comparable device price is used. This source is further elaborated on during the verification of this cost analysis. Secondly, a breakdown cost budget is established using typical system cost averages stated in [47], which is also used for verification. The assumptions for all components are listed below.

[4]URL https://www.statista.com/statistics/183481/united-states-population-projection/ [cited 15 June 2018].
[5]URL http://live.laborstats.alaska.gov/pop/projections/pub/popproj.pdf [cited 15 June 2018].
[6]URL https://www.acl.gov/sites/default/files/programs/2016-11/Hawaii.pdf [cited 15 June 2018].
[7]URL https://www.statista.com/statistics/309732/ad-spend-per-person-usa/ [cited 1 May 2018].
[8]URL http://www.inflation.eu/inflation-rates/united-states/historic-inflation/cpi-inflation-united-states.aspx [cited 29 June 2018].

- For the solar array cost estimation, the integration assembly & testing (IA&T), the propulsion and the project management & systems engineering (PM/SE) costs, the mean values obtained from [47] are used.
- There was no COTS price available for the paint. However, cost estimations for a reference mission's thermal control system could be used. The costs will be less compared to the total thermal budget provided in [47], because the BBIS's thermal control system is passive. Therefore, it is estimated that the costs equal 10% of the thermal budget for the reference communication mission spacecraft.
- For the payload, the mission LightSail with a 3U CubeSat was used as a reference. The satellite had a solar sail spanning $5.6 \times 5.6\ m^2$ with costs equal to $5.5\ mln\ FY2018\ USD$.[9] This mission was started 10 years old and with modern technology advances it is assumed that the solar sail for the BBIS will be able to be produced for the same price. This budget is therefore used to estimate the payload costs.

The final cost estimation for all components can be found in Table 16.1. The total costs per system are printed in bold. By summing these values, the cost of one spacecraft is calculated to be $44.7\ mln\ FY2018\ USD$.

Subsystem	**Element**	**Cost [$FY2018\ USD$]**
A&ODCS	Digital Electronics	11 640
	Sensor[10]	92 000
	Reaction Wheels	20340
	Total	**123 980**
Communications	Antenna[11]	17 277
	Transceiver/Receiver[12]	29 371
	Total	**46 649**
EPS	Solar Arrays (Si)	506 267
	Batteries & Power Distributor	19 900
	Total	**526 167**
Structure	Inner Structure[13]	4 204
	Plates[14]	513
	Adhesive Bonding[15]	33
	Total	**4 750**
CD&H	Data Platform	22 500
	Data Bus	250 000
	Total	**272 500**
Propulsion		**952 402**
Thermal		**1 573 739**
Payload[16]		**5 500 000**
IA&T		**14 478 007**
PM/SE		**21 194 097**
Total		**44 672 291**

Table 16.1: Cost per Subsystem [48] of one spacecraft [47].

[9] URL https://www.scientificamerican.com/article/launch-of-first-private-solar-sail-powered-spacecraft-set-for-wednesday-video/ [cited 24 June 2018].

Verification of Cost

In order to verify the design, development, test and evaluation phase cost per system, the crosschecks given in [47] are used. The crosschecks provide a set of metrics to determine the system costs based on a range of known costs of previous systems. For the BBIS, communication mission spacecraft are chosen as reference. The mission characteristics of communication spacecraft are most similar to those of the BBIS spacecraft; both spacecraft operate in GEO, require a high accuracy, have no primary instruments for research and have a relatively long lifetime. The crosschecks specify the average costs for the attitude determination & control system (ADCS), communications, electrical power system, integration assembly & testing (IA&T), passive sensors, propulsion, project management & systems engineering (PM/SE), structures, thermal control, telemetry tracking and control, and for the total mission. For the BBIS, the cost estimation for the passive sensors and the telemetry tracking and control system are neglected. The passive sensors are already integrated in the cost estimation for example ADCS. Furthermore, BBIS includes a system for command and data handling (CD&H) instead of telemetry tracking and control. Finally, it must be noted that the payload costs are not listed, because communication mission spacecraft do not usually have payload on board. Table 16.2 provides an overview of the estimated costs per system based on the provided reference missions. Here, 'system total' refers to the entire system as stated in [47] and 'total' represents the sum of the system component costs.

Subsystem	Element	Variable	Unit	Cost/Variable $[FY2018\ USD]$	Total Cost $[FY2018\ USD]$
A&ODCS	Digital Electronics	11 640	lb	74 828	122 570
	Sensor	92 000	#	4 702 510	9 405 021
	Reaction Wheels	20 340	#	412 621	412 621
	Total				**9 940 213**
Communication	Antenna	0.22	lb	65 101	14 352
	Transmitter	0.18	lb	88 744	15 652
	Transceiver	0.18	lb	94 670	16 697
	Total				**46 701**
EPS	Solar Arrays (Si)	18.15	ft^2	27 892	506 267
	Power Conditioning & Distribution	2.46	lb	20 758	50 980
	Total				**308 638**
Propulsion	**Weight Based**	22.05	lb	43 200	**952 402**
Structure	**Average Based**	1	#	14 283	**14 283**
Thermal	**Average Based**	1	#	1 573 739	**1 573 739**
IA&T	**Average Based**	1	#	14 478 007	**14 478 007**
PM/SE	**Average Based**	1	#	21 194 097	**21 194 097**
Total	**Total**				**48 508 080**
	Average Based	1	#	45 726 842	**45 726 842**

Table 16.2: Cost for Verification [47].

[10] URL https://www.cubesatshop.com/product/digital-fine-sun-sensor/ [cited 29 June 2018].
[11] URL https://www.isispace.nl/product/antennas/ [cited 29 June 2018].
[12] URL https://www.isispace.nl/product/full-ground-station-kit-for-vhfuhfs-band/ [cited 29 June 2018].
[13] URL https://www.isispace.nl/product/3-unit-cubesat-structure/ [cited 29 June 2018].
[14] URL https://www.onlinemetals.com/merchant.cfm?pid=21943&step4&showunits=inches&id1242&top_cat=60 [cited 29 June 2018].
[15] URL http://www.fastener-world.com.tw/0_magazine/ebook/pdf_download/FW_166_E_306.pdf [cited 29 June 2018].
[16] URL https://www.scientificamerican.com/article/launch-of-first-private-solar-sail-powered-spacecraft-set-for-wednesday-video/ [cited 29 June 2018].

Total Spacecraft Cost Estimation

Reviewing the costs presented in Table 16.1 and Table 16.2, it was chosen to use the cost estimation based on the COTS components. Some system values show a large deviation from those based on the crosschecks. The large error is assumed to be caused by the difference in mission types and spacecraft configurations. The communication spacecraft referred to in [47] typically include specific components that weigh more, resulting in deviating figures. However, for activities related to, for example, IA&T and PM/SE, it is assumed that the costs will be similar. Also, as a validation, it can be seen that this spacecraft cost is comparable to the estimated total system cost based on general communication mission costs stated in Table 16.2. Thus, the final cost estimation per spacecraft, as determined above, equals $44.7\ mln\ FY2018\ USD$.

The total spacecraft configuration costs require an additional analysis. Due to the large fleet of spacecraft that needs to be manufactured, the cost per spacecraft is reduced. This is known as cost improvement. Typically, the learning experience curve for aerospace products is 85% [47]. The spacecraft cost is calculated using Equation 16.1, in which C_n represents the to be determined n^{th} spacecraft cost in $FY2018\ USD$, C_1 is the cost of the first spacecraft of production in $FY2018\ USD$, n is the cumulative volume of production and a is the elasticity of cost with regard to the output. For the aerospace industry, $a = -log_2(0.85) = 0.234$. Using the same parameters, the total mission cost for the fleet of spacecraft is calculated as a sum of the individual spacecraft costs.

$$C_n = C_1 * n^{-a} \tag{16.1}$$

When plotting the spacecraft cost calculated using Equation 16.1, it can be observed that the 900^{th} spacecraft cost is equal to 20.3% of the first spacecraft. The data used to calculate the cumulative costs are based on the costs for one spacecraft determined previously, being $C_1 = 44.7\ mln\ FY2018\ USD$. The reduction in cost related to the fleet size is depicted in Figure 16.1.

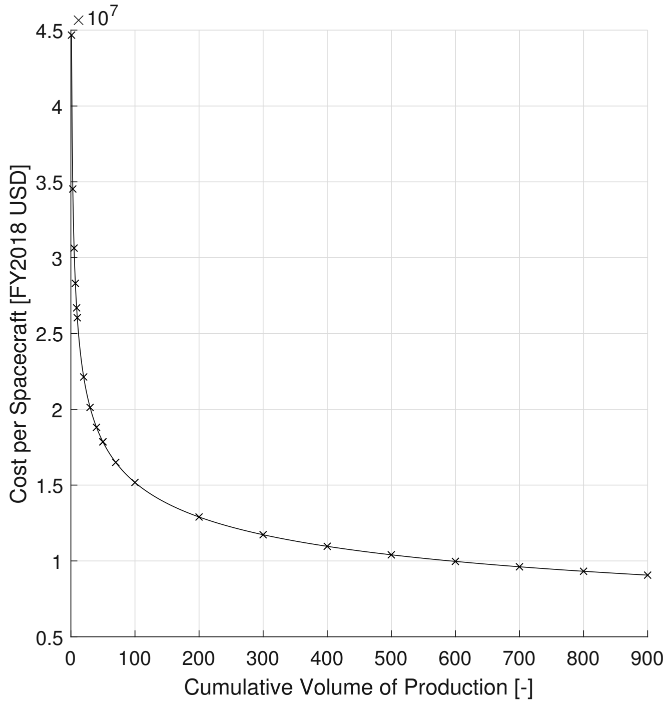


Figure 16.1: Cost per Spacecraft.

Thus, it can be concluded that the final total spacecraft fleet production costs, $C_{tot_{900}}$, will equal $10.6\ bln\ FY2018\ USD$.

16.3.3. Ground Station & Operation

Although the spacecraft are only communicating to the ground once per day, the ground station is needed to keep track of the spacecraft and to transmit new advertisement data to the spacecraft for updates. As was elaborated in Section 8.2.2, a ground station will be bought and used. The ground station cost are based on a typical percentage of management and operational costs equal to 4.2% of the total mission costs.[17] Thus, the operational costs for the BBIS are estimated to be $541\ mln\ FY2018\ USD$. These costs includes maintenance cost [49].

16.3.4. Mission Cost

Finally, the total mission costs can be determined. This cost consists of the launcher costs, the spacecraft costs and the operation costs. In order to account for unexpected costs, a 20% reserve factor for the design, development, test and evaluation phase and 10% for operations is included.[18] For the launch, no reserve factor is included because the number of launches with the Ariane V for the total spacecraft mass has already been overestimated. Thus, the final cost estimation for the BBIS is $15.0\ bln\ FY2018\ USD$ as can be seen in Table 16.3.

Phase	**Cost [$FY2018\ USD$]**	**Reserve factor**	**Total Cost [$FY2018\ USD$]**
Launch	1 635 802 200	1	1 635 802 200
Spacecraft	10 626 689 610	1.2	12 752 027 532
Operations	541 049 745	1.1	595 154 719
Total			**14 982 984 451**

Table 16.3: Mission Cost of BBIS.

16.4. Return on Investment

In Section 5.6, the method to obtain potential views is explained. The number of potential views does not take viewing hours into account. Therefore, this section will discuss the factors which influence this total ideal value and conclude with the total revenue.

In order to see the spacecraft, it is necessary that the sky is dark enough. Therefore it is assumed that the spacecraft are only visible after sunset and before sunrise. The period between sunset and sunrise takes 12 hours, averaged over one year. American people get approximately 8 hours of sleep everyday from 23:00 to 07:00 'o clock.[19] Since the average sunset and sunrise are around 18:00 to 6:00 respectively, this leaves 5 hours of viewing each day. However, there exists a time difference between the western and eastern part of the USA of 3 hours, due to the country being split up into four time zones. This means that when the critical time of 23:00 is reached on the most western time zone, there are still 3 more viewing hours in the most eastern time zone. As a result, every day has an average of 8 viewing hours. This is in line with the requirements stated in **BBIS-Sys-T04-1**.

It should be estimated how many people are actually seeing the advertisement when they are standing in the covered area. Since there is no available data on space advertising efficiency, estimations are based on comparable forms of advertising. A research carried out for aerial advertising yielded that approximately 87% of the people on a beach noticed the aerial advertisement.[20] Although aerial advertising is visible for longer than $14\ s$, it does not involve any illumination. These two effects are assumed to cancel each other out, so the billboard would reach the same efficiency as aerial advertising.

Additionally, the calculation should also take the number of people outside into account. It is assumed that only people outside or in a vehicle are able to see the advertisement. The average spends 7% of its life outside and another 6% inside a vehicle [50]. That means that on average 13% of the Americans are outside.

Moreover, the pointing accuracy is $0.0618°$, 10% of the beam angle, as specified in Section 7.6.1. This has as an effect that at the edges of the covered area, not all spacecraft might be visible as a result of a pointing error. With the current accuracy, this happens within 10% from the edges of the covered area. Therefore, a correction factor of 0.8 is applied to the potential views.

[17] URL http://spaceref.biz/nasa/nasa-fy2015-budget-requests-848m-for-commercial-crew.html [cited 29 July 2018].
[18] URL https://www.nasa.gov/pdf/140643main_ESAS_12.pdf [cited 25 June 2018].
[19] URL http://time.com/4319909/sleep-habits-country/ [cited 14 June 2018].
[20] URL https://study.com/academy/lesson/does-aerial-advertising-work-statistics-effectiveness.html/ [cited 26 June 2018].

Additionally, weather circumstances should be taken into account. The averages of 7 cities is shown in Table 16.4, the table is divide in the east and west of the USA. The spacecraft are able to optimise the place where the advertisement is shown. If one side of the USA is full of clouds, the advertisement will be pointed to the other side. The sample shown in Table 16.4 is too small for an accurate estimation, therefore the averages found are scaled with the global average. Approximately 68% of the Earth is covered with clouds, averaged over one year [51]. Above land this is about 10-15 % lower than above sea [51]. Land is approximately a third of Earth surface. The average covered area above land is then calculated to be $63.83\ \%$. The scaled average percentage where the weather is mostly clear for the west and the east of the USA are shown in Table 16.4. Then statistics is used to determine the percentage where there is mostly clear weather as shown in Equation 16.2. Averaging those values for every month result in a total factor where it's possible to advertise of $60\ \%$.

$$Clear\ Weather = 1 - (1 - West_{Scaled\ Average}) \cdot (1 - East_{Scaled\ Average}) \tag{16.2}$$

The number of potential views obtained in Section 5.6 is $10679\ bln$ per year. After taking into account the mentioned factors from previous paragraphs, this leaves 1.83 % of the potential views; 195 bln. per year. The typical price for a regular billboard is $5.21\ FY2018\ USD$ per 1000 views[21], which results in an annual revenue of $1.02\ bln\ FY2018\ USD$ for the billboard and a total revenue of $19.4\ bln\ FY2018\ USD$. This yields a total profit of $4.4\ bln\ FY2018\ USD$ and a return on investment of 29%. Therefore, requirement **BBIS-Tud-01** and **BBIS-Sys-T01-1** are met.

[21]URL https://dashtwo.com/blog/how-much-does-billboard-advertising-cost/ [cited 2 May 2018].

	West				East					Total
	Coast		Land		Coast		Land			
Month	Seattle	Los Angeles	Salt Lake City	Scaled Average	New York	Miami	Chicago	Nashville	Scaled Average	
January	30%	60%	50%	30%	48%	65%	42%	44%	36%	**55%**
February	32%	58%	48%	29%	48%	68%	45%	48%	37%	**55%**
March	35%	65%	49%	32%	50%	68%	47%	51%	37%	**57%**
April	38%	72%	55%	35%	50%	65%	48%	57%	37%	**59%**
May	45%	78%	58%	38%	50%	55%	51%	55%	33%	**59%**
June	51%	87%	70%	44%	52%	35%	55%	56%	28%	**60%**
July	70%	90%	80%	51%	58%	35%	65%	63%	30%	**65%**
August	70%	90%	80%	51%	62%	35%	68%	69%	31%	**66%**
September	65%	88%	78%	40%	62%	35%	67%	68%	31%	**65%**
October	42%	80%	65%	40%	62%	51%	58%	65%	36%	**61%**
November	32%	70%	55%	33%	55%	62%	47%	56%	37%	**58%**
December	30%	62%	52%	31%	50%	65%	43%	47%	37%	**56%**
Average	45%	75%	62%	38%	54%	53%	53%	57%	34%	**60%**

Table 16.4: Average Time Mostly Clear Weather.[22]

[22]URL https://weatherspark.com/ [cited 2 June 2018].

17. System Verification

This chapter discusses the verification of user-, functional-, system-, and sustainability requirements. First the requirements which have not yet been verified are discussed in Section 17.1, followed by the compliance matrix in Section 17.2.

17.1. Miscellaneous Requirements

All requirements which have not been verified yet, are verified in this sections. They are listed below.

- **BBIS-Sys-P01-1** The spacecraft shall not generate a sound at Earth surface of more than $0\ decibels$ when it is in orbit.
  - There is a vacuum in space, so no sound will travel to Earth surface.
- **BBIS-Sys-G01-3** The spacecraft shall not light pollute other spacecraft.
  - Other spacecraft will be light polluted by the BBIS, however, research has to be done to see what the effect of the light pollution is. The amount of time another spacecraft spends inside the reflected light is small, therefore the effect could be small. This is further elaborated on in Chapter 18.
- **BBIS-Sys-G03-1** The spacecraft shall provide a maximum illumination of $[TBD]\ lx$ at Earth.
  - Requirement **BBIS-Sys-G03-1** is discarded, because the maximum illumination of the spacecraft is not more than that of a full moon. Therefore, the light does not bring any danger to life on Earth which verifies the user requirement **BBIS-Gov-03**.

17.2. Compliance Matrix

Table 17.1 shows the user requirements established at the beginning of the project, along with whether these requirements have been met. Bellow the symbols used in compliance matrix are shown.

- Requirement is met: ✓
- Requirement is not met: ✗
- Requirement is discarded: †
- Requirement cannot be yet verified: ■

System requirements are listed in Table 17.2, 17.3, 17.4 in a form of a compliance matrix. The last column in the matrix shows the status of the matrix and the *reference* column indicates in which section are those requirements discussed.

Table 17.5 and 17.6 list all the functional requirements. Many of those requirements have been discarded and new requirements have been proposed to those listed in [3]. The reasoning is listed in their respected chapters which can be found in column *reference* in the compliance matrix.

Finally, sustainability requirements have been listed in Table 17.7. All of those requirements are discussed in Chapter 15 and majority of the requirements cannot be verified yet as BBIS is only in Phase 0 and those requirements occur in other phases of the mission.

Class	ID	Requirement	Reference	Status
TU Delft	BBIS-Tud-01	The project shall be cost effective.	16.4	✓
	BBIS-Tud-02	The spacecraft shall have a minimal lifetime of 50 years.	4.2.2 & 9.1	†
	BBIS-Tud-03	The spacecraft shall be launched before 1-1-2028.	13.2	✗
	BBIS-Tud-04	The advertisement shall be optimised for the United States of America.	5.3	✓
	BBIS-Tud-05	The spacecraft shall be able to withstand all of the conditions exerted on it during its life.	10 & 11.1	■
	BBIS-Tud-06	The spacecraft shall fly in space.	5.3	✓
	BBIS-Tud-07	The spacecraft shall be sustainable.	15.2.2	✗
	BBIS-Tud-08	The spacecraft shall have a minimal lifetime of 20 years.	4.2.2 & 9.5	✓
Advertisers	BBIS-Ads-01	The perceived billboard size shall at least be equal to the size of a full moon.	5.4	✓
	BBIS-Ads-02	The perceived billboard light intensity shall at least be equal to the intensity of a full moon.	6.4	✓
	BBIS-Ads-03	The advertisement shall be easy to see by public.	5.4 & 6.4	✓
	BBIS-Ads-04	The advertisement shall be recognisable from Earth.	5.4 & 6.4	✓
Public	BBIS-Pub-01	The spacecraft shall not disturb the day to day life of the observers.	15.3	✓
	BBIS-Pub-02	The spacecraft shall not destroy life on Earth.	15.3	✓
	BBIS-Pub-03	Advertisements shall not be provoking.	15.3	✓
Government	BBIS-Gov-01	The project shall follow the legislation of the countries involved in the process.	12.1.2	✓
	BBIS-Gov-02	The billboard shall be able to deliver important announcements during emergencies.	8.2.4	✓
	BBIS-Gov-03	The billboard shall not bring people in danger by distracting them.	15.3	✓
	BBIS-Gov-04	The spacecraft shall comply with all Inter-Agency Space Debris Coordination Committee regulations.	12.3.2	✓
Launcher Company	BBIS-Lan-01	The spacecraft shall be able to be launched by the launcher.	12.1.1	✓
	BBIS-Lan-04	The spacecraft shall not damage the launcher.	12.1.1	✓
Space Agencies	BBIS-SA-02	The spacecraft shall be disposed after service so that the risk of collision is less than 0.1% in the next $10\,000\ years$.	12.3.2	✓
Manufacturer	BBIS-Man-01	The parts from the suppliers shall be delivered at least TBD days before the launch.	13.3	■
	BBIS-Man-02	The assembly of the spacecraft shall be producible.	13.3	✓
Suppliers	BBIS-SUp-01	The order shall be placed TBD days before the delivery.	13.3	■
	BBIS-SUp-02	The spacecraft shall only be designed with COTS components.	13.3	✗

Table 17.1: Verification and Validation of User Requirements.

Class	ID	Requirement	Reference	Status
TU Delft	**BBIS-Sys-T01-1**	The project shall have a return on investment of at least 0%.	16.4	✓
	- BBIS-Sys-T01-1.1	The cost of the total project shall not exceed 1 553 226 216.58 × lifetime USD, with lifetime in years.	16.1	†
	BBIS-Sys-T04-1	The billboard shall be visible from the USA for $905\ hours$ per year under ideal weather conditions.	5.6 & 16.4	✓
	BBIS-Sys-T05-1	The spacecraft shall be able to withstand all the exposed conditions during assembly.	13.3	■
	- BBIS-Sys-T05-1.1	The spacecraft shall be able to withstand all external forces during every stage of the assembly process.	13.3	■
	- BBIS-Sys-T05-1.2	The spacecraft shall be able to withstand all internal forces during every stage of the assembly process.	13.3	■
	- BBIS-Sys-T05-1.3	The spacecraft shall be able to withstand a pressure ranging from $[TBD]\ Pa$ to $[TBD]\ Pa$ during every stage of the assembly process.	13.3	■
	- BBIS-Sys-T05-1.4	The spacecraft shall be able to withstand a temperature ranging from $[TBD]\ K$ to $[TBD]\ K$, during every stage of the assembly process.	13.3	■
	BBIS-Sys-T05-2	The spacecraft shall be able to withstand all the exposed conditions during transportation.	13.3	■
	- BBIS-Sys-T05-2.1	The spacecraft shall be able to withstand a g-load range of $[TBD]$ to $[TBD]$.	13.3	■
	- BBIS-Sys-T05-2.2	The spacecraft shall be able to withstand the vibrations during transportation.	13.3	■
	BBIS-Sys-T05-3	The spacecraft shall be able to withstand all the exposed conditions during launch.		✓
	- BBIS-Sys-T05-3.1	The spacecraft shall be able to withstand a g-load range of $[1]$ to $[6]$ in launch mode.		✓
	- BBIS-Sys-T05-3.2	The spacecraft shall be able to withstand the vibrations ranging from $[8]\ Hz$ to $[27\ Hz$ during launch in launch mode.		✓
	- BBIS-Sys-T05-3.3	The spacecraft shall be able to withstand a pressure ranging from $[TBD]\ Pa$ to $[TBD]\ Pa$ during the launch in launch mode.	11.1	■
	- BBIS-Sys-T05-3.4	The spacecraft shall be able to withstand the change in temperature ranging from $[TBD]\ K$ to $[TBD]\ K$ during launch in launch mode.	10.1	†
	- BBIS-Sys-T05-3.5	The spacecraft shall be able to withstand heat load variation during launch of approximately $250\ W/m^2$.	10.1	✓
	BBIS-Sys-T05-4	The spacecraft shall be able to withstand all the exposed conditions during operation.	10.2.1	✓
	- BBIS-Sys-T05-4.1	The spacecraft shall be able to withstand a g-load range of $[TBD]$ to $[TBD]$ in operational thrust mode.	11.1	■
	- BBIS-Sys-T05-4.2	The spacecraft shall be able to withstand the g-load of $[TBD]$ to $[TBD]$ in the deployed phase.	11.1	■

Table 17.2: Verification and Validation of System Requirements, Part 1.

Class	ID	Requirement	Reference	Status
TU Delft	- BBIS-Sys-T05-4.3	The spacecraft shall be able to withstand the pressure of air at its operating altitude.	11.1	■
	- BBIS-Sys-T05-4.4	The spacecraft shall be able to withstand the electromagnetic field at its orbit.	10.2.1	✓
	- BBIS-Sys-T05-4.5	The spacecraft shall be able to withstand the radiation in its orbit.	10.2.1	✓
	- BBIS-Sys-T05-4.6	The spacecraft shall be able to withstand a temperature ranging from $6.15\ K$ to $573.15\ K$ in its operation mode.	10.6.2	✓
	BBIS-Sys-T05-5	The spacecraft shall be able to withstand a collision with space debris smaller than $0.01\ m$ at a collision speed of $1500\ m/s$ such that it will not lead to total failure of the mission.	13.4	✓
	BBIS-Sys-T06-1	The spacecraft shall orbit at an altitude higher than $100\ km$.	5.3	✓
Advertisers	**BBIS-Sys-A01-1**	The spacecraft shall have a minimum radius of $0.00436\cdot$ altitude of orbit m when it passes over the USA.	5.1	†
	BBIS-Sys-A01-2	The spacecraft shall be able to maintain an altitude of $35\ 786\ km$ with a maximum deviation of $100\ m$.	5.4 & 7.3	✓
	BBIS-Sys-A01-3	The spacecraft shall have an area of at least $\pi(0.00436 h_{sat})^2\ m$ when it passes over the USA.	5.4	✓
	BBIS-Sys-A02-1	The spacecraft shall provide an illumination at Earth of at least $0.13\ lx$.	6.4	
	BBIS-Sys-A03-1	The orientation of the spacecraft shall be controlled with a precision of $0.0618°$.	7.3	✓
	- BBIS-Sys-A03-1.1	The orientation of the spacecraft shall be determined with a precision of $0.0618°$	7.3	✓
	- BBIS-Sys-A03-1.2	The spacecraft shall be able to adjust its orientation with a precision of $0.0618°$	7.3	✓
	BBIS-Sys-A03-2	The spacecraft shall be visible for at least $14\ s$ per flyover.	5.9	✓
	BBIS-Sys-A04-1	The light source shall be focused on Earth's surface.	5.3.1	✓
Public	**BBIS-Sys-P01-1**	The spacecraft shall not generate a sound at Earth surface of more than $0\ decibels$ when it is in orbit.	17.1	✓
	BBIS-Sys-P02-1	The spacecraft shall follow the regulations of the Occupational Safety and Health Administration of the USA department of labour concerning toxic materials.	13.3	†
Government	**BBIS-Sys-G01-1**	The spacecraft shall not be launched from the USA.	12.1.2	✓
	BBIS-Sys-G01-2	The spacecraft shall not interfere with radio signal.	8.2.1	
	BBIS-Sys-G01-3	The spacecraft shall not light pollute other spacecraft.	17.1	✗
	BBIS-Sys-G03-1	The spacecraft shall provide a maximum illumination of $[TBD]\ lx$ at Earth.	17.1	†
	BBIS-Sys-G04-1	The spacecraft shall be able to avoid an "expected" collision with known debris by the Department of Defence of the USA.	13.4	✓
	- BBIS-Sys-G04-1.1	The spacecraft shall have a velocity increment of $[TBD]\ m/s$ available.	5.5.2	†

Table 17.3: Verification and Validation of System Requirements, Part 2.

Class	ID	Requirement	Reference	Status
Launcher Company	**BBIS-Sys-L01-1**	The spacecraft shall have a maximum size of $[TBD]\ m \times [TBD]\ m \times [TBD]\ m$ during the launch.	12.1.1	†
	BBIS-Sys-L01-2	The spacecraft shall have a maximum weight of $[TBD]\ kg$.	12.1.1	†
	BBIS-Sys-L01-3	The total amount of spacecraft per launcher shall occupy a maximum volume of $118.74m^3$.	12.1.2	✓
	BBIS-Sys-L01-4	The total weight of spacecraft per launcher shall not exceed $6000kg$.	12.1.2	✓
	BBIS-Sys-L04-1	The spacecraft shall have a connector to connect the launcher and the spacecraft.	12.1.3	✓
	BBIS-Sys-L04-2	The spacecraft shall have no contact with the launcher except for the connection point(s).	12.1.3	✓
Space Agencies	**BBIS-Sys-SA02-01**	The spacecraft shall dispose itself in a controlled re-entry or in a suitable graveyard orbit.	12.3	✓
	- BBIS-Sys-SA02-01.1	At the end of the mission the spacecraft shall have at least $34\ m/s$ velocity increment left.	5.5.2 & 12.3	✓

Table 17.4: Verification and Validation of System Requirements, Part 3.

Class	ID	Requirement	Reference	Status
EPS	**BBIS-Func-Eps-01**	The electrical power system shall distribute the required power to all subsystems.	9.6	✓
	- BBIS-Func-Eps-01.1	The spacecraft shall be able to adjust the voltage based on the requirements of the subsystems.	9.6	✓
	- BBIS-Func-Eps-01.2	The spacecraft shall be able to adjust the current based on the requirements of the subsystems.	9.6	✓
	BBIS-Func-Eps-02	The spacecraft shall base the generated power on the mode it is in.	9.2	✓
	BBIS-Func-Eps-03	The spacecraft shall generate a peak power of $[TBD]\ W$.	9.2	✓
	BBIS-Func-Eps-04	The spacecraft shall base the generated power on the mode it is in.	9.2	✓
	BBIS-Func-Eps-05	The spacecraft shall be able to store $[TBD]\ Wh$.	9.5	✓
Attitude Control	**BBIS-Func-Att-02**	The spacecraft shall be able to determine the relative position of the Sun relative to the spacecraft with a precision of $0.0618°$ in three axes directions.	7.3	✓
	BBIS-Func-Att-03	The spacecraft shall be able to determine the relative position of the ground station relative to spacecraft with a precision of $0.0618°$ in three axes directions.	7.3	✓

Table 17.5: Verification and Validation of Functional Requirements, Part 1.

Class	ID	Requirement	Reference	Status
Attitude Control	**BBIS-Func-Att-04**	The spacecraft shall be able to determine the direction of the Earth centre relative to spacecraft with a precision of $0.0618°$ in three axes directions.	7.3	✓
	BBIS-Func-Att-05	The spacecraft shall be able to determine the desired attitude.	8.3	†
	- BBIS-Func-Att-05.1	The spacecraft shall determine the required range of attitudes for communication with the ground system with a precision of $[TBD]\ m$.	8.2.2	†
	- BBIS-Func-Att-05.2	The spacecraft shall determine the required range of attitudes for power generation with a precision of $[TBD]\ m$.	8.4	†
	- BBIS-Func-Att-05.3	The spacecraft shall determine the required range of attitudes for the advertisement display with a precision of $[TBD]\ m$.	8.1	†
	BBIS-Func-Att-06	The spacecraft shall be able to accelerate with $[TBD]\ rad/s^2$ around three axes individually.	7.1	†
	BBIS-Func-Att-07	The spacecraft shall be able to accelerate around three axes individually.	7.4	✓
	BBIS-Func-Att-07.1	The spacecraft shall be able to accelerate around the body's x-axis with $5.48 \cdot 10^{-5}\ rad/s^2$.	7.4	✓
	BBIS-Func-Att-07.2	The spacecraft shall be able to accelerate around the body's y-axis with $3.98 \cdot 10^{-9}\ rad/s^2$.	7.4	✓
	BBIS-Func-Att-07.3	The spacecraft shall be able to accelerate around the body's z-axis with $5.42 \cdot 10^{-5}\ rad/s^2$.	7.4	✓
Thermal Control	**BBIS-Func-Temp-01**	The spacecraft shall be able to measure the temperature of all critical subsystems with a precision of [TBD] K.	10.3	†
	BBIS-Func-Temp-02	The spacecraft shall make a prediction of every subsystems temperature per orbit.	10.3	†
	BBIS-Func-Temp-03	The spacecraft shall check if every subsystem temperature complies with indicated mode.	10.6	✓
	BBIS-Func-Temp-04	The spacecraft shall be able to regulate the temperature within the requirements of the subsystems.	10.6	✓
	- BBIS-Func-Temp-04.1	The spacecraft shall be able to increase the average temperature of the spacecraft with a rate of [TBD] K/h.	10.3	†
	- BBIS-Func-Temp-04.2	The spacecraft shall be able to lower the average temperature of the spacecraft with a rate of [TBD] K/h.	10.3	†
Communication	**BBIS-Func-Com-02**	The spacecraft shall be able to receive a command from Earth.	8.2.4	✓
	- BBIS-Func-Com-02.1	The spacecraft shall decode the signal.	8.2.2	✓
	BBIS-Func-Com-04	The spacecraft shall be able to send information to the ground station.	8.2.4	✓
	BBIS-Func-Com-05	The communication system shall not use more than $[15]\ W$.	8.2.4	✓

Table 17.6: Verification and Validation of Functional Requirements, Part 2.

Class	ID	Requirement	Reference	Status
Sustainability	**BBIS-Sust-Mat-01**	The extraction processes shall not cause irreversible damage in the surrounding areas.	15.2.5	■
	BBIS-Sust-Mat-02	The machinery used in the extraction process shall use clean energy.	15.2.5	■
	BBIS-Sust-Mat-03	The extraction sites shall not be inside endangered areas for the fauna and flora according to the Convention on International Trade in Endangered Species of Wild Fauna and Flora (CITES).	15.2.5	■
	BBIS-Sust-Man-01	Manufacturing processes shall have a reduction of 30% of their gas emissions before 2020.	15.2.5	■
	BBIS-Sust-Man-02	The machinery used in the manufacturing process shall use clean energy.	15.2.5	■
	BBIS-Sust-Trans-01	Electric trucks shall be used as the form of transportation of billboard components.	15.2.5	■
	BBIS-Sust-Op-01	The spacecraft shall not emit harmful substances to the environment during its life cycle.	15.2.2	✗
	BBIS-Sust-Op-02	The ground station used for operation shall use clean energy.	15.2.5	■
	BBIS-Sust-EoL-01	The spacecraft shall comply with the Space Debris Mitigation Guidelines of the Committee on the Peaceful Uses of Outer Space.	15.2.5	✓

Table 17.7: Verification and Validation of Sustainability Requirements.

18. System Validation

Since the billboard in space is an innovative project, with no previous missions as reference, the validation of the whole formation is impossible unless all 900 spacecraft are sent to space. However, it is found to be too risky to send all 900 spacecraft before knowing if the individual spacecraft works. Therefore, a plan is made to validate the mission step by step. This plan is explained below.

The first step is to validate the performance of one spacecraft, which is done by sending a pioneer spacecraft to the designated orbit. During the first launch, the launching system, altitude and orbit control and the communication between the spacecraft and ground station can be validated. The deployment of solar sail and the visibility can be validated afterwards. In case of any failure, few changes can be made on subsequent spacecraft.

Next, the effect of requirement **BBIS-Sys-G01-3**, "*the spacecraft shall not light pollute other spacecraft*" should be evaluated. Requirements **BBIS-Ads-01**, **BBIS-Ads-02**, **BBIS-Ads-03** and **BBIS-Ads-04** are met, which means that the BBIS is large and bright enough to be seen on Earth, therefore the light pollution on other spacecraft is inevitable. However, the effect, and therefore the necessity of this requirement, is hard to verify due to the fact that there is no previous mission with the same objective. Furthermore, the light influence could only be measured when all the spacecraft are sent to space, which is too risky and expensive to be actually taken into practice.

In order to have similar experiment, the light pollution effect can be observed at the first step of the process. After the pioneer spacecraft has been sent to the orbit, an experimental spacecraft with Sun and star sensors is sent between the pioneer spacecraft and Earth. During this experiment it is possible to detect the influence of the light pollution on the operation of a spacecraft. Thus experiment should be done in cooperation with all space agencies around the world, otherwise the BBIS could be liable for potential damage to other spacecraft.

It should be noted that the chances that the operational capacities of other spacecraft are affected is low, the time other spacecraft are within the light polluted area is small. The total area covered is $273\ km$ which moves with a maximum ground speed of $19.49\ km/s$, as explained in Section 5.6. Furthermore, spacecraft in a low Earth orbit move with a speed ranging from 7 to $8\ km/s$. Therefore, spacecraft will never be within the polluted area for more than $24\ s$.[1] Therewith, it is expected that the requirement can be discarded after validation.

Next step is the validation of the formation flying of the spacecraft. After the successful operation of the first validation, three more spacecraft are send to space. The formation of these four spacecraft can be checked with an example "advertisement". The visibility and the performance of the swarm is herewith validated.

In case, everything goes well during the previous steps, the final step of the validation process is started; sending all the remaining spacecraft to the orbit. The performance of all 900 spacecraft can be validated, followed by the normal operation of the BBIS.

[1] $t = \frac{s}{v} = \frac{273}{19.49-8} \approx 24\ s$

19. Conclusion

The advertisement industry is characterised by a substantial capital flow and is continuously innovating. In this project, an advanced advertisement method capable of reaching a wide range of viewers by means of a billboard in space (BBIS) is researched. To judge the feasibility of such a project, knowledge of a diverse variety of related fields is required. In this report, the project design for the BBIS, made up of a swarm of spacecraft with reflective surfaces, is investigated. This concept stems from the final trade-off between design concepts in [8].

The biggest challenge for the BBIS revolved around determining a feasible orbit. The previous orbit was chosen to be in low Earth orbit [8]. However, a more in-depth analysis showed that this was not possible due to the harmful environment in this orbit. The new orbit is geosynchronous at $35\ 786\ km$ altitude and is designed for an optimal visibility from the USA, resulting in beneficial profitability prospects. The 900 spacecraft are able to fly in two formations. The first formation has $20\ \times\ 45$ grid of spacecraft and the second formation has a $5\ \times\ 180$ grid. The formations are visible from the USA for 8 hours a day. In total, the BBIS can adjust its formation five times, optimising for text or logo display.

The second challenging design aspect concerns the payload. In order for the BBIS to have the same visibility as a full moon, the minimal illumination is required to be $0.13\ lx$. A solar sail is used to reflect sunlight to Earth. The sail consists of two layers of material, a layer of kapton and a layer of aluminium. Kapton provides the structural strength of the sail, which is especially important during folding and deployment. The layer of aluminium film ensures the solar sail is reflective and can be seen from Earth. The beam angle of the sail is $0.62°$. A total solar sail area of $406\ m^2$ is required such that this beam angle constraint is met. The sail is attached to ultra-light carbon-fibre reinforced booms that extract from the corners of the spacecraft during deployment.

In order to obtain a high pointing accuracy of the payload, the attitude and orbit determination and control system requires a high accuracy. A large safety margin is included, specifying that the system must be able to point with $0.062°$ accuracy equal to $1/10^{th}$ of the beam angle in order for the billboard to be visible from Earth. Four determination sensors, including an additional sensor for redundancy, are chosen such that this requirement is met; a star tracker, a Sun sensor, an inertial measurement unit, a GPS receiver, a GPS-enhanced navigation system and an extended Kalmann filter. .

For the attitude control, reaction wheels are designed to rotate the spacecraft around each axis individually. For 3-axis stabilisation including redundancy, two reaction wheels per plane are included. The required manoeuvrability is determined by the subjected disturbance torques and required rotation actions. For orbit control, a propulsion system is used. In total, there are 12 thrusters present such that the spacecraft is able to perform individual manoeuvres around each of its three axes. These thruster's propellants are monomethyl hydrazine and dinitrogen tetroxide.

Each spacecraft is mounted with three omni-directional antennas for communication. Two antennas are used for crosslink communication, which happens continuously. The spacecraft exchange their location, which is critical to prevent collisions. Additionally, the third antenna is used for downlink and uplink, to communicate with Earth for $12\ hours$ per day. The data is managed by an on-board computer. This computer is also responsible for conducting the communication between all the spacecraft's subsystems.

The spacecraft's power is generated by thin-film solar cells that are attached to the solar sail. In total, a solar cell area of $1.69\ m^2$ is used to generate the required amount of power for the subsystems. Moreover, the solar cells charge the batteries when subjected to sunlight. When the spacecraft is in eclipse, the batteries deliver power to the subsystems.

Throughout the mission, the BBIS experiences a variety of environments. Radiation has the most severe impact on the spacecraft. It affects all electronic components and could cause them to malfunction.

Therefore, each electronic component is protected by aluminium boxes with various thicknesses depending on their resistance to radiation.

The spacecraft's body temperature is predominantly influenced by its exposure to sunlight. The temperature varies depending on the spacecraft's orientation with respect to the Sun. On average, the BBIS's temperature is $21.95°C$. The maximum temperature ($30.44°C$) is achieved on the face of the spacecraft bus that is facing Earth. The minimum temperature ($-15.13°C$) is on the opposite face, located behind the sail. Thermal protection is vital for all components. Subsystems are placed in the spacecraft based on their operational temperatures. Highly conductive paint is used to cover the spacecraft bus and the back of the solar sail has a chromium layer.

Additionally, the spacecraft's structure is designed. The total structural mass is $14.13\ kg$. The primary structure consists of the rectangular bus and three stiffeners, which are all made of aluminium 7075-T73. The structure's thickness is equal to $3.5\ mm$, and its width, height and length are $340\ mm$, $340\ mm$ and $660\ mm$, respectively. The secondary structure includes components such as the 3U CubeSat structure, the bolts and the deployment mechanism of the solar sail. Failure of these elements would not lead to failure of the entire spacecraft. The BBIS is designed for the most critical loading case which occurs during launch. The resulting safety margins for buckling of the bottom plate and yielding of the top plate are 2.06 and 1.05, respectively.

To launch the BBIS, the Ariane V is used. In total, 9 launches are required. The deployment of the spacecraft formed another challenging design aspect. During launch, the spacecraft are fixed to a beam structure that is attached to the spacecraft adaptor. Upon reaching the desired altitude, first, the fairing separates, after which the spacecraft are detached. The individual propulsion systems are activated once the spacecraft are a safe distance apart from each other, bringing the spacecraft to their respective final orbit. At the end-of-life, the spacecraft fleet is disposed to a graveyard orbit which is located $12\ 280\ km$ above the primary orbit of the BBIS.

Risk mitigation is taken into account during the design of the BBIS, and thus, the final design contains redundancies and suitable plans in case of failure. The result is that there are no critical risks left for the BBIS mission. Furthermore, sustainability is taken into account. The spacecraft subsystems that most affect the mission's sustainability are analysed and discussed. For the propulsion system, the option of using green propellants is discarded. This was done, because of the low specific impulse of green propellants, which would cause a propellant mass, and thus total mass, increase. This would lead to the use of more launchers to deliver the swarm to space, which is less sustainable than the use of non-green propellants. The sustainability for the electrical power subsystem is optimised by using solar cells and batteries instead of RTG's. Furthermore, the Ariane V is considered one of the most efficient launchers, minimising the overall environmental impact of the launch. In addition to these technical aspects, the social sustainability elements are also investigated. For the BBIS, ethics-related features such as the fact that the mission should not the intrusive or distracting are verified.

Finally, a market and cost analysis is performed. For the BBIS, multiple investment parties related to a broad variety of advertising companies, as well as scientific research and government-related associations, are identified. The total BBIS mission budget including the launch, spacecraft design, development, test and evaluation phase and mission operational costs sum up to $15.0\ billion\ USD$. Based on the number of views, a total revenue of $19.4\ billion\ USD$ is estimated. This yields a profit and return on investment of $4.4\ billion\ USD$ and 29%, respectively. Thus, is can be concluded that the BBIS is financially feasible.

Recommendations

In order to further improve the design, some recommendations are given. First of all, a total lifetime of $20\ years$ remains a critical requirement to be further investigated. Many COTS components do no have an expected lifetime that meets this requirement, resulting in a high risk of failure. Secondly, in reality, formation flying has never been performed on such a large scale. The execution of such a complex mission still needs to be reviewed extensively before it can be realised. Furthermore, it is important to validate the visibility of the spacecraft pixels on Earth. It is proposed to first attempt formation flying in a geosynchronous orbit with a smaller fleet of spacecraft with reflective surfaces, before attempting to launch the BBIS. Moreover, it is advised to reconsider the launch date. For the BBIS, the launch date is set to be before January 1st 2028. However, due to the high validation risk related to mission aspects and technology readiness level, a launch date delay is recommended. It is estimated that the launch of the BBIS could be in the year 2043.

In addition to these recommendations, two critical mission characteristics require additional investigation. First of all, it must be noted that the design of the BBIS does not meet the requirement concerning the light pollution of other spacecraft orbiting around Earth. However, it is unknown what the consequences of this violation of the requirement are and whether this requirement is killing. Again, a validation of this process is recommended.

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A. Appendix A: Technical Drawings

Technical drawings of the inner arrangement of the spacecraft bus are shown in Figure A.1, A.2. The overall view of the whole spacecraft is shown in Figure A.3.

Figure A.1 scale is 1:15, the figure states it is 1:12, however, the figure had to be scaled down. Same happened with Figure A.3, its actual scale is 1:312.

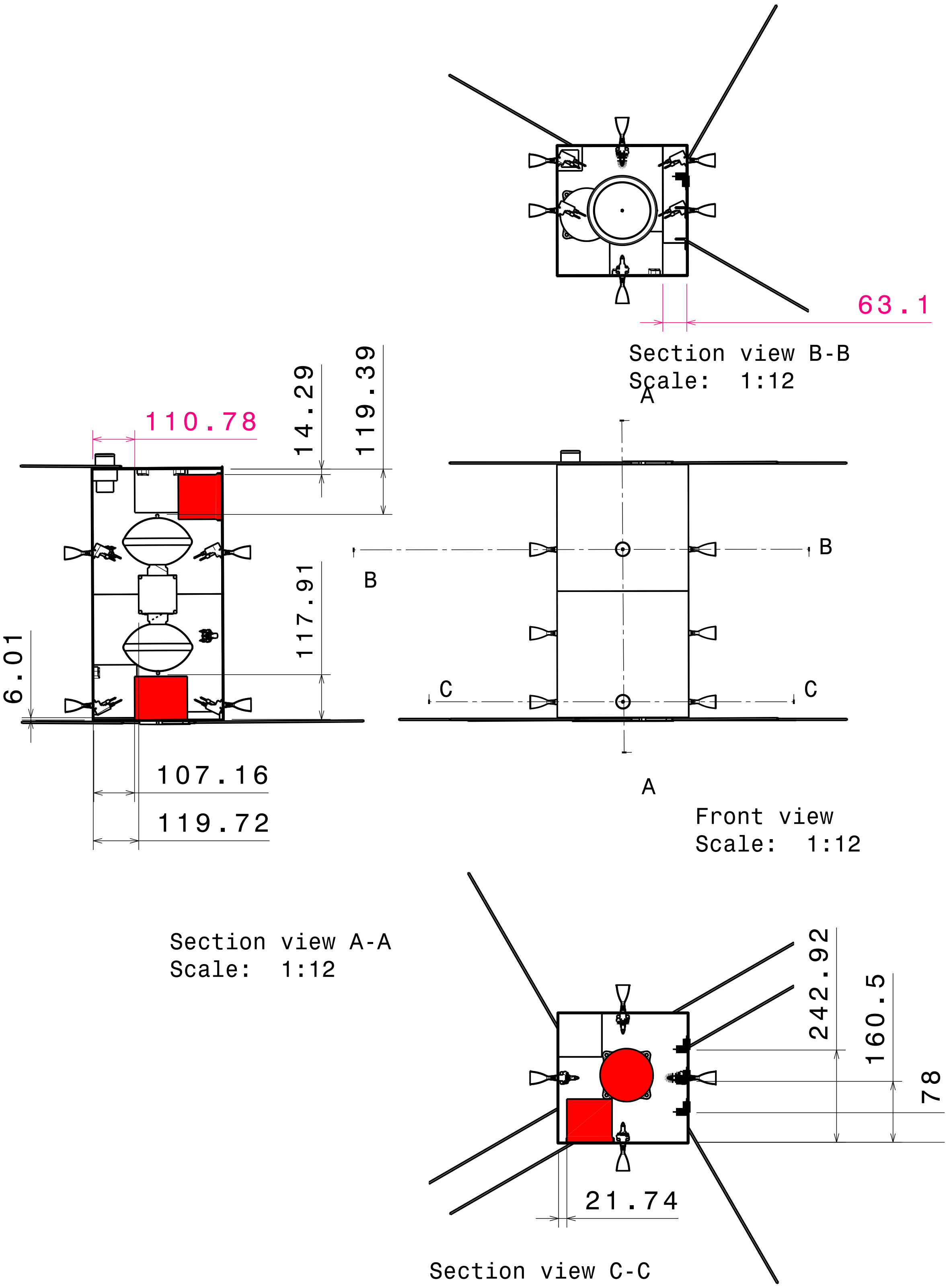

Figure A.1: Technical Drawing of the Components Inside the Spacecraft Bus.

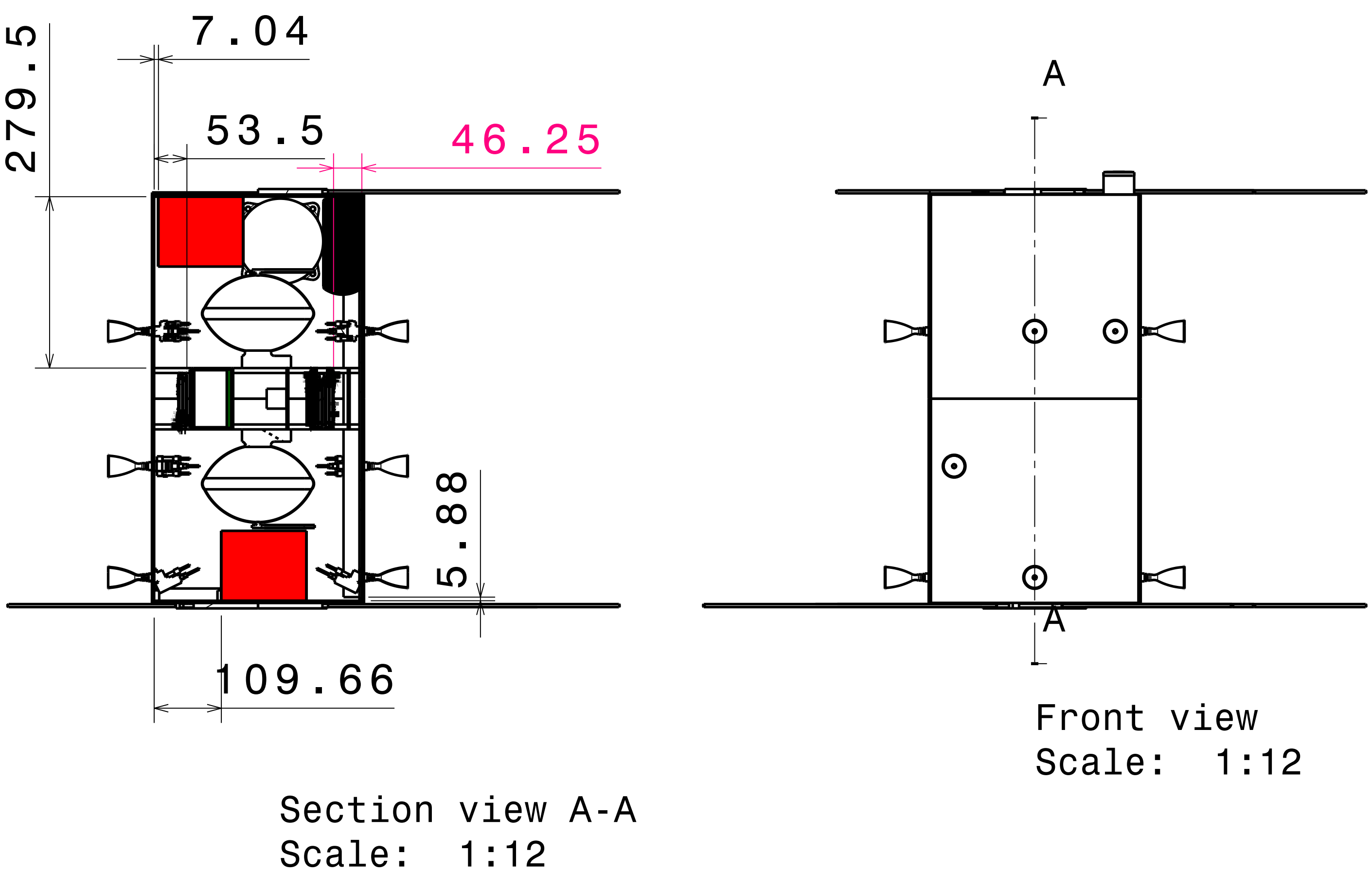

Figure A.2: Technical Drawing of the Components Inside the Spacecraft Bus.

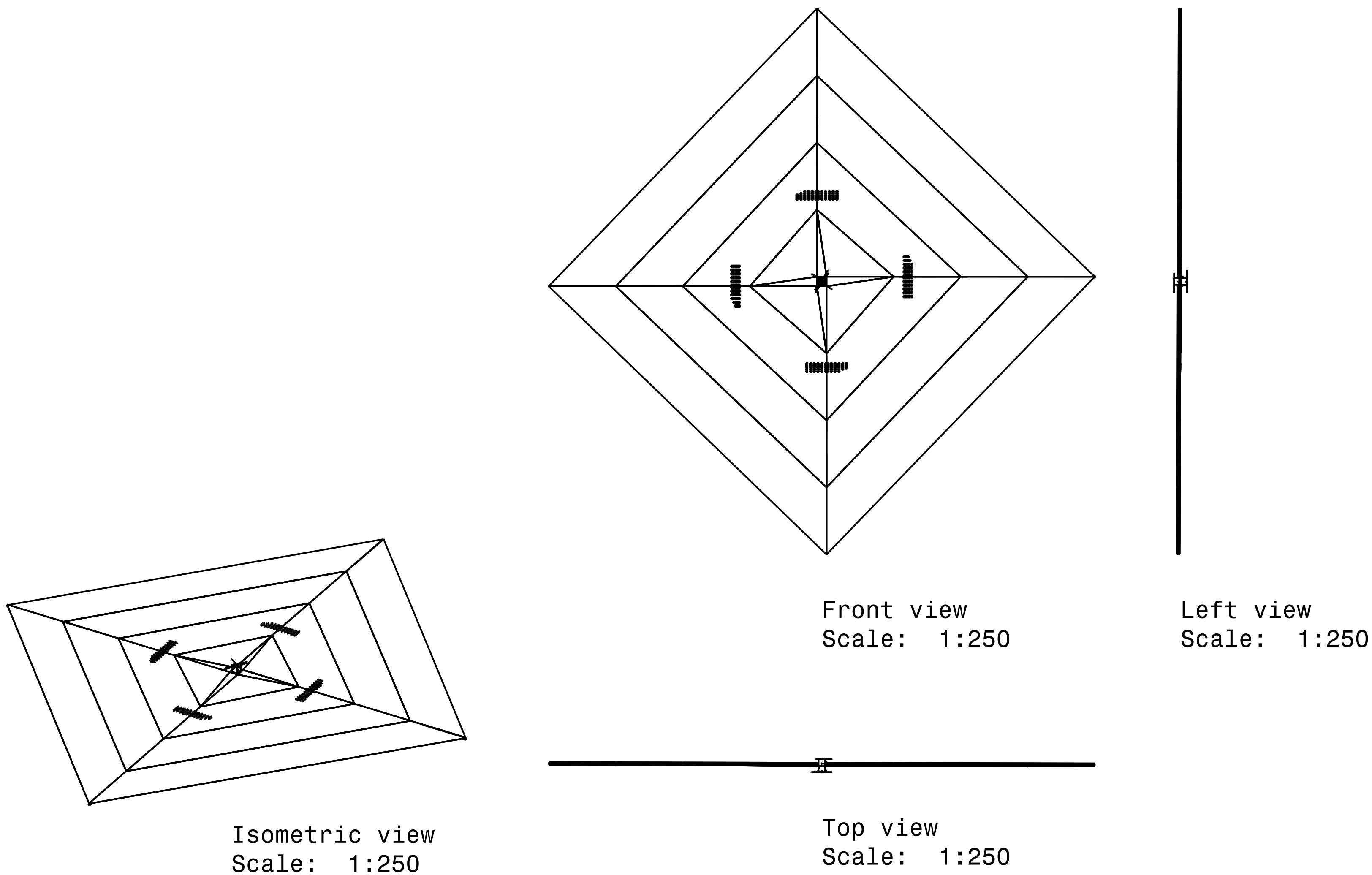

Figure A.3: Technical Drawing of the Overall Spacecraft.