Mercury Orbiter

Work Package 1: Initial sizing, Version 1

Group C4 - T.A. Edward Na – Spacecraft

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Preface

This report was written by a group of eight aerospace engineering students of the Delft University of Technology. This report will be followed by another one in the series of 3 reports that aim to improve team-working as well as engineering behind space missions.

Readers that are interested in functional analysis of the spacecraft, mission elements and objectives are referred to Chapter 1 and the following. The ones interested in spacecraft design requirements, including top level requirements, as well as reasoning standing behind given values shall refer to Chapter 2. If a reader is interested in mission profile, spacecraft size estimation along with orbital parameters of the mission and ΔV estimations should take a look into Chapter 3. The initial sizing of the spacecraft, including mass budget estimations, solar array size calculation and positioning, as well as first engineering sketches of the spacecraft shall consider reading section Chapter 4.

We would like to thank Teaching Assistant Edward Na for the help, feedback and time spent during the research and making of this report. We would also like to thank our responsible lecturers Dr. Angelo Cervone and Dr. Stefano Speretta for the opportunity to work on a space related teamwork engineering project.

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List of Symbols

μ	Standard gravitational parameter	$\mathrm{m}^3 s^{-2}$
ρ	Density	$\mathrm{kg/m}^3$
a	semi-major axis	m
DR	Data rate	bit/s
E	Energy stored in a battery	J
G	Gravitational constant	$\mathrm{m}^3 kg^{-1} s^{-2}$
M	Mass of the orbited body	kg
P	Power	W
r	Radius	m
S	Surface	m^2
T	Orbital period	s
t_e	time of the eclipse (spacecraft not in sunlight)	s
V	Velocity	m/s

Summary

This report intends to give an insight into the initial steps taken in the design of a space mission, more specifically a mission to the innermost planet of the solar system, Mercury. The objective is to devise a feasible mission profile and initial sizing of the spacecraft taking as a foundation previous missions to Mercury as well as statistical relationships derived from preceding planetary exploration missions.

In a first step information on previous missions together with the conditions in the spacecrafts operating environment were collected to be used for estimation calculations for mass and power of the different spacecraft systems. Also an extensive list with mission requirements was created to aid in the further design steps of the spacecraft, all of which were thoroughly explained and justified with the appropriate calculations when necessary. This resulted in preliminary launcher-, delta V-, ADCS-, bitrate and battery requirements for the spacecraft to be used in later calculations. The spacecraft vehicle level estimation was done using two methods of parametric estimation which due to the small amount of data available from the merely three missions that had already left for Mercury by the time of this report, lead to results with very high uncertainty and were therefore unsuitable for further design studies. Using the parametric relationships provided by B.T.C.Zandbergen in the ADSEE manual resulted in more reliable data, this being a dry mass of 731kg, a wet mass of 1553 kg, a power requirement of 125 W and a size estimation of $7m^3$ with a spacecraft cost of 155.17 million USD.

The results from the estimations were then used in the creation of a complete mission profile which employed a Hohmann transfer orbit to Mercury in order to keep the travel time relatively short while harvesting the required energy for its electrical propulsion system to provide the delta V for the maneuvers from the suns radiative power. The optimal launcher was selected to be the new launch vehicle Ariane 6 with a price tag of 115 million euros, launching from French Guyana to a Geosynchronous Transfer Orbit (GTO). After a 102 day cruise the spacecraft would reach Mercury and move to its final polar orbit at a height of 750 km above the planets surface and start its scientific mission of a duration of at least 1 year. The total delta V needed for all the maneuvers was estimated to be 18270 m/s.

In a final step the preliminary mass and power budgets were estimated giving a more detailed breakdown of the mass and power requirements of the various spacecraft subsystems. This allowed for a calculation of the power needed during daytime and nighttime which on the other hand together with the calculations for orbital eclipse period provided the necessary data for a solar array size estimation resulting in $22.25m^2$. All the data calculated previously was merged into three preliminary spacecraft designs. A cubic spacecraft with a bi-array, one with a tri-array configuration and a cylindrical design. For all of them the mass moments of inertia were calculated. The first design concept previously mentioned was turned into a CAD drawing providing a simplified model of the desired outcome.

There were certain challenges in the preliminary design, one of them being the lack of data available from previous Mercury missions. This paired with the wide range of the estimation relationships lead to a very rough design which will have to be fleshed out in further refinement steps through more in depth analysis and calculations.

Introduction

The following report contains the deliverables for the first Work Package (WP1) of the project AE2111-I System Design. This work package consists of the first design steps for a Mercury orbiter, including researching of previous Mercury missions and setting specific mission requirements in accordance with the mission's objectives and top level requirements. Possible spacecraft architectures are discussed too, with mass moment of inertia calculations and a CAD drawing provided for one of the spacecraft.

1

Functional Analysis

This chapter presents research completed on spacecrafts and missions towards Mercury. The mission objectives and general information is presented in Table 1.1 in section D1.1.1. The mission elements with which the design process will interact and what functionalities that each shall achieve are in section D1.1.2. Additionally, the initial design parameters as well as the design parameters which need to be identified are presented in section D1.1.3 in a single list.

D1.1.1. Comparable Spacecraft Missions

In this first section information on previous missions to Mercury has been collected. Only 5 missions are shown because that is the total number of orbiters sent or planned to be sent to said planet as of 2022. However, the existence of relatively recent missions (BepiColombo and Messenger) also facilitates comparable data points, as will be explored in further chapters.

Table 1.1: *Mercury missions comparative table*

	BepiColombo MIO/MMO [3][4][5]	**BepiColombo MPO** [3][4][5]
Launch date	2018	2018
Countries involved	EUROPE /JAPAN - ESA/JAXA	EUROPE /JAPAN - ESA/JAXA
Budget	$2,000,000,000	
Mission objectives	Study the magnetosphere and its interaction with the planetary magnetic field. Investigation of plasma particles and other energetic particles; electric fields, plasma waves, and radio waves.	Study of Mercury's magnetic field and the effects of solar wind.

	Mariner 10 [6][7]	**Messenger**[8][9]	**Mercury-P** [10]
Launch date	1973	2004	2030 approx.
Countries involved	USA - NASA	USA - NASA	Roscosmos
Budget	$100,000,000	$450,000,000	No information
Mission objectives	Measuring Mercury's atmosphere, surface, and body characteristics.	Determine Mercury's surface composition, study its geological history and internal magnetic field, and verify its poles contain dominantly water-ice.	Study geology and planet composition (Orbiter + lander).

D1.1.2. Mission Elements Interacting With the Design Process

To initiate the design process all mission elements with which the design will interact need to be defined. These elements are stated in the list below, as well as their function or characteristics. Block diagram 1.1 has also been provided to highlight primary relations between these elements.

- **Bus**: The vessel housing the payload, provides power, communication and propulsion.
- **Payload:** Sensors and instruments to gather data to perform experiments specified by the mission.
- **Launch Vehicle:** Provides the delta v to reach the target (Mercury).
- **Trajectory and Orbit:** The path that the spacecraft follows to get to the target and around the target respectively.

- **Mission Management and Operations:** The collection of activities required to operate a mission such as monitoring, control, performance analysis, reporting, planning, scheduling and the execution.
- **Ground System:** The network of facilities on the ground supporting the spacecraft and mission. It has the role of receiving and processing data as well as sending data to control the vehicle.
- **Mission Objective:** The goal of the mission. In this case gathering data about Mercury.

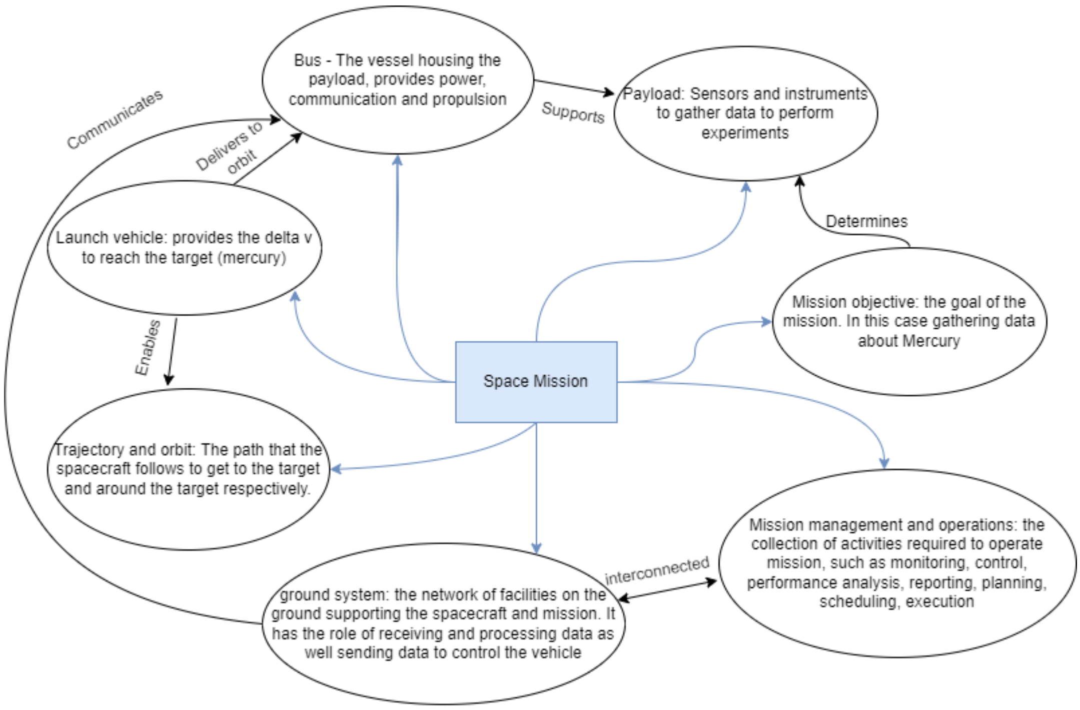

Figure 1.1: *Block Diagram showing Mission Elements and their Relations*

D1.1.3. Design Objectives and Parameters

The main objectives to be achieved in the design process are:

- The orbiter shall be able to carry and permit operation of the payload.
- The orbiter shall also allow for structural support and shielding of the payload.
- The orbiter shall provide enough power for effective operation of the payload.
- The orbiter shall produce enough power to sustain all of its operations.
- The orbiter shall be able to maintain a stable temperature within the range of operations.
- The orbiter shall be capable of achieving a high reliability.
- The orbiter should be able to perform the necessary orbital maneuvers and corrections during its operational life.
- The mission should aim for a launch date no later than 2025.
- The mission should aim for a total cost no larger than 600 million euros.
- The orbiter shall stay in contact with ground control.

2
Spacecraft Requirements

This chapter specifies the requirements for the spacecraft itself as well as the mission overall. This is done by examining previous missions to Mercury as well as inferring requirements from mission's functional analysis presented in Chapter 1. First, the environmental data of the mission objective - Mercury - are presented in Chapter 2. Then, in section Table 2, design data for comparable spacecraft missions is presented. The design requirements drawn from comparable missions and top level requirements are listed in Table 2. The reasoning behind every requirement is explained in-depth under item 2 and its subsections.

D1.2.1 Planet Characteristics

In this section relevant data on Mercury's characteristics will be presented in Table 2.1. This data will be relevant for further steps in the design of the mission. The data was obtained from a Mercury Fact Sheet composed by NASA [11]. Only data relevant for the design was added.

Table 2.1: *Important planet characteristics for the Design*

Characteristic [Unit]	**Value**
Mass $[kg]$	$3.301 \cdot 10^{23}$
Equatorial radius $[m]$	2441000
Polar radius $[m]$	2438300
Gravitational coefficient $[g]$	0.38
Gravitational parameter $[km^3 s^{-2}]$	$2.203 \cdot 10^4$
Surface gravity $[\frac{m}{s^2}]$	3.7
Atmospheric density $[g/cm^3]$	Near vacuum
Bond albedo $[-]$	0.068
Solar irradiance $[\frac{W}{m^2}]$	9082.7
Eccentricity $[-]$	0.205630
Synodic period [days]	115.88
Length of day [hrs]	4222.6

D1.2.2 Design Data for Comparable Spacecraft Missions

Table 2.2 gives data for different spacecraft that have gone or will go to Mercury. For the first four missions, almost all the data was found with some extensive research. However, for Mercury-P, since it's a Russian mission that will be sent in around 10 years, not a lot of data was found. The research was conducted with a mix of google scholar articles, research papers and NASA fact sheets.

Table 2.2: *Comparative Table between different Mercury Missions*

Mission	Mass in Orbit [kg]	Payload Mass [kg]	Dimensions [m]	Power [W]
BepiColombo (MMO)[3][4][5]	285	45	ø1.9 × 1.1	90
BepiColombo (MPO)[3][4][5]	1150	80	3.9× 2.2× 1.7	150
Mariner 10[6][7]	503	79.4	ø1.37 × 0.457	820
Messenger [8][9]	1108	50	1.85 × 1.42 × 1.27	450
Mercury-P [10]	7410	50	-	-

continuation of Table 2.2

Bitrate [$\frac{kbit}{s}$]	Cost [M$]	I_{sp}[s]	Propellant Mass[Kg]	Orbit [km]	Orbit Period [hr]
5	2000	4300	-	Polar orbit, 400 x 11824	9.3
50		4300	-	Polar orbit, 480 x 1508	2.3
118	100	-	29	Flybys, 704; 48069; 327	-
9.9 -104	450	317	607.8	60° - 82.5°, 200 x 9300	12
-	-	-	-	-	-

D1.2.3 Design Requirements

To ensure the design performs as intended, several requirements have been created that the spacecraft needs to fulfil. These are listed below in two lists, separated into top level requirements and those which have been determined to be necessary. The justifications and calculations for the determined requirements are explained in section 2. Requirements that have been deemed central and driving to the design process have been marked with an asterisk *.

Top Level Requirements

1. The spacecraft mission shall have a final reliability of at least 0.9
2. The spacecraft shall have an operational life of no less than a year
3. The spacecraft shall carry at most a payload mass of 150 kg *
4. The spacecraft shall carry at most a payload size of 1 m x 1 m x 0.5 m *
5. The spacecraft shall provide a minimum of 200W of power from BOL to EOL, accounting for array degradation *
6. The spacecraft shall provide 50 W of power to the payload.
7. The spacecraft shall operate between 300 K and 400 K *
8. The spacecraft mission cost shall not exceed 600 million euros

Determined Requirements

Bus

1. The spacecraft shall protect the payload from the harsh environment of space *
2. The spacecraft shall fit in the launcher's payload bay *
3. The spacecraft shall be able to withstand longitudinal launch loads of 4.25-5.5g *
4. The spacecraft shall be able to withstand lateral launch loads of < 0.4g *

Propulsion

5. The spacecraft with its kick stage shall have a delta-v budget of 18.6km/s *

6. The spacecraft shall conserve at its orbit arrival enough propellant for orbit conservation.

   Power

7. The spacecraft shall be able to deliver 423 kJ of electrical energy to its subsystems and payload while the PV-systems are inactive *
8. The spacecraft battery shall have a capacity of (TBD) to support the orbiter during the eclipse phase of the orbit.
9. The spacecraft's batteries shall be able to charge within the daylight time of the orbit (5515.01 s)

   Telecommunication

10. The spacecraft shall be able to transfer and receive data with a bitrate higher than 200 kb/s *
11. The spacecraft shall transfer data with a Bit Error Rate (BER) of maximum 10^{-6}

    ADCS

12. The spacecraft shall be able to point its payload towards Mercury. *
13. The spacecraft shall have an orientation accuracy of 1 arcsec.
14. The spacecraft shall have a pointing accuracy better than 0.1º.
15. The spacecraft shall have a pointing stability of 5 arcsec/msec.

Reasoning

In this section the reasoning behind the requirements of table 2 is explained. These derivations involve comparisons of previous missions as well as calculations.

Payload Protection

The spacecraft has to protect delicate apparatus that is being transported. Space is a harsh environment full of radiation, high-energy particles and meteoroids. All that has to be accounted for in the design process.

Launcher Requirements

The launcher vehicle payload bay has to be large enough to fit the spacecraft inside the rocket. Once the launcher is selected the spacecraft should be designed in such a way that its transportation to space is possible.
Another key aspect is to determine the launch loads the spacecraft must withstand. To estimate them, launch loads of the previous missions can be used, as seen in Table 2.3 below:

Table 2.3: *Launch Accelerations*

Mission	Launcher	Longitudinal accelerations	Lateral accelerations
Bepicolombo [12]	Ariane 5	4.25 g	0.2 g
Mariner 10 [13]	Atlas SLV-3D Centaur-D1A	5.5 g	0.4 g
Messenger [13]	Delta II	5.5 g - 7 g	0 g
Mercury-P [13]	Soyuz 2	4.3 g	-

Important to note are the two longitudinal accelerations stated for the Delta II rocket. These depend on the mass of the spacecraft. To calculate a range of possible loads the range of the previous launchers can be used, which yields 4.25-7g for longitudinal and 0-0.4g for lateral acceleration. However, the high longitudinal acceleration of the Delta II could be considered a statistical outlier. That and the fact that the Delta II was retired in 2018 can result in those values being neglected. This then results in a range of 4.25-5.5g for longitudinal and 0.2-0.4g in lateral acceleration.

Delta V

The value of ΔV present in the design requirements is based on the Hohmann transfer calculations shown in Appendix A. It has a value of 18630 m/s accounting for station keeping.

Table 2.4: *Delta V of several Mercury Missions*

Mission	Delta V [m/s]
Mariner 10	121 [14]
Messenger	1097.75[15]
BepiColombo	3844 [16]

ADCS

The Attitude Determination and Control System is essential in the case of a planetary science orbiter; precise instrument orientation is required, both for the scientific payload and the telecommunications system, which has to exchange data with ground control. These systems have several measurable characteristics, which refer to their overall performance. Among them are pointing accuracy, orientation accuracy, pointing stability and maneuvering rate. Since there are no top-level requirements for these values, they have been obtained from literature [17], choosing numbers similar to those of previous interplanetary missions - always taking into account excessively good accuracy and stability values mean either high energy consumption, high weight, or both. The minimum maneuvering rate was taken to be that which was enough to continuously point towards Mercury and is calculated below. Table 2.5 shows the set of requirements for the Attitude Determination and Control System.

$$\omega_r = \frac{2 \cdot \pi}{T} = \frac{2 \cdot \pi}{7644.5} = 8.219 \cdot 10^{-4} rad/s$$

Table 2.5: *ADCS design requirements*

Requirement	Values
Orientation accuracy	1 [arcsec]
Pointing accuracy	0.1°
Pointing stability	5 [arcsec/msec]
Maneuvering rate	$8.219 \cdot 10^{-4} rad/s$

Bitrate

To estimate the required bitrate for the spacecraft, the following formula can be used:

$$DR_{req} = DR_{housekeeping} + \sum_{i=1}^{n} (DR_i) \tag{2.1}$$

Where DR_{req} is the total bitrate required for the spacecraft, $DR_{housekeeping}$ is the data rate required for the housekeeping of the spacecraft, n is the total amount of instruments and DR_i is bitrate generated by each instrument i after compression. In this case these instruments include a high resolution camera, several spectrometers, several magnetometers and a laser altimeter. Due to low amount of data collected by spectrometers altimeters and magnetometers, most of the bitrate is reserved for the needs of the high resolution camera.

To assess the bitrate, which is mostly data from high resolution camera, the data generated by mapping the surface of the planet is calculated. Different variations of pixel size (in square meters/pixel), as well as different compression rates, bits per pixel (the full color or monochrome image) and time of mission (1 or 2 years) were utilized. The method is as follows:

$$DR = \frac{S_{Mercury} \cdot bits\ per\ pixel \cdot overlap\ factor}{area\ per\ pixel \cdot compression\ ratio \cdot time\ of\ mission} \tag{2.2}$$

The surface area of Mercury used in the aforementioned table is calculated to be 74 796 700 000 000 m^2.

$$S_{Mercury} = 4 \cdot \pi \cdot r^2 = 4 \cdot \pi \cdot 2439700^2 = 74\,796\,700\,000\,000\,\mathrm{m}^2 \tag{2.3}$$

Table 2.6: *Bitrate Data*

Compression	$\frac{bits}{pixel}$	Time of Mission[s]	Overlap	Dimension	$\frac{m^2}{pixel}$	Total Data	Bitrate[$\frac{kb}{s}$]
2	24	31556736	1.20	18.5	342.25	$6.29{\cdot}10^{12}$	99.73
2	24	31556736	1.20	13	169	$1.27{\cdot}10^{13}$	201.97
2	24	31556736	1.20	5	25	$8.62{\cdot}10^{13}$	1365.31
50	24	31556736	1.20	2.6	6.76	$3.19{\cdot}10^{14}$	201.97
2	8	31556736	1.20	7.5	56.25	$1.28{\cdot}10^{13}$	202.27
2	24	63113472	1.20	9	81	$2.66{\cdot}10^{13}$	210.70

The results shown in Table 2.6 are intriguing. Assuming a loss compression ratio of 50, which is very possible, the spacecraft can map the surface of the Mercury with pixel side length of only 2.6 meters (at an average 200 kb/s connection with Earth). When monochromatic mapping is wanted and lossless compression is chosen, then Mercury would be mapped at a resolution of 7.5 meters per pixel.

Battery

To estimate the power stored by the spacecraft, the worst case scenario needs to be considered, where the spacecraft is eclipsed by mercury and power needs to be delivered to both the payload as well as the bus. The required power by the entire system over an arbitrary period of time t is then calculated with:

$$E = (P_{payload} + P_{bus}) \cdot t \tag{2.4}$$

Where E is the required energy, $P_{payload}$ is the power required by the payload and P_{bus} is the power required by the bus. As this is a worst case scenario, $P_{payload} + P_{bus}$ can be considered equal to the total $200W$ stated by the top level requirements. The time used in 2.4 can be considered the maximum time spent in eclipse, which for a polar orbit at 750 km above Mercury is 2115.51 seconds. The calculations for this are shown in Chapter 4.

Finally substituting this value into equation 2.4 yields an energy of $423102J$ or $423kJ$. This means that the spacecraft needs to be able to provide $423kJ$ of energy while its PV arrays are inactive. The actual capacity of the batteries needs to be greater than this value, as these calculations do not include line losses, battery degradation over time nor the fact that batteries are not able to fully discharge.

To ensure the battery can charge at a high enough rate to be sufficiently recharged for a following eclipse the charging rate also needs to be determined. This can be done using the following equation:

$$E_{battery} = P_{charging} * t_{day} \tag{2.5}$$

Where $E_{battery}$ is the energy in the battery, $P_{charging}$ is the charging rate and t_{day} is the time the spacecraft spends in daytime. This value is equal to the total orbital period minus the time spent in eclipse during worst case scenario. Rearranging and substituting the equation yields:

$$P_{charging} = \frac{E_{battery}}{t_d} \tag{2.6}$$

This means that the battery needs to be able recharge within the daylight time of the orbit.

3

Mission Profile

In this third chapter the mission will start to take form. The orbiter's mass, power, size and cost will be estimated using statistical analysis. Afterwards, with this data, it will be possible to establish a preliminary mission profile, to set a trajectory from the Earth to Mercury which will be followed by the orbiter. This chapter, although consisting of preliminary estimations, takes into account different launcher and trajectory possibilities, selecting those with better outlook and more advantageous.

D1.3.1. Spacecraft Vehicle-level Estimation

A first vehicle level estimation of the spacecraft's dry mass, power, size, and cost has been done using two methods of parametric estimation, estimation using regression analysis and relative standard error and estimation based on arithmetic mean and standard deviation. The estimations for total power will be presented and the previously mentioned estimations can be found in appendix B. Additionally, the parametric relationships from the ADSEE manual [1] have been used to compare estimations and retrospectively compare the previous missions in order to select the most reasonable estimations analogously since the calculated estimations are unreasonable and have exceptionally high ranges. This is due to the low amount of missions to Mercury. Reliability wasn't able to be estimated due to lack of available information during research.

Estimation of Total Power using Regression Analysis and Relative Standard Error

A parametric relationship between the total power and payload mass is constructed by computing a trendline in which the sum of the squares of the residuals is the least. The trendline equation is then used to estimate the total power of the spacecraft. The range of the estimation is calculated using the relative standard error (RSE).

The total power of each researched mission has been plotted against their payload mass. Their trendline and R^2 value were computed through Microsoft excel.

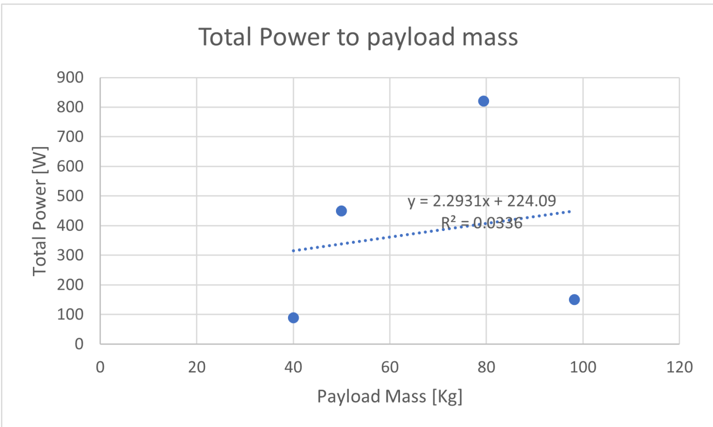

Figure 3.1: *Total Power to Payload Power Relationship and R^2 Value*

The estimated value for total power using the trendline and a payload mass of 150 kg is:

$$P_{total} = 2.2931 \cdot M_{payload} + 224.09 \tag{3.1}$$
$$P_{total} = 568.055\text{W}$$

It is worth to note the exceptionally low R^2 value as it suggests the estimated value for total power is unreliable due to the large residuals. This is further seen in the calculation of the range using the relative standard error which has been calculated as such, where n, m, y_i, $f(x_i)$ are the number of data points, the number of parameters estimated, the real value, and the estimated value according to the trendline respectively. Referring back to table 2.2, the total power and estimated total power are substituted in the sum and equation (the numbers have been rounded for the reader's ease):

$$RSE = \sqrt{\frac{1}{n-m} \cdot \sum_{i=1}^{n} \left(\frac{y_i}{f(x_i)} - 1 \right)^2} \tag{3.2}$$

Applying the formula, the RSE is 1.04. Using the RSE, the range of the possible values for total power are found in which 68%, 95%, and 99.7% of all possible values are included. The lower limit of the 68% range is found by subtracting the estimated value multiplied with the RSE and the upper limit is found by adding the estimated value multiplied with the RSE. For 95% and 99.7%, they are subtracted/added twice and thrice respectively. The range within 68% is calculated as an example, where L_l and L_u are the lower limit and upper limit respectively. The ranges are shown below:

Lower limit and upper limit:

$$P_t = 568.055 - 1.04 \cdot 568.055 = -21.3\text{W} < 0, L_l = 0\text{W}, P_t = 568.055 + 1.04 \cdot 568.055 = 1157.4\text{W}, L_u = 1157.4\text{W}$$

Table 3.1: *Range of total power within 68%, 95%, and 99.7% of possible values calculated through regression analysis*

% of possible values	**Range** [W]
68%	0-1157.4
95%	0-1746.7
99.7%	0-2336.0

Due to the high RSE, the range does not allow for a reliable value for total power. As such, estimation based on arithmetic mean is conducted in search for a smaller range and reliable value for total power, even though it is recommended to have at least 10 data points as pointed in the ADSEE manual [1].

Estimation based on Arithmetic Mean and Standard Deviation

A parametric relationship is constructed

A parametric relationship is established by finding the mean payload mass to total power ratio of the researched missions. This ratio allows to calculate the spacecraft's total power within a range of possible values.

The mean payload mass to total power ratio is calculated as such, with data referring to table 2.2:

$$\frac{\frac{40}{90} + \frac{98.2}{150} + \frac{79.4}{820} + \frac{50}{450}}{4} = 0.33\text{kg} \cdot \text{W}^{-1}$$

The ranges of possible values are found in similar fashion as previously with the linear regression estimation. The standard deviation (SSD) is subtracted/added to find the lower and upper limits of the ranges. Referring to the ADSEE manual [1], the SSD is calculated as such, where N, x_i, and $\bar{x}$ are the number of data points, the ratio of a data point, and the average ratio respectively:

$$SSD = \sqrt{\frac{1}{N-1} \sum_{i=1}^{N} (x_i - \bar{x})^2} \tag{3.3}$$

Applying this formula, the SSD is 0.27 and the range of the power ratio is:

Table 3.2: *Range of payload to total power ratio within 68%, 95%, and 99.7% of possible values*

% of Possible Values	**Range** [kgW^{-1}]
68%	0.06-0.6
95%	0-0.87
99.7%	0-1.14

Multiplying the inverse of the power ratio with the payload mass for each range gives an estimation and the range of possible values for the total power of the spacecraft. An example is shown below:

Lower limit and upper limit

$$P_t = \frac{1}{0.6} \cdot 150 = 250.8\text{W}, L_l = 250.8\text{W}, P_t = \frac{1}{0.06} \cdot 150 = 2702.5\text{W}, L_u = 2702.5\text{W}$$

Table 3.3: *Range of total power within 68%, 95%, and 99.7% of possible values calculated based on arithmetic mean*

% of possible values	**Range** [W]
68%	250.8-2702.5
95%	172.6-∞
99.7%	131.5-∞

The ranges show that the estimation is highly unreliable, comparable to a shot in the dark. The results suggest that an analogical approach would be much more appropriate in this situation.

Estimation based on Parametric Relationships in the ADSEE Manual

Due to the high ranges given by these estimations and unreasonable results when comparing with the previously researched Mercury missions, parametric relationships from the ADSEE manual [1] were used to compare estimations and choose which were more likely.

It is important to note that the ADSEE manual [1] presents relationships obtained with missions that mostly differ from the large scale travelling and maneuvering that takes case in Mercury missions, causing disparity between the actual values.

The relationship used is constructed with mainly "Planetary and Moon orbiters, as well as fly-by spacecraft" [1]:

$$P_t = 2.5 \cdot P_{payload} \tag{3.4}$$

$$P_t = 2.5 \cdot 50 = 125\text{W}$$

This result clearly shows the disparity created by the use of non-similar missions and spacecraft. As such, an analogous power estimation is more appropriate than any of the estimation methods used due to the lack of missions towards Mercury and similar missions/spacecraft.

Estimations of the rest of the parameters which have given reasonable results and align analogously with the researched missions/spacecraft, are listed in the table below:

Table 3.4: *First Vehicle Level Estimations of the Spacecraft*

Parameter	**Estimation**	**Method**
Dry mass [kg]	731.55	ADSEE
Wet mass [kg]	1553.4	ADSEE
Power [W]	125	ADSEE
Size [m^3]	6.99	ADSEE
Cost (Millions US$)	155.17	ADSEE

Even though the power estimated by the ADSEE manual [1] is unreliable, the parametric estimations have exceedingly high ranges and outside of the spacecrafts given constraint of 200W. Similar reasoning is done for the rest of the estimations and as specified in the introduction of section 3.

D1.3.2. Complete Mission Profile

The process of designing a complete mission profile has to take into account many of the results discussed before (dry mass, power budget, celestial bodies, design requirements,...) to find the most suitable trajectory and time frame for the mission. In this process, several possible trajectories were discussed, arriving at the conclusion that although the use of several gravity assists would be ideal for the propellant budget, it would lead to unreasonable timelines (six or seven years of interplanetary travel). It was finally decided that a Hohmann transfer approach using electrical propulsion would be nominal, as the immense power for the delta-V maneuvers could be harvested from the suns radiative power. This mission philosophy ultimately brings immediacy at the expense of weight.

In regards to the launcher model choice contemplating past missions, every probe of those chosen for the statistical study happened to ride different launchers; Mariner 10 was launched in an Atlas-Centaur rocket, MESSENGER was carried on top of a Delta II 7925-9.5 launcher, while BepiColombo was transported to orbit by an Ariane 5 ECA launcher. When taking the launch locations of the probe into account, a clear similarity can be seen: Both MESSENGER and Mariner 10 were launched at Cape Canaveral, whilst BepiColombo's launch location was Europe's Spaceport (French Guiana). Both these locations are on the east coast, making spaceports west of the sea an early preference. Notice that all had gravity-assist fly-byes, which made the use of less powerful rockets feasible, like in the case of Mariner 10, where a more expensive Titan IIIC was proposed at first.

Mariner 10's proposed launchers are, apart from short in capabilities, both retired, so they wont be further discussed. MESSENGER's Delta II is also retired. Bepicolombo's Ariane 5 ECA is not is not only active, but also the only proposed launcher from the EU, which makes this option even more attractive, in terms of ease of communication and bureaucratic matters. With this in mind, and considering there are European launchers perfectly capable of doing the mission, it is a waste of time to consider further international options.

Other active European launchers to be considered are the VEGA family and the new Ariane 6. Both of them pose a more attractive economical alternative to Ariane 5. Unfortunately, no VEGA configuration has the capability to launch the rocket with the current launch mass. Ariane 6 is a promising replacement to its established older brother: Ariane 5, with its first launch scheduled for 2023, promises cutting launch costs in half whilst surpassing the capabilities of its ageing predecessor. This model suits best all the requirements and thus will be the one used for calculations. This launcher (Ariane 6 A64) can deliver 11,500 kg to GTO at a 115 million euro price.

The launch will take place at the French Guyana (6°inclination) and although it offers a final inclination of 0°, this option will not be necessary, as the targeted orbital inclination is 4.7°. After establishing its self in a GTO parking orbit, a bootleg maneuver will be done to get to that inclination (all relevant Delta-V calculations are assesed in Appendix A.2). After that, the kick-stage, together with the electric propulsion system, will be used to do the Hohmann maneuver. After cruising for 102 days (compared to the nearly 7 years a fly-by approach would have taken), the orbiter will establish its self in an orbit around Mercury, where again using EPS, it will perform a bootleg maneuver to get into a polar orbit. After that, the nominal mission can take place. At EOL, the remaining fuel in the EPS is used to de-orbit the satellite, leaving essentially no space debris.

Table 3.5: *Mission Calendar*

Date	Mission event
13 July 2025	Lift-off.
15 July 2025	System check and payload calibration, bootleg maneuver.
17 July 2025	Kickstage burn for Hohmann maneuver.
September 2025(*)	Deceleration maneuver starts.
27 October 2025	Arrival at Mercury.
28 October 2025	System check and recalibration.
29 October 2025	Bootleg maneuver burn starts.
1 November 2025(*)	Entry in desired orbit.
2 November 2025 (*)	Start of nominal mission.
November 2025(*)	End of nominal mission.
November 2026 (*)	End of extended mission.(**).
Nov/Dec 2026(*)	De-orbit burn.(**).
(*) Dependant on EPS performance.	(* *) Possibility of further extension.

Table 3.6: *Launcher Requirements*

Requirements	
Target orbit	The launcher should be able to take the orbiter to a parking GTO
Payload mass	The launcher should lift at least 10000 Kg of payload. (extra heavy)
Payload volume	1.6 x 1.6 x 2.3 m + 5 x 5 x 3.6 m (folded s/c + kickstage)
Success rate	The launcher shall have at least 90% of success deployments in orbit.
Cost	The launcher cost shall not surpass 1/3 of the price of the mission. (200 million euros)
Launch loads	The launcher must have a maximum longitudinal acceleration of 6g, and a maximum lateral acceleration of 2g to ensure structural safety.
Vibrations	Sinusoidal vibrations shall not exceed 100 Hz during the launch.
Operational conditions	The launcher should be able to launch during moderate/ bad weather conditions. (*)
Period frame	The launcher should be ready to operate by 2025
Preferential characterisitcs	
European launchers.	Due to ease of communication and bureaucratic matters, European launchers will have preferential consideration.
Small final inclination	Due to a need to get to a 4.7 degree inclination plane, launchers that can provide small inclination orbits will be favoured.

(*) Due to lack of alternative launch window.

D1.3.3. Spacecraft Trajectory at All Mission Phases

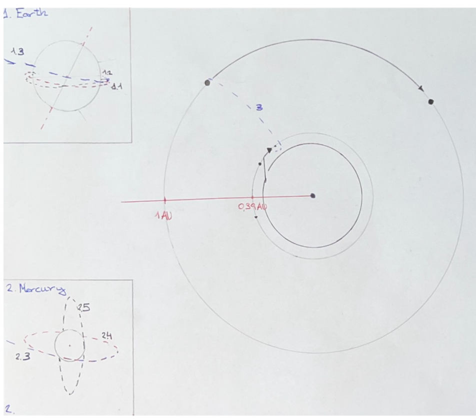

Figure 3.2: *Graphical Sketch of s/c Trajectory*

The Figure 3.2 shows the entire trajectory of the spacecraft during the mission, having close-ups for the Earth (1) and Mercury (2). The trajectory is divided into five parts:

- 1.1.- Launch phase from French Guyana and establishment of parking orbit in GTO.
- 1.2.- Bootleg maneuver to get to a 4.7 degree inclination
- 1.3/2.3/3.- Hohman maneuver in Earth/Mercury/interplanetary frame.
- 2.4.- Stable arrival orbit around Mercury.
- 2.5.- Bootleg maneuver to target polar orbit.

Notice that due to the length of the voyage, Mercury will have completed one full rotation, whilst Earth will have completed nearly a third.

D1.3.4. Orbital Parameters

The goal of the mission is to place the spacecraft in a circular orbit around the planet Mercury. This orbit is to be a polar orbit, enabling the camera to capture a detailed map of the whole surface and the altimeter to measure the elevation changes accurately. The orbits plane of reference is Mercury's equatorial plane from which the inclination of 90 degrees is measured. The period (T) was calculated using Equation 4.2 and the eclipse time (t_e) using the method described thoroughly in Figure 4.1 while the orbital velocity calculation can be found in Appendix A.

Table 3.7: *Orbital Parameters of the Final Orbit*

Orbital Parameters	Values
Semi major axis (a)	750 km
Eccentricity (e)	0
Inclination (i)	90°
Period	7644.5 s
Eclipse time	2115.5 s
Velocity	2621.7 $\frac{m}{s}$

D1.3.5. Total Required ΔV Estimations

As has been explained in a previous section, the Earth-Mercury trajectory followed by the spacecraft includes several maneuvers for which the Delta V of 16900 m/s has been estimated. This Delta V is high compared with other missions, but thanks to high payload capabilities from the launcher (Ariane 6) and high-I_{sp} of electric propulsion it can be achieved. The total velocity increment has been estimated using calculations for a Hohmann transfer orbit from Earth to Mercury, including the maneuvers for orbit inclination change. Table 3.8 below shows the Delta V breakdown. Calculations for part of this preliminary estimations has been added in Appendix A

Table 3.8: *Delta V of several Mercury Missions*

Mission	Delta V [m/s]
GTO to GEO	1520
GEO to Hohmann entry	5615
Mercury Hohmann exit	7682
Mercury orbit inclination change	3710
Station keeping	100
Total	18627

4

Initial Sizing

D1.4.1. Preliminary Mass and Power Budgets

To get a rough view of what the spacecraft should look like, an initial sizing of the spacecraft is of utmost importance to progress the design: between which limits should the possible volume be with the given requirements? What is the area necessary for the solar arrays to provide for all forms of power consumption of the spacecraft? What is the expected mass of the spacecraft? In this chapter the initial sizing of the spacecraft and its subparts will be determined.

Mass Budget

To generate the mass budget for our mission's spacecraft a statistical approach was used. This method was based on the method used on page 69 of the ADSEE manual [1]. The method is pretty straightforward. Gather information on mass distribution on similar spacecraft with a similar mission and then use those values to provide a rough guideline for how much mass should be used per subsystem. This was done in excel as it's more efficient and easier when there are lots of small calculations to do.

To start off, data for an average small satellite was taken from the ADSEE manual [1]. This data can be seen in the second column in table 4.1. The column shows the percentage of the total dry mass that each subsystem takes up on the spacecraft.

Table 4.1: *Mass Budget based on Statistical Data on Small Satellites*

Mass Budget	Average satelite(%)	Normalized Data	Our s/c[KG]	SSD	-2SSD	+2SSD
Payload	23.50	0.235	171.9	7.35	157.21	186.61
Thermal	3.20	0.032	23.4	2.24	18.93	27.89
Communications	7.20	0.072	52.7	4.41	43.85	61.49
Harness	5.60	0.056	41.0	2.05	36.87	45.07
C&DH	7.50	0.075	54.9	5.02	44.83	64.91
AOCS	7.90	0.079	57.8	4.65	48.49	67.09
Structure	24.20	0.242	177.0	6.18	164.68	189.40
EPS	20.90	0.209	152.9	9.16	134.57	171.21
Dry Mass			**731.55**		**649.43**	**813.67**

This data was normalized to give us the fraction of the dry mass that should be used per subsystem. Then by multiplying it by the dry mass found in section 3 we get the mass for each subsystem. These are just estimations and guidelines and are not set in stone yet. The payload mass for example should be 150$[Kg]$ but on table 4.1 its said to be 170$[Kg]$.

The design margins were then calculated by calculating the standard deviation of all the subsystems of the satellite data collected from Table 3 in the ADSEE reader Appendix D [1]. The sample standard deviation (SSD) was calculated using equation 3.3. The SSD was written in the fifth column for each subsystem. Since the design process is still in its earliest stages, a decision was made to take into account a margin of two standard deviations. This is acceptable as the design margins tend to decrease with increase design maturity.

This data is based on a lot of Earth-Orbiting satellites which is related to our mission, but not the most accurate depiction since Mercury is half the distance to the Sun as Earth is. For this reason, we also performed the same procedure but with a Mercury orbiter, BepiColombo's MPO, as can be seen from table 4.2.

Table 4.2: *Mass Budget based on BepiColombo Data [2]*

Mass Budget	MPO[KG] [2]	Normalized MPO [-]	Our spacecraft [KG]
Payload	98.20	0.09	62.65
Thermal	168.90	0.15	107.76
Communications	154.50	0.13	98.57
Propulsion	79.40	0.07	50.66
Harness	87.90	0.08	56.08
Command & Data Handling	40.10	0.03	25.58
Attitude & Orbit Control	63.50	0.06	40.51
Structure	312.40	0.27	199.32
EPS	141.70	0.12	90.41
Dry Mass	**1146.60**		**731.55**

This time, the first column shows the mass of each subsystem found in MPO with the last row being the total dry mass. We normalized these results in the next column by dividing the rows by the total dry mass of MPO. Following the same process that was used in the previous table, the mass for each subsystem was estimated. Since this is based on only MPO's data, the SSD was not able to be calculated so we used a margin value of 25%. This is the same value that was used by Messenger during early stages [18]. This gives us a range of dry mass between $548[Kg]$ and $913[Kg]$.

The range of using the Zandbergen data for small satellites [1] is smaller and more compact than using MPO's data. However this doesn't mean it's more accurate as the data used by Zandbergen [1] is based on Earth orbiters which are almost twice as far from the sun as mercury orbiters. This means that the distribution of weight on say, the thermal subsystem, will be far larger on a Mercury orbiter.

Power Budget

To generate the power budget of our spacecraft, two previous missions had to be looked. After spending some time on trying to find data for a lot of satellites, only the MPO and Ulysees missions provided the necessary data on power budget across each subsystem to be able to make an estimation of how our power should be distributed. This is given in the same format as the mass budget and can be found in table 4.3

Table 4.3: *Power Budget for MPO and Ulysees*

Power Budget	MPO			ULYSEES		
Subsystems	**MPO[W]** [2]	**Normalized**	**Our S/C [W]**	**Ulysees [W]** [19]	**Normalized**	**Our S/C [W]**
Payload	194.3	0.21	41.6	56.4	0.28	55.92
Thermal	177.0	0.19	37.9	41.6	0.21	41.25
COMMs	240.1	0.26	51.3	78.2	0.39	77.54
C&DH	133.1	0.14	28.5	17.3	0.09	17.15
AOCS	190.7	0.20	40.8	8.2	0.04	8.13
Total power	**935.2**		**200**	**201.7**		**200**

As done in the mass budget section, a margin of 25% will be used in each of the subsystems. This totals to a range of total power to be between 150 and 250 [W]. The mission report also states that the maximum payload power we use is 50[W], which roughly aligns with the data we calculated from both MPO and Ulysees in table 4.3. This is the power without taking into account the electrical propulsion system (EPS) that has to be used to get into Mercury orbit. This power budget table starts as soon as we are in orbit and our instruments have started to work.

Table 4.4: *Spacecraft Payload Power Consumption*

Instrument	Daytime Power[W]	Nighttime Power [W]
Laser Altimeter [20]	28.74	28.74
Spectrometer	4.11	-
High Resolution Camera [21]	4	-
Magnetometers [2x] [22]	1	1
Total power	37.85	30.24

In the Table 4.4 a breakdown of the payload power consumption at day- and nighttime can be examined. It was decided upon to use one spectrometer because it would be sufficient to meet the mission requirements, and two magnetometers to increase the accuracy of the measurements of Mercury's magnetic field. The laser altimeters power consumption was the highest of all instruments and is one of two instruments that will be used continuously throughout the science period, the other one being the magnetometer. For the cartography a camera similar to the one used in the Mangalyaan mission, which was especially developed for low power applications, was installed. In the upcoming calculations the absolute maximum payload power of 50 W will be used.

Solar Array

Thanks to being relatively close to the sun, Mercury counts with a very high irradiance from the sun of 9082.7 $W/m2$ as shown in Chapter 2, which even counting in efficiency factors still results in a high value. This means even with a small solar array a relatively high amount of power is obtained. It is important however, to bear in mind complications such as the solar eclipse, which results in the need for bigger arrays to store energy for the eclipse time.

Regarding the different sources of power consumption, there are three main groups: Payload, bus and propulsion. The payload and bus power consumption's are given as a top-level requirement, with the payload requiring 50 W and the bus requiring 150 W. Although for the third group, propulsion, the use of an electric method of propulsion has not yet been fully decided on at this stage, it will most likely be the chosen method. The expected consumption of the 3 groups is shown in Table 4.5. Due to the propulsion the solar arrays will be bigger than required just by the previous 2 groups, but since the engines will only be used at orbital daytime when possible, the requirements and calculations are simplified for battery sizing.

Table 4.5: *Power Consumption per Group*

Power group	Power consumption [W]	Active time per orbit (worst case)
Payload	50	All times - 7631 s
Bus	150	All times - 7631 s
Propulsion	5000	1200 s

To estimate the size of the solar array the following formula was used.

$$P_{SA} \cdot t_d = \frac{t_d \cdot P_d + t_e \cdot P_p}{\eta_d} + \frac{P_e t_e}{\eta_e} \text{[23]} \tag{4.1}$$

P_{SA} stands for the total power delivered by the solar array while P_d and P_e stand for the power required during the daytime and eclipse-time respectively (excluding propulsion); P_p stands for the power required by propulsion. η_d and η_e are the preliminary day/night-time efficiencies with values of 0.8 and 0.6, taken from the reader for the course AE1222-II [23].

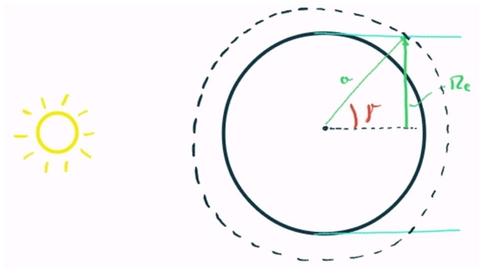

Figure 4.1: *Schematic for Eclipse Time Calculation*

After calculating the angle γ, with a being the radius of the orbit and R_e the radius of mercury, using simple trigonometry and multiplying by two an orbit night-time angle of 49.9 deg can be found. Then, using Equation 4.2

$$T = 2\pi\sqrt{\frac{a^3}{GM_{mercury}}} \tag{4.2}$$

for the orbital period a time $t_d = 5515.01s$ and $t_e = 2115.51s$ can be found. Plugging the values into Equation 4.1 this results in a total of 1737.78 W that the solar array has to deliver. Using specific power data found in table 41 of the reader for AE-1222-II [23], we arrive at a surface of 1.16 m^2 of required solar arrays. However, since electric propulsion has to be used for the injection maneuver exiting Earth a much bigger solar panel area will be required, the calculations are shown below. A power requirement of 5 kW at 1 AU is estimated for these calculations.

$$A_p = 4.45[m^2/kW] \quad P_{esp} = 58.45[W/kg] \quad P_r = 5[kW]$$

$$A_{array} = P_r \cdot A_p = 22.25m^2 \quad m_{array} = \frac{P_r}{P_{esp}} = 85.47kg$$

Although the resulting 22.25 m^2 may seem excessive, it's important to bear in mind the fact that solar irradiance at 1 AU (Earth), is 6.674 times lower than it is in Mercury, and also the fact that during orbit the propulsion system won't be as power hungry as it is in the start of the Hohmann trajectory. This higher requirement at the start has the disadvantage of carrying extra solar arrays for the entire mission which will only be used at the start, however, further down the design line a section of those solar arrays could be designed to be detachable. In any case, the excess surface area on arrival to Mercury means no degradation margin is needed.

D.1.4.2. Preliminary Spacecraft Architectures and Sketches

In this section the different architectures considered for the orbiter will be presented together with explanatory sketches. The main rationale followed when designing these satellite architectures was to minimise the satellites MMOI (mass moment of inertia), while ensuring each component was placed in such a way as to ensure it could perform its functions. Another leading design concern was minimising overall size in launch configuration so the satellite would fit in the launcher's fairing.

As for the dimensions of the satellite, several needs were taken into consideration: The payload, with an overall size of 1x1x0.5 m had to fit in the bus; and the solar array size was determined in section 1.4.1., with a total surface area of 22.25 m^2. The estimated volume for the spacecraft, of 6.99 m^3 was considered too large, taking into account the use of electric propulsion and the reduced volume of the payload. So the final decided volume was of 3.375 m^3, corresponding to a 1.5x1.5x1.5 m cube; this excludes the spacecraft's antenna and solar arrays. Once this volume was decided three main possible architectures were devised: cubic bi-array, cubic tri-array and cylindrical. Below, the three architectures are discussed, with advantages and disadvantages, leading to the chosen architecture, cubic bi-array.

Cubic Bi-array

In this configuration, which can be seen in Figure 4.2, the spacecraft's bus takes a cubic shape, of dimensions 1.5x1.5x1.5 m. As for the payload, it's placed on one of the cube's sides, the one which will point towards Mercury. Then on opposite sides foldable solar arrays, measuring approximately 8 m each. On the side opposite to the payload there's a heat shield, in order to avoid the excessive heating of the bus. On the two remaining sides the antenna and ion engine are placed. The batteries are placed next to each solar array, in order to minimise wiring and providing possibility of one battery failure without requiring mission abort. It's important to note Figure 4.2 is not drawn to scale, since its purpose is just to illustrate the placement of the main satellite's components. As for the pros of this design, it's got the advantage of being relatively simple and symmetrical, with minimal failure points and simple deploying - just the unfolding the two solar arrays. On the other hand, it's true the long solar arrays on each side do contribute to a relatively high MMOI for the satellite.

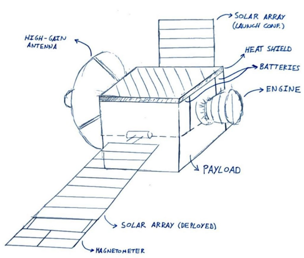

Figure 4.2: *Cubic Bi-array Configuration Sketch*

Cubic Tri-array

This configuration is very similar to the aforementioned, but as its name states, there's one main difference: one more solar array. This results in several changes with respect with the bi-array configuration. On the positives' side, the required solar panel surface area - 22.25 m^2 - can be divided in one more panel, resulting in shorter solar arrays- of approximately 5.5 m - which in turn result in a lower MMOI. This advantage, however, comes with several negatives: first, one more solar panel means one more failure point, but the second negative is more important; the side taken by the third solar array used to be taken by the antenna, which must now be housed in the same place as the heat shield. This new placement of the antenna results in necessarily more time pointing towards the sun, and thus higher levels of signal noise and heat radiation absorption.

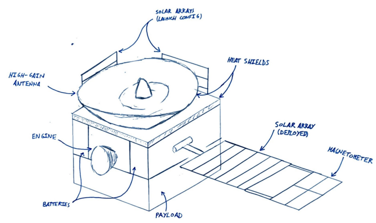

Figure 4.3: *Cubic Tri-array Configuration Sketch*

Cylindrical

This third configuration was devised as a different concept from the other two in order to obtain as low a MMOI as possible: the design philosophy was thus to place everything as close to the rotation axis as possible, this means also the solar panels. The big advantage of this is much easier rotation and maneuvering, but the compromises are too many for it to be feasible, at least for a Mercury orbiter. While the other 2 orbiters had reflective skin all throughout their bodies to avoid excessive heat absorption, this configuration makes such measures impossible, since most of the satellite's surface is covered in solar panel which by design collect the sun's energy, part of it in the form of heat. A radiator was added to the design to cool the spacecraft, but even then, thermal equilibrium as close to the Sun as Mercury's orbit requires of some sort of reflective skin. Another consideration which makes this configuration far from ideal for an electric propulsion thruster is the little surface area available for energy collection from the sun - taking the same volume as the other 2 designs it would result in an effective solar panel area of 3.6 m^2 a value far off the required 22.25 m^2.

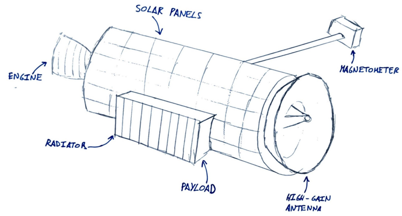

Figure 4.4: *Cylindrical Configuration Sketch*

D.1.4.3. Spacecraft Architecture CAD Drawings

As a visual concept is desirable with the initial sizings calculated and to progress on the with a more improved concept of the final spacecraft, a CAD drawing was made. While the drawing itself is visible in Appendix C, this subchapter will give information on some design choices made, as these need further explanation for full understanding. Another part that comes forward in this section is the assemblies themselves, which are mostly the same as their drawing counterparts, except that the assembly photos are shown more for their exterior design, that is; how would the satellite look from its initial computations, with materials and all?

As this is the first model of the satellite, and thus not close to its final form, some easier, but less precise methods have been used for the CATIA assembly: concerning all the subparts, these have been designed not quite as precise as necessary for the final stages of its design, as this is just an early visualisation. Batteries have been left out of the assembly, as these will be in the body of the orbiter and this body was taken as one whole part for this stage of the design, with the exception of the payload, which has been added as a cube in the bottom of the body. What the assembly does show, is some early sizings that might be key for the satellite to reach Mercury and stay in its orbit. These are the only dimensions that have been given in the drawing. Figures 4.5 and 4.6 show the satellite assembled in CATIA from two different perspectives, both unfolded. Figure 4.7 shows how the satellite is folded when it will be launched. Whilst materials have been added to every subparts, it has to be noted that this is merely for a good visualisation with realistic colors for every subsystem.

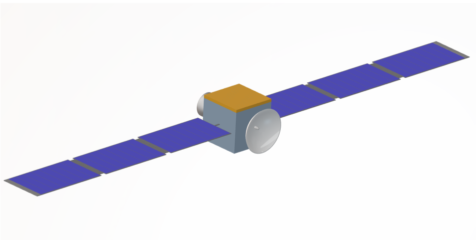

Figure 4.5: *An Isometric View of the Satellite*

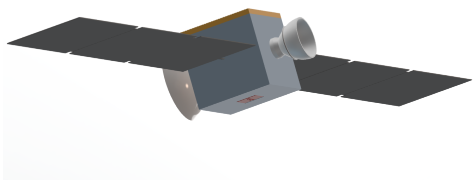

Figure 4.6: *The Satellite Viewed from the Back*

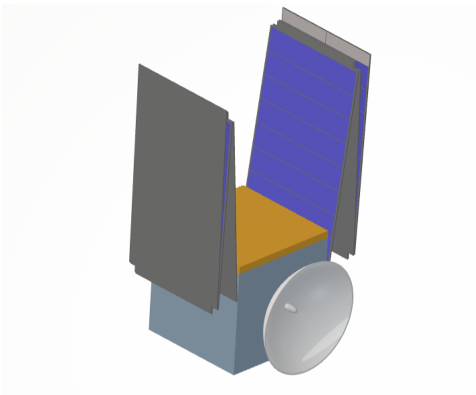

Figure 4.7: *A Folded View of the Satellite*

D.1.4.4. Mass Moments of Inertia of the Proposed Spacecraft Architectures

To evaluate the feasibility of each configuration, several of their aspects can be calculated and compared. One of these is the mass moment of inertia, which represents a bodies inertial resistance along a given axis to changing its rotational motion when acted upon by a moment. It can be calculated using the following equation:

$$I_{aa} = \int_M r_{aa}^2 dm \tag{4.3}$$

Where I_{aa} the mass moment of inertia around a given axis aa, M represents the total mass of the body, dm is the minuscule portion of mass from M and r_{aa} is the distance of each piece dm from the axis aa.

To use this equation to find the mass moment of inertia of a body, the axis aa must pass through the center of mass of that body. To find this the following vector equation can be used:

$$C\vec{O}M = \frac{\int_M \vec{r}\,dm}{M} \tag{4.4}$$

Where $C\vec{O}M$ is the center of mass and $\vec{r}$ is the position of each minuscule portion of mass dm

As evaluating integrals 4.4 and 4.3 over the entire structure of each spacecraft can be complicated, the equation can be simplified by dividing the body into multiple smaller sub-bodies and using a sum. This yields the following equations for the mass moment of inertia and center of mass respectively:

$$I_{aa} = \sum_{i=o}^{n}(m_i d_{iaa}^2 + I_{iaa}) \tag{4.5}$$

$$C\vec{O}M = \frac{\sum_{i=o}^{n}(\vec{r}_i \cdot m_i)}{\sum_{i=o}^{n}(m_i)} \tag{4.6}$$

Here the integer i can represent each of the sub-bodies, n represents the total amount of sub-bodies, m_i is the mass of sub-body i. For equation 4.6, r_i represents the position of the center of mass of body i. For equation 4.5 d_{iaa} is the distance of the center of mass of i from the axis aa and I_{iaa} is the mass moment of inertia of the body i around the axis iaa, which runs parallel to axis aa and through the center of mass of body i.

By dividing the spacecraft into multiple rectangular prisms and cylinders it allows for the following simplifications in the calculation of I_{iaa}

Mass moment of inertia for a rectangular prism:

$$I_{aa} = \frac{1}{12}M(w_S^2 + h_S^2) \tag{4.7}$$

Here the axis aa goes through the center of mass and is perpendicular to the surface S. w_S and h_S are the width and height of S respectively.

Mass moment of inertia for a cylinder through its central axis:

$$I_{aa} = \frac{1}{2}M(R^2) \tag{4.8}$$

Here the axis aa runs along the central axis of the cylinder and R is the radius of the cylinder

Mass moment of inertia perpendicular to central axis:

$$I_{aa} = \frac{1}{12}M(3R^2 + h^2) \tag{4.9}$$

Here the axis aa is perpendicular to the central axis and goes through the centre of mass of the object and h is the height of the cylinder.

Using equation 4.5, the mass moment of inertia around each axis for each spacecraft architecture can be calculated by dividing the structure into several cylinders and rectangular prisms. Depending on the orientation of the spacecraft while deployed relative to the mercury surface and the sun different values for the mass moment of inertia can be obtained, as hinges on the two cubic configurations will have different rotations. Thus to ensure a consistent comparison is made its is assumed that the solar panels are facing the same direction as the heat shield (see figures 4.2 and 4.3.

Cubic Bi-array Deployed

For this configuration the spacecraft has been approximated to a cubic bus flanked by two rectangular prism solar arrays (see figure 4.2), each with a uniform density. The cubic bus includes the mass of the antenna and measures $1.5m$ each side. It has a mass of $m = m_{total} - m_{solararray} = 1738.05 - 85.47 = 1652.58kg$. The two solar arrays each measure $1.5m$ in width, $7.5m$ in height and have been approximated to having a thickness of $0m$. They each weigh $42.735kg$.

Cubic Tri-array Deployed

For this configuration the spacecraft has been approximated to a cubic bus connected to three rectangular prism solar arrays (see figure 4.3), each with a uniform density. The mass and dimensions of the cubic bus are identical to the bi-array configuration. The three solar arrays each measure $1.5m$ in width, $5m$ in height and have been approximated to having a thickness of $0m$. They each weigh $28.49kg$.

Cylindrical

For this configuration, the spacecraft has been approximated to a cylinder with a diameter of $1.2m$ and a height of $3m$ (see figure 4.4). As all components are located directly on the cylindrical bus, its mass is $1738.05kg$, equal to the total estimated mass of the spacecraft, including the antenna and solar array. This mass is again assumed to be uniformly distributed. This approximation holds for both deployed and un-deployed states.

Cubic Un-deployed

For both the cubic tri-array and cubic bi-array architectures the un-deployed state can be approximated by a cube with sides of length $1.5m$. Similarly to the cylindrical architecture the mass is equal to the total mass of $1738.05kg$ and is uniformly distributed.

Results

The calculated mass moments of inertia are presented in table 4.6. The x axis here is the direction of travel for the spacecraft, corresponding to the direction of thrust. The z axis corresponds to the direction of the sun, being the direction that the solar panels and heat shield face for both cubic configurations, and the opposite direction of the radiator for the cylindrical configuration. The y axis is constructed perpendicular to the other two so that a correct axis system is formed.

Table 4.6: *Mass Moments of Inertia*

Configuration	$\mathbf{I}_{xx}$ [$\mathbf{kgm^2}$]	$\mathbf{I}_{yy}$ [$\mathbf{kgm^2}$]	$\mathbf{I}_{zz}$ [$\mathbf{kgm^2}$]
Cubic bi-array	2751.1256	635.74	2767.15125
Cubic tri-array	1345.619	931.736	1711.483257
Cylindrical	312.849	1459.962	1459.962
Cubic un-deployed	651.75	651.75	651.75

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Task Division

Note: all time spent is an indication.

Table 7: *Task Division*

Chapter/Task	Section	Student Name[s]	Time Spent
	Cover	Alfonso Rato	0.5 hours
		Emmanuel Rohrhirs	0.25 hours
	Preface	Jakub Płonka	1 hour
	Summary	Emmanuel Rohrhirs	4 hours
	Introduction	Alfonso Rato	1 hour
Chapter 1			
	Design Data	Emmanuel Rohrhirs	1 hour
		Miantao Zhaos	1 hour
		Alvaro Valderrabano	1 hour
		Alfonso Rato	1 hour
		Nicolas Rodriguez Dfouni	1 hour
		Razvan Herscovici	1 hour
	Mission Elements	Miantao Zhaos	4 hours
		Jakub Płonka	4 hours
		Alvaro Valderrabano	3 hours
Chapter 2			
	Planet characteristics	Jakub Płonka	0.5 hour
	Design Requirements	Miantao Zhaos	8 hours
		Jakub Płonka	8 hours
		Alfonso Rato	2 hours
		Razvan Herscovici	0.5 hours
	Design Data	Yuri ter Denge	2.5 hours
		Nicolas Rodriguez Dfouni	5 hours
Chapter 3			
	Vehicle Estimation	Razvan Herscovici	12 hours
	Complete Mission Profile	Alvaro Valderrabano	10 hours
		Alfonso Rato	6 hours
	Spacecraft Trajectory	Alvaro Valderrabano	2 hours
	Orbital Parameters	Emmanuel Rohrhirs	3 hours
Chapter 4			
	Mass & Power Budget	Emmanuel Rohrhirs	3 hours
		Nicolas Rodriguez Dfouni	10 hours
	Solar array	Emmanuel Rohrhirs	4 hours
		Alfonso Rato	2 hours
	Orbiter Architecture Sketches	Alfonso Rato	3 hours
	Design Architecture Explanations	Yuri ter Denge	1.5 hours
	CAD Design & Drawings	Yuri ter Denge	6 hours
	Mass Moment of Inertia	Miantao Zhaos	6 hours

Table 8: *Continuation of Task Division*

Chapter/Task	Section	Student Name[s]	Time Spent
Appendices			
	Appendix A	Alfonso Rato	3 hours
		Jakub Płonka	1 hour
		Alvaro Valderrabano	2 hours
		Emmanuel Rohrhirs	0.75 hours
	Appendix B	Razvan Herscovici	5 hours
	Appendix C	Yuri ter Denge	0.5 hours
Text & Layout Refinement			
	Chapter 1	Emmanuel Rohrhirs	0.5 hours
	Chapter 2	Emmanuel Rohrhirs	0.5 hours
	Chapter 3	Emmanuel Rohrhirs	0.75 hours
	Chapter 4	Nicolas Rodriguez Dfouni	1 hour
Task Division		Emmanuel Rohrhirs	0.5 hours

A

Appendix A

A.1. Experiments on previous Mercury missions

The individual experiments that are present on each spacecraft sent to Mercury are shown in Table A.1 below.

Table A.1: *Experiments on previous Mercury Missions*

Experiment	Mariner 10	Messenger	BepiColombo
Atmospheric and Surface Composition Spectrometer		✓	
Celestial Mechanics and Radio Science	✓	✓	
Dual Imaging System		✓	
Dust Monitor			✓
Energetic Particle and Plasma Spectrometer	✓		
Energetic Particles Experiment	✓		
Extreme Ultraviolet Spectrometer	✓		
Gamma-Ray and Neutron Spectrometer		✓	✓
Italian Spring Accelerometer			✓
Laser altimeter		✓	✓
Magnetic Field Investigation			✓
Magnetometer	✓	✓	✓
Plasma Particle			✓
Plasma Wave Instrument			✓
Probing of Hermean Exosphere by Ultraviolet Spectroscopy			✓
Radio Science		✓	✓
Radiometer and Thermal Imaging Spectrometer			✓
Scanning Electrostatic Analyzer and Electron Spectrometer	✓		
Search for Exosphere Refilling and Emitted Neutral Abundances			✓
Sodium Atmospheric Spectral			✓
Solar Intensity X-ray and particle Spectrometer			✓
Spectrometers and Imagers for MPO Integrated Observatory			✓
Television Photography Mariner	✓		
Two-Channel Infrared Radiometer	✓		
X-ray Spectrometer		✓	✓

A.2. Transfer Delta V calculations

GTO to GEO

The 1520 m/s ΔV_{orb} used for this first maneuver has been obtained from literature [24],and isn't a definite amount, since depending on the specific GTO orbit the velocity increase needed to reach GEO is different.

Hohmann Transfer Orbit ΔV

The calculations for the Hohmann transfer orbit were done taking the data in Table A.2 as a basis and were as follows:

Table A.2: *Data used for the Hohmann transfer calculations*

μ_{sun}	$1.327 \cdot 10^{11} km^3 s^{-2}$
μ_{earth}	$3.986 \cdot 10^5 km^3 s^{-2}$
$\mu_{mercury}$	$2.203 \cdot 10^4 km^3 s^{-2}$
r_{dep}	$1.496 \cdot 10^8 km$
r_{tar}	$5.791 \cdot 10^7 km$
r_0	42164 km
r_3	3189.7 km

First the departure and target heliocentric velocities V_{dep} and V_{tar} were calculated:

$$V_{dep} = \sqrt{\frac{\mu_{sun}}{r_{dep}}} = 29.78 km/s \qquad V_{tar} = \sqrt{\frac{\mu_{sun}}{r_{tar}}} = 47.87 km/s$$

Then the circular velocities around both the Earth and Mercury were calculated:

$$V_{c0} = \sqrt{\frac{\mu_{earth}}{r_0}} = 3.075 km/s \qquad V_{c3} = \sqrt{\frac{\mu_{mercury}}{r_3}} = 2.628 km/s$$

The semi-major axis of the transfer orbit was calculated in the following way:

$$a_{tr} = \frac{(r_{dep} + r_{tar})}{2} = 1.038 \cdot 10^8 km$$

Then, the orbit's heliocentric velocities at departure and target were calculated:

$$V_1 = \sqrt{\mu_{sun} \cdot \left(\frac{2}{r_{dep}} - \frac{1}{a_{tr}}\right)} = 22.26 km/s \qquad V_2 = \sqrt{\mu_{sun} \cdot \left(\frac{2}{r_{tar}} - \frac{1}{a_{tr}}\right)} = 57.49 km/s$$

With these last values the excess velocities could be found:

$$V_{\infty,1} = | V_1 - V_{dep} | = 7.52 km/s \qquad V_{\infty,2} = | V_2 - V_{tar} | = 9.62 km/s$$

The last values needed before obtaining the ΔV values were the velocity in the pericenter of the hyperbolas:

$$V_0 = \sqrt{\frac{2 \cdot \mu_{earth}}{r_0} + V_{\infty,1}^2} = 8.69 km/s \qquad V_3 = \sqrt{\frac{2 \cdot \mu_{mercury}}{r_3} + V_{\infty,2}^2} = 10.31 km/s$$

Then, finally, the ΔV values were obtained, ΔV_0 refers to the maneuver around the earth, while ΔV_3 refers to the maneuver around Mercury:

$$\Delta V_0 = | V_0 - V_{c0} | = 5.615 km/s \qquad \Delta V_3 = | V_3 - V_{c3} | = 7.682 km/s$$

$$\Delta V_{trans} = \Delta V_0 + \Delta V_3 = 13.297 km/s$$

Inclination Change and Total Velocity Change

After reaching the desired orbital altitude in Mercury the only maneuver for which to calculate the velocity change is the inclination change of 90º to a polar orbit, after this, the total ΔV can be found:

$$\Delta V_i = 2 \cdot V \cdot \sin\left(\frac{\Delta i}{2}\right) = 3.71 km/s \qquad \Delta V_{orb} + \Delta V_{trans} + \Delta V_i = 18.53 km/s$$

A.3. Propellant Mass Optimisation

In order to achieve a Delta V as large as the one required, 18.53 km/s, it is necessary to conduct an exercise of optimization to divide said Delta V between 2 stages: a first stage using liquid propellant and a second stage using a ionic Hall thruster. The chosen launcher, Ariane 6, will be able to launch to the chosen orbit, GTO, 11.5 tonnes of payload. This 11500 kg must be divided between the dry mass of the orbiter (731.55 kg), its propellant (M_P, the dry-mass of the kick-stage M_k, and the mass of the kick-stage's propellant M_{pk}. For preliminary design purposes we will assume the kick-stage mass is equal to 10% of the kick-stage propellant mass Mpk. With these two conditions in mind the following equations can be solved to find our unknown parameters:

$$731.55 + M_P + 1.1 \cdot M_{pk} = 11500 \tag{A.1}$$

$$9.81 \cdot 1500 \cdot \ln \frac{731.55 + M_P}{731.55} + 9.81 \cdot 400 \cdot \ln\left(\frac{11500}{731.55 + M_P + 0.1 \cdot M_{pk}}\right) \tag{A.2}$$

$$M_P = 1006.5kg \quad M_{pk} = 8874.5kg \quad M_k = 887.45kg$$

$$M_{wet,sat} = M_P + M_{dry} = 1006.5 + 731.55 = 1738.05kg$$

A.4. Fairing Configuration

In order to produce a fit launcher choice, the total volume of the payload is needed. For this reason, and, knowing already the s/c volume, this Appendix will dive into the specific kick-stage characteristics.

Kick-stage Size Estimation

The kick-stage mass breakdown, as specified in appendix A, is 887.45kg of dry mass, and 8874.45Kg of propellant. From we can see the mean density is 280 kilograms per cubic meter, the mean volume is, then:

$$V = \frac{8874.45}{280} = 31.69m^3$$

Now, knowing the volume of propellant we can estimate the height of the kick-stage, doing some assumptions: although the fairing has a diameter of 5.4 meters, a 0.2 meter clearance will be assumed due to launch vibrations. Thus, the height of the fuel tanks is:

$$V = \frac{31.69}{\pi \cdot (2.5)^2} = 1.62m$$

Now, assuming the rest of the kick-stage (nozzle, veins, orbiter adapter, bulkheads...) accounts for another 2 meters, the final measurements and the fitting of the whole kick-stage/orbiter in the fairing can be seen in Figure A.1 below.

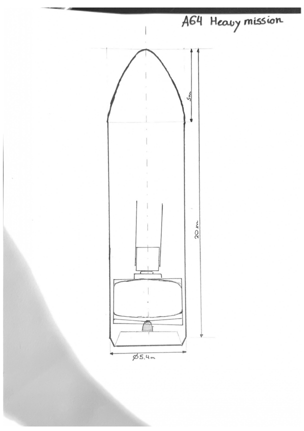


Figure A.1: *Fairing fit ilustration*

B

Appendix B

All the graphs and ranges of the estimations calculated through parametric estimations, including the estimations from the ADSEE manual [1].

B.1. Estimations calculated through parametric relationships

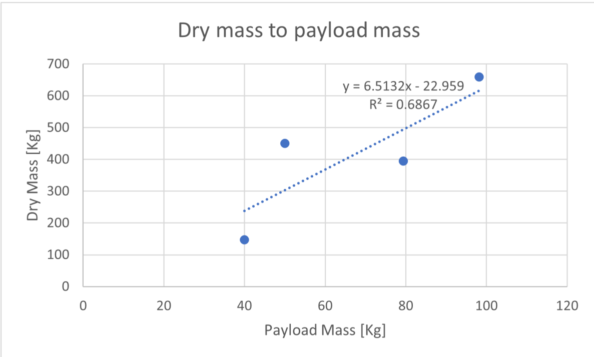


Figure B.1: *Dry mass to payload mass relationship and R^2 value*

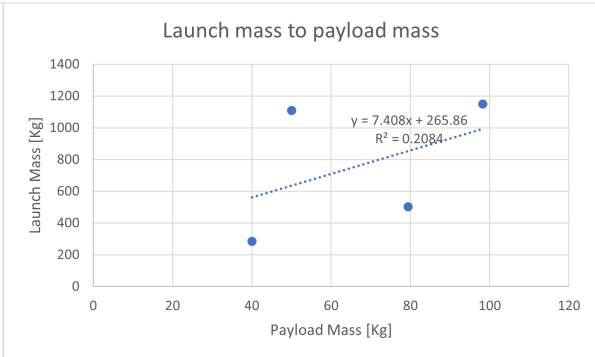


Figure B.2: *Launch mass to payload mass relationship and R^2 value*

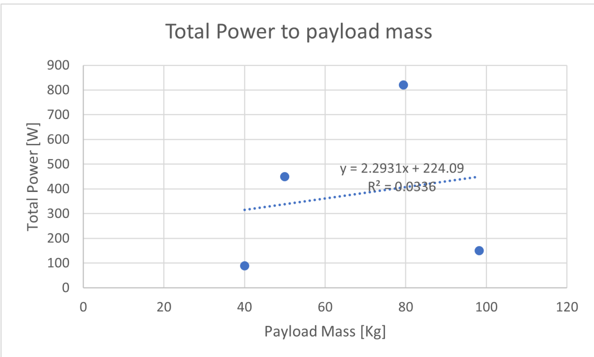


Figure B.3: *Total power to payload power relationship and R^2 value*

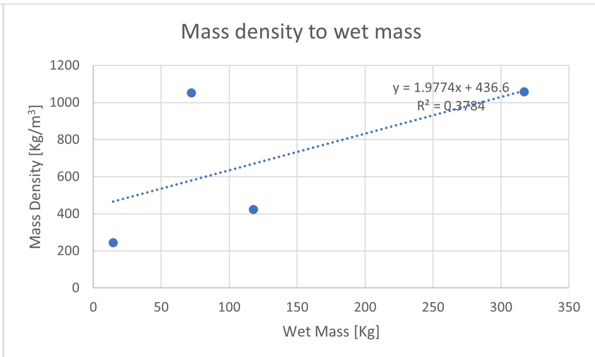


Figure B.4: *Mass density to payload mass relationship and R^2 value*

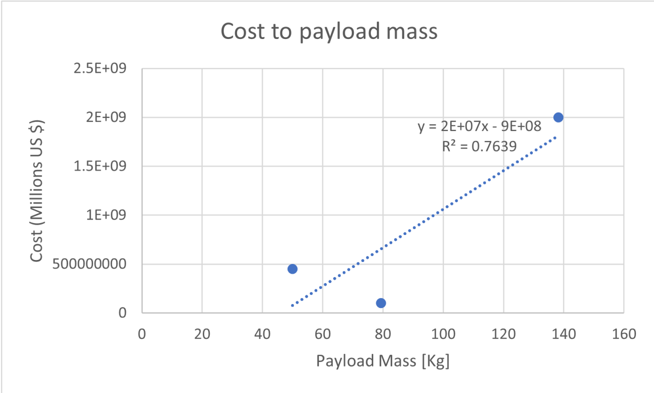


Figure B.5: *Cost to payload mass relationship and R^2 value*

Table B.1: *Dry mass and size estimations and ranges*

	Dry mass [kg]		Size [m^3]
% of possible values	Range (regression analysis)	Range (arithmetic mean)	Range (regression analysis)
68%	513.90-1394.10	594.60-1317.50	0-1.03
95%	73.85-1834.20	466.60-3360.10	0-1.61
99.7%	0-2274.30	infinity	0-2.20
Estimation	954.02	819.40	0.44

Table B.2: *Launch mass and cost estimations and ranges*

	Launch mass [kg]		Cost (US $)
% of possible values	Range (regression analysis)	Range (arithmetic mean)	Range (regression analysis)
68%	132.66-2621.46	944.49-2699.69	7,451,035,703
95%	0-3865.87	712.78-38122.64	12,802,071,407
99.7%	0-5110.27	infinity	18,153,107,110
Estimation	1377.06	1399.39	2,100,000,000

Table B.3: *Total power estimations and ranges*

	Total Power [W]	
% of possible values	Range (regression analysis)	Range (arithmetic mean)
68%	0-1157.37	250.83-2702.53
95%	0-1746.68	infinity
99.7%	0-2335.99	infinity
Estimation	568.06	459.05

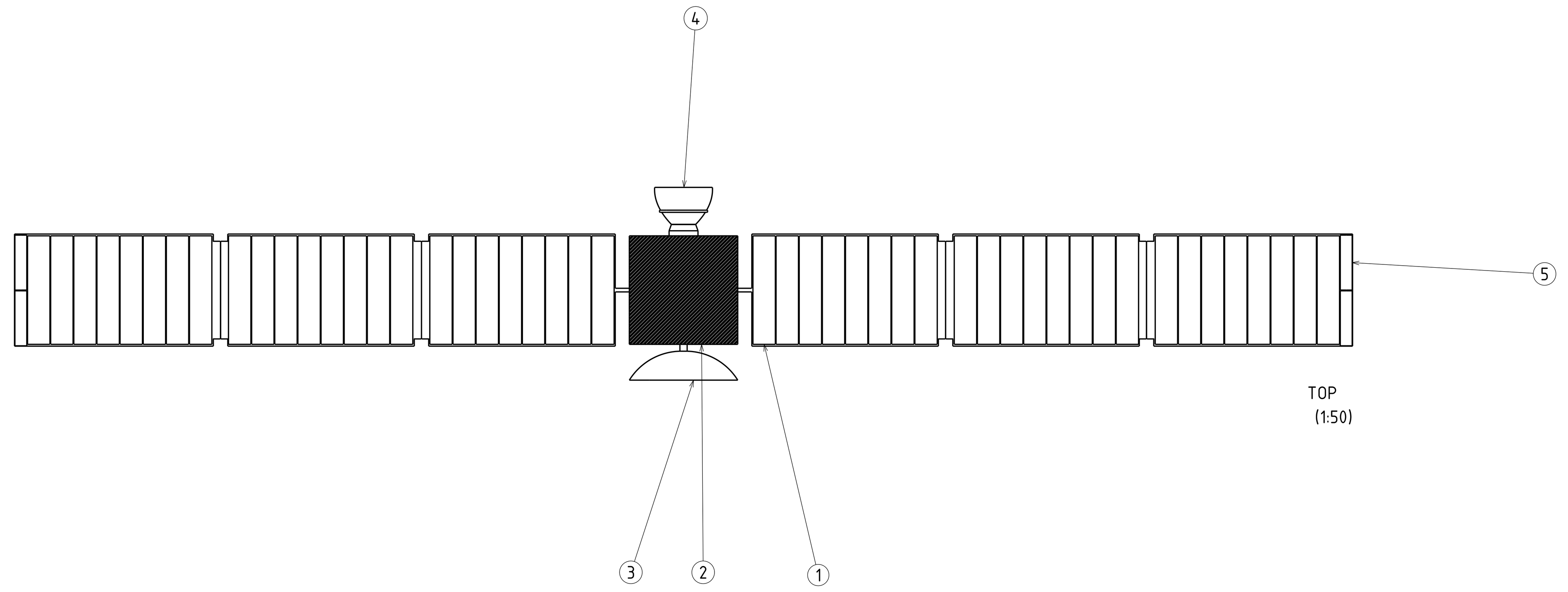
B.2. Estimations calculated through the ADSEE manual [1]

	Dry mass [kg]	Wet mass [kg]	Total power [W]	Size [m^3]	Cost (Millions US$)
Relationship used	$2.233 \cdot M_{pl} + 396.6$	$3.492 \cdot M_{payload} + 1029.6$	$2.5 \cdot P_{pl}$	$0.0045 \cdot M_{wet}$	$0.3531 \cdot M_{dry}^{0.839}$
Estimated value	731.55	1553.4	125	6.99	155.17

C

Appendix C

This appendix presents all the CATIA drawings that were necessary for further research and thus made for this report.



TOP
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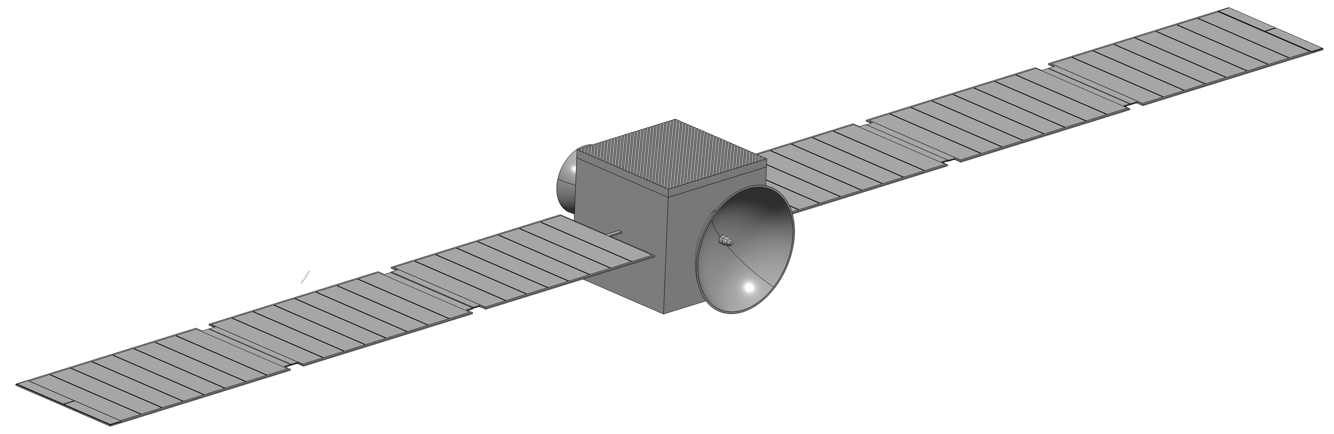

ISOMETRIC
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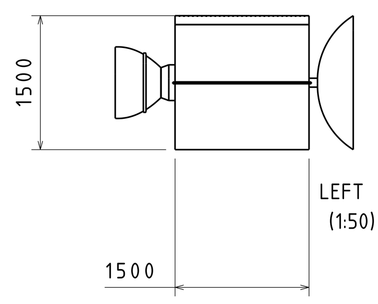

LEFT
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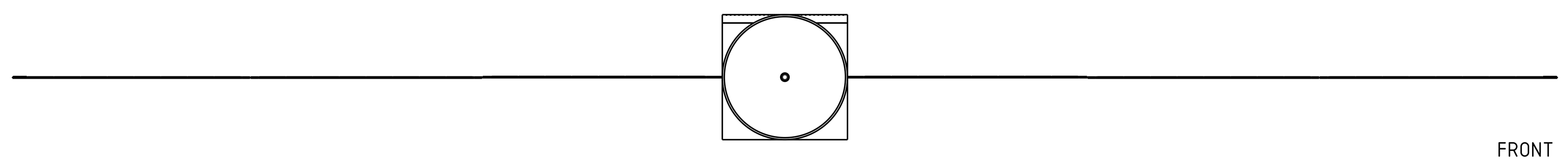

FRONT
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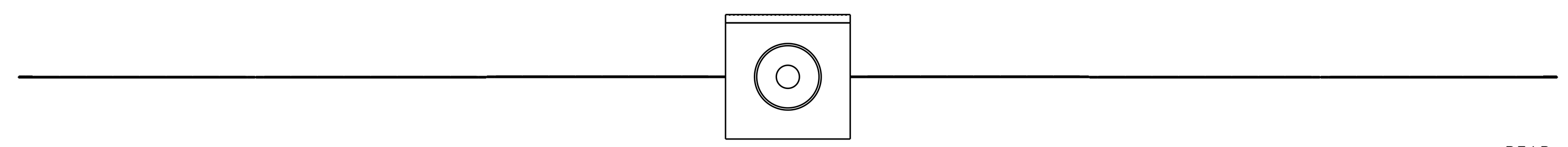

REAR
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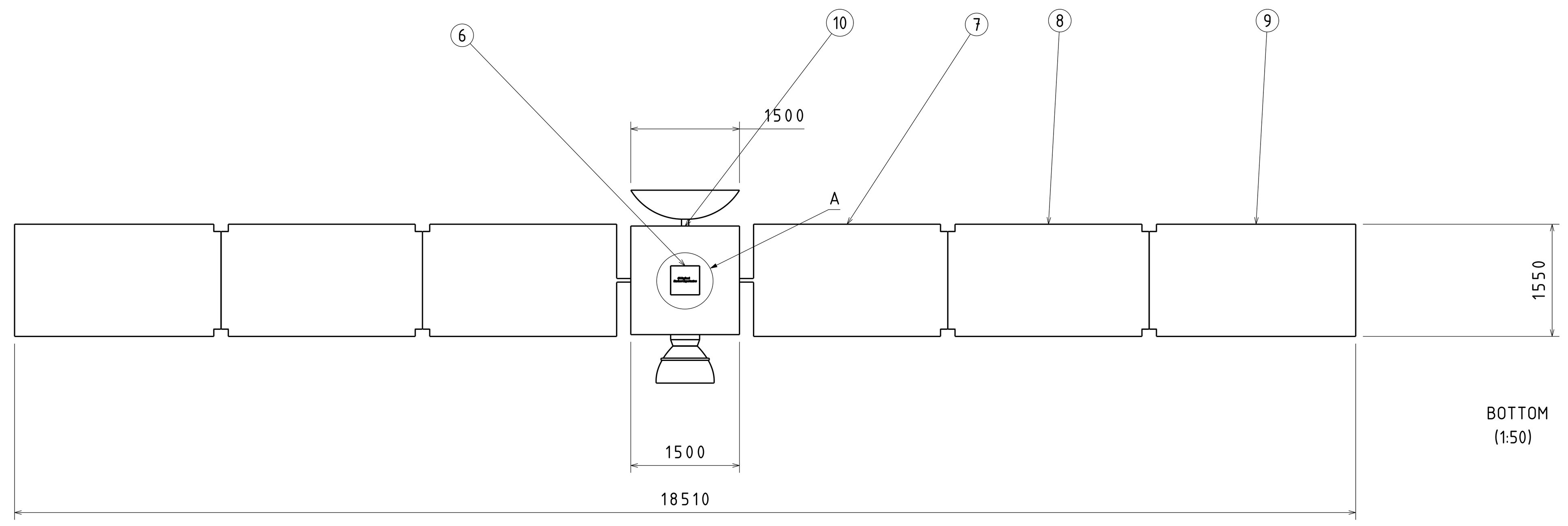

BOTTOM
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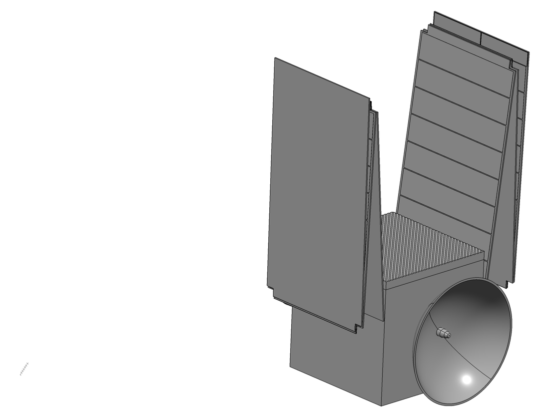

ISOMETRIC
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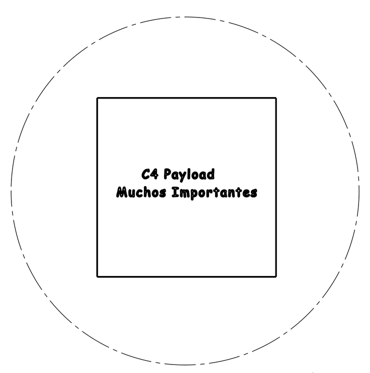

A
(1:10)

Item No.	Quantity	Title	Description
1	48	PanelPlates	Solar Panel Plates
2	1	HeatShield	Heat Shield
3	1	SatAntenna	Antenna
4	1	SatEngine	Engine
5	2	SatMagneto	Magnetometer
6	1	SatPL	Satellite Payload
7	2	OrbiterPanel	Solar Panel Back 1
8	2	OrbiterPanel2	Solar Panel Back 2
9	2	OrbiterPanel3	Solar Panel Back 3
10	1	OrbiterBody	Satellite Main Body

DRAWN BY Yuri ter Denge	DATE 16-9-2022	Description	Initial Design Concept C4 Mercury Satellite
CHECKED BY XXX	DATE XXX	SIZE A0	Title C4Sat, REV X
DESIGNED BY Yuri ter Denge	DATE XXX	SCALE 1:50	WEIGHT(kg) 1553,4 (wet, estimated), SHEET 1/1