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Spacecraft Assignment

Design of a Mars Orbiter Mission

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Group 45

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This report considers the preliminary design of the Electrical Power System (EPS) for the novel B15 Mars Mission. This will be broken down into a number of principal sections. First, section 1 presents the role distribution between the team members, and outlines the key information about the mission and the assignment itself. Following from this, section 2 provides more detail about the nature of the mission orbit, which in turn will be useful for the sizing of key EPS components. Based on the accumulated information, section 3 describes the key functions and requirements of the EPS, and the means by which these will be verified will also be detailed. section 5, section 6, section 7 and section 9 explains in detail the first design iteration of the EPS, including an explanation of the key EPS trade-off criteria, a power storage system selection, the description of the key EPS components and a mass budget for both the EPS and the spacecraft as a whole. Next, based on the new design requirements, section 10 provides the second iteration of the EPS design. Finally, section 11 reflects on the role distribution of the group and how this impacted the functioning of the group throughout the assignment.

1 Role Distribution and Key Information

This section gives a preliminary role distribution of the team members, as well as the key information about the mission and assignment itself. Table 1 presents the role distribution chosen by the team to perform the project.

Table 1: Role Distribution

Member	**Role**	**Role Description**
Oliver Ross	Team Leader	Manage the team, make sure that everyone has a task and summarize the state of the tasks at the beginning and end of the work sessions.
Lorenz Veithen	Discussion Moderator	Make sure that discussions are constructive and that everyone participates. Take note of the meeting outcomes.
Julie Paddeu & Niek Zandvliet	Research Responsible	Find the required sources and references online and in relevant textbooks, and document all the results such that they can be used by the other team members.
João Rodríguez & Oleksandr Krochak	Time Manager	Make sure that the time is used efficiently and the deadlines are met.

The mission being designed aims to map the entire Martian surface, including parts that have not been covered by past missions [1]. To do so, an orbiter in low Mars orbit (<800 km) is considered. This report considers the preliminary design of the Electrical and Power Subsystem (EPS) which shall provide a minimum of 800 Watts during the sunlight and 400 Watts during the eclipse. Key information for the following design steps is outlined in Table 2.

Table 2: Key information for the mission

Parameter	**Symbol**	**Value**	**Units**
Mars Radius	R_m	3389.5	[km]
Mars Mass	M_m	$6.39 \cdot 10^{23}$	[kg]
Distance Sun-Mars	$d_{S,M}$	1.52	[AU]
Power Required During Sunlight	P_d	800	[W]
Power Required During Eclipse	P_e	400	[W]
EPS Efficiency During Sunlight [2]	X_d	0.8	[-]
EPS Efficiency During Eclipse [2]	X_e	0.6	[-]
Orbital height	h	≤ 800	[km]
Maximum Inclination Angle	i	5	[deg]
Mission Duration	t	6	[years]

Furthermore, a set of assumptions during the design of the system, those are documented in Table 3.

Table 3: Assumptions made during the design

Assumption ID	Description
EPS-AS-01	The nominal (design) mission lifetime is estimated to be six years. This was chosen based on similar past missions [3, 4].
EPS-AS-02	The effects of moons on the eclipse time is neglected.
EPS-AS-03	The spacecraft bus runs on 28 V voltage, which is a standard in the aerospace.
EPS-AS-04	The orbit is equatorial, maximizing the eclipse time, which is conservative for the design of the EPS.

2 Description of Mission Orbit

In order to establish the key requirements for the EPS, first it is important to know more about the nature of the mission orbit, and how that impacts key parameters such as time in eclipse, time in sunlight and therefore, the power which needs to be generated and stored during these phases. These details will be covered here.

In most planetary exploration missions, the orbital altitude which is selected depends upon the required observation instruments to scan the planet in satisfactory sharpness. However, given that information regarding these instruments are unavailable, an altitude will be assumed. This is chosen to be 800 km, in compliance with the limit specified in the assignment. However, it is important to point out that for most EPS sizing equations, the radius of the orbit matters and therefore, the orbit altitude forms a very small part in comparison to the radius of Mars. Therefore, any selection of altitude less than 800 km should have little impact on the overall sizing.

Based on the selection of the orbital altitude, the orbital period of the spacecraft can be readily computed using Equation 1 under EPS-AS-04. Using this, combined with Figure 1, the eclipse period can be found as shown in Equation 2. The sunlight period then simply follows from subtracting T_e from, T_{orbit} as shown in Equation 3. For this mission in particular, the exact calculated values are shown in Table 4. These are key in order to determine the power which has to be generated by the solar arrays per orbit in order to accommodate for the requirements stipulated in the assignment. These will also be an important driver in the amount of energy storage which is provided during the eclipse period. These elements will be discussed in the definition of the requirements (see section 3) and when determining the solar array type (see section section 4).

$$T_{orbit} = 2\pi\sqrt{\frac{(R_m+h)^3}{GM_m}} \quad (1) \qquad T_e = \frac{2\arcsin\left(\frac{R_m}{R_m+h}\right)}{2\pi} \cdot T_{orbit} \quad (2) \qquad T_d = T_{orbit} - T_e \quad (3)$$

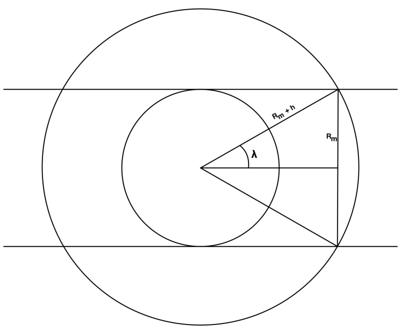

Figure 1: Eclipse Period around Mars

Table 4: Mission Profile Characteristics

T_{orbit}	8250.44	[s]
T_d	5775.18	[s]
T_e	2475.26	[s]

3 Preliminary Key Requirements

With the detailed orbit profile determined, the key requirements for the EPS may now be presented in the SMART format [5]. The expected top-level power subsystem functions are first presented, followed by the related top-level requirements of the system. The following functions were determined for the EPS based on guidelines given in [2].

- **EPS-FUNC-001:** The EPS autonomously provides continuous electrical power to the spacecraft and its payload throughout the entirety of the mission duration.

- **EPS-FUNC-002:** The EPS regulates and distributes the generated electrical power to the spacecraft and payload.
- **EPS-FUNC-003:** The EPS is compatible with the spacecraft bus and instruments.
- **EPS-FUNC-004:** The EPS provides energy storage capabilities for operations when the power generation units are unavailable.
- **EPS-FUNC-005:** The EPS protects the spacecraft payload in the event of an electrical failure.

Based upon the top-level functions identified above, the key requirements which drive the design of the EPS are readily derived and presented below. In addition, the validation (the requirement is VALID [6]) and verification of the requirements are also outlined. Note that it is assumed that the 800 W and 400 W requirements provided by the customer is already conservative.

- **EPS-REQ-0100:** The EPS shall provide an average power of at least 400 W at mission end-of-life during the eclipse period of the orbit.
  - This requirement was directly given by the project description.
  - This requirement can be verified through demonstration of the energy storage system for beginning of life operations. This can then be coupled to analysis to extrapolate the results to the expected end of life operations, based on knowledge from past missions.
- **EPS-REQ-0200:** The EPS shall provide an average power of at least 800 W at mission end-of-life during the direct sunlight period of the orbit.
  - This requirement was directly given by the project description.
  - Similarly to the EPS-REQ-0100, this requirement can be tested through by considering the solar panel power generation capabilities under Martian simulated environment in facilities such as the ESA SPACE POWER LAB [7]. The derived data can then be extrapolated through analysis to the end of life capabilities by considering degradation based on knowledge from past missions.
- **EPS-REQ-0300:** The EPS shall provide a peak power of at least 2000 W at mission end-of-life during the direct sunlight period of the orbit, for a maximum 10 minutes.
  - According to [2], the peak power can be roughly estimated as 2-3 times the average power at the preliminary design phase, a factor of 2.5 was chosen. Furthermore, 10 minutes is a conservative first estimate for the peak power duration.
  - Similarly to the previous two requirements, this can be verified by means of extensive testing using facilities such as the ESA SPACE POWER lab, in which the performance of the solar arrays can be measured under simulated sunlight. The extrapolation of the results to end-of-life conditions (in the same way as above) can be used to determine whether the requirement was satisfied.[7]
- **EPS-REQ-0400:** The EPS shall have an operational lifetime of at least 6 years.
  - This requirement was derived based on assumption EPS-AS-01, which was made earlier.
  - The verification of this requirement must be carefully done through analysis. Based on documentation of the individual components and information on their reliability throughout the operational lifetime, a risk assessment can be made on the probability of failure, and what measures can be taken in the event of a certain malfunction.
- **EPS-REQ-0500:** The power storage subsystem of the EPS shall supply the required energy to the spacecraft and its payload throughout at least 23,000 discharge cycles.
  - This requirement was derived from the orbital period and the assumed mission lifetime, knowing that only one eclipse happens per orbit.
  - This can be tested in facilities such as the ESA SPACE POWER LAB [7] where they propose "Long-term multi-year space battery testing including thousands of charge/discharge cycles".
- **EPS-REQ-0600:** The power storage subsystem shall store at least 1950 kJ of energy per one orbit at mission end-of-life.
  - This requirement was derived from the fact that the battery must have sufficient storage capacities to satisfy the power requirements for the duration of one eclipse (the exact value of which is provided in Table 4). During the eclipse period, 600 seconds must be at peak power (2000 W) while the remaining time is at average value (400W).
  - The capacity of a battery can be readily measured via battery testing. However, given that a battery performance is very sensitive to temperature, environmental testing of the battery must also be conducted. The results of these tests must be extrapolated using empirical methods to determine the end-of-life performance. This will require some analysis.

- **EPS-REQ-0700:** The EPS shall be capable to regulate the generated such that it is directly usable by the spacecraft and payload subsystems.
  - This requirement was derived directly from the top-level function EPS-FUNC-002, which was derived earlier.
  - This requirement can be verified by a combination of investigation, analysis and testing. By making use of investigation, the power, voltage and current requirements of the spacecraft and payload subsystems can be determined. This can then be directly matched with the capabilities of the EPS and analysis can be used to check if the current combination of regulators can match the demands. Finally, testing at the ESA Space Power Lab using "Power Converter Design Validation" can also be used to verify that the regulation system matches the needed specifications [7].
- **EPS-REQ-0800:** The EPS shall be capable to distribute the generated power to the spacecraft and payload subsystems.
  - This requirement was derived directly from the top-level function EPS-FUNC-002, which was derived earlier.
  - Similarly to the above, this requirement is can be verified through demonstration in a dry run of the PCDU capabilities to handle the wide range of possible power consumptions by the spacecraft at the specified voltage (400-2000 W, 28 V from EPS-AS-03).
- **EPS-REQ-0900:** The EPS shall be compatible with the other subsystems in the spacecraft.
  - This requirement was derived directly from the top-level function EPS-FUNC-003, which was derived earlier.
  - This requirement can be through demonstrations of the different component relationships in the EPS system and the system relationship to its environment.
- **EPS-REQ-1000:** The EPS shall protect the spacecraft and payload subsystems from failures in EPS system.
  - This requirement was derived directly from the top-level function EPS-FUNC-005, which was derived earlier.
  - Verification of this requirement implies testing of the entire EPS architecture and simulating initiating failures of the electrical system. Although such operation is deemed to be expensive, this is necessary due to the central role of the EPS.

4 Solar Array Type Selection

Following the establishment of the requirements, the first step in the development of the EPS is the determination of the power generation method. It is known from the assignment that this must be by means of a solar array; however, the type of solar array is left open to decision by the group. For the choice of solar array, the single most important driver is the array area, as a smaller array will lead to a more weight and volume efficient design, which are both tremendously important factors in the development of any new space vehicle. Other elements of consideration such as reliability and cost are fairly similar amongst all panel types, therefore a trade-off in this scenario is unnecessary: the most efficient solar panel will be selected.

In order to determine the required solar panel area, it is first important to determine how much power these must generate per orbit. It is known that the required end of life (EOL) power during the sunlight and eclipse period are 800 and 400 Watts respectively. Furthermore, given that power can only be generated during the sunlight part of the orbit, it is crucial that the solar panels generate enough power during the sunlight period of the orbit (T_d) to be able to provide the necessary power required during eclipse (T_e). This will result in a higher required power generation during sunlight periods. This is shown in Equation 4 [2]. The EOL power is the constraining factor for the minimum solar panel array area, which depends on the panel degradation (L) and the mission duration (t) as shown in Equation 5 [2].

After having determined the EOL power, the required solar panel array (A_{req}) is calculated by dividing the $P_{SA_{BOL}}$ by the power provided by the Sun at Mars multiplied by the arrays' efficiency (P_{gen}, Equation 8) whilst ensuring that this multiplication is not higher than the maximum power the array can provide per square meter (Equation 6 [2]). The solar power, provided by the Sun, decreases with the distance (in AU) squared and the incidence angle (i). This relationship can be seen in Equation 7 [2].

With a relation for the solar panel area based on EOL power, the solar panel area can be determined for a variety of different solar panel types. This was done with the six different types given in Table 5. By making use of the procedure above, the required solar panel areas are therefore determined to be the values shown in Table 6. Based on those results, Option 6 was found to be the best option, and therefore, this is the type which will be used for this mission.

$$P_{SA_{EOL}} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right) + \frac{(P_{peak} - P_d) T_{peak}}{X_e}}{T_d} \tag{4}$$

$$P_{SA_{BOL}} = \frac{P_{SA_{EOL}}}{(1-L)^t} \tag{5}$$

$$A_{req} = \frac{P_{SA_{BOL}}}{P_{gen}} \tag{6}$$

$$P_{Sun} = \frac{P_{Earth}}{d^2_{S,M}} \cdot \cos(i) \quad (7)$$

$$P_{gen} = P_{Sun} \cdot x_{array} \quad (8)$$

Table 5: Key information for different solar panel arrays [2]

Option	Solar Panel Type	Efficiency (x_{array})	Degradation per Year
1	Si	0.140	0.0375
2	GaAs (SJ)	0.185	0.0275
3	GaAs(MJ)	0.226	0.0275
4	GaAs (Improved MJ)	0.260	0.0275
5	GaAs (ultra MJ)	0.280	0.0275
6	QJ 4G32C - Advanced	0.320	0.0275

Table 6: Outputs for different solar panel array options

Option	P_{gen} [W]	$P_{SA_{BOL}}$ [W]	A_{req} [m^2]
1	82.1	1617.1	22.9
2	108.5	1519.9	16.3
3	132.6	1519.9	13.3
4	152.6	1519.9	11.6
5	164.3	1519.9	10.7
6	187.8	1519.9	9.4

5 Trade-Off Criteria for Power Storage Systems

With the key preliminary requirements derived and validated, and the solar array type now selected, the next step in the design process is to identify and develop the key system criteria for the trade-off selection of the power storage system. These must reflect the primary objectives and requirements of the mission, such that the subsequent subsystem trade-offs yield meaningful results. The main trade-off criteria have been developed based on constructive group discussions and reflection, as well as external literature such as the course notes from past and current courses, the Space Mission Analysis and Design book (see [2] and [8]), and documentation from previous missions. These elements will be outlined and motivated in the list below:

- **Mass:** Mass is a key design factor that needs to be taken into account. Sending anything to space is expensive, thus minimizing mass is a necessity.
- **Volume:** Volume is another crucial design parameter, since the size of spacecrafts is limited by the launchers payload bay. When minimizing the mass, a smaller launcher can be used, which can lead to significant cost reductions and launch options.
- **Cost:** It is in the customer's interest to get the desired data from the mission for the lowest cost, meaning that cost should be taken into account when designing a spacecraft.
- **Durability:** Every system in the spacecraft has to survive the rough conditions of launch, it is therefore very important that all the systems are durable. The entire mission is a fail if key subsystems, such as the EPS, fails.
- **Reliability:** When designing a spacecraft, one has to ensure that the equipment lasts for the entire duration of the mission. Reliability is therefore an important factor for the design of a spacecraft. When equipment fails prematurely, the mission will be a failure.

All the above-mentioned criteria relate to each other is many ways, of which a few will be elaborated in this section. First, it can be reasoned that reducing the total volume of the EPS will most likely reduce the mass of the system. However, reducing mass and volume whilst keeping the same functionality comes at a cost. More advanced, and often more expensive, electronics have to be used, which increases the cost of the system. This miniaturization of equipment can have a positive effect on the durability of the entire system. An example of this would be the difference between a hard drive, making use of spinning disks, and flash storage that uses small flash memory modules. In spite of this, making use of these novel electronics reduced the reliability of the total system, since it's often not tested for long periods in space.

6 Power Storage System Selection

Having outlined the trade-off criteria in the previous section, it is now time to look at what methods can be used for storing the necessary power used during eclipse. There are three different methods that are often used during space missions [9]: rechargeable batteries, capacitors and fuel cells. The usefulness of these three different methods of storing energy will be discussed below, making use of the trade-off criteria.

Rechargeable batteries are often used for space missions, due to their high specific power and power density when comparing to capacitors and fuel cells. The fact that they are often used in space mission makes them also a relatively cheap option; they can be seen as off-the-shelf parts, eliminating RD costs. A downside of using rechargeable batteries is their durability, they have a limited lifetime that is closely related to the depth of discharge (DoD). This DoD results in the need of bringing extra battery weight with you, to ensure that the battery does not need to fully discharged every cycle, resulting in a heavier battery configuration.

Capacitors are another, often used, method to store energy. A significant benefit of capacitors is their durability. They can handle more charging cycles when comparing to rechargeable batteries. However, the specific power and power density of capacitors is often ten to twenty times lower than that of rechargeable batteries [9]. This low specific power and power density results in a heavy power storage system, which is undesirable for space missions. Capacitors are mainly used to aid in delivering peak power due to their ability to supply high pulses of power.

The final energy storage method is making use of a fuel cell. Fuel cells are, up to this point, mostly used for manned space missions that require significant power for longer durations. Fuel cells have adequate specific power and power density, just like rechargeable batteries [9]. The main downside of using fuel cells is that they are yet to be used in unmanned science missions like ours. This means that their reliability and durability over longer periods of time is unknown. This is a concern, since losing the ability to store power will be detrimental for the mission.

Figure 2: Trade-off Table

The findings, presented above, are summarized in a trade-off matrix shown in Figure 2. Rechargeable batteries have been chosen as the power storage system, due to their high specific power and power density. They are also commonly used in other space missions, thus it is a relatively cheap and reliable option.

7 EPS Key Components and Architecture

In this section, the different key components of the EPS are presented based on [2] and [5]. The system considered is based on photovoltaics combined with a battery as a secondary power generation unit, furthermore, a power regulation and distribution unit is necessary to ensure compatibility and control over the produced power. Those three elements together fulfil the functions EPS-FUNC-001 to EPS-FUNC-005; in Table 7, an appropriate component for each element is presented such that it complies with the requirements presented in section 3.

Table 7: Selected components for the EPS

Element	Component	Justification
Solar panels	QJ Solar Cell 4G32C - Advanced	At the time of writing this report, this is the most advanced type of solar cell on the market for spatial applications [10]. Produced by AZUR SPACE, its efficiency is estimated to be 32%. Furthermore, it was documented as the solar panel choice for the FAHRENHEIT mission orbiter B [11], which has an estimated maximum average power consumption of 926.4 W (similar to EPS-REQ-0200). Power density: 0.861 kg/m^2.
Battery	VL 10ES - SAFT	It was found that this space-qualified battery is the latest version presented by the European SAFT company in 2019 and has an energy density of 230 Wh/kg [12]. This battery is indicated to be mostly used for GEO, MEO and LEO orbit, the former usually implying operational lifetimes of around 15 years. Per 44 Wh, cylinder of 102 mm diameter and 33 mm height.
Power regulation & distribution	Magellan Aerospace PCDU	This PCDU uses the 'by the slice' design which permits to optimize for mass, power and volume by scaling to the mission. It is indicated to be able to handle from 100 W to 3 kW [13], which fits EPS-REQ-0100 to EPS-REQ-0300.
Wiring	ESCC 3901 021	Due to the long exposition to space radiation, a component that is classified as "excellent radiation resistance" was selected. Those wires have a mass of 3.4 g/m, an operating temperature range of -200°C up to +200°C and a voltage rating of 600 VAC max.

Having those components in mind, the architecture of the EPS and its relation with its environment is presented in Figure 3. This architecture was inspired from [14, 2]. Furthermore, the mentions "Appropriate Converters" relate to the DC to AC conversions depending on the components used in other subsystems.

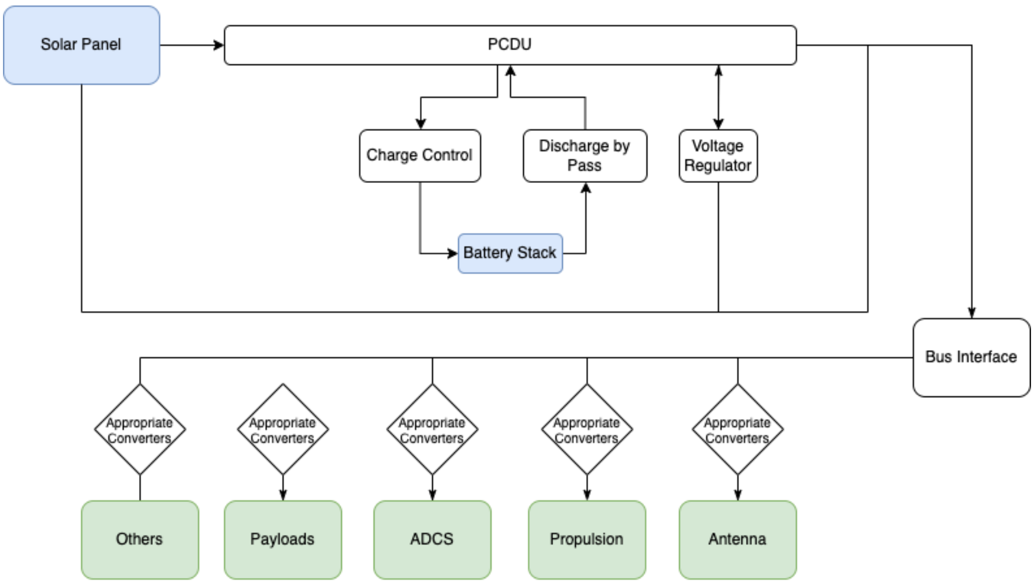

Figure 3: EPS architecture

8 Mass Budget of the EPS

Identification of the key components of the EPS from commercial off-the-shelf sources now allows for a preliminary mass budget to be drafted of the system as a whole. This is done in order to firstly provide an estimate for how heavy the entire system, and secondly to identify which elements contribute the most to the total. The procedure to generate a mass budget is as follows: first, based on the commercial off-the shelf components identified in the previous section of this report, the individual component masses must be identified. Now, this step is not always as straightforward as it seems, as many suppliers for space applications tend to customize their products based on the requirements that the customer needs. Therefore, this will be addressed on a case by case basis below:

- **Solar Panels:** Based on the information provided on the data-sheet of the supplier identified in the previous section, it is known that per unit of product, the cell area is $30.18cm^2$ and that the associated weight of this area is $1780mg$. Now, based on earlier calculations, it is known that the required solar panel area is $9.4m^2$. Therefore, the number of solar panel units required is as follows:

$$Units_{solarpanel} = \frac{9.4m^2}{30.18cm^2} = \frac{94000cm^2}{30.18cm^2} \tag{9}$$

$$Units_{solarpanel} = 3114.645 \approx 3115 \tag{10}$$

  Therefore the total mass of the solar array will be given as 3115 times the unit mass, thereby leading to a final mass of 5.547 kg.

- **PCDU:** In the case of the PCDU, it was found that upon contacting Magellan Aerospace (the supplier of the proposed PCDU for this mission), the weight of the PCDU is not fixed, but rather determined based on the customized product for the specific application. Therefore, to determine the approximate PCDU weight, PCDU data from page 128 of the ADSEE course of year one (see [5]) will be used in order to create a regression of PCDU weight as a function of peak power (which in the case of this mission is 2 kW). This was found to be the following:

$$Mass_{PCDU} = 5.3603 \cdot ln(Power_{PCDU}) + 15.382 \tag{11}$$

  This yielded a regression value of 0.7704, which is sufficient. Using this, the approximate PCDU weight is estimated to be 19.09 kg.

- **Battery:** The determination of the battery weight is rather straightforward, as it is known that the required energy capacity is 1950kJ (see requirement EPS-REQ-0600), the depth of discharge is 0.3, and that one battery can contain 44Wh of energy. Therefore, the number of batteries required is given according to the following equation:

$$No._{batteries} = \frac{\frac{1950kJ}{0.3}}{44Wh} \tag{12}$$

$$No._{batteries} = \frac{6500kJ}{158.4kJ} \tag{13}$$

$$No._{batteries} = 41.03 \approx 42 \tag{14}$$

  Now, based on the number of batteries, and the energy density of one battery being $230\frac{Wh}{kg}$, the mass of one battery is given to be:

$$m_{battery} = \frac{44Wh}{230\frac{Wh}{kg}} \tag{15}$$

$$m_{battery} = 0.1913kg \tag{16}$$

  Then the total mass of the batteries must be:

$$m_{batteryassembly} = 42 \cdot 0.1913 = 8.0347kg \tag{17}$$

- **Wiring:** The mass estimate of the wiring is probably the trickiest, as the mass of the wire provided by the supplier is provided per metre of length. Therefore, an estimate for the length of wiring which is used in this orbiter has to be provided. Based on the information provided in [15], a large satellite generally uses approximately 20 km of wiring for its harness. Now, this satellite design is generally categorized as small to medium-sized based on the sizes of the other components, therefore a wiring length of 10km will be used instead. This in combination with a mass per unit length $3.4\frac{g}{m}$ leads to a final wiring mass of 34 kg of wiring.

With the component masses calculated, the total mass of the EPS can be simply determined by adding the individual masses together. In order to account for any expected design changes, a further safety margin of 5% is added on to the mass of each component of the EPS. This is common amongst most spacecraft EPS designs in industry, including in the Project AIM and Nebula from the European Space Agency.[16][17]. The mass budget and breakdown of the component masses is provided in Table 8. Note that here the masses are expressed in grams for the sake of increased accuracy.

Table 8: Mass Budget of the First Iteration of the Mars Orbiter Mission

System	Acronym	Description	Quantity	Unit Mass (g)	Margin (%)	Unit Mass with Margin (g)	Total Mass with Margin (g)	
EPS								
	BTA	Battery Assembly	42	191.304	5	200.869	8436.522	
	PCDU	PCDU	1	19097.476	5	20052.350	20052.350	
	SLA	Solar Array	3115	1.780	5	1.869	5821.935	
	WG	Wiring	10000	3.4	5	3.57	35700	
							TOTAL	70010.807

9 Mass Budget of the Spacecraft

With a mass budget of the EPS drafted, the last step in this first design iteration is to provide a rough estimate of the spacecraft mass as a whole. Given that there is no information regarding the remaining elements of the spacecraft, this estimate can only be provided based on historical trends regarding the fraction of the total mass which the EPS takes up. Based on Spacecraft Mission Analysis and Design, the EPS generally takes about 21% of the total spacecraft mass. However, given that there is no information regarding any of the remaining subsystems, this percentage will be reduced to 16%. Using this margin, the total spacecraft mass is the following:

$$Mass_{spacecraft} = \frac{70.010}{0.16} \tag{18}$$

$$Mass_{spacecraft} = 437.567kg \tag{19}$$

10 Second Iteration

This first iteration was sent to the customer, who was very happy about the work produced, but then announced that two design changes arose.

- Only 300 W of power are necessary to sustain the spacecraft during the eclipse.
- A peak power of 900 W during 6 minutes will occur due to data transmission just after the eclipse.

The first iteration being more conservative with respect to those changes, this will result in a more desirable EPS with respect to mass and cost. First, the following requirements are rewritten:

- **EPS-REQ-0100:** The EPS shall provide an average power of at least **300** W at mission end-of-life during the eclipse period of the orbit.
- **EPS-REQ-0300:** The EPS shall provide a peak power of at least **900** W at mission end-of-life during the direct sunlight period of the orbit, for a maximum **6** minutes.
- **EPS-REQ-0600:** The EPS shall store **960** kJ of energy per one orbit at mission end-of-life.

Those requirement changes result in a change in the required solar panel area to $7.71\ m^2$. This however does not change the result of the trade-off performed on the energy storage system. Furthermore, the PCDU which was previously selected is also scalable and was scaled to the previous peak power, meaning that a smaller version of the same PCDU can be selected. The rest of the components also do not require any change and simply need to be scaled accordingly, hence the EPS architecture presented in Figure 3 is still valid. The mass budget of the EPS and the spacecraft change according to Table 9.

Table 9: Mass Budget of the Second Iteration of the Mars Orbiter Mission

System	**Acronym**	**Description**	**Quantity**	**Unit Mass (g)**	**Margin (%)**	**Unit Mass with Margin (g)**	**Total Mass with Margin (g)**	
EPS								
	BTA	Battery Assembly	21	191.304	5	200.869	4218.253	
	PCDU	PCDU	1	14817.236	5	15558.097	15558.097	
	SLA	Solar Array	2555	1.780	5	1.869	4775.295	
	WG	Wiring	1000	3.4	5	3.57	35700	
							TOTAL	60251.645

Using the same ratio of EPS mass to total spacecraft mass (16%), the total spacecraft mass is given to be 376 kg.

11 Role Distribution & Reflection

With the two design iterations complete, it is now time to briefly reflect upon the role distribution which was given to all members of the team at the beginning of the assignment. Although the idea of assigning different roles to team members is conceptually a good idea, it was found that in this assignment in particular, the roles outlined in the beginning were not strictly adhered to for the majority of the project. The reason for this was mainly due to the small group sizes and the smaller size of the assignment in comparison to projects such as the Design Synthesis Exercise. As a result of this, it was found that work could be done more effectively if all the group members kept an overview of the project overall while being completely responsible for the completion of their own tasks. As a result of this, the roles of "Team Leader", "Research Responsible" and "Time Manager" fell upon all the members equally, thereby not requiring individuals to be solely responsible for it. The accumulation and distribution of resources, as well as documenting the results of group discussions, was the responsibility of one team member, which is essentially a fusion of "Team Leader" and "Discussion Moderator". Therefore, for this assignment in particular, a strict hierarchy was not necessarily required apart from the one unique role of "Team Leader and Resource Responsible" was required. It is important to stress that for the sake of organization, however, a more rigid structure will be required for larger projects as it will not be possible for all members to have a complete project overview and there will be a loss of critical resources and group results otherwise.

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