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OSTM/Jason-2 Mission

Assignment 6: Electrical Power Subsystem

Group 25

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Acronyms

ADCS Attitude Dynamics and Control System. 6

AMR Advanced Microwave Radiometer. 6

BEU Battery Electronic Unit. 5, 6

BM Battery Management. 5

BOL Beginning of Life. 4

CARMEN-2 Environment Characterization and Modelisation 2. 6

DHU Data Handling Unit. 4, 5

DOD Depth of Discharge. 7

DORIS Doppler Orbitography and Radiopositioning Integrated by Satellite. 6

EOL End of Life. 8

EPS Electrical Power Subsystem. 2, 4, 7

ESA European Space Agency. 8–10

GPSP Global Positioning System Payload. 6

LEO Low Earth Orbit. 4

LPT Light Particle Telescope. 6

NASA The National Aeronautics and Space Administration. 10

NOM Nominal Mode. 6

OBDH On Board Data Handling. 6

OBSW On Board Software. 5

PCE Power Conditioning Equipment. 4–6

POSEIDON Positioning,Ocean,Solid Earth, Ice Dynamics, Orbital Navigator. 6

PROTEUS Plateforme Reconfigurable pour l'Observation, les Télécommunications Et les Usages Scientifiques. 4, 5, 10

SA Solar Array. 4, 8

SADM Solar Array Drive Mechanism. 4, 7, 8

T2L2 Time Transfer by Laser Link. 6

TCS Thermal Control System. 6

TTCM Telemetry Tracking Commanding and Monitoring. 6

1 Architecture

The electrical power subsystem of a satellite is a critical component that ensures the satellite's functionality by generating, storing, and distributing electrical energy to all the other subsystems of the spacecraft. For Jason-2, considering its placement on a LEO, the choice of the primary energy source easily fell on a solar array, while during eclipses power will be provided by batteries, which makes them the secondary power source.

Analyzing specifically how the electrical energy is generated in the satellite, it can be noted that there are two symmetric solar arrays located near the center of mass of Jason-2 itself, each equipped with single-axis stepping motors, in order to control the orientation of each wing.

A Lithium-Ion battery is used to store the energy which is then distributed through a single unregulated primary electrical bus, which has an average voltage of 28V and can vary between 23 and 36 V depending on the state of charge of the battery.

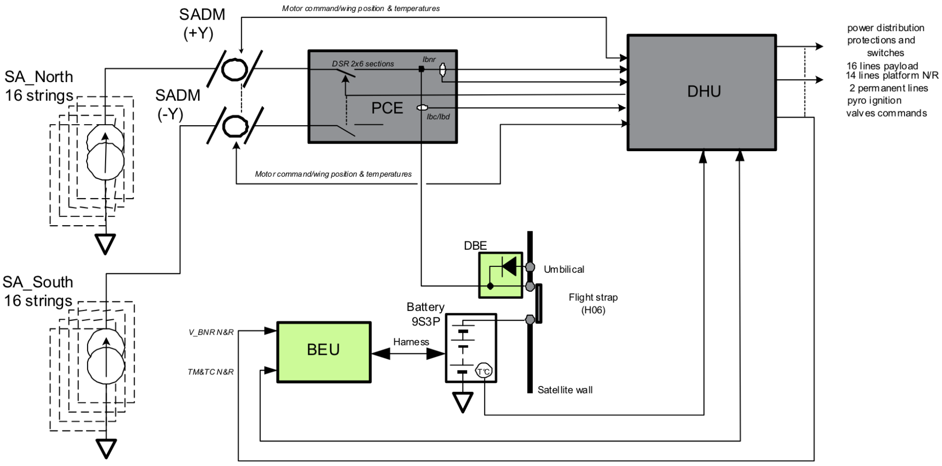

Fig. 1: EPS overall architecture

The average power required by the PROTEUS platform in nominal mode is $300\,W$ while the payload average consumption is $250\,W$ [3]. Payload power interface is given by the DHU, which typically provides 16 switchable unregulated power line on the unregulated bus, with a maximum current for each line of 5 A. This distribution permits the usage of 16 lines simultaneously or a complete cold redundancy for up to 8 lines at the time. [2]

1.1 EPS Components

1.1.1 Solar Array

The solar arrays of the Jason-2 satellite consist of 32 strings, each containing 102 standard Silicon solar cells connected in series, which are organized into 12 digital sections at the input of the Power Conditioning Equipment. The current output from each string is affected by the bus voltage, sunlight exposure, temperature, and the radiation dose received by the cells. Covering a total area of $9.6\,m^2$, the solar array is capable of delivering a maximum power of $1250\,W$ at the beginning of its operational life (Beginning of Life (BOL)).

The orientation in space of the SA is regulated by the Solar Array Drive Mechanism (SADM), which aligns the panels using an assembly consisting of a motor and an appropriate transmission.

Fig. 2: One wing of the SA on Jason-2

1.1.2 Power Conditioning Equipment

The PCE manages the electrical power of the solar array connecting and disconnecting the solar array cells to the bus. The number of strings connected to the bus can change from 2 to 32 by turning On and Off the electrical sections. To ensure the satellite receives the minimum required current, at least two strings are to be always kept connected. The PCE delivers the required charge current, according to the control commands coming from DHU,

1.1.3 Battery

For the electrical functional chain of PROTEUS a single battery permanently connected to the main bus was used, and its charge is ensured by the on board software, which adjusts the number of solar sections connected to the bus.

The battery used for the platform is a 9S3P VES 100 Lithium-Ion with a capacity of 26Ah. The battery is composed of:

- 9 cell packages connected in series, where each package is made up of 3 cells in parallel to provide a nominal capacity of 78Ah.
- 9 safety devices, called Bypass, each dedicated to a cell package.
- 18 shunts, used to compensate for the dispersion of the state of charge among all the cell packs of the battery and to balance the cell packs, both the 9 nominal and the 9 redundant ones.
- 2 heaters and 3 thermistors for Battery Management (BM) thermal control.
- A harness to connect the battery to the Battery Electronic Unit (BEU).

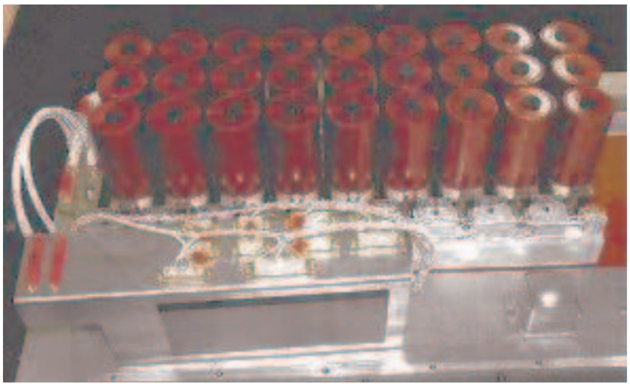

Fig. 3: Battery Pack on Jason 2

1.1.4 Battery Electronic Unit

The BEU is a crucial component that monitors and manages the voltage of individual cells in the battery. Its functions include measuring cell voltages, controlling communication between cells, activating bypasses when needed, converting primary voltage, and interfacing with other subsystems.

1.2 Battery Management

The battery management system of the Jason-2 satellite ensures autonomous battery charge control after each eclipse, it is performed by the On Board Software (OBSW). This system operates in two distinct modes, sequenced at a period of 1Hz.

Mode 1: Constant Current - This mode is accessed during initialization, the initial phase of battery charging, and during discharge. Its primary objective is to provide a constant current when the solar array (SA) is exposed to sunlight.

Mode 2: Constant Voltage (Tapering Mode) - This mode is activated when the maximum battery cell voltage reaches a predefined limit (adjustable up to 4.1 V). In this mode, the battery management system calculates the number of PCE sections to switch On or Off to maintain the battery voltage between two thresholds $[V_{limit} - \Delta V_{dis}]$ and $[V_{limit}]$. If the voltage drops to an adjustable lower level, the system reverts to Mode 1 to ensure optimal battery performance and longevity.

2 Power Budget and phase analysis

This section will estimate the power budget needed in the Nominal Mode (NOM) for the eclipse and daylight phases, which are the two key phases. This process requires unifying all the results from the previous chapters and conducting a thorough literature analysis. Not all subsystems had a well-defined power budget, so some values were reasonably assumed, knowing the total power budget and some of the other subsystems' power budget.

<table>
<tr><th></th><th>Daylight [W]</th><th>Eclipse [W]</th></tr>
<tr><td>Proteus</td><td>300</td><td>225</td></tr>
<tr><td>Propulsion</td><td>-</td><td rowspan="5">225</td></tr>
<tr><td>TTCM</td><td>60</td></tr>
<tr><td>ADCS</td><td>100</td></tr>
<tr><td>TCS</td><td>40</td></tr>
<tr><td>OBDH</td><td>100</td></tr>
<tr><td>Payload</td><td>250</td><td>75</td></tr>
<tr><td>POSEIDON</td><td>78</td><td rowspan="7">35</td></tr>
<tr><td>AMR</td><td>31</td></tr>
<tr><td>DORIS</td><td>42</td></tr>
<tr><td>GPSP</td><td>17.5</td></tr>
<tr><td>CARMEN-2</td><td>10</td></tr>
<tr><td>LPT</td><td>15</td></tr>
<tr><td>T2L2</td><td>48</td></tr>
<tr><td>Other Systems (TCS, TTCM...)</td><td>10</td><td>40</td></tr>
<tr><td>Total</td><td>550</td><td>300</td></tr>
<tr><td>Margin +20%</td><td>+ 110</td><td>+ 60</td></tr>
<tr><td>Total Margined</td><td>660</td><td>360</td></tr>
</table>

Table 1: Power Budget Analysis

3 Reverse Sizing

3.1 Introduction

In this section of the report a preliminary sizing of the primary and secondary electrical power subsystems will be performed. This sizing will take into account the data available of the subsystems and some reasonable assumptions of everything which could not be found.

The sizing was performed at the worst possible power consumption case, both under sunlight exposure and in eclipse, applying a 20% safety margin.

Once the total power requirements are known, the primary power source may be sized. For Jason-2, the primary power source are the two solar arrays, which provide all the power for the system and for the recharging of the batteries while exposed to sunlight.

As a secondary power source, batteries were chosen to provide power to the system during the eclipse periods. A major change was performed with respect to the previous Jason-1 mission as the batteries were changed. The satellite upgraded its batteries from the previous single Nickel-Cadmium battery to a Lithium-Ion, due to their improved charging efficiencies and low leakage. [8]

Furthermore the system utilizes a quasi-regulated power system, similar to the one utilized in the Jason-1 mission. This meant that the discharge of the battery was unregulated, while the charge of the battery was regulated by the Power Conditioning Equipment (PCE) and more specifically each cell was supplied with the correct voltage through a Battery Electronic Unit (BEU). [8]

3.2 Establishing Initial Parameters

Before beginning the reverse sizing process, it is crucial to define key parameters in accordance with the datasheet and eviromental conditions.

The environmental evaluation takes into account the satellite's performance under nominal conditions. The orbit period during this phase, is typically around 112 minutes. The duration of eclipses is then calculated using the following trigonometric formulas:

$$T_{eclipse,max} = 2\sin\left(\frac{R_e}{R_e + h_{s/c}}\right)^{-1} \cdot \sqrt{\frac{(R_e + h_{s/c})^3}{\mu_e}} = 34.82\,\text{min} \tag{1}$$

Table 2 provides a concise summary of the environmental and system parameters:

Environmental Data		**System Required Data**	
Sun Irradiance	$P_0 = 1366.1W/m^2$	**Daylight Power**	$P_d = 660W$
Time in Eclipse	$T_e = 34.82$ min	**Eclipse Power**	$P_e = 360W$
Time in Daylight	$T_d = 77.60$ min	**Expected Life Time**	$T_{life} = 5\,y$
Inclination Angle	$\theta = 25$ deg	**Voltage**	$V_{sys} = 28V$
		Daylight Efficiency	$X_d = 0.85$
		Eclipse Efficiency	$X_e = 0.65$

Table 2: Initial Data EPS

Another crucial aspect to consider is the expected mission's duration. Originally, the mission was envisioned for a 3-year period, however, according to the Proteus manual, sizing of the Electrical Power Subsystem for a 3-year mission with 4 solar array failures is comparable to the sizing of a 5-year mission without failures. Consequently, all reverse sizing is done based on a 5-year duration, despite the actual mission lasting nearly eleven years.

Jason-2 utilizes a Saft VES 100 battery model, with a capacity of 26 Ah chosen instead of the specified 27 Ah indicated in the datasheet [5]. This decision is consistent with findings from a study analyzing the changes destined for the EPS, following the Jason-1 mission[8]. In the same document, it's notable that a very low DOD was deliberately chosen to prevent battery degradation, given the low orbital period and high cycle count, and from our point of view, in anticipation of potential mission time extensions, as observed with Jason-1.

Although nothing was found on the solar panel cells' precise model, it was taken into account the use of a space-grade model used in that period, which had similarities with data found on other sources, such as the use of Si cells and similar absorbance and emissivity coefficients. [2]

According to the Platform manual [2], attitude control and the Solar Array Drive Mechanism (SADM) can achieve a 90% recovery of sunlight in the case of a non-sun-synchronous orbit, such as that of Jason-2. This indicates that, in the worst scenario, there will be a $\theta = 25$ deg between the sun and the solar panel.

Table 3 highlights the data of Jason-2's solar array and battery pack:

Solar Array: Spectrolab Silicon K4702 [6]		**Battery**: Saft VES 100 [5]	
Efficiency at BOL	$\epsilon = 0.133$	**Specific Energy**	$E_m = 118Wh/kg$
Degradation per year	$dpy = 0.028$ [4]	**Density Energy**	$E_v = 230Wh/dm^3$
Inherent Degradation	$I_d = 0.76$	**Efficiency**	$\mu_{batt} = 0.80$
Cell Voltage	$V_{solar,cell} = 0.585V$	**DOD**	$DOD = 15\%$ [8]
Cell area	$8cm^2$	**Cell voltage**	$V_{batt,cell} = 3.6V$
		Cell Capacity	$C_{cell} = 26Ah$ [8]

Table 3: Solar Cells and Battery Data

3.3 Calculation and sizing

3.3.1 Solar array

In accordance with Section 2, the worst-case scenarios for power consumption occurs during the nominal phase in daylight when the system is fully operational and requires charging the battery for the eclipse. The first step is to calculate the total power requested by the solar array as:

$$P_{sa} = (\frac{P_e T_e}{X_e T_d} + \frac{P_d}{X_d}) = 1025W \tag{2}$$

Then the specific power at the beginning of life is found:

$$P_{BOL} = \epsilon_{solar} P_0 I_d \cos(\theta) = 146.72 W/m^2 \tag{3}$$

Then the specific power at EOL is evaluated as:

$$P_{EOL} = P_{BOL} L_{life} = 127.29 W/m^2 \qquad \text{with: } L_{life} = (1 - dpy)^{T_life} = 0.87 \tag{4}$$

The surface area of the Solar Array (SA) can be calculated using the SA power requirement and the specific power output at the EOL. According to ESA standards [1], it is advisable to size the array considering at least one string failure. Considering the 8 panels (one string per panel) with 4 panels per wing, the sizing is based on 7 panels and then the eighth is added.

$$A_{SA} = \frac{P_{SA}}{P_{EOL}} = 9.08 m^2 \qquad A_{panel} = A_{SA}/7 = 1.3 m^2 \tag{5}$$

Furthermore, the calculation of the number of cells in series needed to achieve a voltage of $28V$ can be straightforwardly determined, followed by the evaluation of the total number of cells required using the formula:

$$N_{series} = \text{ceil}(\frac{V_{sys}}{V_{cell}}) \qquad N = \text{ceil}(\frac{A_{SA}}{A_{cell}}) \qquad N_{tot} = \text{ceil}(\frac{N}{N_{series}}) N_{series} \tag{6}$$

Afterward, a faster recalculation is required to determine the new value following the selection of the number of cells.

$$V_{real} = N_{series} V_{cell} \qquad A_{SA} = N_{tot} A_{cell} \tag{7}$$

Finally, it is also important to make a rough estimation of the mass. Considering a silicon layer of $200 \mu m$ [7] ($\rho = 2330 kg/m^3$) and an aluminum honeycomb of $3cm$ ($\rho = 123 kg/m^3$) the final table can be easily compiled comparing the panel system without redundancy, the panel system with redundancy, and the actual panel mounted on Jason-2.

SOLAR ARRAY	NO Redundancy	WITH Redundancy	Jason2
N_{cell}	11376	13002	N/A
Voltage [V]	28.08	32.09	23-36 [2]
Area [m^2]	9.06	10.4	9.6 [2]
Mass [Kg]	37.82	43.23	43 [4]

Table 4: Solar Panel Sizing and Comparizon

It is interesting to note the different values compared to the real Jason-2 case. The difference can be attributed to the choice of solar array efficiency, which significantly affects the sizing. Another difference could be the use of ESA's redundancy guidelines, which have been in place since 2017 and surely were not followed by the Jason-2 missio, launched in 2008.

Assuming that the SA modeled in hte reverse sizing is exactly the same as the one mounted on Jason-2, the calculation without redundancy is the closest to the real scenario. The area evaluation appears plausible as it acknowledges the empty gaps between the cells, present in the real panels utilized for the Jason-2 mission, which increase the panel's actual area without adding power. Additionally, the mass evaluation seems plausible as it does not consider other components like the junctions between panels and the SADM.

3.3.2 Battery

During the analyzed mission, the spacecraft passed through many solar eclipse phases, necessitating a secondary power source.

The first step is to determine the battery capacity required to meet the power demand, calculated as:

$$C = \frac{T_e P_e}{(DoD) N \eta} = 1638.59 \text{ Wh} \tag{8}$$

where $N = 1$ is the number of battery packs and $DoD = 15\%$ is the maximum depth of discharge.

Then the number of cells in series can be evaluated as:

$$N_{series} = \text{ceil}(\frac{V_{sys}}{V_{cell}}) = 8 \tag{9}$$

This value must be increased according to ESA guidelines [1] from 8 to 9, in order to add redundancy, considering one string failure. It is important to note that the system has the ability to bypass the single string. Then, a new value for the bus voltage can be obtained:

$$V_{sys,new} = N_{series} V_{cell} = 28.8 \text{ V} \tag{10}$$

$$C_{string} = \mu_{batt} C_{cell} V_{sys,new} = 599.04 \text{ Wh} \tag{11}$$

where $\mu_{batt} = 0.8$ represents a typical value for the package efficiency. To achieve the desired capacity, the number of cells needed can be calculated as $N_{parallel} = \text{ceil}(\frac{C}{C_{string}}) = 3$.

Afterwards, it is important to estimate the mass and volume of the battery, which can be obtained using the values provided in the battery datasheet [5] and in Table 3:

$$m_{batt} = \frac{C_{batt,real}}{E_m} \qquad V_{batt} = \frac{C_{batt,real}}{E_v} \tag{12}$$

where $C_{batt,real}$ is the real capacity of the battery and can be obtained as: $C_{batt,real} = N_{parallel} C_{string}$.

Moreover, from the datasheet, the real values for each battery can be considered for a comparison with the obtained mass and volume. Each battery has a mass of $m_{cell} = 0.81$ kg and dimensions of $d = 53$ mm in diameter and $h = 185$ mm in height, resulting in a total volume of $V_{cell} = 0.41$ dm^3.

Finally, a comparison can be made between the actual data and the reverse sizing using the summary table.

BATTERY	NO Redundancy		WITH Redundancy		Jason2 [8]
Total Cell	8s-3p: tot 24		9s-3p: tot 27		9s-3p: tot 27
Capacity [Wh]	1797.12		2021.76		N/A
Max Voltage [V]	28.80		32.40		36.9
Volume [dm^3]	From E_v	7.81	From E_v	8.79	29.7
	From $N_{battery}$	9.84	From $N_{battery}$	11.07	
Mass [Kg]	From E_m	15.23	From E_m	17.13	28.4
	From $N_{battery}$	19.44	From $N_{battery}$	21.87	

Table 5: Battery Sizing and Comparizon

As expected, the results obtained differ slightly from the actual values. The additional mass and volume account for the casing and other circuitry of the battery cell, as shown in Figure 3.

References

[1] ESA. Concurrent design facility studies standard margin philosophy description, 2017.

[2] NASA & Alcatel. PROTEUS User's Manual, 2004.

[3] CLS, CNES, NASA & EUMETSAT. OSTM/Jason-2 Product Handbook, 2011. https://www.ospo.noaa.gov/Products/documents/J2-handbook.pdf.

[4] Jean-Michel Cresp, Eric Boutelet, Jean Massot, and Pierre Tastet. Proteus Electrical Functional Chain: Flight Return Experience. In *9th European Space Power Conference*, volume 690 of *ESA Special Publication*, page 127, October 2011.

[5] Saft. Rechargeable lithium battery VES 100. https://alpha-energy.ru/D/0000009803/EB_Saft_VES_100_33017-2-0608_200806_en.pdf.

[6] Spectrolab BOEING. Silicon K4702 Solar Cells. https://www.spectrolab.com/DataSheets/K4702/k4702.pdf.

[7] Thales Alenia Space. Solar Generator Family. https://www.thalesgroup.com/sites/default/files/database/d7/asset/document/solar_generators_family.pdf.

[8] V. Michoud, J. Digoin, P. Tastet, J.Loubeyre. Proteus Electrical Functional Chain: From Nichel-Cadmium to Li-Ion Battery. https://articles.adsabs.harvard.edu/pdf/2005ESASP.589E.106M.