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# Reverse Engineering of Juno Mission

## Homework 6

Course of Space Systems Engineering & Operations
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## Group 5


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# Contents

# Notation

| | |
|---|---|
| **BOL** | Begin of Life |
| **BOM** | Begin of Mission |
| **EPS** | Electric Power System |
| **LEOP** | Launch and Early Orbit Operations |
| **ME** | Main Engine |
| **RCS** | Reaction Control System |
| **GRAV** | Gravity science |
| **MWR** | Microwave Radiometer |
| **EPM** | Earth Pointing Mode |
| **SPM** | Sun Pointing Mode |
| **DSM** | Deep Space Manoeuvre |
| **PJ** | Perijove |
| **BAT** | Battery |
| **PS** | Propulsion Subsystem |
| **TMTC** | Telemetry and Telecommand |
| **AOCS** | Attitude and Orbit Control Subsystem |
| **TCS** | Thermal Control Subsystem |
| **EPS** | Electric Power Subsystem |
| **P/L** | Payload |
| **C&DH** | Command & Data Handling |
| **SDST** | Small Deep Space Transponder |
| **TWTA** | Traveling Wave Tube Amplifier |
| **KaTS** | Ka-band Translator |
| **SSPA** | Solid State Power Amplifier |
| **SRU** | Solar Reference Unit |
| **SSS** | Spinning Sun Sensor |
| **IMU** | Inertial Measurement Unit |
| **MLI** | Multi Layer Insulation |
| **Li-Ion** | Lithium Ions |
| **AC** | Alternating Current |
| **DC** | Direct Current |
| **DoD** | Depth of Discharge |
| $D$ | Distance of Juno from the Sun [AU] |
| **EOL** | End of Life |
| **EOM** | End of Mission |
| **EGA** | Earth Gravity Assist |

| | |
|---|---|
| **IC** | Inner Cruise |
| $yrs$ | Years elapsed from BOM |
| **LILT** | Low Intensity and Low Temperature |
| **OC** | Outer Cruise |
| **PDDU** | Power Delivery and Drive Unit |
| **RTG** | Radioisotope Thermoelectric Generator |
| **S/C** | Spacecraft |
| **SA** | Solar Array |
| **SASU** | Solar Array Switching Unit |
| **SPE** | Sun Earth Probe angle [°] |
| **UTJ** | Ultra Triple Junction |
| **SoC** | State of Charge |
| $A'_{sa}$ | SA area with continuous cell distribution [$m^2$] |
| $A_{sa}$ | SA area with discrete cell distribution [$m^2$] |
| $A_{cell}$ | Cell's area [$m^2$] |
| $C$ | Battery capacity [Ah] |
| $n_{series}$ | Number of cells in series in a string [-] |
| $n'_{cells}$ | Number of cells with no ESA margins[-] |
| $n_{cells}$ | Number of cells with ESA margins[-] |
| $q_{sun}$ | Solar flux [$W/m^2$] |
| $q_0$ | Solar flux at 1 AU [$W/m^2$] |
| $P_{ch}$ | Absorbed power for recharging [W] |
| $P_{req}$ | Required power before margins [W] |
| $P_{sa}$ | Electrical power provided by SA [W] |
| $P_{bat}$ | Electrical power given by the battery [W] |
| $p_{py}$ | Annual degradation [-] |
| $I_d$ | Inherent degradation factor [-] |
| $T_{pj}$ | Time at perijoves [h] |
| $T$ | Orbital period of Juno in nominal mission [h] |
| $t_{av}$ | Time available to recharge the batteries [h] |
| $t_{ch}$ | Time needed to recharge the battery [h] |
| $V_{nom}$ | Nominal voltage of the system [V] |
| $V_{cell}$ | Voltage of the single cell [V] |
| $\varepsilon$ | Efficiency of SA [-] |
| $\theta$ | Aspect angle [°] |
| $\eta$ | Line efficiency [-] |

# 1 Introduction of EPS

The Electric Power System of Juno adopts a solar-based energy source to provide enough power through the various conditions encountered during the mission, which ranges from low to high energy request around Jupiter, periods of eclipse, high radiation environment and more. This chapter will study firstly the complex requirements coming from the other subsystems and from the environment encountered, then a brief rationale of the adopted architecture will be treated. In the end, a reverse sizing of the primary and secondary sources (solar panels and batteries) will be carried out to check the compliance with the mission.

# 2 Analysis of power requirements along the mission

In the following section the power budget is presented (Table 1). All the subsystems have different requirements throughout the whole mission, which has been dived in multiple phases and modes.

| | LEOP | Cruise | ME Man. | RCS Man. | GRAV PJ | MWR PJ | BAT Charge |
|---|---|---|---|---|---|---|---|
| **PS** | 0 | 0 | 20 | 49.5 | 0 | 0 | 0 |
| **TMTC** | 71.8 | 71.8 | 71.8 | 71.8 | 114.3 | 71.8 | 71.8 |
| **AOCS** | 10.4 | 10.4 | 43.4 | 10.4 | 10.4 | 10.4 | 10.4 |
| **TCS** | 170 | 170 | 201 | 201 | 170 | 170 | 170 |
| **EPS** | 15.2 | 15.2 | 17.4 | 19.2 | 21 | 18.9 | 56.6 |
| **P/L** | 39.6 | 39.6 | 0 | 39.6 | 114.3 | 114.3 | 39.6 |
| **C&DH** | 12 | 12 | 12 | 12 | 12 | 12 | 12 |
| **Total** | 318.7 | 318.7 | 365.4 | 403.5 | 441.8 | 397.2 | 348.1 |
| **Tot. w/ margin** | 382.5 | 382.5 | 438.7 | 484.2 | 530.2 | 476.6 | 417.8 |

Table 1: Power budget [W]

Where for each subsystem the following assumptions and considerations were made:

- **PS:** only the power required to operate the valves of the RCS thruster or the ME was considered. For the RCS six thrusters were assumed to be on simultaneously at any given time. It can also be noted that the RCS require power only in short burst as the thrusters are turned on and off during a manoeuvre, so the average power request is actually lower than the one reported above.
- **TMTC:** for this subsystem one X/X SDST and one TWTA are supposed to be on for the entire mission while also presuming that the amplifiers are always operating at their maximum capability. During the GRAV PJ passes the KaTS and the SSPA are turned on. The redundant X/X/Ka SDST was not considered in this analysis.
- **AOCS:** the 10.4 W value is recovered by considering only one SRU and SSS active (nominal AOCS setup), while the 43.4 W power request during ME manoeuvres is due to the utilization of one IMU instead of the SRU.
- **TCS:** only the PS heaters were considered, while neglecting additional TCS hardware required by the other subsystems. In particular RCS valves and catalyst beds heaters, ME valves heaters and tanks heaters were focused on. Both of the RCS requirements were available in their datasheet.[1] For the ME valve heaters no data was found, so their requested power was assumed to be the same of the RCS. A rough estimation of the tanks heaters was performed considering MLI-covered titanium tanks partially radiating into deep space with an operating internal temperature of 20 °C during manoeuvres and a survivability temperature of 0 °C for all other phases/modes. Overall values of 133 W and 102 W were obtained, respectively, which include all propellant and pressurizer tanks. This approach ignores the whole insulating body of the spacecraft, so the aforementioned results are overestimated. Both values also neglect the large variations of the Juno-Sun distance throughout the mission, which has a significant impact on the thermal behavior of the S/C and the power request of the TCS.
- **EPS:** for this subsystem a reasonable 95% global efficiency was assumed in lieu of more accurate specifications which weren't available. Its consumption is therefore directly proportional to the power request of the rest of the S/C during each phase/mode. An additional 40 W are required to charge the batteries,[2] supposing that only one battery is being charged at any time.
- **P/L:** during the real mission numerous instrument checkouts were carried out. To estimate the power request, it was assumed that only the sensors were turned on, while the electronics was off. This value is also overestimated since the various tests were accomplished separately in time for each P/L. Similar to the RCS valves operation this power requested is also infrequent and not continuous over the whole mission. Concerning the science modes (GRAV PJ and MWR PJ) the stated value considers all P/Ls operating at maximum power for the entire PJ pass, while in reality this request is only achieved for a narrower time period.[3]

- **C&DH:** only one of the two redundant RAD750 single board computers is assumed to be on along the entire mission, the specific value was obtained from its datasheet.[4]

A 20% was then applied to the total values as required by standard guidelines[5] to obtain the values highlighted in the last row of Table 1.

# 3 Architecture and rationale of EPS

## 3.1 Available alternatives

The endeavour that Juno faces to generate enough electricity to sustain science operations at approximately 5.45 AU required particular attention in designing an efficient and reliable electric control system. Particularly, different options were present to generate the amount of power required, each of them with advantages and disadvantages:

- **RTG:** the choice of a radioisotope to generate electricity could be considered. To generate the amount of power required around Jupiter, during the planetary phase (Table 1), one RTG of the same size of the one present on board New Horizons spacecraft would have been sufficient ($\approx$ 250 We of production at BOL).[6] Electric power requirement would also be lower as some heat could have been routed to the propulsion section to heat up the tanks and fuel lines. This choice however had some problems, mainly with respect to the safety during Earth EGA, radiation contamination, heat dissipation at distances lower than 2 AU from the Sun and weight distribution. Particular problems could have rose as the said RTG generates around 4.4 kW of heat, requiring a very efficient dissipation system for the first years of the mission, oversizing it with respect to the nominal orbit around Jupiter. Availability of Plutonium-238 was also critical as suppliers could not guarantee the needed amount of fuel with the needed power output, given the stop in the production of the said isotope during the 80s.[7]
- **Solar panels:** no spacecraft equipped with solar panels has ever been tested at 5.45 AU from the Sun. This choice would have required a very large surface area, and thus precautions had to be taken into account inside the fairing during launch operations, to provide enough power for safe operations at Jupiter. A complex management system is also required in order to not discharge too much current inside the electronics during the ICs and the OC as the amount of solar flux hitting the S/C during different parts of the mission dramatically reduces. More stringent pointing requirements are also present as not having a clear view of the Sun could have led to the need of bigger and heavier batteries.

Considering also the driving requirement of utilizing as much as possible off the shelf components, budget constraints, the limited supply of plutonium and the different possible configurations offered, solar panels were chosen. This choice led to a particular design of the satellite, where mass distribution was exploited to grant more stability throughout the different phases of the mission.

## 3.2 Components and distribution

The flown spacecraft is fitted with 3 solar arrays as the primary source and 2 Li-Ion batteries as the secondary one. SASU are used to manage the generated power and a PDDU is responsible of managing and distributing electricity to the various subsystems. The whole system works at 28 V DC, which is a standard for low-power consumption system as Juno.

### 3.2.1 Solar arrays

The arrays are mounted on the side of the main body, spaced 120° apart, and are composed by a different number of panels: A1 presents only 3 panels while A2 and A3 present 4 panels each. The total mass of the arrays is 340 kg, with a total area of 60 $m^2$[8] and an active area of 49.77 $m^2$, composed by 18.698 solar cells.[9] Each cell, produced by Spectrolab[10], is in an UTJ configuration, to obtain optimum packing and high performances in LILT environments like the one around Jupiter. The three layers, Ge for the bottom cell, GaAs for the middle cell and $GaInP_2$ for the top cell, are placed on top of a Germanium Kapton substrate and connected with tunnel junctions. A multi junction configuration optimizes the electromagnetic spectrum exploitation, focusing on specific wavelength bands. An anti-reflective Si coating protects the cell from the harsh environment of Jupiter.

Solar panels are linked one to the other and to the main body with electro-actuated hinges and the first panel of each array is supported by struts. All these elements are needed to both extend the panels soon after separation from the Centaur and to control the position of the arrays during all the operations. This is necessary to take into account the bending of the arrays during large manuevers and their thermal expansion. Moreover it is necessary to align the main inertia axis (Z-axis) with the spin axis due to the particular mass distribution of the panels and of the instrumentation. Power produced by each panel is varying in each phase, depending on incidence and distance from the Sun. Values at 1 AU are in the range of 14 kW and about 400 W EOL. This value consider all the active surface to be working. Fixing the attitude of the spacecraft, as the distance increases, and thus the solar flux decreases, a lower current is produced at the same voltage. However with the increase of distance, and thus the decrease of the temperature, the current decreases, the cells' efficiency increases and an higher voltage is produced. Cell efficiency at 28°C is 28.4%. Given that Juno's trajectory ranges between 0.88 AU and 5.45 AU from the Sun, the cells are grouped in strings of three different lengths, whose connection must be adjustable to satisfy power, voltage and current requirements in each phase. When not in use, the power generated by each cell is left on the panels to be dissipated as heat. This is crucial to ensure a high level of efficiency, increasing as temperature decreases, throughout the whole journey. The rationale for the distribution of the cells in each panel is shown in Figure 1 and explained as follows. Once fixed the cell type used for the whole mission and known the power, voltage and current required in each phase, exploiting the characteristic curve of each cell, the number of cells that must be connected in series and in parallel can be obtained and so the required string length type. Then, as the cells' dimension is known and the length of the required strings has been retrieved, the area of each string can be computed. The mass of the panels needs to be distributed so that the inertia matrix becomes as diagonal as possible, with the constraint that only one string type can be fitted in each panel. Indeed in the case of a panel composed by all three types of strings, hot spots due to the activation of only a single type of string could lead to uneven internal stresses and thus damaging the whole panel. Moreover, damages to a single panel in this case would lead to the loss of multiple types of strings, reducing the available power during different phases of the mission. However, given all the aforementioned requirements and considering also the limited amount of space inside the Atlas V's fairing, A3P1 had to feature both medium and short strings. The resulting string configuration is reported in Table 2. The maximum number of strings in parallel and the minimum activation distance are limits that ensure the survivability and correct operation of the system: exceeding these limits, and thus generating a current above 7A, would blow the fuse on that string, compromising the whole mission. The critical points of the mission to prevent middle string turn on is 1.2AU ÷ 1.5AU and 1.5AU ÷ 1.9AU for short strings.

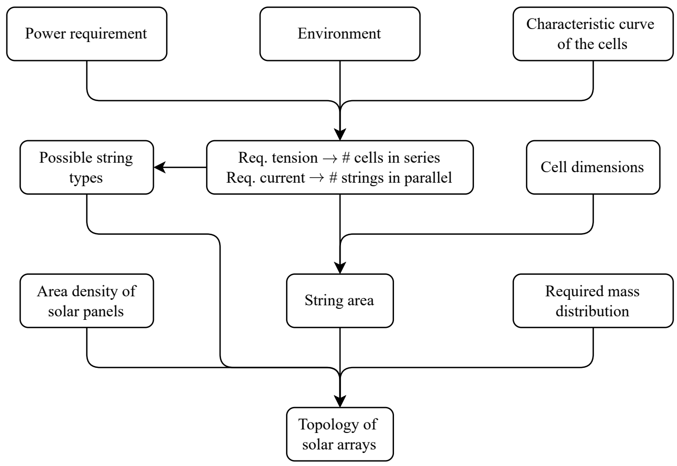


Figure 1: Cell distribution rationale

| String type | # cells in series per string | # strings | Max # strings in parallel | Minimum distance [AU] |
|---|---|---|---|---|
| **Long** | 22 | 114 | - | 0.88 |
| **Medium** | 14 | 369 | 40 | 1.8 |
| **Short** | 13 | 848 | 64 | 3.8 |

Table 2: Strings description

Consequently, as can be seen in Table 3 and Table 4, A2P1, A2P2 and A2P3, and the same in A3, are slightly smaller than the A1 correspondents while the presence of A2P4 and A3P4 counterbalances the presence of the MAG boom on the tip of A1.

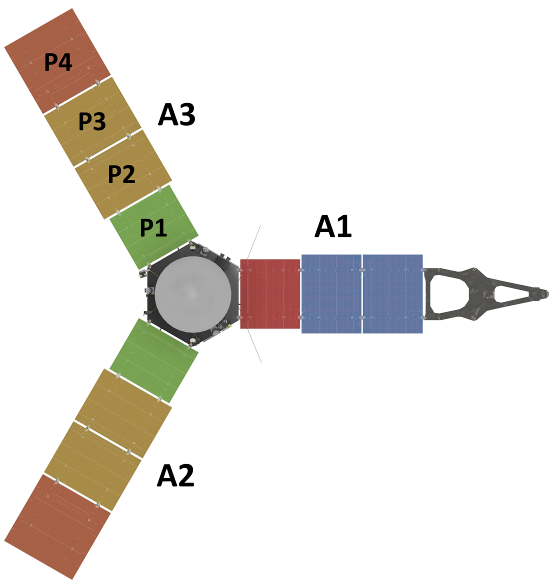


Figure 2: Juno's panel configuration

| | P1 | P2 | P3 | P4 | Array's area |
|---|---|---|---|---|---|
| **A1** | 4.92 | 5.60 | 5.60 | - | 16.11 |
| **A2** | 4.81 | 5.46 | 5.46 | 6.29 | 22.02 |
| **A3** | 4.81 | 5.46 | 5.46 | 6.29 | 22.02 |

Table 3: Panels areas [$m^2$]

| | P1 | P2 | P3 | P4 | Array's mass |
|---|---|---|---|---|---|
| **A1** | 27.81 | 31.62 | 31.62 | - | 91.05 |
| **A2** | 27.17 | 30.89 | 30.89 | 35.54 | 124.47 |
| **A3** | 27.17 | 30.89 | 30.89 | 35.54 | 124.47 |

Table 4: Panels masses [kg]

### 3.2.2 Batteries

Juno is equipped with two Li-Ion batteries in cold redundance: these ensure complete operability throughout the whole mission, even when the power generated by the solar panels is not sufficient. The mounted batteries are produced by EaglePicher, a proven and already flown design[11]. Particularly each of the two cells is able to hold up to 55 Ah in a range between 24.0 V to 32.8 V. Voltage at 50% SoC is 29.4 V.[12] The choice of Lithium Ions batteries is related to their high energy density, longevity (as Juno mission had to last at least 7 years), limited self discharge, higher number of cycles and wider range of operating temperatures with similar DoDs with respect to other types of batteries. This battery is characterized by a particular high value in terms of specific power as its primary usage is to cover the peaks of the system during science operations. The total mass for the batteries, without considering the applied MLI and radiation shielding is about 32 kg, 16 kg each. Considering that the available datasheet[11] is not the one for Juno mission but for MAVEN, a mission around Mars with various cycles of sunlight and eclipses, a significant difference in the DoD and life cycles must be made: Juno's trajectory was carefully planned to avoid as much as possible eclipses, with the exception of EGA. The batteries are more than capable to handle that $\approx$ 19 minutes eclipse[13] and are heavily used to perform science operation around Jupiter, when the power request exceeds of about 110 W the power generated by the solar arrays.

This design mission required 33 science orbits around Jupiter, one EGA and a total of four manuevers where Juno's panels did not face directly the Sun: the cycles the batteries have to sustain is much lower than the 40.000 cycles at 40% DoD of MAVEN. This difference allowed to discharge more the batteries and thus to reduce the weight of the system while guaranteeing its correct operability. Since the mission is still ongoing seven years after its planned decommissioning, as a consequence of the numerous failures of the propulsion system which did not allow to perform the critical PRM, the sturdiness and effectiveness of the batteries as the generated power from the solar panels is lower than the planned EOL one is proven. Given the previously described limits of Amperes the system is capable to handle, the batteries are charged at C/50 and continuously kept at 50% SoC during ICs and OC and only charged to the needed percentage prior to the manuever. Different panels' strings are capable of reaching the correct voltage to charge the batteries at 50% SoC and beyond, in particular the middle strings from 1.2 AU and the short strings from 1.5 AU from the Sun. Full charge voltage of the said string is reached at 1.5 AU and 2.5 AU respectively, while long strings are capable to fully charge the batteries at closer distance from the Sun as they are the first to be turned on. Strings can provide the needed voltage sooner than required, but the described safety procedure in subsubsection 3.2.1 must be taken into account.[12] The positioning of the batteries is critical given the harsh environment Juno faces: temperature range for the on board batteries is tighter than the solar panels' one. The latter are rated to work between -133 °C and +96 °C, the batteries instead are only capable to withstand temperatures between -20°C to +40°C. As the batteries are mounted on top of the propulsion module, and linked to all the instruments by external and shielded cables, on the exterior of the S/C, they need to be protected thermally, while they are rated to withstand the predicted radiation along Juno's orbit around Jupiter.[14] A MLI blanket over a Beryllium box is present to help the heaters keeping the batteries inside their operating range. Given the size of the batteries, 30.1 cm × 20.56 cm × 23.85 cm each, they could not be placed inside the vault, where all the electronics is positioned: dimension needed for the vault (now at $\approx$ 80 cm × 80 cm × 70 cm ) would have almost doubled and thus its mass would have been significantly higher, at over 200 kg.[14]

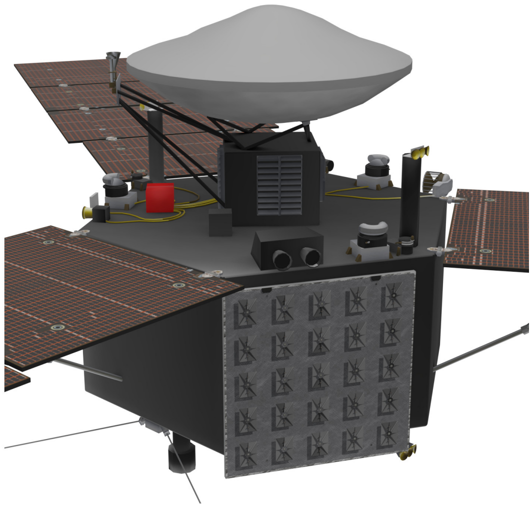

Figure 3: Position of the 2 batteries (red box)

### 3.2.3 Power distribution

The EPS of Juno handles the generation of power from the solar array in order to cope with the large Sun range variation. Before 2000s, few missions were capable to produce such energy with merely solar panels. The increased efficiency of solar cells technology, reliability and methods to cleverly rearrange cell strings configuration have allowed the access of this technology also to deep space missions such Juno. Due to lack of information about the primary source distribution method, the Solar Array Switching Unit (SASU) patent developed by Nasa and Lockheed Martin in 1998 was assumed to be the one used.[15] As previously said, the need to have three different strings is driven by the different radiation and temperature conditions encountered along the mission that change the performances of the cells. The patent of the SASU comes from the need to manage in a clever way the power generation, in order to furnish always a constant voltage to the main bus and fulfil the power demand of the subsystems. The switching unit is composed by:

- switch circuits to connect the strings as imposed by the control system;
- the control system;

- external shunt power card, which shunts the excess of power generated;

The bus voltage is battery-regulated, which means that no BCR nor BDR are present on-board.[16] This design choice is typical for NASA unmanned spacecrafts.[17] The absence of a dedicated regulator improves the efficiency and lowers the mass of the system. The PDDU is responsible for delivering the required power to all the loads which are then self-regulated in order to generate the required tension and current. This is particularly true for the some of the on-board instruments: for example JEDI[18] and JADE[19] require voltages up to 10 kV while the MAG[20] suite requires AC current to safely conduct measurements.

# 4 EPS operation

During the cruise of Juno, different conditions are encountered: prior to the arrival at Jupiter (about 5 years after launch), the solar aspect angle (Figure 4) varies between 0° and 90°. For the majority of time (as reported in previous chapters) Juno is in EPM, while during DSMs the angle increases. This variation cannot be appreciated because the resolution of the graph is not enough to show the peaks. At around year 2, the angle is zero as due to thermal requirement the S/C is in SPM. Science operations are nominally performed in EPM, but peaks can be observed after 5 years as Juno performs MWR PJ passes. This condition is not treated in section 5 since the sizing is conducted on the nominal mission, where such values of aspect angle were not expected. Despite this, the batteries are capable of handling the real situation since their capacity presents a reasonable margin.

In Figure 5 the distance of Juno from the Sun and the type of string used by the solar arrays is displayed, as described in subsubsection 3.2.1. In particular, the long strings are used during the whole mission, the medium strings from 1.8 AU and the short strings from 3.75 AU. A relevant region is between 1.8 and 1.9 AU, in which Juno can function with or without the medium strings enabled. The region between 1.2 and 1.9 AU, highlighted by the red lines, is critical as any trigger of the short or medium strings could lead to the loss of these (as explained in subsubsection 3.2.2).

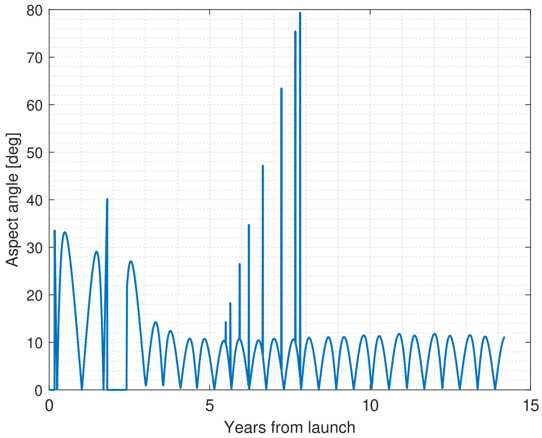


Figure 4: Aspect angle during the mission

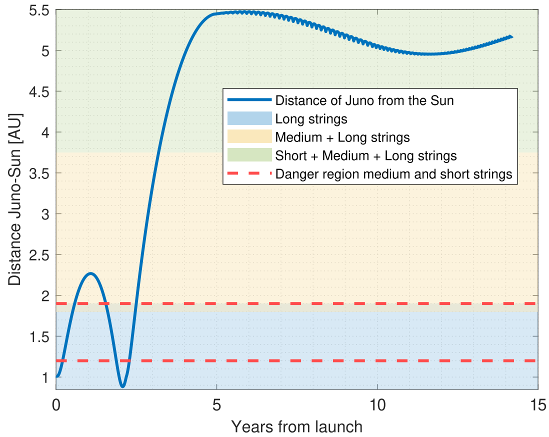


Figure 5: Strings utilization

# 5 Reverse sizing of EPS

As discussed in section 2, the most demanding phases are the GRAV science perijoves, which have a duration of about 6 hours each. In particular, the most critical one occurs when Jupiter is at its aphelion. This specific point is the one chosen for the sizing of the EPS, considering the planned EOM of $yrs$ = 6.20 years for the computation of the solar cells degradation. Firstly, the dimensioning of the solar panels will be carried out without considering the presence of the secondary power source. This is done to overestimate the required surface at Jupiter in order to satisfy the whole power requirement. Later, the real solar panels are assumed and the batteries are sized in order to fill the actual gap between the primary energy source and the power required by the whole system.

## 5.1 Solar panels

To compute the solar flux incident on the panels at the design orbit point, the following equation has been adopted:

$$q_{sun} = \frac{q_0}{D^2} \cos\theta \quad [\mathrm{W/m^2}] \tag{1}$$

where $D$ = 5.4543 AU is the distance of the S/C from the Sun, $q_0$ is the solar flux at the distance of 1 AU from the Sun and $\theta$ is the angle between the Sun direction and the normal from the panel surface. Since during GRAV the S/C is Earth pointing, $\theta$ coincides with the SPE angle, which can be found from ephemeris (≈ 7.48°).

From $q_{sun}$ and the power required in this condition ($P_{req}$ = 530.2 W from Table 1), the total area required to satisfy the power demand at perijove is computed as:

$$A'_{sa} = \frac{P_{req}}{q_{sun}\,\varepsilon\,(1-dpy)^{yrs}\,I_D} = 62.83\ \mathrm{m}^2 \tag{2}$$

Equation 2 takes into account the degradation of the panels during the mission. Typical values for GaAs UTJ panels are assumed and reported in Table 5:

| $\varepsilon$ [-] | $dpy$ [-] | $I_D$ [-] |
|---|---|---|
| 0.3 | 0.0350 | 0.77 |

Table 5: Properties assumed for solar arrays

It is worth noting that $A'_{sa}$ does not take into account the discrete distribution of areas due to cells. Moreover, an additional string of cells must be added to satisfy the official margin by ESA.[5] A more refined calculation is shown in Equation 3.

$$n'_{cells} = \left\lceil \frac{A'_{sa}}{A_{cell}} \right\rceil \qquad n_{series} = \left\lceil \frac{V_{nom}}{V_{cell}} \right\rceil \qquad n_{cells} = \left\lceil \frac{n'_{cells}}{n_{series}} \right\rceil \cdot (n_{series}+1) \qquad A_{sa} = n_{cells} \cdot A_{cell} \tag{3}$$

The area of a single cell is taken from the technical sheet[10]: $A_{cell} = 26.6\ \mathrm{cm}^2$. The voltage of each cell at Jupiter is taken from a model[12] that takes into account the distance from the Sun and the low operative temperature and no difference have been considered between cells belonging to different strings. The used value is thus $V_{cell} = 2.77$ V.
The solar arrays have a complex distribution of cells in three different types of series, hence with different voltages (as already discussed in subsubsection 3.2.1). To keep the calculation simpler, an average on the number of cells in the series has been computed through the nominal voltage of the system ($V_{nom} = 28$ V). The results of computation are compared to the real arrays in Table 6.

| | $n_{cells}$ [-] | $A_{sa}$ [$\mathrm{m}^2$] |
|---|---|---|
| **Sizing results** | 25776 | 68.56 |
| **Real values**[9] | 18698 | 49.74 |

Table 6: Results and comparison of the solar arrays

As can be seen in Table 6, the sized arrays result to be noticeably larger with respect to the real panels. This is due to the fact that this phase utilizes both the primary and the secondary sources and the batteries are not considered in this first preliminary sizing.

## 5.2 Batteries

For the battery sizing, the real active area of solar arrays was assumed (Table 6). From this, the power required from the battery was computed as the difference between the required power in GRAV mode (from Table 1) and the one delivered by the solar panels:

$$P_{req} = 530.16\ \mathrm{W} \qquad P_{sa} = q_{sun}\,\varepsilon\,(1-dpy)^{yrs}\,I_D\,A_{sa}^{real} = 419.69\ \mathrm{W} \qquad P_{bat} = P_{req} - P_{sa} = 110.47\ \mathrm{W} \tag{4}$$

This power has to be delivered by the batteries in proximity of the perijove for approximately $T_{pj} = 6$ h. From Equation 5 it is possible to obtain the capacity required by the battery in order to satisfy this request:

$$C = \frac{T_{pj}\,P_{bat}}{\eta\,DoD\,V_{nom}} = 49.84\ \mathrm{Ah} \tag{5}$$

where the line efficiency $\eta$ is assumed to be 95% and the $DoD$ is assumed 50% to be conservative and to not excessively reduce the battery life cycles along the mission. The result is compliant with the chosen battery, whose capacity is 55 Ah.[21] Moreover, since the battery is a 6s1p type, hence has only one series of cells, an additional battery is added to the system for cold redundancy as requested from ESA margins.[5] Subsequently, a series of calculations were conducted to verify the capacity of the solar panels to recharge the batteries in the orbital region where scientific operations are not conducted. The time available to recharge the batteries was computed as the difference between the nominal orbital period of 11 days and the 6 hours period passed at the perijove while performing science operations. The time necessary to recharge the batteries is then calculated with Equation 6:

$$t_{av} = T - T_{pj} = 258\ \mathrm{h} \qquad t_{ch} = \frac{C\,DoD\,V_{nom}}{P_{ch}} = 17.44\ \mathrm{h} \tag{6}$$

where $P_{ch}$ is the value shown in section 2. The solar panels are capable of recharging the batteries since $P_{sa} = 419.69\,W$ is higher than the power requested in BAT mode (Table 1) and $t_{ch}$ is widely less than $t_{av}$.

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