Politecnico di Milano

`Sixth Assignment`

Space Systems Engineering and Operations

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Change log	
§3.1	pp. 5: added results of the intermediate parameters, added explanation to the power margin, table 4 revised, added clarification about OSR area with respect to the solar cells
§3.2	pp. 6: added results of the intermediate parameters

1 EPS components

The MESSENGER power system consists of the Power System Electronics (PSE), the Power Distribution Unit (PDU), the Solar Array Junction Box, the battery, the two solar array (S/A) panels, and the Solar Array Drive Assembly (SADA) 1. The nominal bus voltage is 28 V and can vary between 22 and 35 V depending on the state of charge of the battery.
The mission design called for an interplanetary phase that did not require any excursion beyond 1.08 *AU* with relatively short eclipses, making solar panels and battery a much easier and simpler choice compared to the use of an RTG. Open bus design was also dictated by ample power availability and to reduce subsystem complexity.

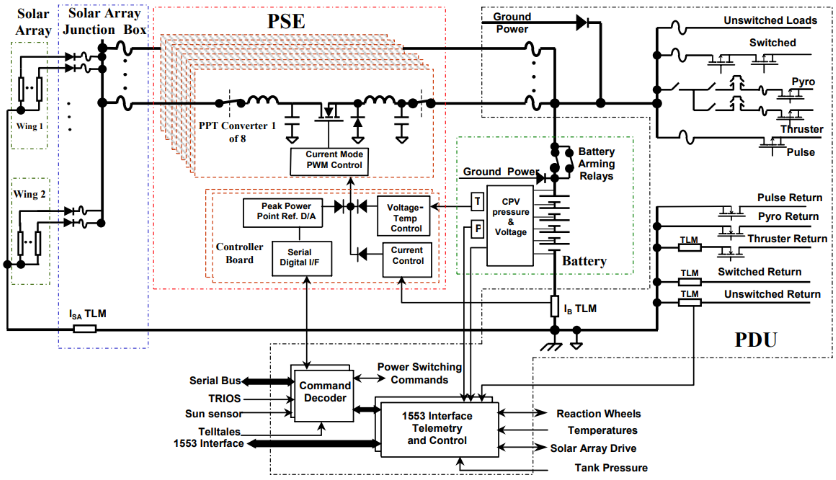

Figure 1: Simplified Block Diagram of the MESSENGER Spacecraft Power System

1.1 Solar Array

The MESSENGER solar array consists of two deployed single-panel wings. Each panel is 1.54 m wide and 1.75 m long and contains 18 strings of 36 cells each. The strings were placed between Optical Solar Reflector (OSR) mirrors with a cell to OSR ratio of 1:2 to reduce the panel absorbance. The panels are located on the +X and -X faces of the spacecraft, and they stick out of the sunshade and can pivot around the X axis independently from each other.
The panel substrates are 18-mm thick aluminum honeycomb with RS-3/K13C2U composite face sheets. The face-sheets are 0.6-mm thick with local 0.5-mm doublers. The graphite-cyanate-ester materials on the panel face sheets were chosen for their high thermal conductivity, but their mechanical strength is relatively low. Doublers and triplers are therefore required in areas of high stress due to the large panel cantilever in the stowed configuration. The panel front cell side is insulated with 0.05-mm Kapton, co-cured with the graphite fiber face sheet. The back face sheet is covered with co-cured square-shaped aluminized Kapton 30.5 cm x 30.5 cm sheets. The aluminized Kapton is used to lower the absorbance of the backside of the solar panel to a level comparable to the solar cell-OSR side. This ensures that the panel can survive solar illumination with normal incidence to either side at the closest approach to the Sun, even in case of loss-of-pointing of the spacecraft.
The solar cells are 0.14-mm thick, 3 cm by 4 cm triple junction gallium-arsenide cells, with minimum efficiency of 28%. The cells use a standard one-Sun cell top metallization grid design. The cover glass on each cell is 0.15-mm-thick cerium-doped microsheet. The cell interconnects utilize silver plated Kovar material. The wires that carry power and temperature signals are routed along the titanium boom to connectors at the SADAs. Both the wires and the boom are wrapped with multi-layer insulation. The solar panel temperatures are sensed using platinum wire sensors, placed beneath the solar-cell-side face sheet in small bored cavities. To minimize the magnetic field induced by the currents in the strings, adjacent strings are placed with alternating current polarity, and the strings are back wired such that each string return runs under its cells.

1.2 Battery

The MESSENGER spacecraft battery consists of 11 two-cell 23 *Ah* Nickel Hydrogen CPVs (Common Pressure Vessel) housed in a three-piece clam-shell-type mechanical design that meets all thermal-gradient requirements imposed by the mission profile. It was designed and built at APL. The cells are capable of

supporting the required 8-year mission life, as well as enable for a mission extension. Bypass switches were placed across each CPV to eliminate the potential of a spacecraft single-point failure caused by an open circuit of a pressure vessel. In the event of a cell open-circuit failure, small diodes provide a current path to blow a fusing element in the activating coil of the bypass switch. This circuit provides protection against an open-cell fault for both charge and discharge operation.
The battery chassis is electrically insulated from the spacecraft structure and connected to the S/C ground through two parallel 20-kΩ resistors. The battery cells were demagnetized, and the inter-cell wiring is designed to minimize the induced magnetic fields. Approximately 300 eclipses are expected during the first year in Mercury orbit, with the maximum eclipse duration during the mission being around 60 minutes. Power is managed during Mercury eclipses to allow adequate reserve battery energy to recover from any attitude anomaly. The maximum Depth-of-Discharge (DoD) expected during Mercury orbital operations is approximately 55%. The battery DoD from launch to Sun acquisition was 18%. The battery is also rated for a maximum discharge current of 14.5 A, and charging current was limited to a trickle charge of $\sim 0.15\ A$ during the final 5% of charge to decrease battery degradation.
The battery is located on the -Y face of the spacecraft, closest to the sunshield, to keep the sensitive battery in a location where it could operate for the whole mission in a range of $-10°C < T_{Battery} < 0°C$, with only minor excursions above this range.

1.3 Power System Electronics

The PSE contains eight buck-type peak-power-tracking converter modules, each designed to process around 100 W of output power. Each card can dissipate about 8 W. The control loop of the buck converters varies the duty cycle to maintain the input voltage from the solar array wings to the reference value set by the spacecraft command and data handling (C&DH) processor. The peak power tracking converters process all the power from the solar array. The maximum power the PPT trackers can process is around 800 W. The primary and backup controller circuits of the PPT converters and the housekeeping DC/DC converter are on two printed circuit (PC) cards. The maximum PSE power dissipation is 65 W.
The PSE is packaged in slices with one card per slice. There are eight PPT converter slices and two controller board slices. The slices are machined aluminum 6061-T6, in a tongue-and-groove design with venting through joints.
A heat spreader with thermal vias distributes heat dissipations among the slices and provides thermal interface from chassis to heat pipes that are located underneath the spacecraft deck. It is made of machined aluminum 6061-T6, 0.32 cm thick. There are sixteen thermal vias that are 1.9 cm in diameter and 2.86 cm long.

1.4 Solar Array Junction Box

The Solar Array Junction Box contains the isolation diodes in series with each string of the two solar array panels, the current shunt resistor of each wing, the peak power tracker module solar array side fuses, and solar array voltage telemetry buffer resistors. The maximum dissipation in the solar array diodes under worst-case conditions is 26 W.
The fuses placed in series with each string inside the solar array junction box protect the input and output fuses of the peak power tracker modules, in case of a short in one of the string isolation diodes, during the fault condition when the battery voltage is higher than the solar array voltage. This condition can occur during a spacecraft attitude anomaly when the spacecraft is closest to the Sun and the solar panels are pointed normal to the sun.

1.5 Power Distribution Unit

The Power Distribution Unit contains the circuitry for the spacecraft pyrotechnic firing control, power distribution switching, load current and voltage monitoring, fuses, external relay switching, reaction wheel relay selects, power system relays, Inertial Measurement Unit (IMU) reconfiguration relays, Integrated Electronics Module (IEM) select relays, solar array drives, propulsion thruster firing control, and propulsion latch valve control. There are two independent sides to the PDU: A and B. Onboard telemetry is also collected by the PDU. Each side of the PDU can command all of the PDU's circuitry.

1.6 Solar Array Drive Assembly

Each solar panel is independently rotated with a stepper motor actuator with harmonic drive gearing. A cable wrap that allows rotation from -20 to 200°around the X-axis, with zero being the panel normal in the -Y (sunshade) direction, is used to transfer the solar array power and panel temperature signals to

the power system electronics. Initially both panels were moved symmetrically to maintain an equal surface temperature, however scientists figured out a way to orient the panels asymmetrically to generate a sufficient SRP torque to reduce the needed number of momentum dumps needed over the course of the mission. This choice allowed for fuel savings that were proven useful during the extended missions [5].
The closer to the sun the mission got, the slimmer the window of allowable operating angles became, as both thermal and power generation limits got to each other as shown in figure 2. This made the SADAs a key element to manage during operations near the Sun and in Mercury orbit.

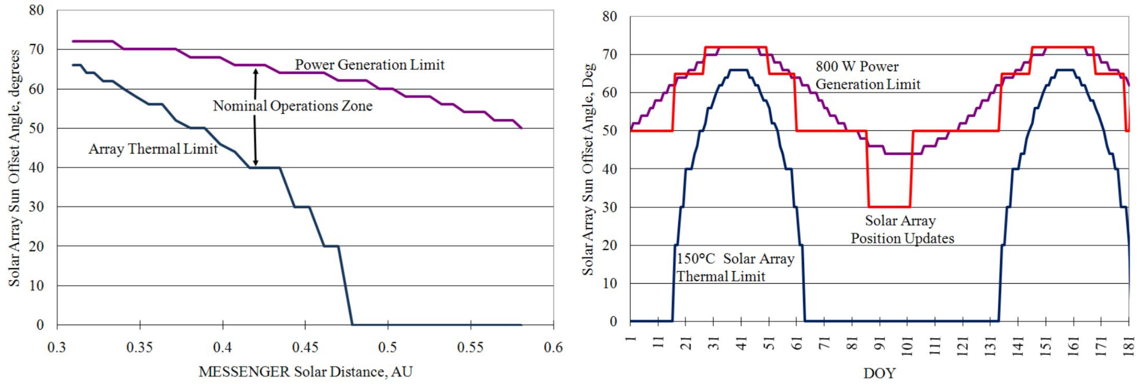

Figure 2: Example of limit curves and panel position

2 Architecture

2.1 Electrical power per phase

Throughout its journey to Mercury, MESSENGER reaches a distance of roughly 0.3 AU from the Sun: this implies strong variation of the intensity of the solar radiation during the mission. Furthermore, during science operations, the spacecraft operates in highly eccentric orbits, with eclipse periods which can sensibly change, meaning not only in discrepancies in the solar power input but also in the requirement of energy supply coming from the batteries. In Tab 1 and are displayed the spacecraft's power budget by subsystem [4] :

	Outer cruise (>0.95 AU)	**Inner cruise (<0.95 AU)**	**Orbit** [1]	**Eclipse phase (>35 min)**	**Eclipse phase (<35 min)**
Instruments and heaters	13.2	15.4	95.1	42.6	58.9
IEM	32.7	34.2	33.5	30.2	30.2
Power system	25.6	25.6	105.6	20.9	20.9
TTMC	65.5	65.5	110.5	22.8	25.4
G&C	70.3	70.3	119.7	119.7	119.7
Thermal	10.2	62.2	58.4	10.0	23.8
Propulsion	21.9	95.6	74.6	34.0	47.8
Harness	3.7	7.6	12.0	4.4	5.5
TOTAL	**247.8**	**382.7**	**597.4**	**284.6**	**332.2**
Power input from solar arrays	490.0	528.2	720.0	-	-
Maximum DOD	-	-	-	55	36

Table 1: Worst case scenario power budget for different subsystems [W]

The main difference between the phase of outer and inner cruise stands in the body's orientation, as if the spacecraft is far enough from the Sun, the sunshade points in the opposite direction in order to reduce the heating power required; once the sunshield is flipped back towards the Sun during the inner cruise and Mercury orbit, power consumption jumps up due to the need for more heating required by delicate components such as the tanks, valves, plumbing and delicate electrical components to maintain an adequate temperature.
During the orbit phase pointing requirements become more stringent to allow for the correct pointing of the

[1] Non-eclipsed time window

various scientific instruments, and the instruments themselves are activated and brought into full operation. Telecommunications also ramp up during this phase in order to transmit the collected data back to Earth.

2.2 Eclipse management

An important aspect of the mission was power management during eclipse periods. Two main types of eclipses occurred during the mission, each with it's set of challenges: the first one, during $< 35\ min$ eclipses is linked to the $14.5\ A$ maximum discharge current, while the other is due to the maximum allowed DOD of 60% during $> 35\ min$ eclipses.
This second condition is the limiting one of the battery sizing, as some eclipses last up to one hour.

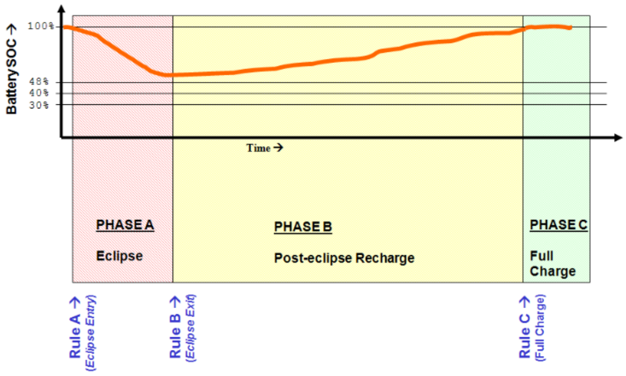

Figure 3: Expected battery state of charge during eclipse [5]

2.2.1 Example of short eclipse: Mercury eclipse

During an average Mercury eclipses, lasting about $15 - 20\ min$, battery discharge is not a problem in and of itself, however minor power-downs were still programmed in order to reduce the discharge current of the battery to below the maximum allowed value: this mainly involved spacecraft heaters and non-critical components, allowing scientific data collection to continue unaffected by the eclipse. This lowers the current draw of the spacecraft from the nominal $15 - 16\ A$ to below the $14.5\ A$ limit.

2.2.2 Example of long eclipse: Venus FB1

The first Venus flyby was challenging because the associated eclipse was 56 minutes long, so battery depth of discharge was the primary concern. For this eclipse, a major power-down was designed to keep the battery state-of-charge above 48%. It included the instruments, communications equipment, star trackers, and spacecraft heaters. The power-down and recovery were written into three on-board autonomy protection rules, one for eclipse entry. The first powered off the spacecraft loads at eclipse entry; the second powered on select loads at eclipse exit; the third powered on the remaining loads once the battery reached full charge.

2.3 Power regulation

The large variations in the solar distance and in the duration of the eclipses implies a large variation in the solar array maximum power point voltage: for this reason, the power system uses a Peak Power Tracker (PPT) topology which isolates the battery and the bus from the voltage and current large fluctuations, maximizing the solar array power [2].

3 Reverse Sizing

3.1 Primary source selection and sizing

For the purpose of the mission the selected primary source are solar panels, considering that the spacecraft will always be less then 1 AU of distance from the Sun. The cells chosen are triple junction gallium-arsenide cells, given their relatively high efficiency, radiation resistance and low risk.
In order to size the panels some data are shown in table (2):

Efficiency at beginning of life [-]	$\epsilon_{BOL} = 0.28$	**Degradation per year [-]**	$dpy = 0.0375$
Inherent Degradation [-]	$I_d = 0.77$	**Line efficiency in daylight [-]**	$X_d = 0.8$
Line efficiency in eclipse [-]	$X_e = 0.6$	**Power request in daylight [W]**	$P_d = 597.4$
Power request in eclipse [W]	$P_e = 284.6$	**Spacecraft time in daylight [h]**	$T_d = 11$
Spacecraft time in eclipse [h]	$T_e = 1$	**Mission duration [y]**	$T_{life} = 10.75$
Sun irradiance [W/m^2]	$I_0 = 9116.4$	**Sun-panel inclination angle [°]**	$\theta = 65$

Table 2: *Solar Array Data [3] [4]*

The sizing takes into account the most power demanding scenario, which is verified to be during the orbit phase around Mercury, when the array has to satisfy the high power demand, while also recharging the battery.

The first step is to calculate the total power requested by the solar array as:

$$P_{SA} = s\left(\frac{P_e T_e}{X_e T_d} + \frac{P_d}{X_d}\right) = 947.85\ W \tag{1}$$

Where s is equal to 1.2 and represents the ESA system level power margin of 20% [1]. Then the specific power at the beginning of life is found:

$$P_{BOL} = \epsilon_{BOL} \cdot I_0 \cdot I_d \cdot cos(\theta) = 830.65\ W/m^2 \tag{2}$$

The value of the critical Sun-panel inclination angle θ, which is the maximum one during the course of the mission, is very high due to the necessity to rotate the solar panels every time they reach a temperature of 150°, to dissipate the heat.
The specific power at the end of life can be obtained as:

$$P_{EOL} = P_{BOL} \cdot L_{life} = 550.78\ W/m^2 \qquad L_{life} = (1 - dpy)^{T_{life}} = 0.66 \tag{3}$$

Knowing the SA power request and the specific power output at EOL, the SA cells surface and the total surface of the panels (which takes into account the OSRs through a multiplier 3 in the calculation for the total array area due to the ratio of 2:1 OSR to solar cells), can be computed as:

$$A_{cells} = \frac{P_{SA}}{P_{EOL}} = 1.72\ m^2 \qquad A_{SA} = 3A_{cells} = 5.16\ m^2 \tag{4}$$

Finally, knowing that each cell is 3 by 4 cm, the number of cells is found:

$$N = ceil\left(\frac{A_{cells}}{A_{cell}}\right) = 1434 \tag{5}$$

The Solar Arrays are made of different layers of materials : Honeycomb Aluminium for the biggest part but also Kapton used for the OSR mirrors and different carbon fibers positioned at different ply angles to improve the mechanical properties [6]. Here densities and thicknesses retrieved from literature have been summarized:

	Density [kg/m³]	**Thickness [mm]**
Honeycomb Aluminium Alloy 5056	72	14
K13C2U/RS-3 Carbon Fiber	1750	1.152
K13C1U/RS-3 Carbon Fiber (Doublers)	1750	2.032
RS-4A	1210	0.510
Kapton	1420	0.102

Table 3: Solar array layers and thickness

A possible value for the density of the Solar Arrays can be deduced: $\rho_{SA} = 412.56\ kg/m^3$ taking into account the proportions of each material to the total thickness. The total thickness for the Solar Arrays is $t = 1.8\ cm$. The mass of the solar arrays can be computed as: $m = \rho_{SA} \cdot A_{SA} \cdot t$.

	A_{SA} [m²]	**N_{Cells} [−]**	**m_{SA} [kg]**
Reverse sizing	5.16	1434	37.90
Real	5.39	1368	34.12

Table 4: Sizing results and comparison

The calculated size of the solar array takes into account both solar cells and OSRs areas with perfect packing efficiency, while the area of the real solar array is retrieved from literature and includes gaps between adjacent elements. The value of the effective area of solar cells and OSRs in the real case is 4.92 m^2, that is coherent with the lower number of cells with respect to the computed one.
The mass discrepancy is likely due to the fact that no information was found on the specific locations where the doubler layer of composite was used. The computation supposes the layer is used everywhere on the panel and results in a preliminary result, which is not too dissimilar from the real result.
The cell number and panel area is linked to the fact that MESSENGER's solar array was designed to accommodate the loss of a line of cells, while still providing enough power for the orbital phase of the mission.

3.2 Secondary source selection and sizing

During the analyzed mission the spacecraft has passed through many solar eclipse phases, therefore a secondary power source was needed. The selected source is a Ni-H_2 battery, well suited for long lasting missions, as Lithium ion batteries were still not mature enough and too expensive at the time of launch.

Technology	Energy density (Wh/l)	Specific Energy (Wh/kg)	Specific power (W/Kg) (short pulse)	Cycle number (100% DOD)	Optimum operating temperature range (°C)	Energy efficiency	Self discharge	Comments
Ni-Cd	90	30	200	2000	0 to 40	70-75%	Moderate	
Ni-H2	50	55	150	2500	-10 to 30	70-75%	High	
Li-ion	250	150	2000	3000	10 to 40	>95 %	Low	Low Thermal dissipation
Advanced Li	>300	>200	>2000	>2000	-10 to 40	>95 %	Low	

Figure 4: Possible secondary battery choices

In order to size the battery some data are shown in table, retrieved from literature and from an analysis of the orbit (5):

Spacecraft time in eclipse [h]	**Power request in eclipse [W]**	**Specific energy [Wh/kg]**	**Energy density [Wh/dm^3]**
$T_e = 1$	$P_e = 284.6$	$E_m = 60$	$E_v = 50$

Line efficiency [-]	**Cell voltage [V]**	**Battery capacity [Ah]**	**Bus voltage [V]**
$\eta = 0.85$	$V_{cell} = 1.55$	$C_{cell} = 23$	$V_{bus} = 28$

Table 5: Battery data [4] [3]

The battery capacity required to fulfill the power demand is calculated as:

$$C_r = \frac{T_e P_e}{(DoD)N\eta} = 608.77\ Wh \tag{6}$$

where $N = 1$ is the number of batteries and $DoD = 55\%$ is the maximum depth of discharge. This value, for the expected number of eclipses (around 300), falls well within the working range of a Ni-H_2 battery. The number of cells in series is then obtained as:

$$N_{cells} = ceil\left(\frac{V_{bus}}{V_{cell}}\right) = 19 \tag{7}$$

This value needs to be increased from the resulting 19 to 22, in order to add the possibility to bypass one CPV (containing 2 cells) in case of failure. Then it is possible to retrieve a new value for the bus voltage and finally the total string capacity.

$$V_{bus,new} = N_{cells} V_{cell} = 34.10\ V \qquad C_{string} = \mu C_{cell} V_{bus,new} = 627.44\ Wh \tag{8}$$

Where $\mu = 0.8$ is a typical value of the package efficiency. The string capacity is already higher than the required capacity, which means that only one string is needed. Mass and volume of the battery can be found as:

$$m_{batt} = \frac{C_{batt}}{E_m} \qquad V_{batt} = \frac{C_{batt}}{E_v} \tag{9}$$

where : $C_{batt} = C_{string}$ because there is only one string needed.

The sizing results and a comparison with the real values are shown in table (6).

	$\mathbf{C_r}$[Wh]	$\mathbf{N_{Cells}}[-]$	$\mathbf{C_{batt}}$[Wh]	$\mathbf{m_{batt}}$[kg]	$\mathbf{V_{batt}}[\mathbf{dm^3}]$
Reverse sizing	608.77	22	627.44	10.46	12.55
Real	N/A	22	655.50[2]	24.50	41.09

Table 6: Sizing results and comparison

The capacity of the real battery is computed as : $C_{batt-real} = 23\ Ah \cdot V_{batt}^{Avg}$. The mass and volume values are much lower than the real ones, because they don't include the 11 common pressure vessels and the rest of the structure.

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[1] European Space Agency. *Margin philosophy for science assessment studies*. Tech. rep. 2012.

[2] George Dakermanji C. Jack Ercol and Binh Le. *The MESSENGER spacecraft power subsystem thermal design and early mission performance*. Tech. rep.

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[2] Supposing 23 Ah at an average of 28.5 V