Spacecraft design and system engineering
Jupiter's aurorae analysis

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Contents

1 Introduction

Due to its fast rotation and size, Jupiter has the most impressive northern lights in the solar system. Its fast rotation around itself (one full rotation is done in 10 hours) creates strong electric fields which attract charged particles from Io, one of its satellites, as a magnet. It therefore creates permanent aurorae around its poles, approximately a hundred times stronger than Earth's ones. Hubble telescope could take some ultraviolet pictures of these northern lights, completed by the probe New Horizons. The main goal of this mission is to send a satellite to Jupiter to be able to take pictures and analyse this amazing natural phenomenon.

Figure 1: Aurorae on Jupiter

2 Mission Definition

2.1 Mission type and objectives

Since the main objective of this mission is to caption Jupiter's northern lights, the most appropriate mission would be to design and send an orbiter with relatively low orbit around Jupiter. In the space mission's history, only two spacecrafts (Galileo and Juno) were sent on orbit to Jupiter to analyse its composition and main characteristics. To summarize, we will need a spacecraft on a low orbit around Jupiter to analyze:

- its composition
- its electric and magnetic field around the poles
- if the constraints are not too high, its deep composition (below the clouds up to its core)
- analyse the magnetic field around the planet to be able to map it
- measure the energetic particle density around it

2.2 Project Organization

Due to the impossibility to find any data on Juno's mission to base our research on, we will consider the New Horizon project organization ([2]), that lasted 5 years in total, which would correspond more or less to the amount of time we need for organizing our project. The project will be composed by 4 main phases with several milestones. The project organization will start on January 2020 and finish in June 2025, with the satellite's launch. The project organization is as it follows:

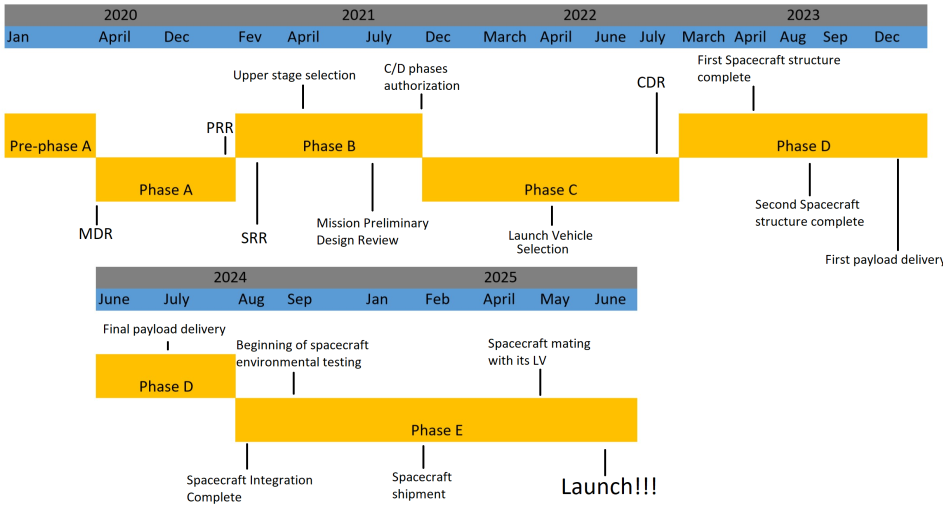


Figure 2: Project Organization

2.3 Launch window

From our own Hohmann transfer's calculations, the synodic period between Jupiter and the Earth is 1.0919 years. As seen in the project organization, the satellite cannot be launched before June 2025. We have decided to take the starting launch date at January 20th, 2025 to have a round number of months since the vernal equinox (March 20th) to facilitate our calculations. The launch windows are the following:

- 08-Jun-2025 06:43:48
- 11-Jul-2026 20:11:25
- 14-Aug-2027 09:39:02

We are considering the following approximation: we consider Juno's trajectory, which begins with deep space manoeuvres and a quiet phase to reach Jupiter. The deep space manoeuvres take more or less 2 years and then the quiet phase another two. This phase can be approximated by a Hohmann transfer for which we did the calculations previously. Since we have 3 launch dates and the deep space manoeuvres finish in august 2027 according to Figure 3, we will consider the launch window 14 August 2027 to be more realistic in the trajectory duration.

2.4 Duration and phases of the mission

As seen in the previous section, the launch will be performed in June 2025. We are considering Juno's trajectory for this mission, so obviously we will have the same steps to reach Jupiter. Furthermore, we are considering an 8 year mission duration: since it takes around 5 years to reach Jupiter, we want to have around 3 years to be able to make scientific measurements, because it is the main point of flying a spacecraft. The duration of the mission is as follows:

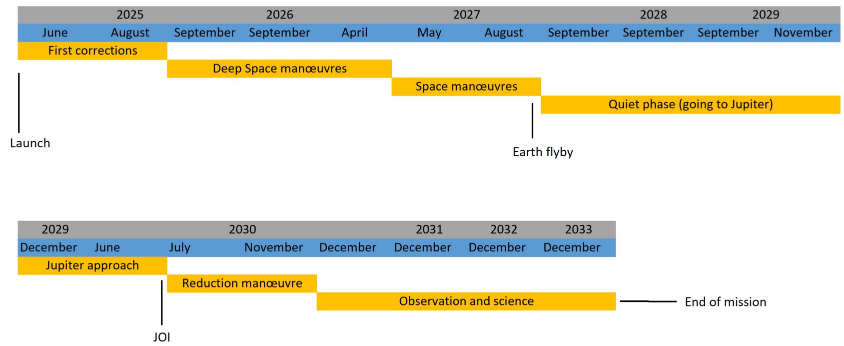


Figure 3: Phases of our mission

2.5 Mission class

Exploring Jupiter requires several years of work and huge infrastructure and technological costs. Furthermore, as we will discuss in a next section, orbiting Jupiter using solar panels adds an important challenge to our mission, from either the material selection as well as a different spacecraft structure. Taking in consideration the previous assumptions, we can estimate the mission as an high cost mission, which will cost between 1 and 1.5 billion $.

2.6 Redundancy

Redundancy is fundamental to decrease the risks of failure during a mission, especially a long term mission like ours. It is suggested that each subsystem considers redundant elements. For our spacecraft, all the engines (main and secondary) as well as the batteries are redundant in case of any problem (for example a strong radiation doses or an electronics failure).

2.7 Other constraints

The first and main important parameter that needs to be considering while orbiting around Jupiter is the planet's harsh spatial environment. It can irremediably affects all the spacecraft's components and damage considerably the electronics. The mission design shall have to consider the following parameters:

- harsh radiation environment
- material degradation: especially the electronics and the loss of efficiency in our solar panels
- the spacecraft needs to produce enough electrical power to be able to work properly during the orbital eclipses
- since it will require a huge amount of propellant and heavy motors and power generators, the launch vehicle shall be able to put the satellite on the correct trajectory to acheive its mission

2.8 Heritage

Since Jupiter has always been fascinating mankind, it is not surprising that many space mission had the objective to analyse the gas giant over the years. In this report, several space missions will be taken in consideration, but the Juno mission from NASA will be the main source of data and inspiration. Fortunately, there are several papers and scientific articles written about it, from NASA as well as other research groups.

3 Mission Design

3.1 Orbit selection

The first point to take in consideration is how to get to Jupiter. Since it is far away from Earth (almost 600 million kilometers at perigee), different trajectories can be planed and compared to find the most time and fuel efficient.

3.1.1 Hohmann Transfer

Considering a pure Hohmann transfer without orbit change in a first time, we need

$$V_d = \sqrt{(\mu_{Sun} * (\frac{2}{ae} - \frac{1}{a}))} \tag{1}$$

$$V_e = \sqrt{(\mu_{Sun}/ae)} \tag{2}$$

$$V_d^{\infty} = V_d - V_e = 8.7926 \ \ [km/s] \tag{3}$$

Thus, our C_3 would be 77.3096 $[km^2/s^2]$ to reach Jupiter.
At arrival, we would have to following velocity equations, where V_a is ur arrival heliocentric velocity and v_a^{∞} our arrival excess velocity:

$$V_a = \sqrt{\mu_{Sun}(\frac{2}{a_{Jupiter}} - \frac{1}{a})}; \tag{4}$$

$$v_a^{\infty} = V_{Jupiter} - V_a \tag{5}$$

Finally, we are now considering, since we are in Jupiter's sphere of influence, that we are not in heliocentric referential anymore but in Jupiter's referential. The δ_V that we need to consider is the substraction of the hyperbolic velocity we have when we arrive to Jupiter (i.e, at perijove) and the elliptical velocity we want for our final elliptical orbit. As we are going to discuss in the next subsection, we need to choose an orbit that avoids Jupiter's radiation belts at maximum. We are going to consider the following ellipse:

- The perijove is at 1.8*$R_{Jupiter}$ from Jupiter's center
- The apojove is at 6.5*$R_{Jupiter}$ from Jupiter's center
- The eccentricity is then $e = 0.5663$

These informations allows us to derive the ΔV needed for the orbit insertion

$$\Delta V = v_{hyp-perijove} - v_{ell-perijove} = \sqrt{(v_a^{\infty})^2 + 2\frac{\mu_{Jupiter}}{r_{perijove}}} - \sqrt{\mu_{perijove}(\frac{2}{r_{perijove}} - \frac{1}{r_{apojove}})} = 3.55[km/s] \tag{6}$$

3.1.2 Plane change

Furthermore, we need to change the orbit plane before reaching Jupiter due to its tremendous radiation belt that would considerably damage the satellite's electronics. We need to arrive to Jupiter with a polar orbit before receding to a point where the radiations are not strong enough to irremediably damage the electronics. An article of Jupiter's magnetosphere has assembled data from Galileo's mission around Jupiter to draw a graph of the high energy particles around Jupiter as a function of its radius:

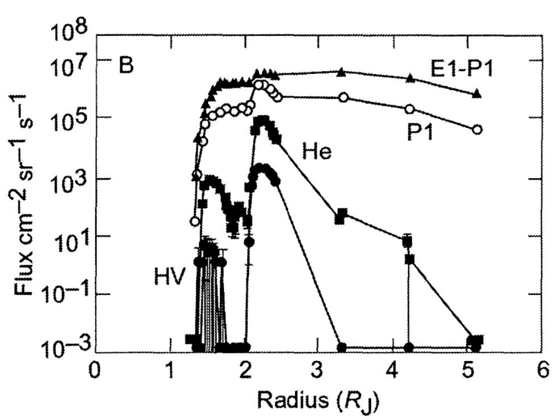


Figure 4: Flux of high energy particles around Jupiter, taken from [4]

We will consider the following approximations:

- the Hohmann transfer is performed near Cap Carneval (latitude $\Phi \pm 28°$)
- Earth's and Jupiter's equators are aligned

To perform a polar orbit around Jupiter we have chosen an elevation degree of $85°$. So, since we launch at $28°$, we will need $\alpha = 57°$. Furthermore, since we are gravitating in an elliptical orbit, the optimal change plane will be at the apojove, where the spacecraft's velocity is minimal. Thus, the ΔV required to perform a change of orbital plane is given by:

$$\Delta V_{orbitalchange} = 2V_{apojove} sin(\alpha/2) \tag{7}$$

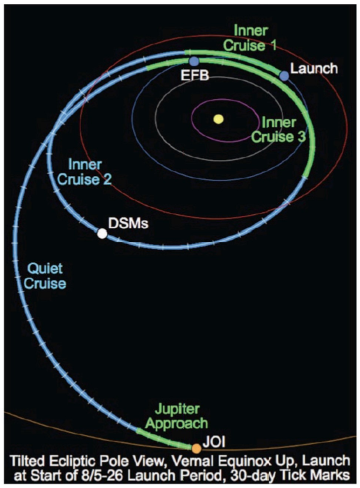
Combining all the conditions above we finally get $\Delta V_{orbitalchange} = 328.8[m/s]$, for a final $\Delta V = 3.878[km/s]$. However, regarding the previous result, the Homann transfer is clearly not the most fuel efficient trajectory to follow. As for NASA's Juno mission to Jupiter, we are going to take in consideration a trajectory with an Earth Gravity Assist.

3.2 Final orbit

From Figure 5 we can expect a maximal C_3 of 30.8 $[km^2/s^2]$ and a ΔV of 2.044 $[km/s]$ for Juno's mission on Jupiter, which is much less than what we did find from our transfer's calculations. This is due mainly to the Earth's gravity assist ($\Delta V - EGA$) performed during its mission and several simulations of the most fuel-efficient trajectory. We will consider $\Delta V = 2.044[km/s]$ and $C_3 = 30.8[km^2/s^2]$

Maximum C_3 (km^2/s^2)	Injected Mass (kg)	1st Day of Launch Period	ΔV (m/s)	Post-JOI Clean-up Mass (kg)
31.3	3590	8/12/2011	2044	1860
31.1	3605	8/12/2011	2044	1868
31.0	3610	8/12/2011	2044	1870
30.9	3615	8/12/2011	2044	1873
30.8	**3625**	8/12/2011	2044	**1878**
30.7	3630	8/11/2011	2061	1871
30.6	3635	8/11/2011	2069	1868
30.5	3645	8/11/2011	2078	1868
30.4	3650	8/11/2011	2087	1865

Figure 5: Mass comparison across entire launch period for various C_3 limits, taken from [5]



Earth–Jupiter trajectory

Figure 6: Juno's orbit to reach Jupiter, taken from [6]

We have taken Juno's trajectory as a model for its fuel efficiency, as discussed in the previous parts. However, our spacecraft's final orbit won't be the same, it will have the following parameters:

Parameter	Symbol	Value
Semi-major axis	a	6.5*$R_{Jupiter}$
Eccentricity	e	0.5663
Inclination	i	85°
Ascending node	Ω	TBD
Argument of perigee	ω	TBD
True anomaly	Θ	TBD

3.3 Eclipses, orbital period, coverage

3.3.1 Eclipses

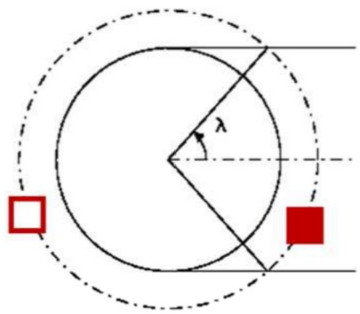


Figure 7: Eclipse angle

Considering that the light coming from the Sun arrives in straight lines, we can deduce the angle λ during which the satellite will be in the shadow. Considering the maximum radiation avoiding case ($perijove = 0.8 * R_{Jupiter} + R_{Jupiter}$). We have then, at hte perijove:

$$\lambda = arcsin(\frac{R_{Jupiter}}{r_{perijove}}) = 33.749[deg] \tag{8}$$

$$T_{eclipse} = \frac{2\lambda}{360} * T_{orbit} = 9.18[h] \tag{9}$$

3.3.2 Orbital period

As discussed in the previous part, we do consider an elliptical orbit where

- The perijove is at 1.8*$R_{Jupiter}$ from Jupiter's center
- The apojove is at 6.5*$R_{Jupiter}$ from Jupiter's center
- The eccentricity is then $e = 0.5663$

Thus, the orbital period is given by

$$T_{orbit} = 2\pi\sqrt{\frac{r^3_{apojove}}{\mu_{Jupiter}}} = 48.96[hours] \tag{10}$$

3.3.3 Coverage

As discussed in the previous parts, one of the simplest ways to avoid radiation as much as possible is to orbit around Jupiter following a polar orbit, i.e having an orientation angle of almost 90°respect to the equatorial plane. Furthermore, the polar orbit allows us to have a close look to both of the poles and generally of all Jupiter's surface.

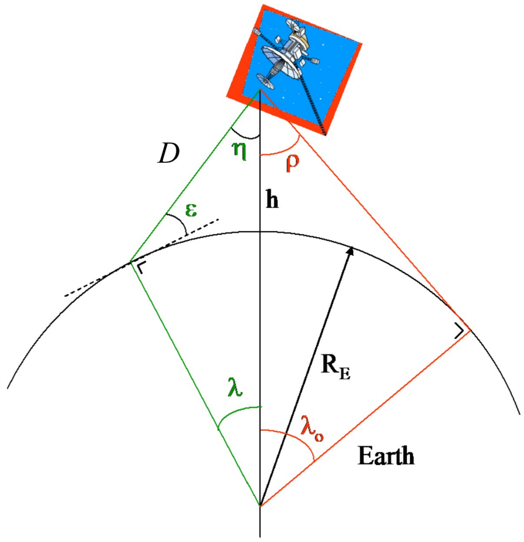


Figure 8: Jupiter's coverage, taken from the lecture

The instantaneous footprint area, which quantifies how much of Jupiter's surface we can see at one particular moment from our chosen orbit, is given by:

$$F_A = \frac{\pi}{4} L_F W_F = \frac{\pi}{4} D \frac{sin(\eta)}{sin(\epsilon)} R_{Jupiter} arcsin(\frac{Dsin(\eta)}{R_{Jupiter}}) \tag{11}$$

We are going to make, at this point, some heavy approximations. We are considering $\epsilon = 20°$, $\eta = 30°$, which matches more or less with our instrument's FOV. With those values, we obtain $F_A = 3.593 * 10^9$ $[km^2]$, which may seem too much but considering the immense size of Jupiter and that we are more than 10^5 km from its surface it is "only" one twentieth of the planet's area.

3.4 Planetary insertion

The planetary insertion will be performed the following way:

- the corrections done during the manoeuvres will put the spacecraft in a polar orbit
- the spacecraft will perform an aerobraking when Jupiter's sphere of influence is reached (ΔV_{main})
- further corrections will be performed to keep the orbit around Jupiter (perijove raise, counteract the ascending node drift etc...). These corrections are included in the ΔV budget

3.5 Launch Vehicle selection

We shall consider for this mission a $\Delta V = 2.044[km/s]$ and a $C_3 = 30.8[km^2/]$, and Figure 9 gives an overall aspect of the possible launch mass respect to the C_3 obtained.

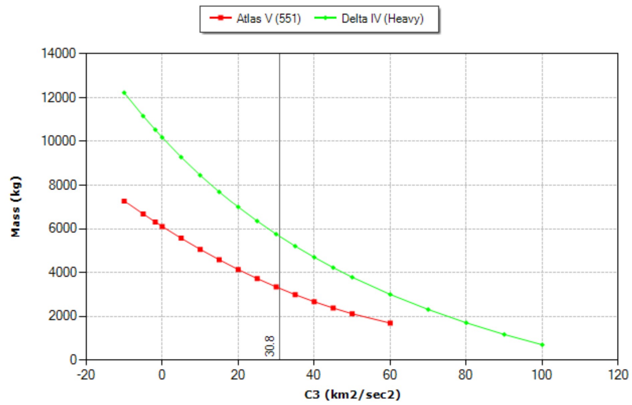


Figure 9: C3 spacecraft performances

From Figure.9, website [3] and the Juno mission overview, the best choice for reaching Jupiter via a Earth gravity assist transfer seems to be the Delta IV Heavy spacecraft. The Delta IV Heavy launch vehicle uses a 1194-mm payload separation ring with 1575-5 payload attach fitting (PAF) as adapter. From the Delta IV website, we know that the Delta IV Heavy offers *a 5m diameter elongated carbon-composite bisector or metallic trisector PayLoad Fairing*. At $C_3 = 30.8[m^2/s^2]$, the Delta IV Heavy is capable to lift up to 5670 [kg].

3.6 Launch site

The launch is performed from Cape Canaveral Air Force Station Space, Florida, USA which is the simplest launch site election since it's hosting Delta IV series launch vehicles for space missions.

3.7 Mission operations scenario at mission level

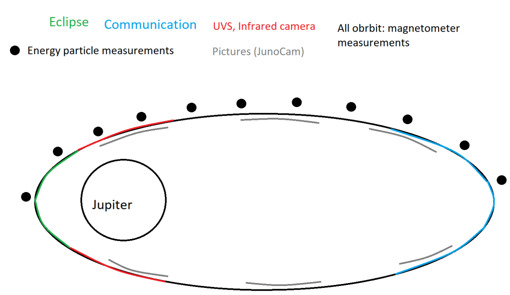


Figure 10: Different phases during an orbit

. . . To be continued . . .

3.7.1 Time in view of a ground station

The spacecraft will perform its mission in deep space. Thus, it will communicate with the ground stations on Earth by the Deep Space Network. This network is the best communication option for deep space missions because since it has three ground stations located at 120°on Earth, the spacecraft will be able to communicate with at least one of those three at any moment.

4 Propulsion subsystem

Figure 5 gives us an overall aspect of the injected mass for various C_3 limits for Juno's mission. These injected masses are for the Atlas V 551, the same launch vehicle we are using for this project. We will consider Juno's mission injected mass of 3625 kg and that the maximal acceleration is $0.1 * g$ $[m/s^2]$, so $m_{wet} = 3625[kg]$.

4.1 Spacecraft size

From the SMAD, we know that we can give a proper estimation of the spacecraft size once we know m_{wet}. We can calculate the following parameters:

Characteristic	Estimate
Volume	$V = 36.25[m^3]$
Linear dimension	$s = 3.8404[m]$
Body area	$A_b = 14.7486[m^2]$
Moment of inertia	$I = 8.554 * 10^3[kg * m^2]$

4.2 Number of thrusters

The first step is to determine how many thrusters the spacecraft will need. from the article [7], this number is given by the equation [7]:

$$J \geq 1 + 2LR + D \tag{12}$$

where LR is the redundancy we need (or want) and D the dimensional space in which the spacecraft will perform. In space, we will have $D = 6$ since the spacecraft has 6 degrees of liberty. If we choose a redundancy of $LR = 2$, we will need at least 11 thrusters. We will go for a total of 14 thrusters: two main thrusters for the consequent burns (aerobraking) and 12 for the torque and attitude control. Furthermore, from the article [7] which compares 14 different thruster configuration, we have chosen the following configuration:

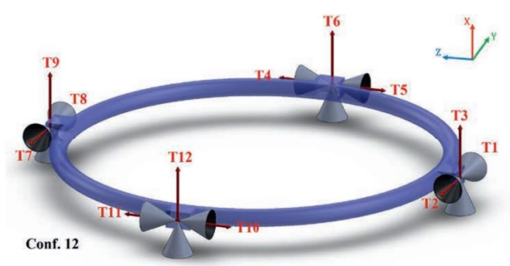


Figure 11: Thrusters configuration, taken from [7]

The preliminary thrusters configuration is given by the following figure, where we have our 14 thrusters, two main engines and 12 side thrusters for torque and attitude control, following Figure 11 spatial configuration.

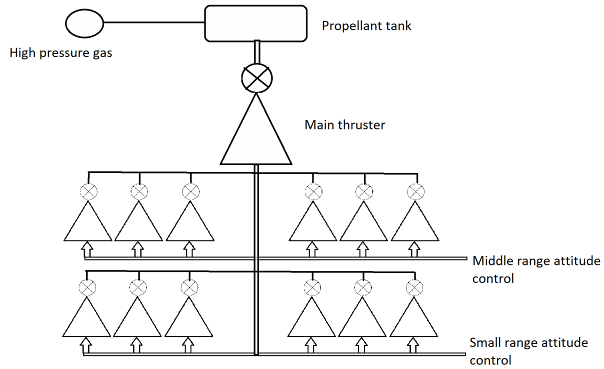


Figure 12: Propulsion subsystem diagram

Note: we are considering only one main propulsion engine in this hardware schematic, even there is one more for redundancy.

4.3 Propellant and engine choice

4.3.1 Main engine

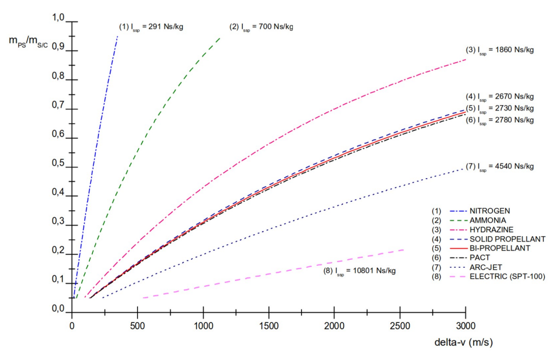


Figure 13: Mass ratio as a function of ΔV

From literature and Figure13, the best choice seems to be liquid propellant, preferably bipropellant. Indeed, since many burns are going to be done to reach the optimal orbit around Jupiter, the solid propellant is discarded. Furthermore, electrical propulsion seems to be a good option but it needs a much more complex trajectory that takes more time. We will first choose the R-42 890N (200 lbf) Bipropellant Rocket Engine by Aerojet, compute our calculations with its data and then check if it fills all the spacecraft constraints. Indeed, it has the following characteristics that match quite well our mission design:

- high thrust, in the same range of value that the thrust used for the Juno mission (890 [N])
- high Isp (305 [s])
- low engine mass (4.53 [kg])
- bipropellant: MMH/NTO(MON-3)
- flight proven (but currently in production)

We will consider 2 main engines: one in function and the other for redundacy.

4.3.2 Secondary engines

From Figure 11, we already know how we will configure the disposition of the secondary thrusters. From the lecture and the SMAD, we know that the corrections will not take a high importance in the final ΔV. We will consider that, taking in account our resulting ΔV, the correction will imply up to 200 $[m/s]$ of ΔV and the burns will not last more than a few minutes, as discussed in previous sections. Thus, we will need thrusters with a low nominal thrust. From [8], a good choice would be the MONARC-22-6 Monopropellant Hidrazine developped by Moog. Indeed, its thrust is 22 [N], is light (720 [g]) and has a satisfyingly $I_{sp} = 240$ [s]. From [9] and equation 12, *it is common to use a set of 3 wheels, with their spin axis not co-planar, to provide three-axis attitude control. To provide redundancy in the design, often a fourth is added*, as seen in Figure 12.

4.4 Mass calculations

We are now able to derive the mass budget required for our mission. Since we are taking the ΔV needed for the Juno mission, we do not know how many burns the spacecraft did in its entire mission (which is still going on). We are supposing the following approximations: the article about Europa exploration [10] gives us a good estimation on the corrections performed to reach Europa. Obviously, since we are not going to Europa, we will consider the following correction.

Activity	ΔV [m/s]
Deep Space manoeuvres	152.9
Earth Biasing	55
JOI	1189.1
Gravity Losses	29.7
Perijove raise	821.7

Which gives us a total $\Delta V = 2248.4[m/s]$ that corresponds to the ΔV found for Juno's mission plus a required 10% margin.

The Tsiolkovsky equation gives us:

$$m_{prop} = (m_{wet}) * (-exp(\frac{-\Delta V}{g_0 * I_{sp}}) + 1) \tag{13}$$

Iterating over the different corrections made (each time subtracting the mass obtained to m_{wet}), our final propellant mass would be $m_{prop} = 2071.5$ [kg].

The dry mass of the spacecraft is then given by:

$$m_{dry} = m_{wet} - m_{prop} = 1553.5[kg] \tag{14}$$

Thus, the maximal thrust supported is given by

$$F = m_{dry} * 0.1 * g = 1524[N] \tag{15}$$

From these results, our motors seem to be a good option. The total m_{dry} is in the same range as Juno's. Furthermore, the maximal thrust we get is 1524 [N], so with 890 [N] from the motor we are below this value.

4.5 Tanks sizing

The final step in the propulsion subsystem design is to choose which tanks we will use to keep our propellant. From the data sheets presented in [11] and [12], we will need two kind of fuel tanks: one for the main engine and another kind for the secondary thrusters. A good tank choice for the main engine would be the SURFACE TENSION PROPELLANT TANK OST 22/X. Indeed, it can contain up to 1100 liters of propellant, weights less than 50 kg and can endure a pressure up to 26 Pa, which is beyond the pressure that will be operating on our spacecraft during its mission. Since we will need more than 1 ton of propellant for the JOI burn and perijove raise, we will have to take two of those tanks to fulfill correctly the mission (one for the propellant and one for the oxydizer).
Note: as the perijove raise implies an important propellant correction, we will consider the propellant used will come from the main tanks. Even if it is not totally correct, it would imply otherwise to have too many secondary tanks.
For the secondary thrusters, our choice will be orientated towards the EPDM - BLADDER TANK BT 01/0. Indeed,

it can carry almost 40 liters of propellant, has the same maximal pressure than the main tanks and it is relatively light (less than 8.5 kg).
Having a total correction propellant mass of 334.3 [kg], we will then need 9 secondary tanks.

4.6 Propulsion system dry mass

The empirical formula for the propulsion subsystem is given by:

$$mass_{propulsion} = 26.4 + 0.077 * m_{prop} = 185.9[kg] \tag{16}$$

The last step is adding the error margin. The "worst case" would be adding 25% of $mass_{propulsion}$, giving:

$$mass_{propulsion} = 232.38[kg] \tag{17}$$

Now, we will compare with our subsystem design's weight, which contains:

- 2 main engines
- 12 secondary engines
- 2 main tanks
- 9 secondary tanks

and weights a total of 187.16 [kg], so, as we can see, the predicted propulsion weight is almost the same as the weight we did find with our chosen components and, therefore, it fits in the interval of [$mass_{propulsion} \pm 25\% *$mass$_{propulsion}$].

4.7 Duration of burns

The duration of the burns are given by the following equation, considering that it is the first burn and the mass is m_{wet}:

$$\Delta T = \frac{m_{wet} g_0 I_{sp}}{F} * (1 - exp(\frac{-\Delta V}{g_0 I_{sp}})) \tag{18}$$

Our total burn duration will end as it follows:

1. ΔT_{JOI} = 1436.00 [s]
2. ΔT_{corr-i} = 329.99, 118.38, 42.65, 707.07 [s] for $i = 1, 2, 3, 4$
3. ΔT_{Total} = 2634.1 [s]

Once again, since the R-42 890N Bipropellant Rocket Engine has a Steady State Firing (Single Firing) of 3940 [s], it is beyond our worst case calculations, so this engine is indeed a good motor choice. For the secondary engines, since it was commented before, the correction burns will not last more than a few minutes, so the secondary engines are well designed to endure these burns as well.

4.8 Spacecraft heating

For the main engine and a 28V standard bus, the valve needs $46[W]$.
The secondary engines need $30[W]$ to activate their valves. Taking in consideration the fact that we will perform only attitude manoeuvres or orbit corrections, they won't be all activated at the same time. A good approximation for the worst case is to say that we will activate up to 6 of them at once, therefore they would need a total of $180[W]$ at maximum.

4.9 Final propulsion mass budget

To resume all the propulsion subsystem in a table, we have

Element	Mass [kg]
m_{wet}	3625
m_{prop}	2071.5
m_{dry}	1553.5
$m_{main-engine}$	2*4.53
$m_{secondary-engines}$	12*0.72
$m_{main-tanks}$	2*49
$m_{secondary-tanks}$	9*8.5

5 Power subsystem

Since Jupiter is five times further from the Sun than the Earth, it receives 25 times less sunlight, and, thus, solar energy. However, based once again on Juno's mission to orbit Jupiter and technology improvements, we are now able to say that making a "green" spacecraft powered by solar energy is possible, even around Jupiter.

5.1 Power source

As discussed before, previous mission show that solar energy was used to power Juno spacecraft around Jupiter. In this project, we are going to try to approximate as much as possible the power subsystem with our own calculations. So our spacecraft will be powered by **solar arrays**. In the next sections the power budget is going to be determined and all of its main components defined and sized.

5.1.1 Requirements and constraints for power subsystem

At this phase of the project, we can consider the following requirements:

- mission lifetime of around 8 years
- during eclipses, our science instruments won't perform measurements
- the average power needed is considered to be 300 [W] during the day and 270 [W] during eclipses
- the orbit altitude and the eclipse duration are given in the previous sections

5.1.2 Power provided in a full orbit

The amount of power the solar array must provide during daylight to power the satellite during the entire orbit is given by the following equation:

$$P_{sa} = \frac{k}{T_d}(\frac{P_d T_d}{\eta_d} + \frac{P_e T_e}{\eta_e}) = 448.80[W] \tag{19}$$

where we consider P_{day} = 300[W] and $P_{eclipse}$ = 270 [W]

5.1.3 Solar cells selection

Once we have the power that needs to be generated by our olar panels we can start to dimension them. We have to take in account several parameters that reduce the efficiency:

- degradation over the years in orbit
- angle with the Sun
- initial efficiency
- temperature

All of these parameters have been "confined" in the Inherent Degradation, that equals in general around $I_d = 0.77$. A good solar cells choice would be the triple junction GaInP/GaInAs/Ge SC solar cell, according to [14]. From Figure 14 we get the initial efficiency (i.e at BOL) and the decay of efficiency over the years, for a 3 MeV radiation. We suppose that, in our polar orbit, that we can avoid the main radiation belts (i.e the maximum radiation would be around 3MeV), so that Figure 14 gives accurate values for our spacecraft mission. Furthermore, multijonctions are designed to resist 33 years to 1MeV radiation and up to 6 years to 10MeV radiations, which is beyond our mission's lifetime.

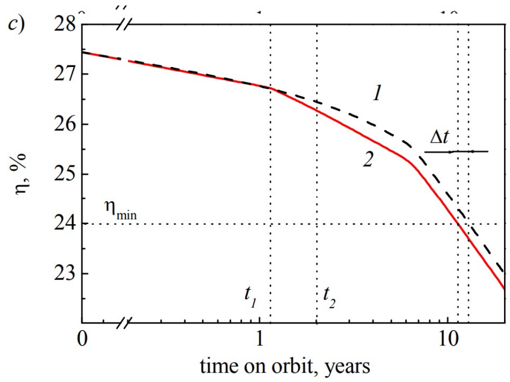


Figure 14: Efficiency degradation for triple-junction solar cells, taken from [14]

Thus, at the beginning of life (BOL), we will have a power given by:

$$P_{BOL} = P_o I_d cos\theta = 9.64[W/m^2] \tag{20}$$

where P_o is the ideal solar cell output performance, given by the solar constant in Jupiter times the efficiency of our triple-junction solar cell and $\theta = 25°$is the Sun inclination angle.
Over the years in orbit, we will have a degradation. From Figure 14 we can see that we loose 2.5% of efficiency over 8 years. Thus, the end of life (EOL) output power is given by

$$P_{EOL} = P_{BOL} L_d = P_{BOL}(1 - degradation/yr)^{satlife} = 9.399[W/m^2] \tag{21}$$

The last step is to determine the area of the solar array needed to produce this power. It is given by

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} = 47.75[m^2] \tag{22}$$

To compare, if the spacecraft was launched around Earth (i.e the solar constant would be 1361 $[W/m^2]$) we would only need $A_{sa} = 1.77[m^2]$ for the same solar cells and trajectory.
Note: at launch, the solar panels will be folded to improve the aerodynamics and thus giving an optimal ΔV and also fit in the LV's upper stage.

5.1.4 Area offset and moment of inertia

The SMAD gives us some formulas to take in account the area offset and the moment if inertia of our solar arrays. The results have to be added to the spacecraft size seen in Table **??**. We will consider that we have 2 rectangular solar panels. The solar panels will greatly increase the spacecraft's moment of inertia, particularly about the axes perpendicular to the array axis.

Characteristic	Value
Solar array area offset	$L_a = 8.20[m]$
Solar array moment of inertia $[kg * m^2]$	
Perpendicular to array face	$I_{ax} = 855.36$
Perpendicular to array axis	$I_{ay} = 831.48$
About array axis	$I_{aa} = 23.88$

5.2 Energy storage and battery sizing

In a first step, we need to consider that we are doing a mission with solar arrays powering our entire spacecraft and that we will have some eclipse time during our orbit. Thus, the choice of a primary and secondary battery is primordial in our power design. We are considering a second order redundancy for the batteries: we will put a redundant battery for each one we choose.

5.2.1 Primary battery

From the SMAD, the best primary battery choice is a lithium battery. Indeed, it has a higher specific energy that zinc or thermal batteries and is conceived for longer applications. For now, we will consider taking a lithium monoflouride battery, because it has the longest life, a low rate and a high specific energy density. Furthermore, it will work perfectly for memory backups. Those will be performed after the JOI, when we are in the elliptical orbit and start collecting scientific data. We can, for example, imagine that we put a binary trigger on the battery that is activated when the data rate reaches a certain threshold, meaning that we are starting to collect the data with the measurement instruments and activates the battery for the memory backups once we collect interesting data.

5.2.2 Secondary battery

For the kind of mission we are performing, the choice of the secondary battery may be even more crucial than the primary battery. Indeed, even if the secondary batteries have much lower specific energy densities, their ability to be recharged make them ideal for backup power in a solar cell powered spacecraft, where power will be needed during eclipses.

Following Juno's mission system design choices, we are going to take a Lithium-ion battery. Indeed, it has a high specific energy density (100–265 $[Wh/kg]$), lasts between 400 and 1200 cycles for a DoD of more or less 80%, according to [15] for 1000 battery cycles (the lifetime of a Lithium-ion battery is around 1200 cycles).

5.2.3 Battery capacity

The last step in battery sizing is indeed to determine its size, i.e its capacity. It is given by the following formula:

$$C_r = \frac{P_e T_e}{DOD * N * n} = 1.721[kWh] \tag{23}$$

where P_e and T_e have already been given for the P_{sa} calculation. N is the number of batteries (2 for our spacecraft) and n the transmission efficiency between the battery and the load (we are considering $n = 0.9$ here).

5.2.4 Battery charging

From the SMAD, we know that for missions during more than 5 years, the batteries charging must be done independently. We will then charge our two batteries separately.

5.3 Power regulation and control

5.3.1 Electrical control subsystem

From previous parts, we have seen that the power needed at EOL is not very far away from the power needed at BOL. Furthermore, since our mission lasts around 8 years, the best power regulation subsystem seems to be, according to the SMAD, a Direct Energy Transfer (DET) power regulation subsystem. Indeed, in the direct energy transfer, a shunt regulator will operate in parallel to our solar array and will shunt its current away when there is no power needed from the batteries or the loads. Furthermore, this system implies lower mass, fewer parts and higher efficiency at EOL respect to a Peak Power Tracking system. At last but not at least, the bus voltage control, if it is quasi regulated or fully regulated implies a strong electromagnetic interference if the electrical subsystem is controlled by Peak Power Traking. Since the spatial environment is already tough enough around Jupiter, it does not seem a good idea to add more interference inside the spacecraft itself.

5.3.2 Electrical bus voltage control

For the bus voltage, we are considering a standard 28V transmission bus. Indeed, from the SMAD, *most spacecrafts have demanded low power* $< 2000W$ *so power distribution has relied on a standard, 28V bus*, so we can deduce that our spacecraft will be adequate for a 28V standard bus.
The second step is to decide if our bus voltage will be controlled by a regulated, quasi regulated or unregulated technique. At this point of the design, we do not know enough about the command and telecommunication system to determine with certitude which control method is the most adapted to our system. So, for now, we are supposing that our spacecraft will be controlled by a fully regulated bus. We do know however that our measure instruments will need a highly regulated power, during either daylight and eclipses to work properly and give us the data we need. The best choice seems then a fully regulated bus, even if it becomes a more complex power system. The The power regulation subsystem will look as the following diagram, for a fully regulated bus in a Direct Energy Transfer system:

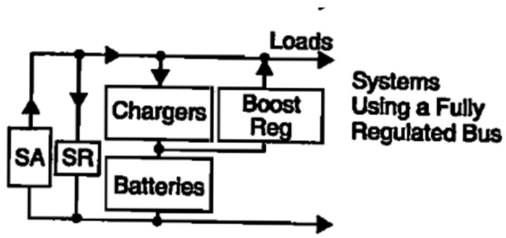


Figure 15: Power regulation technique (taken from SMAD)

5.4 Mass and power budget

From the SMAD, p334 we have all the required elements to give a mass and power budget for our spacecraft's power subsystem, considering a required power of 390W.

Component	Weight (kg)	Power (W)	Comments
Solar arrays	12	448.80	Produced power (P_{sa})
Batteries	4*38.25	-	For 4 NiH_2 batteries
Power control unit	6	-	-
Regulators/Converters	9.75	60	Converted power
Wiring	62	15	Taking worst case

6 Science and instruments

6.1 Scientific objectives

At this point we already know that our mission is heavily inspired by the Juno mission. For our mission, the main objective is to provide a complete (or at least try) analysis of the Jupiter's aurorae, as discussed previously in the introduction. Our polar orbit fits perfectly this purpose, since we will have a good view of both Jupiter's north and south poles.

6.2 Jupiter's aurorae

There is a main difference between the aurorae on Earth and the ones on Jupiter. On Earth, they are produced when the charged particles coming from a solar eruption collide with the magnetosphere. They can be canalized by the Earth's magnetic field in the extreme North or South of the globe. When the collisions occur, the charged particles will excite the atoms in the ionosphere. The excited atoms will not remain in the excited state and, to return to a non-excited state they will emit a photon in a characteristic wavelength (which depends of the atom), creating the aurorae.
On Jupiter, the aurorae do not depend on the "solar weather". Jupiter, due to its fast angular velocity (one rotation around itself in 10 jours) and it's enormous mass, creates a strong electric field, like a magnet. These electric fields will attract the charged particles emitted by satellite Io's strong volcanism, creating continuous and powerful aurorae. Since the particles do not emit radiation is the visible spectrum, the majority of the aurorae in Jupiter can only be seen with UV or X-ray instruments.

6.3 Desired science instruments

As discussed in the previous part, we will need instruments allowing us to see phenomena in shorter than the visible wavelengths. Juno's Ultra Violet Spectrograph (UVS) has already provided satisfactory data and images of the aurorae, so we will rely on the data provided on this instrument. Furthermore, it would be interesting to measure a set of Jupiter's characteristics, such as magnetic field, gravity or the energetic particles, since they all have an influence on the aurorae phenomenon.

- The magnetosphere in Jupiter was formed due to its fast rotation around itself. It is the biggest magnetosphere in the solar system (Sun excluded) and the currents inside it produce the aurorae, as seen previously. Furthermore, the magnetosphere action traps and accelerated the particles, creating the intense radiation fields that the spacecraft will try to avoid with its elliptical orbit. It is then interesting to provide the spacecraft with a magnetometer to be able to quantify the magnetosphere. From Juno's mission, we know that the magnetometer of the spacecraft measures the strength and the direction of Jupiter's magnetic field lines It relies on magnetic sensors mounted on the magnetometer at one of the spacecraft's extremities, to avoid the instruments from confusing the spacecraft's magnetic field with Jupiter's one. *The spacecraft magnetic field is further separated from Jupiter's field by the use of two magnetometer sensors: one 10 meters from the center of the spacecraft and one 12 meters from the center. By comparing measurements from both sensors, scientists can isolate the magnetic field of Juno from Jupiter* [6].
- To be able to measure and see properly the aurorae, we are going to use and infrared camera. Indeed, we know that the aurorae can be seen mostly in small wavelengths (UVs and X-rays) but they can also be seen in infrared wavelengths. However, the main objective of the infrared camera is to probe the upper layers of Jupiter's atmosphere down to pressures of 5–7 bars, to determine it's "below cloud" and inner composition. It will finally broadens our "vision range", adding the infrared spectrum to it.
- It is also interesting to measure the charged particles around Jupiter. Juno's energetic particle detector will measure the particle beams going through space and how they interact with Jupiter's magnetic field.
- For aesthetic and obvious reasons, we will add a camera on the spacecraft to take pictures during the mission.
- Another idea is adding an X-ray camera to be able to collect some data that the previous instruments cannot get (i.e, particle emissions in the X-ray range of wavelength). Unfortunately, based on the Chandra X-ray observatory mission, we can say that our spacecraft is not well fit to carry this kind of instrument. Indeed, it would need to be much more massive and long to be able to carry all the parabolic and hyperbolic mirrors that allow to focus the X-rays in a tiny point to analyse them.

6.4 Instruments characteristics

Note: all of our instruments, based on the ones used for Juno's analysis of Jupiter, have very low power consumption. Indeed, the solar panels cannot produce a considerable amount of power that far from the Sun, so the instruments had to be optimized to consume as less as possible.

6.4.1 Ultra Violet Spectrograph (UVS)

As discussed before, the UVS is the key instrument of our mission, since it is the perfect instrument to see the aurorae on Jupiter. The article [17] gives us an overall aspect on the UVS used onboard on the Juno mission. The main characteristics are show below.

Characteristic	Value
Mass	21.6 kg
Dimension	44x24x22 + 13x6x9 cm^3
Power	9.8 W
Spectral Bandpass	68–210 nm
Field of view	3 segments projected onto the sky: 0.2°x2.5°, 0.025°x 2°, 0.2°x 2.5°
Data volume	1810 Mb per orbit

The spectrograph is composed by two main parts: one is the optical particle detector and the other is the electronic box processor. Light enters the UVS telescope via a flat scan mirror and passes through an input aperture. The scan mirror is used to target specific auroral features by moving the mirror up to +/-30°. The light is then collected, focused and coated to maximize the wavelengths superior to 100 nm (i.e, as a low pass filter that will cut the highest frequencies). The light is then dispersed and passes through the detector, to finally obtain the data that we can analyze. All the UVS, as well as the other instruments, is shielded to protect it from the heat and the radiation (this will be discussed later).

6.4.2 Magnetometer

The Juno magnetometer (MAG) is going to be used, as discussed previously, to give an accurate magnetic field map of Jupiter, determining Jupiter's interior dynamics and give a representation of Jupiter's polar magnetosphere, according to [20]. These functions will be acheived by employing two Flux Gate Magnetometers and two Advanced Stellar Compasses to provide accurate data on the orientation and location of Juno's magnetometers, as depicted in Figure 16.

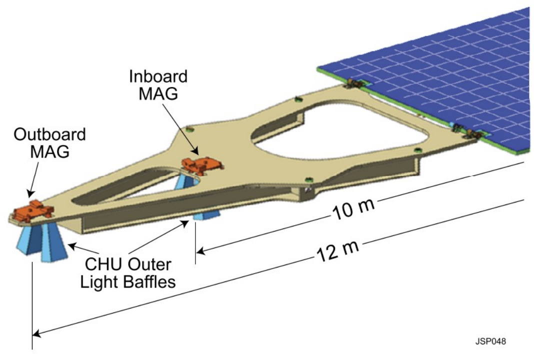


Figure 16: Magnetometer disposition on Juno

The FluxGate Magnetometers, according to [21] are mounted in the remote part of the solar arrays. They are composed by high magnetic permeability ferromagnetic: indeed, the nonlinearity of the magnetization properties of these materials give an indicator of the local field strength.

The compasses will give us information on where the spacecraft is pointing, by providing data to a Data Processing Unit. In Juno mission, they are able to provide an accuracy of one arc second.
Due to the impossibility to find any Juno MAG datasheet, we are basing our characteritic table on [22], which provides us data on a different magnetometer which is however, very similar to the one used for the Juno mission.

Element	Value
Mass	3.3 [kg]
Power	4 [W]
Instantaneous bandwidth (kbps)	4
Telemetry interface	SpaceWire
Field of view	N/A

6.4.3 Infrared Auroral Mapper

Juno uses the Jovian Infrared Auroral Mapper (JIRAM) to be able to perform infrared measurements on Jupiter. JIRAM combines 2 data channels in one instrument: the imager and the spectrometer, which are housed in the same optical subsystem. It has the main characterics depicted below, based on the article *JIRAM, the Jovian Infrared Auroral Mapper* [24]:

Element	Value
Mass	12.9 kg w. shielding
Power	16.7 W
Observation range	2–5 micron wavelength
Data volume (compressed)	0.3 Mbits (L-M bands)
Frame size	256 × 432
FOV (telescope)	13.7x13.7°
FOV (imager)	3.66x6.24°

Similarly to the UVS, the infrared camera is composed by two instruments: a spectrometer and a telescope. The light enters into the telescope, is focused and passes into a beam splitter that creates two optical paths and allocates one in each focal planes. As the reflected separated beams enter in the spectrometer, it will apply a different pass-band filter to each beam: L-Band at 3.3 to 3.6 micrometers and M-Band at 4.55 to 5.05 micrometers (i.e, infrared domain). We are now able to get, with those filter's results, a 2-D infrared picture. The whole light treatment is show in Figure 17.

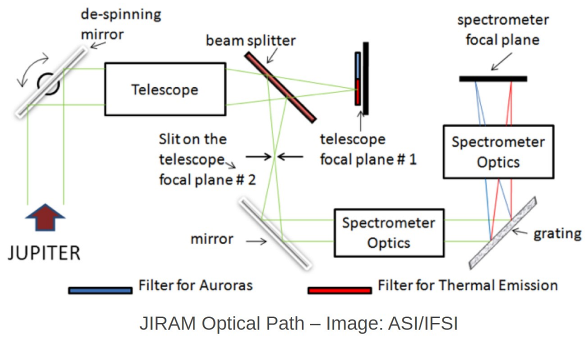


Figure 17: Infrared Auroral Mapper

6.4.4 Energy particle detector

The Energy Particle Detector instrument will measure energetic particles and their interaction with Jupiter's magnetic field, investigating Jupiter's polar space environment with special focus on the physics of the intense Jovian aurorae. Energy Particle Detector measures the energy, spectra, mass species (H, He, O, S), and angular distributions of the higher energy charged particles. From [23], we get the following characteristics for the energy particle detector.

Element	Value
Mass	6.4 [kg]
Power	3.1 [W]
Volume	23.3 × 16.9 × 12.8 cm
Data rate	16 kbps
Data volume	1780 Mb per orbit
Field of view	160°× 12°
Angle resolution	26.7°x 12°

The energy particle detectors allows to measure a wide range of electron/ion energies. The results are shown below (credits to NASA JPL).

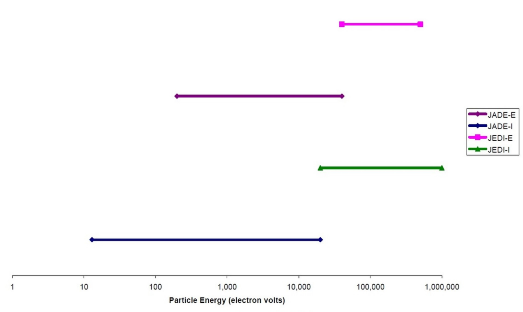


Figure 18: Detected particles as a function of their energy

6.4.5 JunoCam

The easiest way to proceed is to take the camera that is integrated in the Juno spacecraft. It is called JunoCam and has the following characteristics (the data is taken from [18] and [19]).

Element	Value
Mass	2.642 kg
Power	4.7 W—idle, 5.9 W—imaging
Dimensions	3.8 × 3.9 × 7.5 in Camera Head
Field of view	58°; 0.7 mrad/pixel
Picture rate	2 per minute
Spectral range	400-900 nanometers
Spectral filters	3 RGB color, 1 methane [878-899 nm])
Location	Side of spacecraft
Image size	1600 x 4800 pixel 3-color image

6.5 Pointing requirements

The science instruments for science experiments purposes are the main reason the mission is flown. For this reason, the science instruments have to be perfectly calibrated, and thus, need to have a very accurate pointing. Furthermore, due to the strong doses of radiation around Jupiter, the pointing must be very accurate to be able to perform measurements during the 3 years orbiting. Finally, to be able to be as precise as possible, the instruments will be disposed, if possible, in the front face of the spacecraft (i.e facing Jupiter) or in the solar panels (for example the magnetometer).

6.6 Data volume and data rate

The data in Juno's mission is transmitted and received by antennas connected to the Deep Space Network. It has an inboard computer capable of transmitted up to 50Mbps of data. In our spacecraft, the transmitted data rate (downlink) will be dependent on where the spacecraft is situated in the orbit (i.e what is its distance respect to Earth) and also how many subsystems are used at the same time. Indeed, the most important transmitted data will come from the instruments results, and this will depend on how many science experiments are operating simultaneously (for example, during the eclipses there is no need of taking pictures with the camera). From Figure 19 we can expect a downlink data rate of at least 40kbps, which is coherent with the data rates we have found in our instruments.

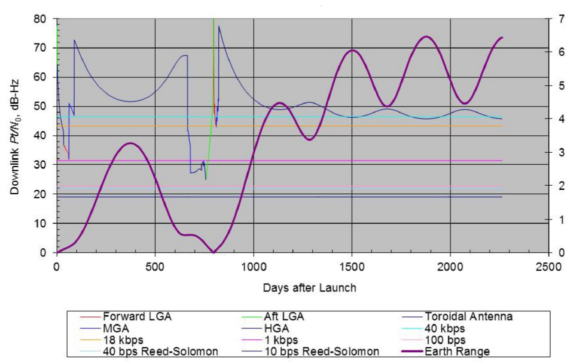


Figure 19: Juno's downlink performance, taken from [25]

6.7 Strategy for data collection

It is inefficient and not recommended to send all the data we get instantaneously. For this end, we will distinguish two main phases: one where the satellite is able to communicate with the Earth (i.e day time) and the other when it cannot (i.e during eclipses). Obviously, the data transmission rate will be higher during the time we communication is operating, so the strategy would be to send all the data collected with the high rate measurement instruments, for example the Energy Particle Detector or the UVS. During eclipses we will send low bit rate information, so the critical data (for example orbit correction or emergency signals) can be sent at any time. Note that our primary battery can storage information and send it at any time. We can count on the battery to reduce the size of the needed transmission buffer (for example, storage the pictures taken with the camera that do not have science or critical importance to send them later).

7 Mass budget

Before computing the mass budget, wee need to calculate the launch vehicle adapter mass. It is given by the empirical formula :

$$m_{LV-adapter} = 0.0755 m_{launch,max} + 50 = 478.09[kg] \quad (24)$$

where our $m_{launch,max} = 5670[kg]$ for a Delta IV Heavy.
Furthermore, the total wiring mass is given by

$$m_{wiring} = 10\% m_{dry} \quad (25)$$

Subsystem	# elements	Mass [kg]	Contingency %	Contingency [kg]	Mass w. contingency [kg]
Payload		46.84	5%	2.342	49.18
UVS	1	21.6			22.68
Magnetometer	1	3.3			3.465
EPD	1	6.4			6.72
Infrared Cam	1	12.9			13.545
JunoCam	1	2.642			2.7741
Propulsion		166.7	20%	33.34	200.04
Main engine	2	9.06			9.97
Secondary engine	12	8.64			10.368
Main tank	2	98			117.6
Sec. tank	6	51			61.2
Power		178.507	10%	17.85	196.358
Solar arrays	2	12			13.2
Batteries	4	153			168.3
PCU	1	6			6.6
Regulators	?	7.5			8.25
Propellant	-	2102.6	10%	210.26	2309.72
Nominal		2071.5			2278.65
Residual 1.5%		31.07	-		-
Wiring	-	155.53	20%	31.11	186.64
Dry mass	-	1553.5	-	84.642	1638.1
Wet mass	-	3625	-	294.84	3947.8
Boosted mass	-	4103.1	-	294.84	4397.9
LV capability	-	5670	-	-	-
Margin	-	22.44%	-	-	-

8 Power budget

Subsystem	# elements	Power [W]	Comments
Payload		39.5	
UVS Magnetometer EPD Infrared cam JunoCam	1 1 1 1 1	9.8 4 3.1 16.7 5.9	
Propulsion		15	Takes $\pm$ 5% of total power
Power		75	Takes $\pm$ 25% of total power
Solar arrays Batteries PCU Regulators Wiring	2 4 1 ? -	448.80 - - 60 15	P_{sa} 20% of converted power losses
Total		129.5	
30% margin		129.5 + 38.85	
Margin from P_{req}		131.65	43.88% margin

Note: this budget (like the mass budget) is preliminary since several subsystems are still missing. The margin will be reduced as more subsystems will be added (and if the margin is still acceptable, other science instruments might be added as well).

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