Mount Olympus Mons Ascension Mission Mission Design - Team Red

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***Abstract*—Designing manned missions to Mars, due to its demanding requirements, is not an easy task. One of the main issues is finding trajectories that will give short overall mission duration, but at the same time won't require too much from the spacecraft in terms of ΔV.**

This report presents the way this problem was solved for the Tantalus mission. The report also contains information about the overall logistics of the mission and communication considerations.

The study concludes that the designed mission would require 5 launches with 4 configurations of rockets and have a total duration of 1752 days. The manned mission would last 645 days. The manned spacecraft would follow opposition class trajectories, with a Venus flyby during the outward trip to Mars. The total ΔV cost including Mid-Course maneuvers for the manned flight would be 11.446 km/s.

Keywords: Manned Mars mission, Lambert's problem, Porkchop plot, Opposition class trajectories, Venus flyby

***Sammanfattning*—Att designa bemannade rymdfärder till Mars är ingen enkel uppgift med tanke på de högre krav som ställs. En av de stora utmaningarna är att finna banor från Jorden till Mars med korta flygtider men samtidigt inte kräver mycket ΔV.**
Denna rapport presenterar det sätt som detta problem löstes för Tantalus uppdraget. Rapporten innehåller också information om de övergripande logistiska lösningarna samt kommunikationssystemen. Studien visar även att det designade uppdraget kräver 5 raketuppskjutningar med 4 olika konfigurationer av raketer samt har en total tidsram av 1752 dygn, varav 645 är bemannade. Den bemannade rymdfarkosten skulle följa en oppositionsklassad omloppsbana, med en passage förbi Venus på väg mot Mars från Jorden. Den totala ΔV kostnaden för den bemannade resan beräknades till 11.446 km/s inklusive korrigeringsmanövrar.

NOMENCLATURE

LEO	Low Earth Orbit
LMO	Low Mars Orbit
SOI	Sphere Of Influence
MCM	Mid-Course Maneuvers

I. INTRODUCTION

The goal of this mission is to reach Mars, spend enough time there to climb Olympus Mons and come back to Earth. The time to climb Olympus Mons was estimated to be around 25 days (with a number of margins) by our Mars Operations Team [1]. The desired trajectory is thus one that would provide a stay on Mars of at least 25 days and transfer times (from Earth to Mars for the outward trip and from Mars to Earth for the return trip) of minimal duration, because this is a crewed mission.

However, time is not the only factor here. In fact the required payload for this mission is quite heavy: a total of 170 tons has to be brought to Low Mars Orbit (LMO), as estimated by our other teams [1] [2] [3]. This cargo includes propellant for diverse maneuvers such as the landing and the hop, supplies, rover, crew vehicle and crew station for the transfers. Two cargo flights of 45 tons each will be flown to Mars orbit prior to the mission, and the main crewed flight, of a payload of 55 tons will be flown to Mars orbit. For the return trip, only the crewed flight is concerned of course. The payload here will be lighter and with the possibility of refuelling on Mars, the return trip is not the limiting factor of the mission. In any case, with high payload, the propellant needed will be quite high meaning that the ΔV costs of the different maneuvers must be kept as low as possible.

In this context, the different maneuvers will be considered impulsive, using high thrust. Given the long transfer time (several hundreds of days), this assumption seems quite reasonable. Moreover the Patched Conics model will be used, splitting the trajectory in 3 phases. First a departure burn from a parking orbit around the Earth (at 400 km of altitude), then a transfer orbit to Mars in the Sun's Sphere of Influence (SOI) after leaving the Earth's SOI. Finally an arrival in Mars' SOI with an arrival burn to catch up to Mars and put ourselves in a parking orbit around Mars (at 230 km of altitude). For the return, the considerations are the same but from Mars to the Earth.

In this report, a numerical solution to Lambert's problem will be implemented in order to study all possible direct transfers from Earth to Mars and from Mars to Earth. Finding these trajectories will enable us to know the different costs (ΔV and duration) of each trajectory. This information can be summed up in a series of porkchop plots. A first set of trajectories will be chosen, corresponding to the planets ideal alignment for a transfer. Then the use of a gravity assist with Venus will be considered, to see if this opens up any opportunity, especially time wise. Finally a trajectory will be chosen and optimized as much as possible.

II. MISSION LOGISTICS

The logistics of the mission is presented in diagrams in the Appendix A at the end of this document. To perform the mission 5 launches in total will be made. Due to the high ΔV requirements it was decided that mainly expandable rockets will be used. The configuration and the detailed information about the launchers can be found in the Space vehicles report [2].

Firstly, approximately 1100 days before the manned launch, 3 rockets with necessary supplies will be sent to Mars. One of the rockets will be carrying a capsule with the necessary supplies (rover, food etc.) and an orbital refueler. The supply capsule will land on the Martian surface at the Olympus Mons landing site (detailed information about localization of the site in the Mars Operations Team Report [1]). The orbital refueler will be kept into low Mars orbit. The other two rockets will be transporting the surface refuelers - these will land at the same site as the supply capsule. All three rockets will travel to Mars via the most fuel efficient direct transfer orbit (will be explained further in section III).

Secondly, 30 days prior to the manned mission, the crew station will be launched into low Earth orbit. As the crew station weighs only 18 tons, the rocket for this launch was not decided (a lot of already existing rockets can deliver such payloads, therefore it is not necessary to specify the launcher at this stage). Finally, the rocket with the crew lifts off. After reaching LEO the crew capsule performs a rendezvous and is assembled with the crew station waiting in LEO. At this point the spacecraft consists of 5 modules: the Mars transfer stage, tank T1, tank T2, the crew capsule and the crew station. It is worth noticing that T2 tank is empty throughout the majority of the mission, but later on, with refueling, it will be vital for the journey back home.

On November 19th 2042, the spacecraft is inserted into the Mars transfer orbit. What led to the selection of this trajectory will be discussed in section III. On that journey, a gravity assist at Venus is performed. After 389 days, low Mars orbit is reached. There, the spacecraft is there refueled thanks to the orbital refueler already waiting in LMO. The crew station is then separated and waits until the return trip in the parking orbit. The rest of the spacecraft lands on the Olympus Mons landing site.

Next, the main part of the mission - reaching the summit of Olympus Mons - starts. These activities are described in detail in the Mars Operations Team report [1]. In the mean time the spacecraft is refueled again, by the surface refuelers. After reaching the mountain a hop to the Martian base at Gusev crater is performed. There the spacecraft is refueled again by the on-site infrastructure.

Finally, the return journey starts. The rocket is launched into LMO, rendezvous and docks with the crew station that was left in the parking orbit. The Mars departure burn is then performed and the spacecraft is on its direct transfer trajectory back to Earth. After 198 days the spacecraft reaches Earth. Only the crew capsule needs to land on the surface, therefore all other modules are jettisoned. As the spacecraft will have significant speed, an aerobraking maneuver will be performed (as discussed in section V). After that the capsule performs one orbit around Earth and then performs a ballistic reentry. In the final stage of the descent the capsule is decelerated with the help of parachutes, allowing for a safe landing.

III. Trajectory study

A. *Solving Lambert's Problem*

Connecting two position vectors r_1 and r_2 with a conic trajectory around a central body of a given gravitational force μ with a specified transfer time $t_2 - t_1$ is Lambert's Problem. This is illustrated in Figure 1.

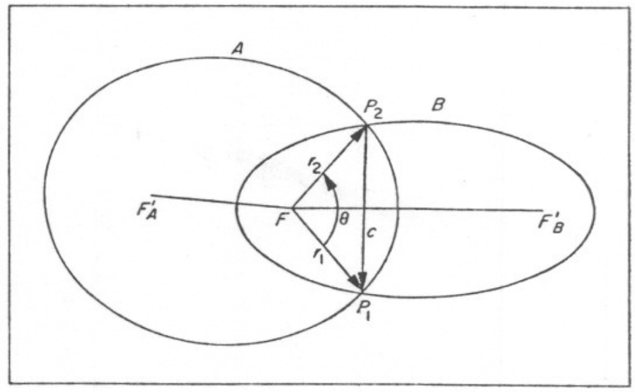

Fig. 1. Lambert's Problem for an elliptical orbit [4]

To put it differently, it is the boundary value problem for the differential equation of the orbital motion of the two-body problem [5]:

$$\ddot{r} = -\mu \cdot \frac{\hat{r}}{r^2} \tag{1}$$

Solving this problem means finding all the different trajectories possible between these two vectors that satisfy the time condition and the focal point. For example two given possible trajectories A and B intersecting the two positions are given in Figure 1.

Numerous numerical solutions exist to this problem and one was selected [4] to be used in an algorithm form to solve this problem precisely in the case of an Earth to Mars transfer.

To know the position of the different bodies of the solar system, SPICE date from JPL (NASA) in the form of kernels was used [6]. Given any date between 1850 and 2150, one can know the precise location and heliocentric velocity of any body in the solar system. Using this data, we will be able to take into account the fact that the planet's orbits are elliptical and each have a different inclination relative to the Sun's ecliptic plane.

The different inputs will be the departure date and the arrival date (or the time of flight). The dates will give us the position of Earth (for departure) and Mars (for arrival), while the time of flight is simply the time between the two dates.

Then Lambert's problem for these inputs can be solved numerically, resulting in a initial velocity vector and arrival velocity vector. From this the spacecraft trajectory can be propagated by solving the orbital motion equation 1.

Knowing the departure and arrival velocities, the different ΔV costs can also be computed. Considering Patched Conics, the departure cost from the considered parking orbit around the Earth can be obtained. The same considerations enable us to find out the arrival cost for an arrival parking orbit around Mars.

The same procedure can be repeated for a Mars to Earth transfer simply by considering the departure at Mars' position at the departure date and the arrival at the Earth's position at the arrival date. For now, an arrival burn is considered around the Earth, but the possibility of aerobraking is studied in section V.

An example of this trajectory solver is found below on figure 2 and table I. The dates were chosen arbitrarily, purely to demonstrate what the algorithm outputs.

TABLE I
DATA CALCULATED FOR THE EXAMPLE TRAJECTORY

	Date	ΔV cost	Mission time
Earth departure	2038-01-01	7.847 km/s	0
Mars arrival	2039-02-06	6.385 km/s	400 days
Mars departure	2039-05-06	4.748 km/s	650 days
Earth arrival	2039-11-26	3.752 km/s	850 days

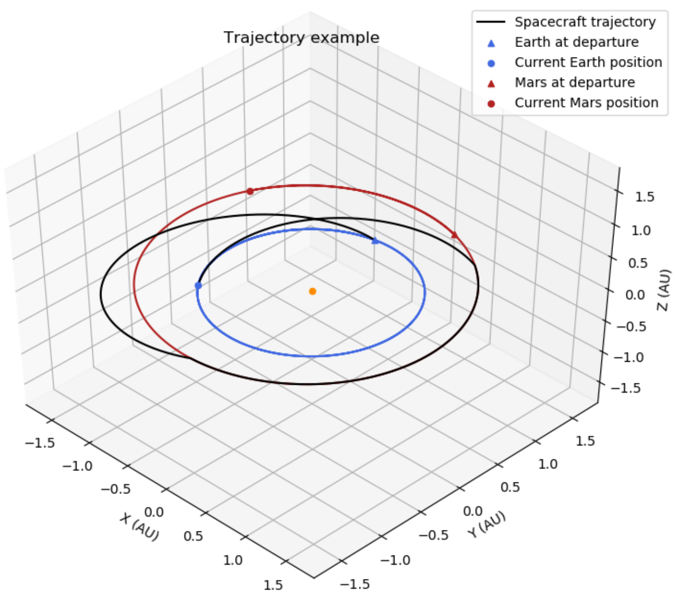

Fig. 2. Example of a trajectory found by the algorithm

B. Porkchop plots

Now that the values of interest (ΔV costs and dates of departure and arrival) can be determined for any trajectory, the idea is to find a way to present it succinctly and find the best one. A way to do that is to use a so called Porkchop plot. There are different kinds of these plots, but for this report: on the x-axis is the departure date, on the y-axis the time of flight, or duration of the transfer.

For any trajectory determined by its departure date and time of flight, the desired ΔV cost is calculated. It can be the Earth departure cost or the Mars arrival cost for example. The cost shown here is the total cost of the transfer (both the departure burn and the arrival burn), to get an idea of how expensive the transfer is overall. This value is indicated by a color on the plot. All Δv costs are from and to the defined parking orbits: 400 km around the Earth and 230 km around Mars.

The idea is thus to identify the trajectories of minimal ΔV costs and minimal time of flight. The following Porkchop plots (Figures 3 and 4) will focus on the Earth to Mars transfer cost and the Mars to Earth transfer cost respectively (from and to the defined parking orbits). All time windows will now refer to 2039-01-01, because there is no interesting trajectory opportunity in 2038, the starting year of the mission. The plot windows are changed to see properly the pattern that appears for each transfer, which is repeating about every 800 days. About a million trajectories were studied to make each plot. A larger window plot is also available in the appendix D to have a better view of the repeating pattern.

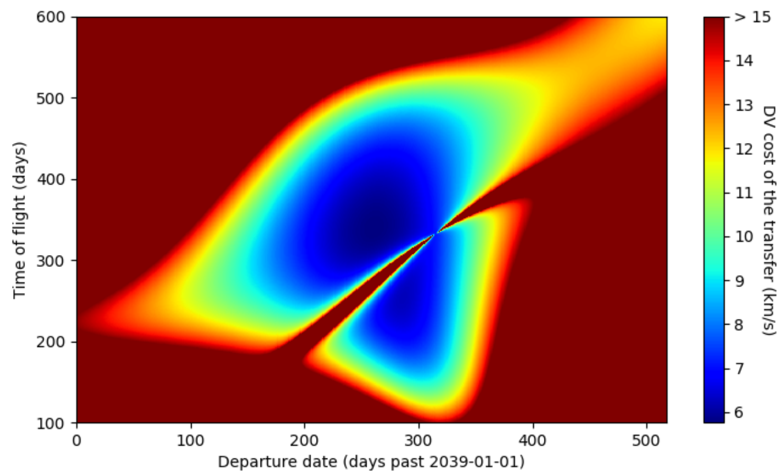

Fig. 3. Porkchop plot for the Earth to Mars transfer

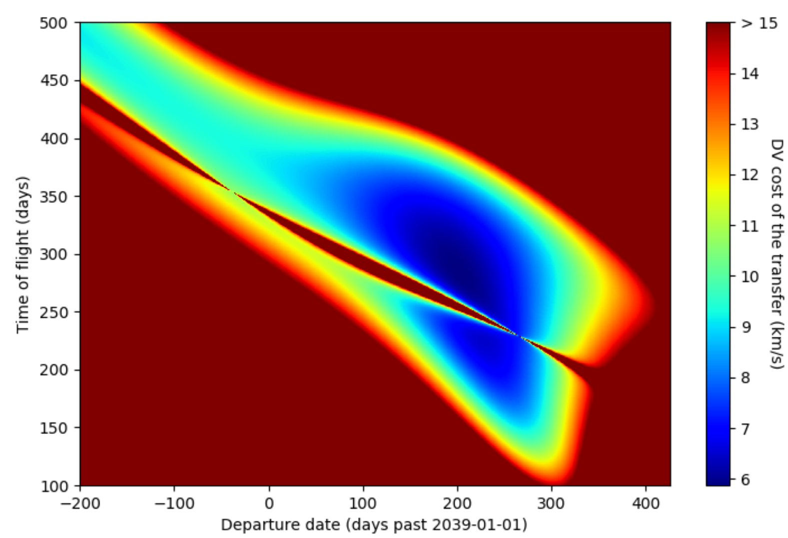

Fig. 4. Porkchop plot for the Mars to Earth transfer

The dark red area corresponds to values above 15 km/s (the transfer cost can go up to 100 km/s in the worst cases, and as they are not of interest for the mission, anything above 15 km/s is not registered in the plot).

For the outward trip from Earth to Mars, presented on figure 3, two minimum points (in dark blue) of about 5.8 km/s can be identified. In more details, the Earth departure maneuver costs 3.8 km/s and the Mars arrival maneuver costs 2 km/s. The time of flight is around 375 days for the smaller minimum (in the upper part of the pattern) and 200 days for the slightly higher minimum (in the lower part of the pattern). These two trajectories would be of interest for the mission.

For the return trip from Mars to Earth presented on figure 4, again two minimum points can be identified for the pattern at a value of around 5.8 km/s, in dark blue. In more details the Mars departure is around 2 km/s and the Earth arrival around 3.8 km/s. Again one is slightly cheaper but has a time of flight of 300 days, the other one of 200 days.

In both outward and return trips, the second minimum (in the lower part of the plot) will be considered for the crewed mission, because its ΔV cost is only slightly higher than the other one, but for a flight time of around 100 days less, which is of critical interest for our crewed mission.

This will not be the case for the trajectory of the cargo missions, in which the time of flight is not a problem, as long as the cargo arrives before the crewed mission launch (which is the case). Indeed the payload mass has to be maximized, meaning the transfer cost minimized. The trajectory chosen for the cargo is thus a direct Earth to Mars transfer of minimal energy, to maximise the payload mass, despite the transfer time of 300 days.

A different kind of porkchop plot uses contours and arrival date on the y axis. They can be found in the appendices B and C. Instead of showing the total transfer cost, they only show the cost of each maneuver (for example the Earth departure burn), meaning we have 4 of them. This can be of interest to optimize the trajectories by tweaking the maneuver dates. Additionally, the time of flight lines are plotted every 50 days.

C. First retained trajectories

Using these complex plots, one is able to assess with only two plots which trajectories for the outward and return trips are possible and how expensive they are.

The very first thing that was noticed is that in order to use the minimum for both trips (outward and return), one has to wait for the second window of opportunity to start in order to use the minimum in the return trip as well, so around 800 days, meaning around 600 days on Mars for a transfer of 200 days. This is seen better on the larger view of the porkchop in appendix D. Such a trajectory, with a long stay on Mars, is called a conjunction class trajectory, or ”long-stay” trajectory.

The conjunction class trajectory is using the lowest energy transfers but has a long waiting time on Mars (600 days) and despite some relatively short transfers (200 days) makes up for a long total mission time, of around 1000 days.

For an Earth departure in 2039, a conjunction class trajectory is presented in Table II and Figure 5.

On the other hand, one could consider using points on the Porkchop plots that are not exactly the minimum points, in order to arrive at Mars early enough to still be in the same window of opportunity, enabling us to leave without waiting for the next one, 800 days later.

Such a mission would result in a higher energy cost, but a very short waiting time on Mars (only a few days) for an overall mission duration of around 350 days. These trajectories are called opposition class trajectories, or ”short-stay” trajectories.

For an Earth departure in 2039, an example of an opposition class trajectory is presented in Table III and Figure 6.

TABLE II
RETAINED CONJUNCTION CLASS TRAJECTORY

	Date	ΔV cost	Mission time
Earth departure	2039-10-01	4.010 km/s	0
Mars arrival	2040-04-30	2.904 km/s	212 days
Mars departure	2041-08-15	2.190 km/s	584 days
Earth arrival	2042-05-10	3.752 km/s	952 days

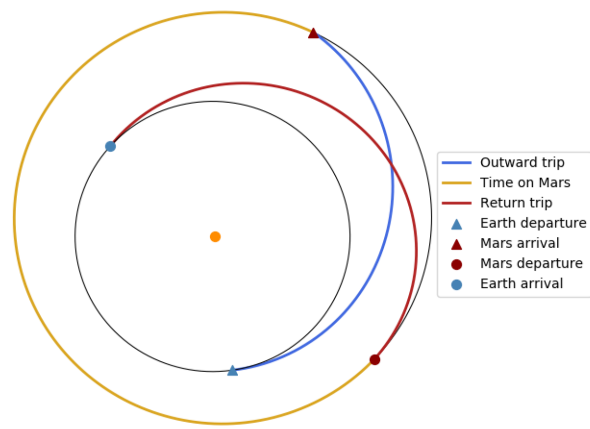

Fig. 5. Retained conjunction class trajectory

TABLE III
RETAINED OPPOSITION CLASS TRAJECTORY

	Date	ΔV cost	Mission time
Earth departure	2039-04-14	9.171 km/s	0
Mars arrival	2039-10-28	5.297 km/s	197 days
Mars departure	2039-11-12	8.495 km/s	212 days
Earth arrival	2040-03-13	4.468 km/s	334 days

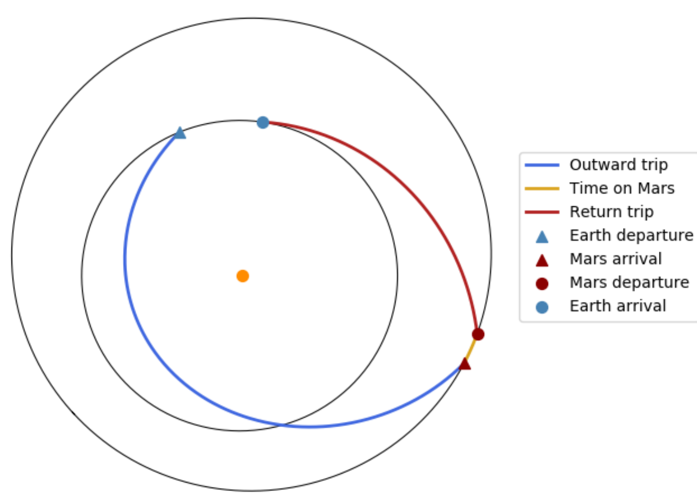

Fig. 6. Retained opposition class trajectory

D. The opportunity of a Venus flyby

There is a rough correlation between the required mission time and the necessary energy needed to perform the mission. The shorter the mission, the higher the energy one needs to spend for it. To shorten the mission without spending too much fuel one should therefore seek other ways of gaining that amount of energy. One way is a flyby: by entering the gravitational wheel of a third planet like Venus, a spacecraft can gain momentum in the heliocentered inertial reference frame in a slingshot fashion. The theory for flyby design is given by [5]. To design a flyby within the context of a Lambert problem, many possible trajectories targeting Venus are tested, and the flyby geometry is recovered a posteriori. The more precise optimization of the trajectory is discussed in section III-E.

The choice of exploiting the flyby in the Earth-Mars transfer instead of doing it on the way back was made for safety concerns. If something goes wrong on the departure trajectory (which is the most complex part of the trajectory), there is still plenty of time and resources to devise a way back. Moreover the flyby trajectory was found to be slightly more expensive than the direct transfer, meaning that more fuel would be needed for the transfer. Doing the most expensive maneuvers at the beginning of the mission keep us from having to carry the extra fuel for all the mission. Finally the flyby on the outward trip gives us 2 full windows for sending the cargo missions (including the arrival at Mars before the beginning of the crewed mission), which is quite useful in case of launch delays of the cargo missions.

An Earth to Mars trajectory including a Venus flyby requires a certain alignment of the planets, which was found to happen about every 6 years and a half. The first occurrence after 2038 is in 2042-2043, so the trajectory is studied in this time frame.

The mission trajectory around the Sun is detailed in figure 8 and table IV. The trajectory in Venus' SOI for the flyby is illustrated in figure 7.

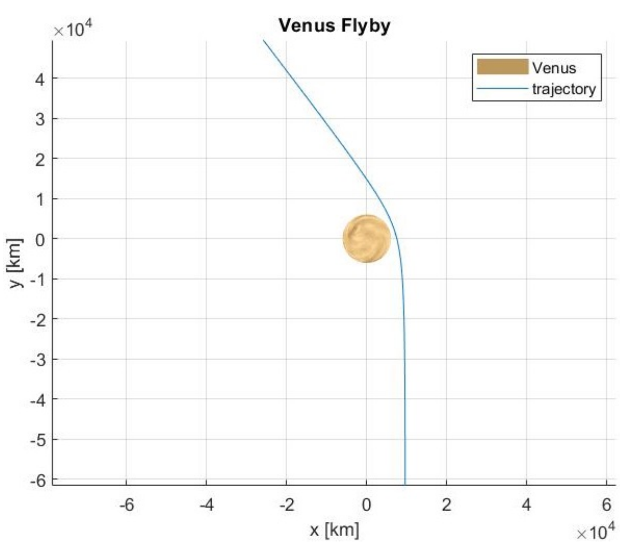

Fig. 7. Flyby Trajectory at Venus, Spacecraft Velocity Reference Frame

TABLE IV
RETAINED OPPOSITION CLASS FLYBY TRAJECTORY

	Date	ΔV cost	Mission time
Earth departure	2042-09-14	4.005 km/s	0
Venus Flyby	2043-03-29	0 km/s	196 days
Mars arrival	2043-11-10	3.054 km/s	387 days
Mars departure	2043-12-05	4.121 km/s	412 days
Earth arrival	2044-06-20	4.008 km/s	610 days

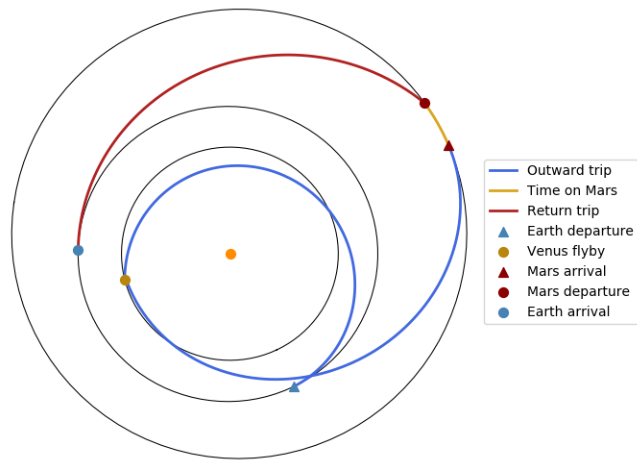

Fig. 8. Retained opposition class flyby trajectory

E. Final trajectory optimization

The inward flyby trajectory has 3 parameters: the departure date from Earth, the Venus flyby date and the arrival date at Mars. Changing these values will give different trajectories, meaning different times of flight and ΔV costs. In this case the goal is of course to minimize both times of flight and ΔV costs. The trajectory should also be such that the transfer window for a direct Mars to Earth transfer is lined up about 30 days after Mars arrival, leaving enough time for Mars operations.

But the optimization is in fact slightly more complicated: the Earth departure maneuver is the dimensioning burn for our mission, because it carries all the propellant for the rest of the mission (excluding refueling). Thus it's actually more interesting to decrease the cost of the Earth departure while slightly increasing the other costs.

Moreover the optimization has to take into account the return. Indeed the window for the direct Mars to Earth transfer is actually already open for most of the possible Mars arrival dates. So the later the Mars arrival date, the later and the more expensive the return will be, meaning also a bigger velocity differential at Earth arrival. This could become a limiting factor too in the case of aerobraking, which will be discussed further in part V.

An optimization of the trajectory was thus done considering all these factors in order to choose a trade-off between the different costs, resulting in the trajectory that has been presented previously in part III-D.

IV. MID-COURSE MANEUVERS

So far the trajectories studied only consider 4 maneuvers: a burn in LEO to leave the Earth's gravitational influence and reach Venus/Mars (the Earth departure burn), a burn near Mars to put ourselves into LMO (the Mars arrival burn), a burn again in LMO to leave Mars' gravitational influence (the Mars departure burn) and finally a burn near Earth to put ourselves into LEO again (the Earth arrival burn). However some additional burns might be required to make some precise corrections.

As an example, the flyby is only successful if the periapsis of the hyperbola in Venus' SOI is at the correct altitude (which will rotate our heliocentric velocity by the correct angle, to aim us at Mars). In real life it would be difficult to get exactly the correct periapsis with only the departure burn at Earth. This is why some corrections during the travel are necessary. They are called mid-course maneuvers (MCM). The different kinds of required mid-course maneuvers are discussed in the following subsections.

A. Flyby corrections

As explained before, the flyby requires us to have a precise periapsis altitude on our hyperbola trajectory in Venus' SOI, as shown in figure 7. For the optimized flyby trajectory, the periapsis altitude would be about 6900 km (Venus' radius being about 6052 km). It is possible to model this trajectory with Patched Conics, considering each body's SOI one at a time and patching together all the parts of the trajectory. However this would be quite far off from the actual trajectory we would observe in real life. Indeed, seeing as we cross several SOI (the Earth's, the Sun's, Venus'), the Patched Conics model is nowhere near precise enough to predict a correct periapsis altitude around Venus. To predict a trajectory closer to real life, we would need to consider a more than 2-body problem in order to take into account the different gravitational perturbations felt by the spacecraft, especially at the crossing of the SOI borders.

This is of course quite complex, considering any problem with more than 2 bodies has no analytical solutions. That is why in this report, we decide to stay in the Patched Conics approximation and instead estimate the cost of a mid-course maneuver which would give us access to a range of possible periapsis altitudes around Venus. It is important to at least consider a change of 7000 km (Venus' radius, taking into account its atmosphere and a margin) in case we are on a collision course with Venus, which would be catastrophic for the mission.

We actually decided to opt for a margin the size of the SOI radius (which is around 617 000 km for Venus). This will give us an idea of how expensive it would be to correct an error which would be of the order of the SOI radius of the targeted body. This is quite a large error and it is reasonable to assume that we would at least be able to aim at Venus' SOI from the Earth departure burn. We would then adjust our trajectory to aim more precisely at the correct periapsis around Venus thanks to the MCM, whose cost is estimated in this report.

The idea behind the MCM to correct our Venus periapsis is illustrated by figure 9. If the initial trajectory (in solid blue) gives us a given periapsis, it can be increased or decreased by raising or lowering very slightly our transfer orbit to Venus around the Sun (in dotted blue).

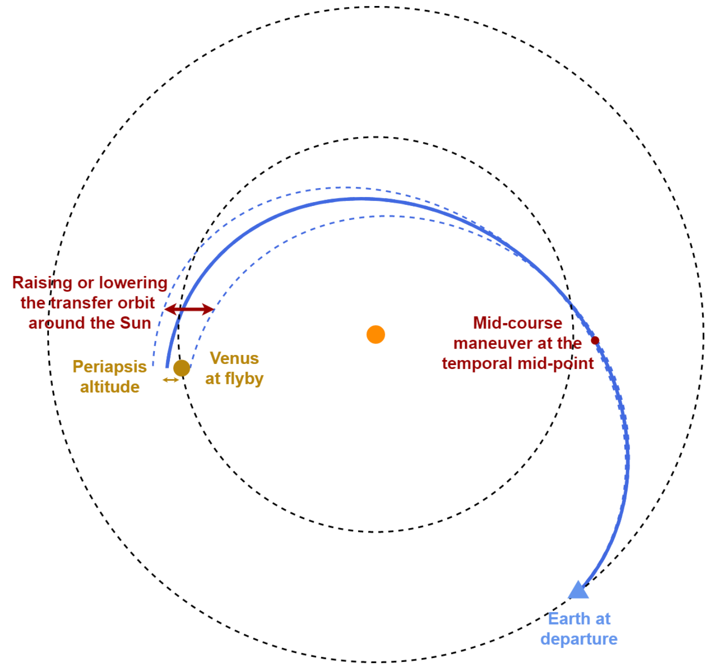

Fig. 9. Mid-Course Maneuver for the Venus flyby (not to scale)

For this MCM, the mid-point chosen is the temporal mid-point (i.e. the middle point of the transfer duration, so after around 80 days). This is an arbitrary choice and in a more precise study, the MCM would be done at the optimal point, i.e. where the desired MCM is the cheapest possible. This requires a quite complex optimization process. Because the idea here to approximate the cost of such an MCM, the temporal mid-point was chosen to simplify the model.

For our given transfer trajectory around the Sun, which is an ellipse of semi-major axis a, energy E and for a given point with velocity v and heliocentric distance r [5]:

$$E = -\frac{\mu}{2a} = \frac{v^2}{2} - \frac{\mu}{r} \tag{2}$$

where $\mu = 1,327 \cdot 10^{11}\ km^3 s^{-2}$ is the standard gravitational parameter of the Sun.

Raising or lowering our orbit corresponds to increasing or reducing the semi-major axis a, meaning increasing or reducing the velocity at a given point.

Considering equation 2 at the mid-point of heliocentric distance r_{MCM} and heliocentric velocity v_{MCM}, it can be deduced that an increment of Δa would require an increment in velocity of Δv. The result is given by equation 3:

$$\Delta v = \sqrt{2\left(\frac{\mu}{r_{MCM}} - \frac{\mu}{2(a+\Delta a)}\right)} - v_{MCM} \tag{3}$$

We assume that changing a by Δa will also change our Venus periapsis by a similar order of magnitude. By applying the margin considered earlier (Venus' SOI, so $\Delta a = 617000\ km$) to our change of a, the required velocity increment can be calculated.

The state of the spacecraft at the mid-point (heliocentric distance and velocity) are given by the simulation in table V.

Finally with equation 3, the cost of the MCM to change our Venus periapsis $\Delta v_{MCMtoVenus}$ can be estimated to be around 80 m/s.

B. Mars arrival corrections

After the Venus flyby, there would be a need for another MCM during the transfer from Venus to Mars. Indeed it would be difficult once again to predict accurately our trajectory to reach Mars on a hyperbola with a periapsis of 230 km (the altitude of the LMO considered for our mission).

Thus the considerations are the same as before: a correction of the order of magnitude of the target body's SOI, which is about 578 000 km in the case of Mars.

Using the same equation (3) and once again taking the temporal mid-point (given in table V), it can be found that the velocity increment (or decrement) needed would be $\Delta v = 49\ m/s$.

However there is another correction that might be needed for our insertion around Mars. Indeed our arrival orbit has to be at an altitude of 230 km, but it also has to pass above the landing site, Olympus Mons. Indeed Olympus Mons is not on the Mars equator but at a latitude of 18.65°, so an orbit of an inclination lower than that (with respect to Mars' equator) would not pass above Olympus Mons. This means that our arrival orbit must have an inclination of at least 18.65° relative to Mars' equator in order for us to reach our targeted landing site.

Moreover, Mars has an axial tilt, like the Earth. It is of around 25.19° to its orbital plane and its orbital plane has an inclination of 1.85° to the Sun's ecliptic plane, so in the worst case Mars has an axial tilt of 27.04° with respect to the Sun's ecliptic.

Let's consider the worse case scenario for our arrival trajectory: at arrival around Mars, our spacecraft has an equatorial orbit, meaning of an inclination of 27.04° relative to the Sun's ecliptic plane. This is illustrated by the solid red line called ”Trajectory in Mars' SOI” on figure 11.

To be able to reach Olympus Mons, we need an inclination change of at least $\Delta i^M = 18.65°$ with respect to Mars' equator, so the correction needed for such a trajectory would be to aim our SOI encounter slightly higher (or lower) in order to increase (or decrease) our inclination to Mars' equator. The minimal change is represented by the red dotted line called ”Minimum required inclination” on figure 11 and would be achieved by moving our SOI encounter up by $\Delta z = 195080\ km$, given by equation 4:

$$\Delta z = R_{SOI} \tan \Delta i^M \tag{4}$$

In order to move our SOI encounter up (or down), an inclination change of our trajectory around the Sun is needed, as illustrated in figure 10. How high of an inclination change will be determined by when the maneuver is done. Considering the temporal mid-point as before, the Spacecraft-Mars distance $d_{S/C,M}$ is around 0.264 AU, and the required inclination

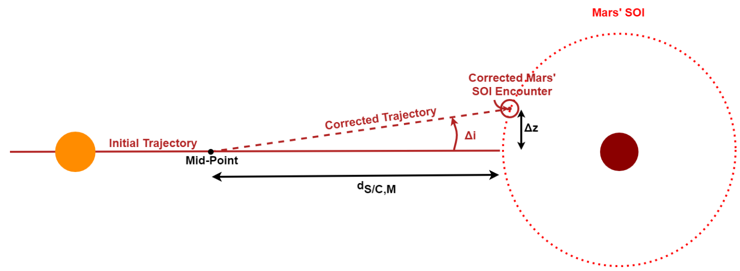

Fig. 10. Mid-Course Inclination Maneuver for Mars arrival around the Sun (not to scale)

change around the Sun is $\Delta i = 4.926 \cdot 10^{-3}\ rad$, as given by equation 5:

$$\Delta i = \arctan\left(\frac{\Delta z}{d_{S/C,M}}\right) \tag{5}$$

For high thrust maneuvers, the cost of the inclination change can be estimated by $\Delta v_{inclination} = 2v \sin \frac{\Delta i}{2} = 125\ m/s$ but considering we also need to change our semi-major axis to correct our Mars periapsis, one can estimate the Δv cost of doing both maneuvers at the same time with equation 6, which ends up being cheaper.

$$\Delta v_{MCMtoMars} = \sqrt{v^2 + (v + \Delta v)^2 - 2v(v + \Delta v)\cos \Delta i} \tag{6}$$

where Δv is the velocity increment previously calculated needed to correct our periapsis around Mars.

So the total estimated maximal cost for the MCM on our way to Mars is $\Delta v_{MCMtoMars} = 134\ m/s$.

C. Earth arrival corrections

The final part of our trajectory design is the Mars to Earth direct transfer, or return trip. Once again a correction would be needed at the mid-course to correct our arrival periapsis around the Earth (perigee), either to reach the initial LEO at 400 km of altitude, or to dive a precise depth into the atmosphere to perform an aerobraking maneuver (which will be discussed further in section V).

The considerations are the same as before: the margin taken is the target body SOI radius, about 925 000 km in the case of the Earth and the temporal mid-point is taken for the MCM (spacecraft state given in table V). This results in a MCM of $\Delta v_{MCMtoEarth} = 52\ m/s$.

TABLE V
SPACECRAFT STATE AT THE DIFFERENT MID-POINTS

Given transfer	r_{MCM}	v_{MCM}
Earth to Venus	0.722 AU	36.573 km/s
Venus to Mars	1.265 AU	25.401 km/s
Mars to Earth	1.297 AU	27.072 km/s

D. Summary of the various MCM

To conclude on the mid-course maneuvers, the different values are given in table VI. The corrections for the periapsis

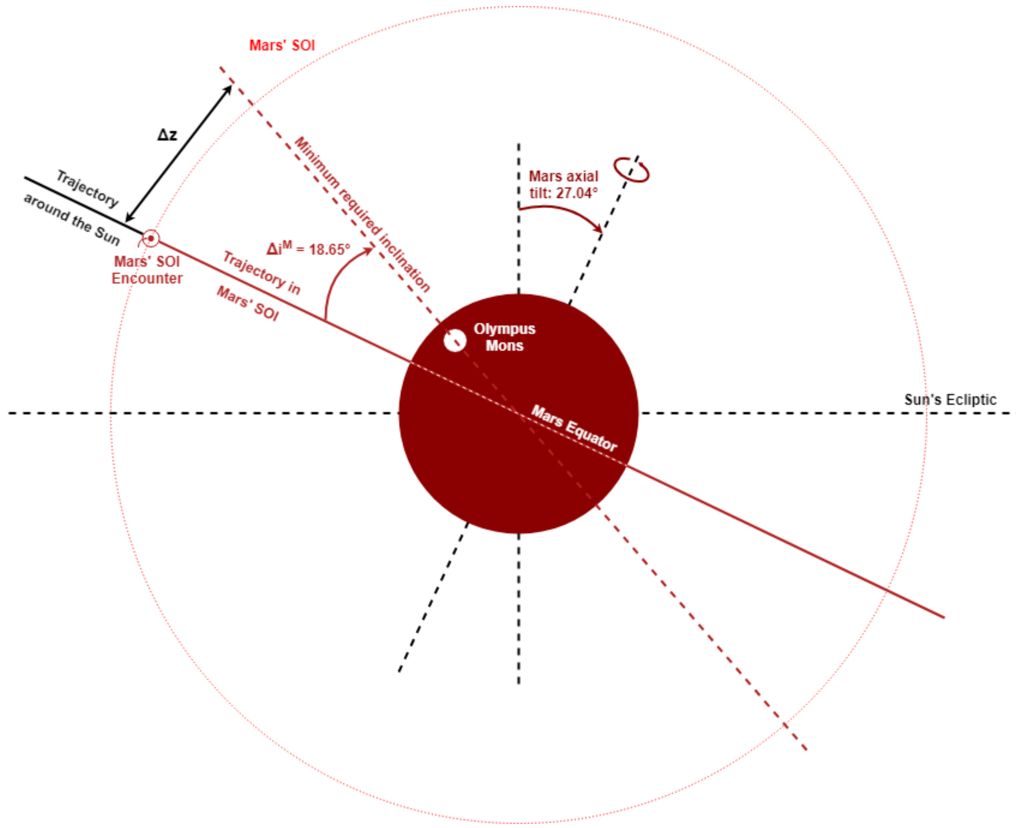

Fig. 11. Mid-Course Inclination Maneuver for Mars arrival (with respect to the Sun's ecliptic) (not to scale)

consider a margin of the SOI radius of the targeted body, which make our values higher than what the real mid-course corrections might be. The date of each maneuver (in mission time) is also given.

TABLE VI
ESTIMATED MID-COURSE MANEUVERS

Correction	**Maximum Δv estimated**	**Mission time**
Venus periapsis	80 m/s	80 days
Mars periapsis and inclination	134 m/s	274 days
Earth periapsis	52 m/s	511 days

V. AEROBRAKING

As can be noticed from previous sections, upon arrival back to Earth the spacecraft will not be put in low Earth orbit, but will rather head straight into the Earth's atmosphere. Therefore, as the spacecraft is arriving from outer space with a 4.33 km/s excess velocity (upon entering the Earth's SOI), it is obvious that its speed at the point of reaching the upper atmosphere will be higher than Earth's escape velocity. More precisely its speed at that point will be approximately 11.87 km/s. The fastest ever reentry was performed by the crew of Apollo 10. They entered the atmosphere with speed of 11.09 km/s and during reentry were exposed to load of 6.78g [7]. It was decided that in the case of the Tantalus mission, the speed of reentry should not exceed Apollo's record. Therefore it was decided that prior to the final reentry, an aerobraking maneuver shall be performed, which would reduce the final reentry speed and thus g loads and heat loads that will be acting on the vehicle during the final descent. To estimate the results of the maneuver, a very simple model assuming the spacecraft is moving in a straight line across the atmosphere was established (see fig.12). The atmosphere was assumed to span up to 150 km altitude with its density exponentially distributed and drag force was assumed to be the only force acting on the vehicle.

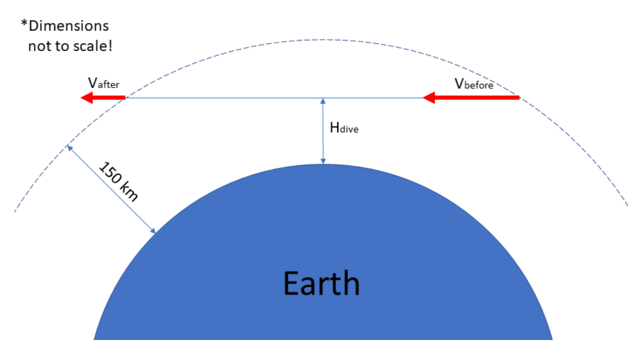

Fig. 12. Simplified model for the aerobreaking simulation

Calculations showed that a dive into an altitude of 45 km should allow to reduce the speed of the spacecraft by 1.60 km/s. While performing it the astronauts would experience a maximal load of 3.32 g. Such a reduction in speed would keep the spacecraft in orbit around Earth and on the next leap enable us to perform a ballistic reentry with a speed that is more comparable with speeds of typical vehicles coming back from low Earth orbits.

VI. Communications

For communication to be available at all times, a network for different types of transmitters and relay stations will be necessary. Text based communication will only be possible between Mars and Earth due to the distance that will create a delay from 5 to 20 minutes for one way communication. On the surface of Mars an audio connection will be able to work without any noticeable delay between the astronauts.

A. Mars Surface Communication

On the surface of Mars a normal VHF audio radio at around 100 MHz will be enough for line of sight communication between the astronauts, the base and the rover. These radios are so small so that spare ones can be carried for redundancy.

B. Deep Space Communication

The deep space network is a communication network established by NASA and JPL in 1958 that has laid a good foundation that this communication network will continue on. The latest proven frequencies lay in the Ka-band with 34.2-34.7 GHz uplink and 31.8-32.3 GHz downlink with a bandwidth of about 2.8 Msps (Mega symbols per second) [8] giving about 6 Mbps with an 8PSK modulation with a forward error correction of 3/4 [9]. All vehicles will be equipped with a Ka-band type radio setup for transmitting telemetry and communication.

C. Relay Link

If any vehicle would to lose its ability to communicate directly with Earth, then a relay link between all vehicles will be used. A UHF radio at around 400 MHz will be used to be able to establish the relay link with a bandwidth of about 2 Mbps. It is the same type of relay link that the 2020 Mars mission Perseverance rover uses as a relay link to the MRO (Mars Reconnaissance Orbiter) [10].

VII. Off-nominal scenarios

The addressed off-nominal scenarios are graded according to their possibility to occur and the severity of the event. The grades range from 1 to 5, where 1 is very low and 5 is very high.

TABLE VII
Off nominal scenarios

	Event	Possibility (1-5)	Severity (1-5)
1	Cargo Launch Delay	3	2
2	Mars Landing Failure	2	4

In the event of a delayed cargo launch there will be another launch window such that the cargo can reach Mars before the crew lifts off from Earth. This gives a new launch opportunity without delaying the mission. The cause of the delay can range from something simple like weather conditions, which only results in a wait until the next launch window, to the loss of the cargo rocket, which will cost the price of a whole new cargo rocket. If the money for a new cargo rocket is not available then the mission is going either to be delayed until the planets realign for the Venus flyby (6 years and a half later) or cancelled.

The first case for landing failure would be a system failure at Mars arrival. In this case it is possible to exploit a flyby at Mars for an almost free return. With a moderate burn of 0.24 km/s at Mars' periapsis, it is possible to come back to Earth in 279 days, reaching our destination on August 15 2044 with a velocity compatible with aerobraking. This solution comes however at the price of a big stress on the crew due to the scarcity of the life support related resources on board. A more detailed study should iterate further to guarantee the success of this strategy, but within the scope of this preliminary study our results were considered enough to consider it feasible.

An aborted landing on Mars is a hard problem to solve, since the fuel from the orbital refueling is enough to reach a ΔV just shy of 4.8 km/s with the crew station disconnected. A direct return trajectory from Mars at that time requires a ΔV of about 4 km/s as seen in table IV.

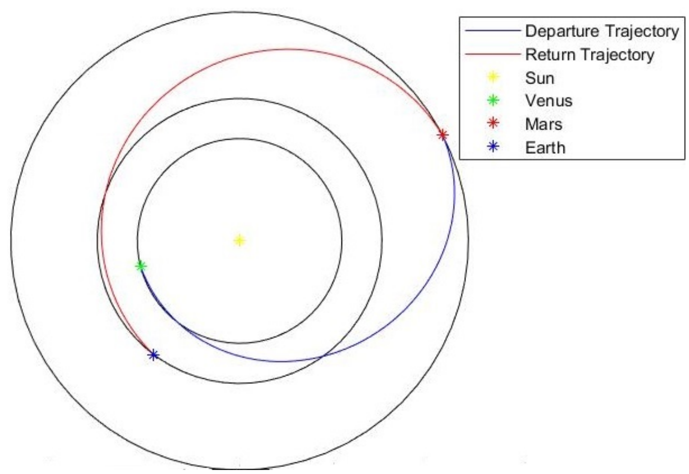

Fig. 13. Off-Nominal Return Trajectory

This allows for a possibility which is however limited by life support requirements such as food, water, air and waste. To account for this, another option could be a direct landing on Mars without refueling first, accounting for the refueling capabilities on ground. Such a emergency landing can't be aborted due to the risk of remaining stranded in an orbit around Mars.

Based on the above, the consequences of a Mars landing were considered quite severe but not disastrous and were therefore assigned a grade 4.

VIII. Conclusions

The mission design study for Tantalus shows that it is possible to make a round trip to Mars, in a relatively short and sustainable mission duration, with enough supplies that could support crews to accomplish the climbing task on Olympus Mons. Main design considerations include carrying sufficient supplies, catching the different transfer windows, ensuring a short Mars stay, lowering the total ΔV costs, building reliable communications, and safety planning.

The idea is to separate the whole mission into different phases: In total, 4 rockets will be launched from Earth. The first two are cargo rockets, transporting refuel tanks and rover and a supply capsule to Mars. The third rocket carries the crew station. The manned mission launch happens at approximately 1100 days after mission starts. As a result of solving the Lambert's Problem, an opposition class trajectory with a Venus flyby is found to be the optimal choice, the flyby enabling a shorter mission duration. In addition, the flyby takes place while travelling outward to reduce the ΔV cost, and aerobraking is used at Earth arrival to further reduce the ΔV cost. The entire mission will last for 1752 days, with 645 days as manned. The total ΔV cost is calculated to be 11.466 km/s, including mid-course maneuvers.

In order to communicate with ground station at anytime, Ka, UHF and VHF frequency bands are planned to be used. Voice radio at 100 MHz is for communication on Mars; Ka band of 34 GHz uplink and 32 GHz downlink is for deep space communication; Relay link at 400 MHz is for supplementary communication between all space vehicles.

Another crucial step in mission design is to make the risk analysis and alternative plans for emergencies. In this study, the conditions of delay of cargo launch and failure during crewed landing at Mars are included, including an almost free return trajectory from Mars to Earth.

Overall, this study shows the feasibility of the Tantalus Mission in the mission design aspect, and it used an analytically optimal way to achieve it with satisfying all restrictions.

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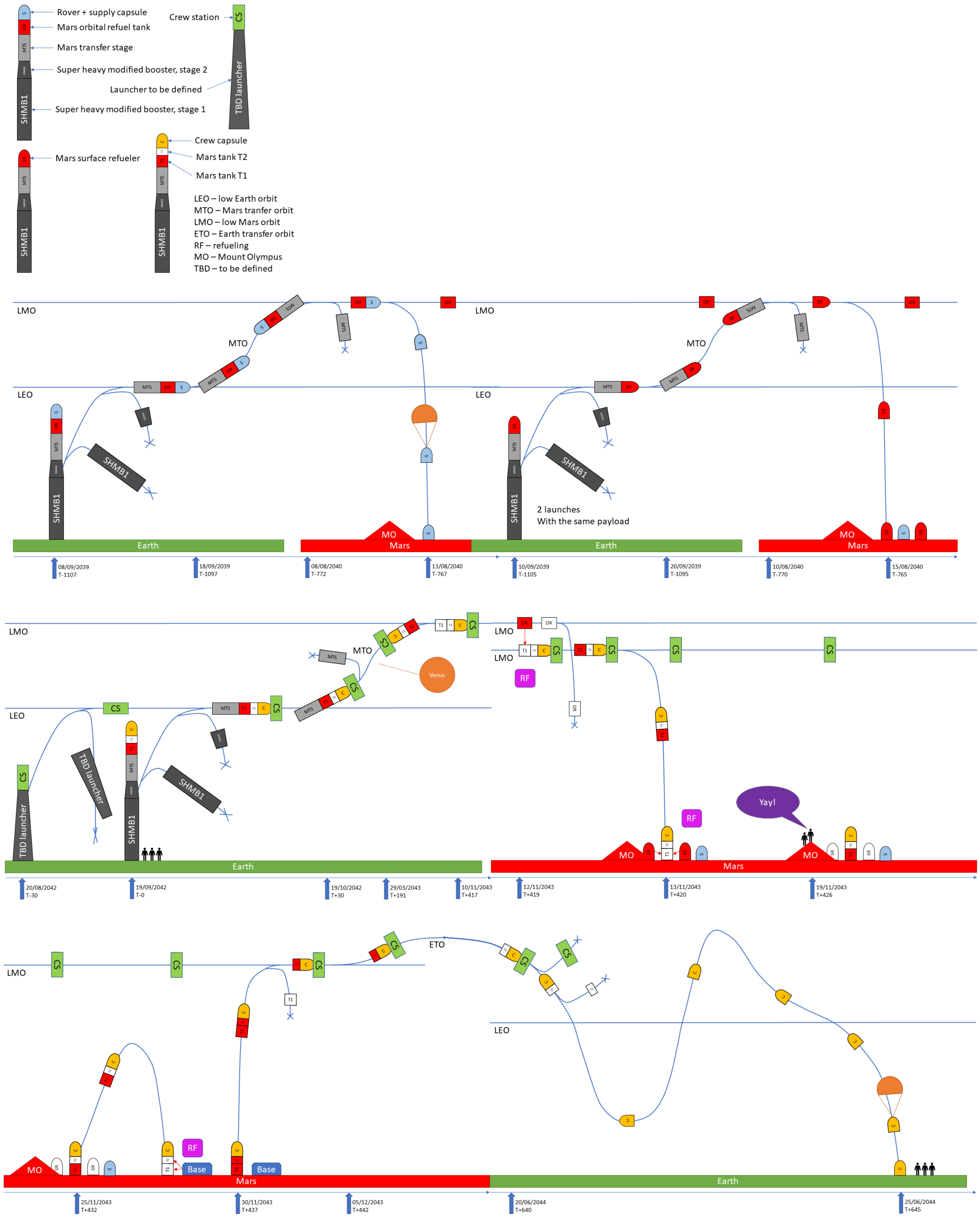
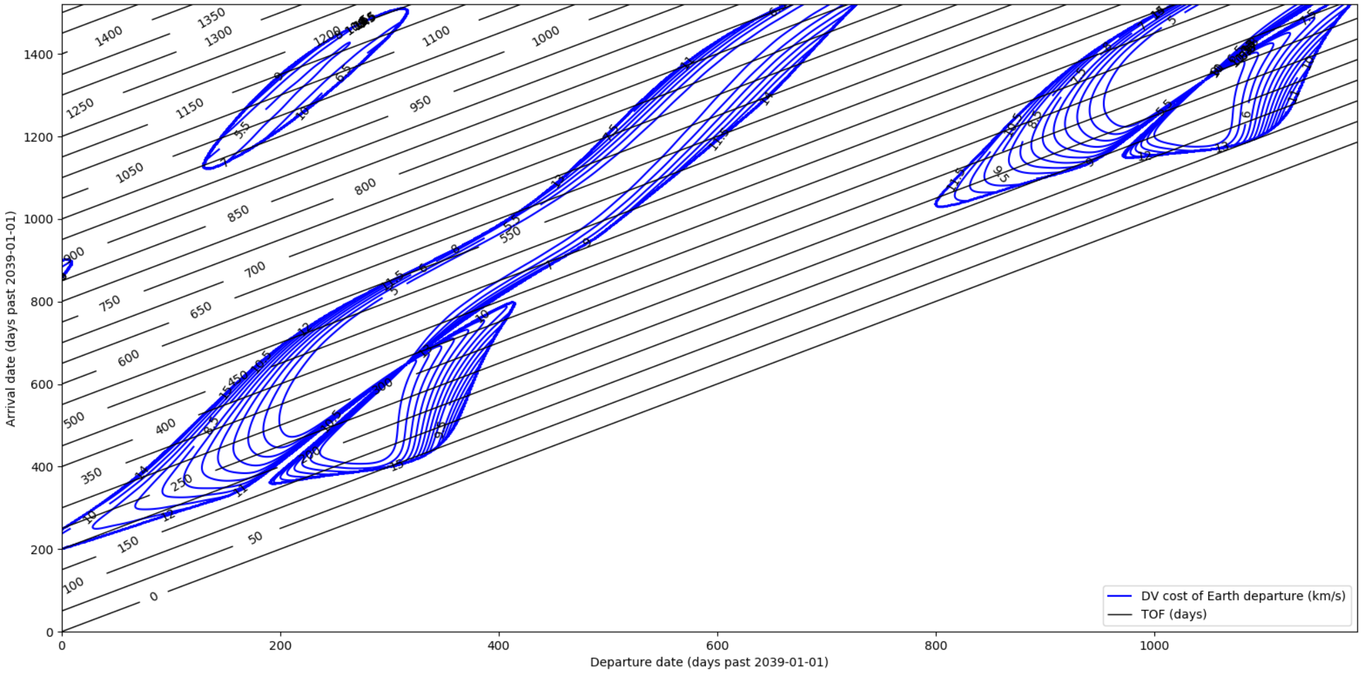
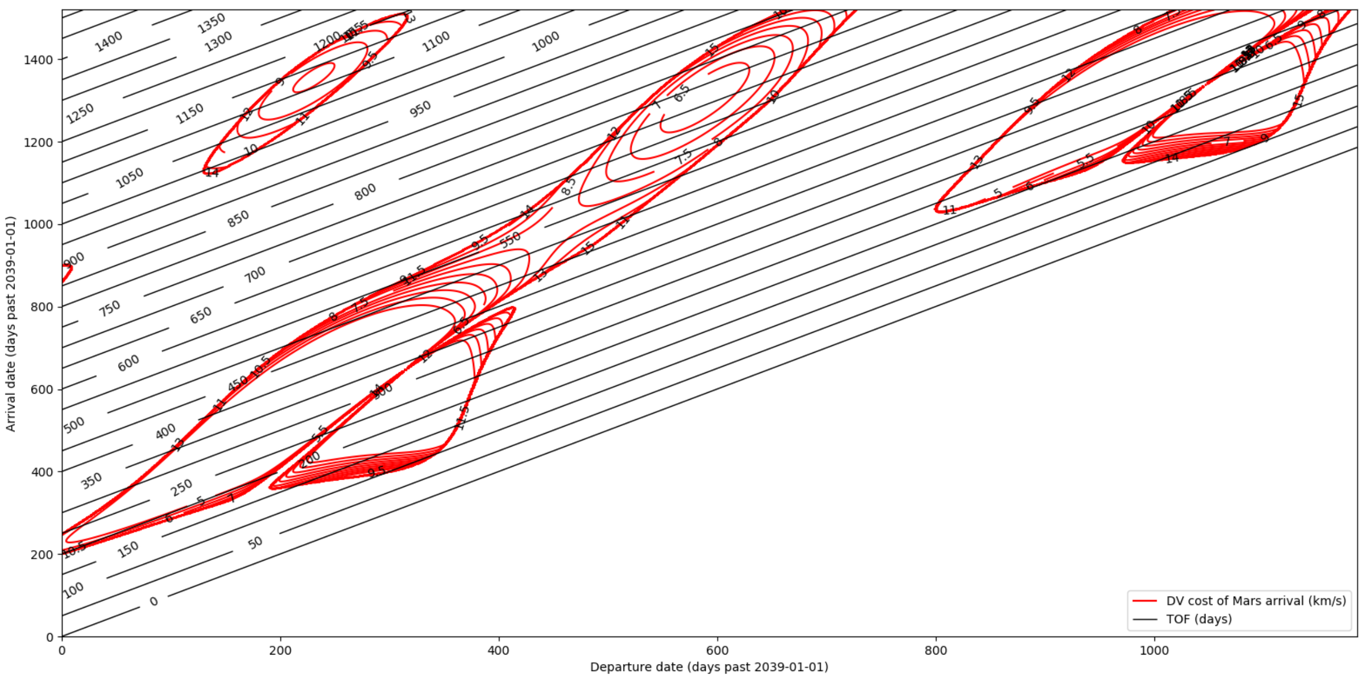
Appendix A
Mission Diagram

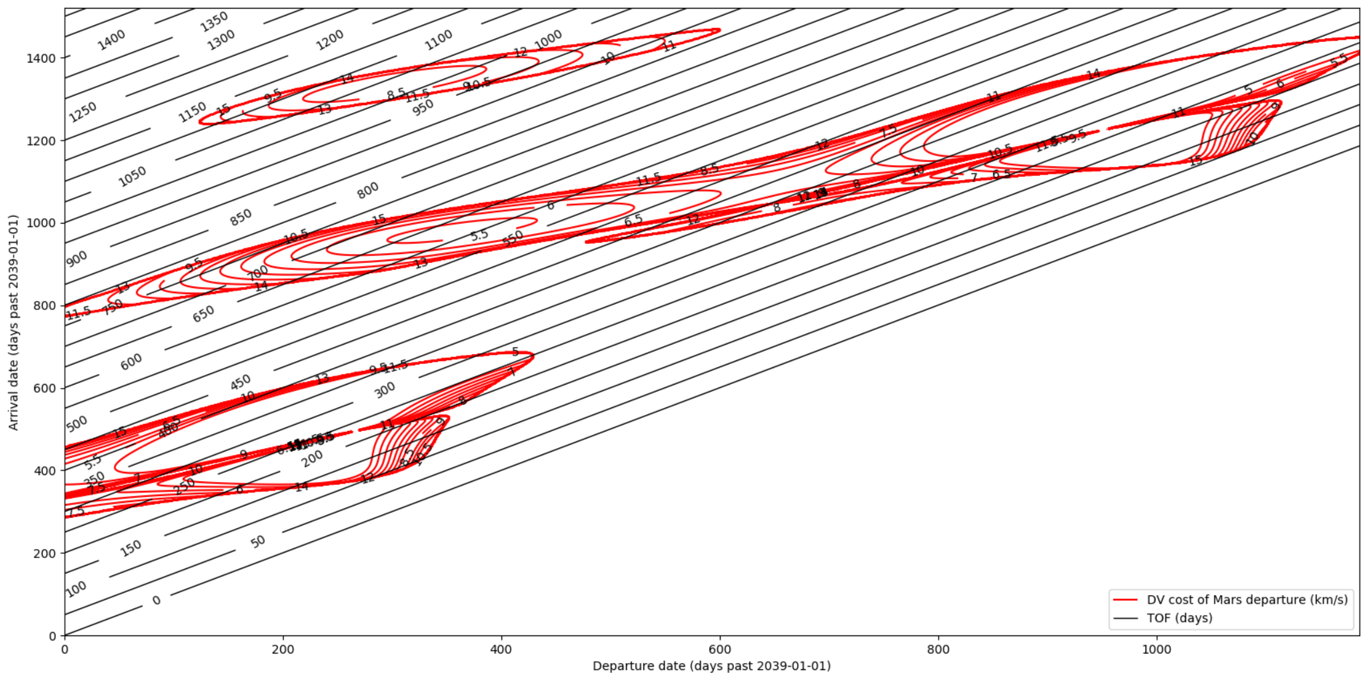


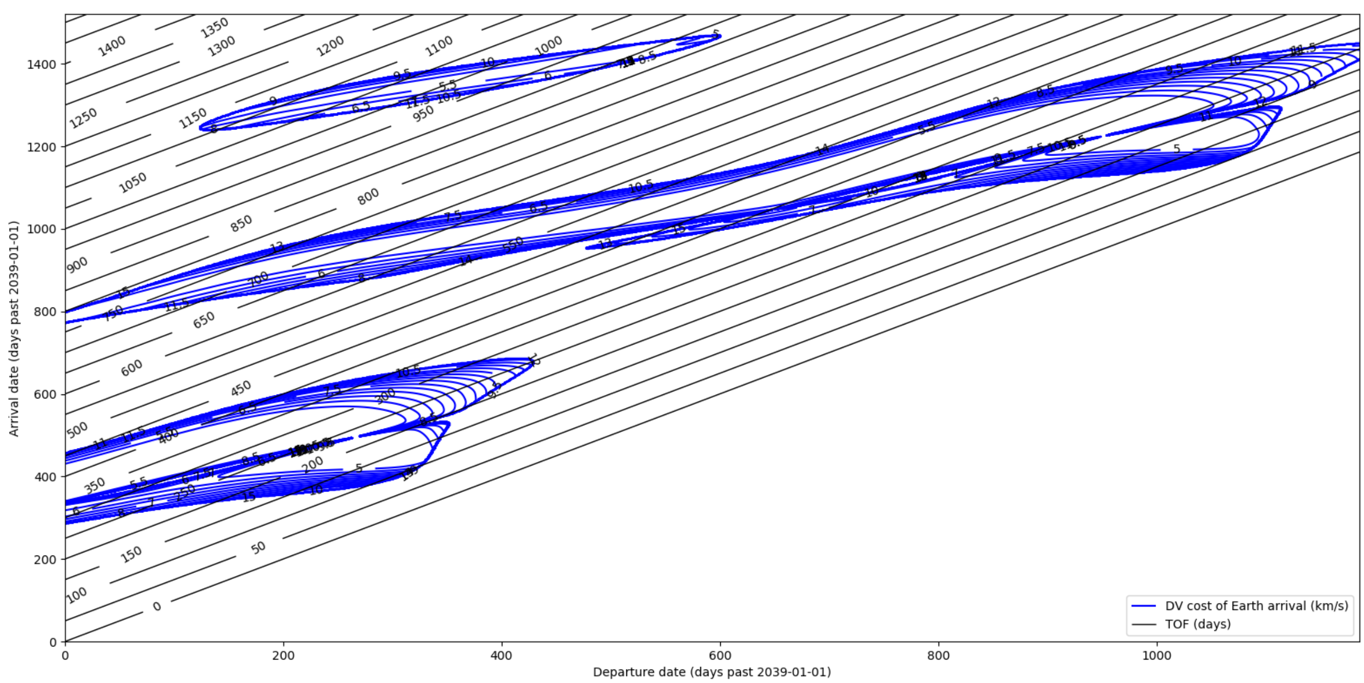
APPENDIX B
CONTOUR PORKCHOP PLOTS FOR THE EARTH TO MARS TRANSFER





APPENDIX C
CONTOUR PORKCHOP PLOTS FOR THE MARS TO EARTH TRANSFER





APPENDIX D
LARGER PORKCHOP PLOTS