Mars Expedition
Mission Design - Team Red

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***Abstract*—The mission design main task was to plan the trajectory for the mission to Mars and back to Earth. For this to be successful, it had to had a low ΔV and a low travel time. To achieve this task, different trajectories were analysed. Gravity assists and Lagrange points were studied. This was mainly using MATLAB to simulate interplanetary travel. With the help of MATLAB, the Lambert's problem was solved for different launch dates and travel times to plot pork chop plots. The pork chop plots showed where the optimal launch dates are. This information was given to the rest of the team so the final trajectory was chosen based on factors relevant to the whole mission. The final trajectory was then analysed based on off-nominal situations.**

NOMENCLATURE

DEL Direct entry and Landing
EMD Earth to Mars Direct
LEO Low Earth Orbit
LMO Low Mars Orbit
SOI Sphere of Influence
TOF Time of Flight
TRL Technology Readiness Levels

I. INTRODUCTION

There are different entities that are planing missions to Mars. One of these entities is Pythom Space. They are planing a minimalist mission and we were given the task to design this mission. The mission design main task was to plan and analyze the planetary trajectory. In addition to this, the different capture maneuvers as well as the communications and radiation through the mission were covered.

The trajectory can be calculated attending to different approaches. For this study, the ones performed include: *Lambert's Problem*, *Flyby around Venus*, *through Lagrange points*. For the initial and final altitudes, all of them will consider a departure from Low Earth Orbit (LEO) ($h = 530$ Km) and arrival to Low Mars Orbit (LMO) ($h = 230$ Km). Additionally, all of them where coded in MATLAB and subjected to an optimization process to obtain the best solution in terms of both Time of Flight (TOF) and total ΔV.

In terms of *Capturing Methods* and analysis was done to cover the arrival possibilities in both Mars and Earth. These methods include aerobraking or a ballistic entry.

Once all possible solutions for both the transfer and the arrival at the desired planet were calculated, a trade-off analysis was done to select the optimum solution for the chosen criteria. The variables included in this trade off were ΔV, TOF, risk, feasibility and time in mars.

Moreover, an off-nominal analysis was done for the chosen trajectory to cover the effect on the trajectory of the surrounding planets i.e., Mars, Earth, Jupiter and Venus (in case of the flyby being considered). In addition, the assumptions used previously to compute the trajectory were reduced to try to achieve the closest mission profile when compared to reality. Lastly, in this analysis, a study of the possible trajectories in case of a mission abortion were also included.

Lastly, a study of the communication black-outs in the capture orbit was done. With this, both the *Logistics* and *Human Aspects* teams can further develop possible solutions to have a mission within the desired margins.

II. TRAJECTORY STUDY

In this section, a detailed explanation of how the trajectory of the transfer vehicle was calculated will be presented to the reader. Moreover, the assumptions taken in the different methods will be introduced as well. Finally, the different parts of the trajectory alongside with their own approaches will be described.

For the different approaches considered the following assumptions were made:

- The time for getting out of the Sphere of Influence (SOI) is considered to be negligible in comparison with the TOF.
- Departure and capture positions are estimated as the planet's position.(SOI radius is negelected.)

Phases of the mission

There are several phases of the mission. The first one is to depart from LEO, followed by the transfer to Mars. Then the arrival and departure from Mars will be the next steps of the mission. Finally, the transfer and arrival to Earth will end the mission. Overall, they can be grouped in departure, transfer and capture phases.

A. Departure Phase

The departure orbit was selected by the *Logistics' team* to ensure a feasible assembly of the transfer vehicle. As a result, the desired orbit is a circular LEO with a height of $h = 530$Km

Additionally, for the mission being considered there are two departure phases. There is first the departure from Earth and then departure from Mars. Departure from Earth is of more interest as there are two possible alternatives i.e., a flyby (to reduce ΔV) or a a normal propulsive burn to leave the orbit. In contrast, on Mars the only possible approach is to perform a normal propulsive burn to leave the orbit.

1) Lunar Flyby

This option does not seem to have been researched as much as the later discussed Venus flyby but studies of this option have been made. According to a paper published in 2020 [1], the TOF for such a mission would be approximately 309 days with a $\Delta V = 8.006 km/s$, for a one way trip. An Earth to Mars Direct (EMD) approach would instead mean a TOF of 306 days and a $\Delta V = 5.708 km/s$, which is very close to the value calculated in this report [1]. The fact that the values for the EMD approach are similar to those calculated in this report should be considered as a good sign. However, the values for the Lunar flyby are not as enticing. Nevertheless, this depends on the mission's goals and in some cases the benefits of a Lunar flyby could absolutely overcome the increase in ΔV.

Nonetheless, considering that this mission has a minimal mass approach, the option for a Lunar flyby does not seem to be a valid option since the ΔV is higher than the other options considered in the study.

2) Low thrust trajectory

Low thrust acceleration rocket engines have a large specific impulse. However, this is not the case with the propulsion system given in the project and therefore will not be considered. A low thrust trajectory would make it more likely that a correction burn later will be smaller/not needed as it could be made more precise.

B. Transfer Phase

For the transfer part of the mission design, different approaches can be considered i.e., *Lambert's Problem*, *Venus Flyby* or *Lagrange points*. In this section, the methods followed to solve for each of them will be presented.

1) Lambert's Problem

Lambert's Problem solves for the trajectory needed to connect two different points in space with a conic curve around a central body of a given gravitational force μ provided a specific TOF. This means that when solving Lambert's problem one can find all possible trajectories that join the two points while satisfying the time constraint.

To find the trajectory the method of *Patched conics* was employed. In it, some intrinsic assumptions are made [2]:

- The SOI of a given planet is considered to have infinite radius when observed from the planet and zero when observed from the Sun.
- Within the SOI the problem is considered to be two-body with the planet as the main body. Outside the SOI the main body is considered to be the Sun.

The first step towards the final solution is to select both the departure and arrival planets, dates and times. With this, making use of Algorithm 8.1 from Orbital mechanics for engineering students [3] the code is able to obtain the orbital elements of both planets at the desired instant of time.

The next step is to solve the previously mentioned *Lambert's Problem*. To that end, Algorithm 5.2 from Orbital mechanics for engineering students [3] was used. In it, an assumption of a prograde trajectory was made. With this, the velocity vector of the spacecraft both at the departure and arrival are known. With these values, the orbital parameters of the trajectory can be computed with the use of Algorithm 4.2 from Orbital mechanics for engineering students [3].

By now, the trajectory followed by the spacecraft is known. However, the required Δv for both the departure from the departure circular parking orbit (LEO) and the arrival at the circular capture orbit are still missing. They can be computed with the difference between the speed at the periapsis of the departure/arrival parabola and the speed of the circular parking orbit.

The solution given by this method can be seen in Table I and in Figure 1.

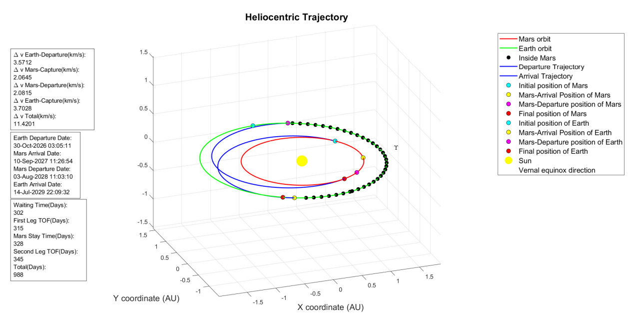

Figure 1. 3D Lambert Earth to Mars transfer

Table I
TRAJECTORY DATA (OPTIMUM LAMBERT'S SOLUTION).

State	Date	Δv [Km/s]	Mission time [days]
Earth departure	1-Nov-2026	3.57	0
Mars arrival	4-Sep-2027	2.08	308
Mars departure	7-Aug-2028	2.08	646
Earth arrival	16-Jul-2029	3.71	988

a) Porkchop plot

A *Porkchop plot* is a chart that illustrates how the required ΔV for a given trajectory relates with the TOF.

The trade-off between this two parameters is crucial for the whole mission. For instance, having a shorter mission will be beneficial for the human aspect team while it could make the mission non-feasible due to an impossible mass budget (huge increase in terms of propellant mass). Thus, the porkchop plots presented here helped to make a decision on the most suitable TOF and ΔV for the mission.

The cost in both graphs (see Figures 2 and 3) represents the ΔV both for the departure and arrival burns. It is important to mention that the time window studied cover from 1/1/2024 since it is the first year in the period that the mission wants to be developed when interesting trajectories opportunities appear.

Moreover, it is worth to mention that the range of ΔV retained for the study cover from 5Km/s $\leq \Delta V \leq$ 15Km/s.

The values lying above this value appear in the graphs as the dark red area.

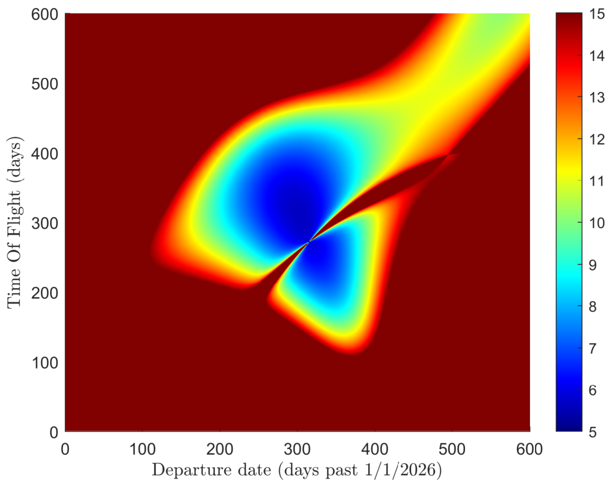

Figure 2. Porkchop plot for the Earth to Mars transfer

From Figure 2 one can see that two points represent the minimum ΔV required for the trajectory from Earth to Mars (dark blue in the graph). It's value is approximately, $\Delta V = 5.8$Km/s with a departure date around *November 1st 2026* and TOF of approximately 330 days. Therefore, it can be observed that the optimum data coming from the Porkchop plot coincides with the optimum solution given by Lambert's Problem solution (see Table I).

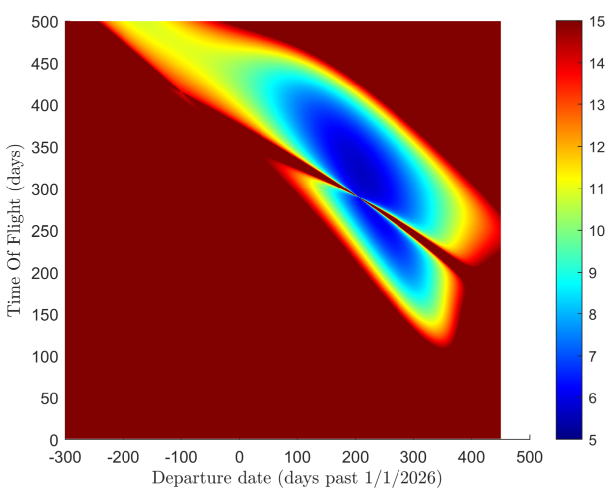

Figure 3. Porkchop plot for the Mars to Earth transfer

From Figure 3 one can see that two points represent the minimum ΔV required for the trajectory from Mars to Earth (dark blue in the graph). It's value is approximately, $\Delta V = 5.6$Km/s with a departure date around *July 16th 2029* and TOF of approximately 340 days. Therefore, it can be observed that the optimum data coming from the Porkchop plot coincides with the optimum solution given by Lambert's Problem solution (see Table I).

Additionally, other solutions in terms of shorter TOF could be considered for both legs of the mission. However, due to the huge ΔV that they will need and one of the mission's requirements to be around a thousand days those trajectories were discarded.

b) Optimization

Among all the possible trajectories given by solving *Lambert's Problem*, an optimization was done to get the preferred one in terms of both total ΔV required for the mission and its TOF.

To that end, a multi-objective optimization was done using a genetic algorithm. This gives as a result a Pareto front, see Figure 4 (containing non-inferior solutions, i.e., solution in which improvement of one objective implies deterioration of another).

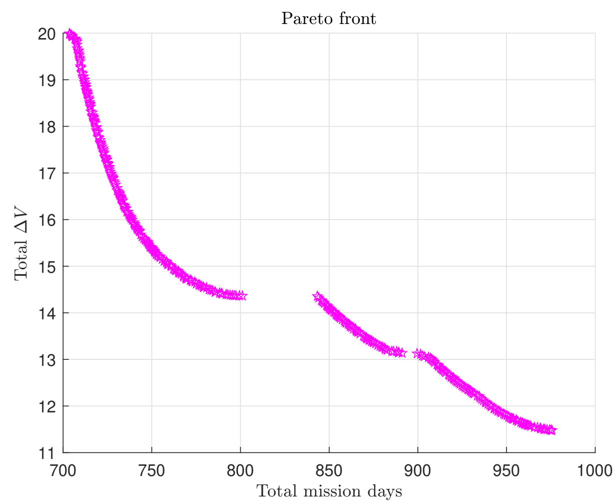

Figure 4. Pareto front

2) Venus Flyby solution

This mission has a minimal mass approach, which means all variables such as fuel, ΔV etc should be kept as low as possible. For a space mission, one can draw a correlation between the ΔV needed and the time it will take to complete the mission. If a low ΔV is desired, mission time will increase and vice-versa. Because of the minimal mass requirement, the ΔV should be kept as low as possible. In order to accomplish this, without reaching unreasonable mission times, solutions have been investigated to see if that energy can be gained from other sources. When entering a orbital body's SOI, a spacecraft could perform a slingshot maneuver around the body to gain energy using the body's gravitational force [3]. In terms of flyby options for a trip to Mars there are only two reasonable options, a Venus flyby and the previously explained Lunar flyby.

Performing a Venus flyby during a trip to Mars would, according to multiple researchers, result in a decrease in fuel, weight and cost for the mission [4]. An example of the slingshot maneuver that would be performed during such a flyby can be seen in Figure 5. There is a lot to like with the Venus flyby and multiple experts have studied about the benefits of performing such a maneuver. Performing a flyby of Venus on the way to Mars could even result in a shorter mission, which would be beneficial both for the crew but also for the overall cost of the mission. Additionally, if something were to go wrong during the mission, a Venus flyby also provides the option of aborting the mission and returning to Earth. This abortion trajectory would require a significantly smaller change of course than if the mission was headed straight for Mars. [4]

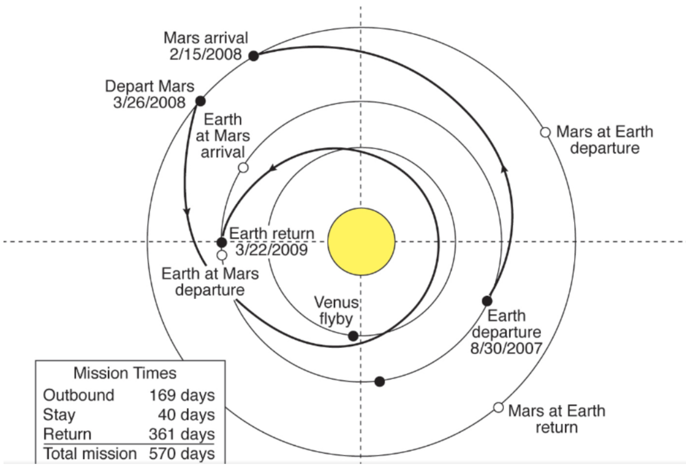

Figure 5. The trajectory for a Venus fly-by [5]

For a Venus flyby, the total ΔV for the mission is about 10-12 km/s which is slightly higher than a direct Earth to Mars transfer (see Table I for more detailed values). Moreover, the total mission time about 600 days with 40 days in orbit around Mars [3].

a) Trade-Off analysis

As discussed previously in this section, there are a few trade-offs that has to be considered and they are based on the mission requirements. The option of a Venus flyby comes with a slightly higher ΔV, so one could think this option is out of the question. For a stand alone Mission Design perspective, that would be true when considering the minimal mass requirement. However, there are more factors to take into account. For example, a Venus flyby would provide an easier return to Earth if the mission would fail during the early stages. Although this would benefit the crew, the crew would also be exposed to a higher amount of radiation because the trajectory of the Venus flyby is a lot close to the Sun than the EMD approach.

3) Lagrange points solution

The goal of this method is to take advantage of the stable and unstable manifolds of the Sun-Earth and Sun-Mars systems (see Figures 6 and 7). The stable ones help with the transfer of the spacecraft into the halo orbit. In contrast, the unstable ones help to escape from the given orbit [6].

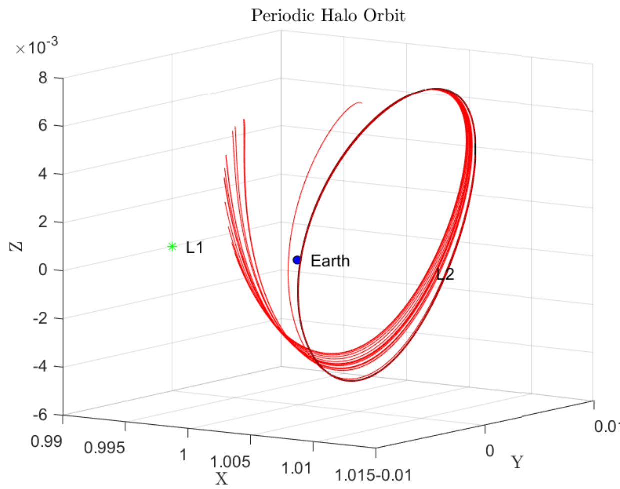

Figure 6. SEL2 halo orbit stable manifolds.

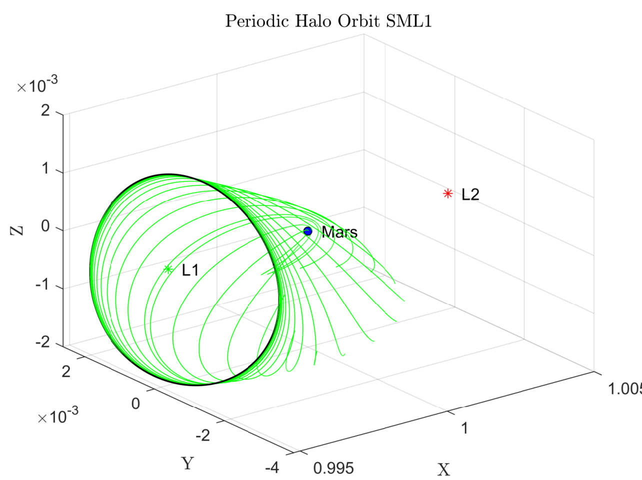

Figure 7. SML1 halo orbit unstable manifolds

Additionally, to make use of the *Lagrange's points* for both Earth and Mars a trajectory based on the invariant manifolds and Lambert's problem was designed (see Figure 8).

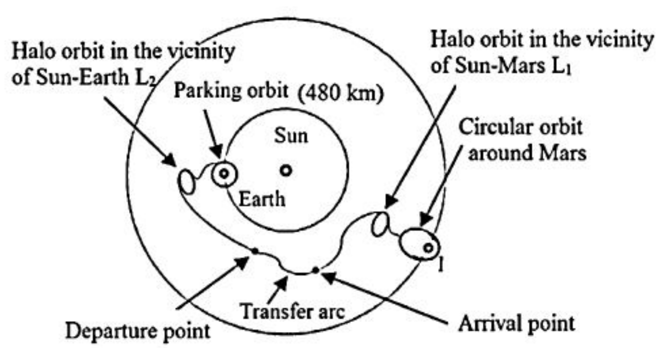

Figure 8. Earth to Mars trajectory. Lagrange points approach [6]

The trajectory can be divided into four different steps. Firstly, a maneuver from LEO to the halo orbit in the vicinity of Sun-Earth L_2. Then a transfer arc with a following maneuver to enter the halo orbit in the vicinity of Sun-Mars L_1. Finally, arrival to the desired LMO.

It is important to mention that the data retrieved from the literature [6] considers a LEO with an altitude of 480Km and that the final orbit around Mars is circular.

Table II
ΔV BUDGET EARTH-MARS TRAJECTORY. LAGRANGE APPROACH [6]

Trajectory part	ΔV [km/s]	TOF [days]
EPO to $SE - L_2$	1.1843	32.5
$SE - L_2$ *to* $SM - L_1$ *(Halo orbit)*	3.2755	89.3
$SM - L_1$ *to LMO*	0.1605	–

From the Table II, one can see that the total ΔV needed with this approach is 4.6203km/s. It is important to keep in mind that this ΔV represents only the cost to go to Mars, and additional ΔV should be added for the return trip to Earth.

Additionally, the main drawback of this approach is that in order to be able to perform the necessary maneuvers to enter and exit the halo orbit a huge waiting time is needed. Therefore, the total TOF for only one leg of the mission i.e., going to Mars could be around 700 to 900 days. Because of this, the *Lagrange point* method was discarded.

C. Capturing Methods

After performing the interplanetary leg of the mission, the transfer vehicle needs to lower the speed significantly for it to orbit the planet. In this report a specific analysis of aerobreaking and propulsive breaking will be presented.

Moreover, it is important to specify that the capture orbit was selected to be a circular LMO with a height of $h = 230$Km. The reason behind this orbit is to ensure good performance in terms of both landing on Mars and communications.

1) Aerobreaking and Aerocapture
Aerobreaking would reduce the ΔV by using friction from the atmosphere to reduce speed. This is however hard to preform as one need the correct trajectory and the vehicle needs to be designed structurally for such scenario and it needs to have a shield.

An aerocapture means that one would use a planets' atmosphere after entering the planets' SOI. With this, the speed is reduced after interplanetary travel to such a degree that the spacecraft get captured in to orbit around the planet(See Figure 9).

No kind of aerobreaking will take place on Earth entry upon leaving Mars for this mission. This is because the spacecraft has no heat shield. A shield would need to be constructed in space on the spacecraft. This would be extremely risky and has a low Technology Readiness Levels (TRL). It would also add additional mass and therefore it could increase the fuel needed for the mission more than its possible reduction.

When looking at the same approach for landing on Mars, its thin and low density atmosphere make it not an useful solution.

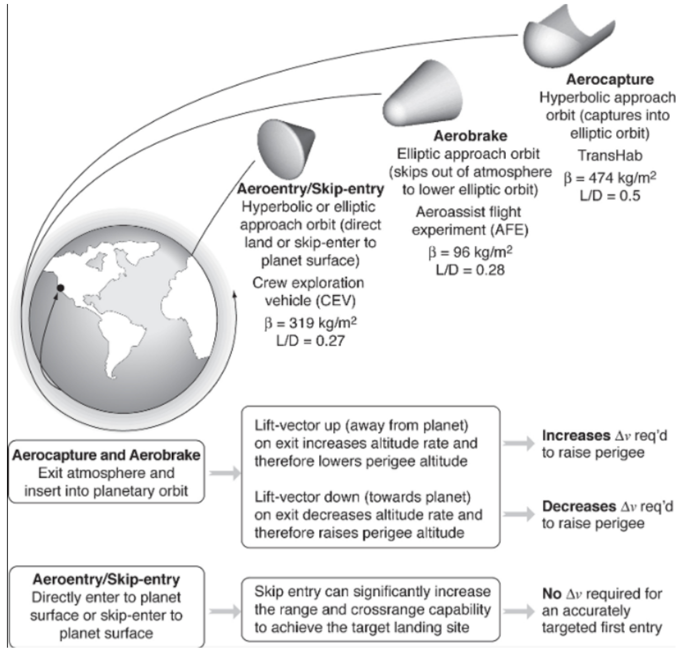

Figure 9. Different types of Aerodynamic Braking Maneuvers [5]

2) Propulsive Breaking
Propulsive breaking is done by doing a retrograde burn at the periapsis. This is done until the correct orbit is archived. This requires ΔV to preform. However, it does not require a heat shield. It is also safer as it will not interact with the friction of the atmosphere. The cost of this maneuver in terms of ΔV can be observed in Table I as the value given for the *Earth arrival.*

III. TRADE-OFF OF CURRENT TRAJECTORIES

The aim of this section is to present a trade-off analysis for all the phases of the mission.

When concerning the departure from either Earth or Mars propulsive burn was the chosen option among the ones considered (for further details of the options and trade-off see Section II-A)

Table III
TRADE-OFF TABLE FOR INTERPLANETARY TRAVEL

Mission Phase	**Planetary body**	**Propulsive Delta V (km/s)**
Lagrange points	Transfer	4.6(one way)
Venus flyby	Transfer	about 10
Lambert's	Transfer	11.4

Table III show the three different methods of interplanetary travel that were considered by this report. When comparing the Lambert's problem transfer to the Venus flyby the advantages in ΔV are almost negligible and the direct transfer present a wider range in terms of launch windows. Additionally, when the Lambert's Problem is compared to the Lagrange solution the ΔV needed reduces by approximately a 21% but with a penalty of an increase in the total mission time of about a 200%.

Therefore, the Lambert's solution was chosen for the computation of the interplanetary transfer. The detailed information in term of ΔV and TOF can be seen in Table I. Finally, it can be remarked that the other two methods, specially the Lagrange approach, can be of huge interest for different mission concepts such as unmanned mission to Mars due to the increase in TOF.

Table IV
TRADE-OFF CAPTURE EARTH TABLE

Misson Phase	**Planetary body**	**Propulsive ΔV (km/s)**	**Propulsive Mass fraction**
LEO insertion (Aerocapture)	Earth	0.039	0.988
Direct entry and landing(DEL)	Earth	0.000	1.000
LEO insertion-propulsive	Earth	5.130	0.195

Table IV show the different methods for capture when returning to Earth. Both the Aerocapture and Direct entry and Landing (DEL) use aerobreaking and this will not be used as it will overall increase the mass of the mission (due to the need of special heat shield to ensure a successful maneuver). Therefore a propulsive capture in LEO will be used even if it has the largest ΔV among the methods being studied.

Table V
TRADE-OFF CAPTURE MARS TABLE

Misson Phase	**Planetary body**	**Propulsive ΔV (km/s)**	**Propulsive Mass fraction**
Mars ordbit insertion (MOI) -propulsive	Mars	2.866	0.401
Mars orbit insertion(MOI) -aerocapture	Mars	0.111	0.965

Table V has the propulsive and aerocapture capture on Mars. As it was stated for the Earth analysis, the lack of heat shield makes the propulsive burn method the only possible solution for the mission being designed.

IV. FURTHER ANALYSIS. ABORT TRAJECTORY

The aim of this section is to cover the possible trajectory back to Earth in case of having an emergency leading to an early end of the mission. For this, a literature study was performed.

When looking at abortion trajectories from a trip to Mars there are three different possibilities:

- Direct return
- Free return
- Powered flyby

The second one i.e., the *Free return* needs for a specific alignment of the planets and thus it is rare to be a possible solution. Moreover, for the third approach i.e., the *Powered flyby* the trajectory to be performed will required both more ΔV and TOF that the nominal solution. Therefore, based on this analysis the only remaining solution will be to do a *Direct return.* [7]

Additionally, to perform an abortion trajectory the time at which it is needed is crucial. According to the days after the departure from LEO, the trajectory can be considered of type A1 (if the trajectory is needed up to a month after departure) or type A2 (if the trajectory is needed up to approximately a hundred days after departure). However, if the emergency arises later than a hundred days it is better both in terms of ΔV required and TOF back to Earth to continue as planned with the nominal trajectory [7].

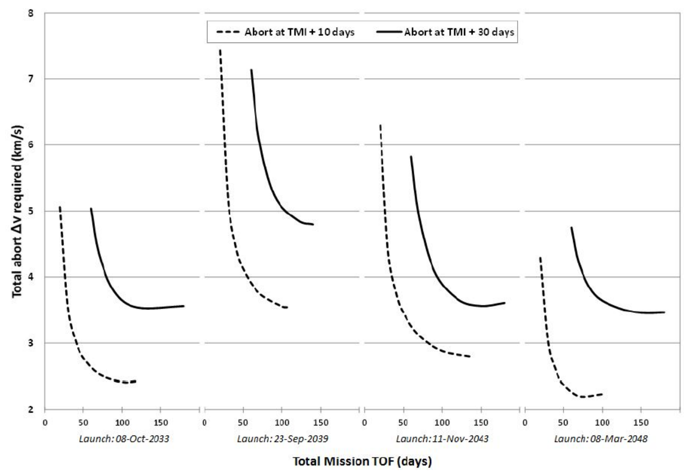

Figure 10. Abort trajectory ΔV analysis [7]

Lastly, in terms of ΔV required to perform this trajectory (see Figure 10, the cost is up to $\Delta V = 5.5$Km/s [7]. Which is less than the nominal trajectory and therefore it does not represent a problem for the *Vehicle design team* in terms of propellant needed.

V. OFF-NOMINAL ANALYSIS

The aim of this section is to analyze the effect of the assumptions considered during the calculation of the trajectory. To that end, both solutions and ΔV estimations for the correction maneuvers will be presented.

A. *Off-nominal scenarios*

The impulsive burns needed for the trajectory were considered to be perfect. However, this is not the case and there is a short delay between the time you apply the burn and the response in terms of acceleration given by the engines. Lastly, the error coming from this aspect will grow with time.

Moreover, the effect of other planets/satellites such as Venus, Jupiter or the Moon should be taken into consideration. This can be done by solving the *N-body* problem. By doing so, the shift with respect to the nominal trajectory is at most approximately 400Km. This value can be considered small when compared to the interplanetary distances used among the trajectory computation.

B. Correction maneuvers

During a space flight, correction maneuvers will always be a factor. A perfect space flight (as calculated) is nearly impossible to achieve and therefore correction maneuvers will be needed to adjust the positioning of the spacecraft. For example, a Venus flyby can only be accomplished if the periapsis of the hyperbola is at the correct altitude in Venus' SOI, which would result in a rotation of the heliocentric velocity and aim the spacecraft at Mars. This would be extremely difficult to achieve through the use of only the Earth departure burn and therefore correction maneuvers might be needed. The transfer to Mars itself is also difficult to accomplish in a sufficient way from Earth meaning that correction maneuvers could be needed to slightly adjust the orbit and hit the desired Mars target. Both the Mars and Earth arrivals could need correction maneuvers to reach their desired orbits and does therefore also have to be accounted for.

1) TCMs

To analyze the cost of the previously explained scenarios a literature research was done. In it, the mission studied was *Mars Pathfinder* (launched December 4, 1996) [8] was used to determine the in orbit corrections cost as $\Delta V = 50m/s$

This value can be considered to be negligible when compared to the cost of the mission (see Table I) of $\Delta V = 11.4km/s$. However, it is important to asses the cost to be able to account for suitable margins in terms of propellant carried for the mission.

2) Off-nominal solutions

The first step towards a more accurate trajectory will be to improve the modeling for both the performance and trajectory. Additionally, In-flight Real Time Optimization of Correction Burns can be analyzed. With this, the previously mentioned cost of *50m/s* could be reduced.

Lastly, a ΔV margin was considered by using the linearized Relative Motion Equations. This was done to give room for possible discrepancies from the calculated trajectory and the real one. With it, the amount of propellant calculated by the *Transfer Vehicle team* will already account for a safety margin needed for the mission.

C. Methodology

To solve for the ΔV required to solve for the off-nominal cases the following system of equations was addressed inside the code:

$$\dot{\mathbf{y}} = \mathbf{f}(t, \mathbf{y})$$

where

$$\mathbf{y} = \left\{ \begin{array}{c} \delta x \\ \delta y \\ \delta z \\ \delta \dot{x} \\ \delta \dot{y} \\ \delta \dot{z} \end{array} \right\} \dot{\mathbf{y}} = \left\{ \begin{array}{c} \delta \dot{x} \\ \delta \dot{y} \\ \delta \dot{z} \\ \delta \ddot{x} \\ \delta \ddot{y} \\ \delta \ddot{z} \end{array} \right\} \mathbf{f}(t, \mathbf{y})$$

$$= \left\{ \begin{array}{c} y_4 \\ y_5 \\ y_6 \\ \left(\frac{2\mu}{R^3} + \frac{h^2}{R^4}\right) y_1 - \frac{2(\mathbf{V}\cdot\mathbf{R})h}{R^4} y_2 + 2\frac{h}{R^2} y_5 \\ \left(\frac{h^2}{R^4} - \frac{\mu}{R^3}\right) y_2 + \frac{2(\mathbf{V}\cdot\mathbf{R})h}{R^4} y_1 - 2\frac{h_0}{R^2} y_4 \\ -\frac{\mu}{R^3} y_3 \end{array} \right\}$$

As it is a linear system of equations, the boundary value problem can be solved analytically and thus the corresponding ΔV for the corrections can be calculated. Its value is approximately $\Delta V = 50$m/s which was exactly the value founded in the literature study.

VI. FINAL TRAJECTORY

A. Interplanetary

The final interplanetary trajectory of the mission can be seen in IX. The trajectory was chosen since it was found to be the optimal solution with the previously mention requirements in mind. The final trajectory is of a heliocentric nature and fits the mission description well in time span while also requiring the most optimal ΔV. The trajectory is made up of two Lambert's transfers for both departure and arrival at Mars. As A Venus flyby trajectory was also considered but it did not provide enough upsides to validate the use of a flyby.

As can be seen in IX optimal arrival and departure points of the planets were calculated to maximize efficiency. A Venus flyby trajectory was also considered but it did not provide enough upsides to validate the use of a flyby.

B. Departure & Arrival Earth

For the departure from Earth, the method chosen was propulsive burn with a cost in terms of ΔV of 3.57 km/s (see Table I). The trajectory followed by the transfer vehicle in this maneuver can be seen on Figure 11. Moreover, in this figure one can also observe the arrival to Earth which has a cost in terms of ΔV of 3.71 km/s (see Table I)

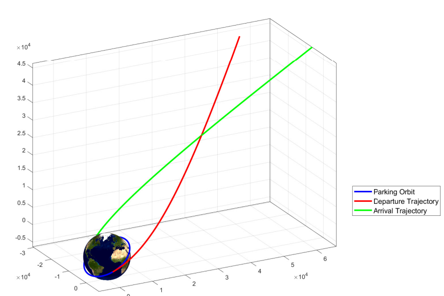

Figure 11. Departure from Earth

C. Departure & Arrival Mars

For the arrival to Mars, the method chosen was propulsive burn with a cost in terms of ΔV of 2.08 km/s (see Table I). The trajectory followed by the transfer vehicle in this maneuver can be seen on Figure 12. Moreover, in this figure one can also observe the departure from Mars which has the same cost as the arrival in terms of ΔV i.e., 3.71 km/s (see Table I)

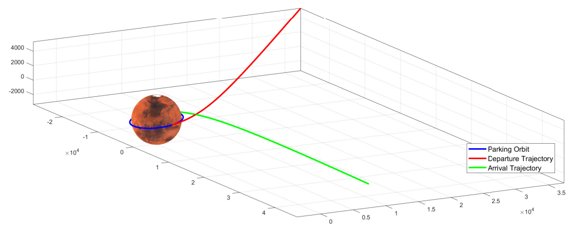

Figure 12. Departure from Mars

VII. MISSION ANALYSIS

The aim of this section is to analyze the mission in terms of communication i.e., possible black-outs due to the trajectory followed by the spacecraft.

A. Ground Track

It is important to the mission to know how the ground track of the transfer vehicle is behaving. This is important for communication purposes as well as emergencies, and ascent trajectory planning. Figure 13 provides the ground track on the Mars surface. Location of Tom & Tina is also located on the map. A zoomed version of ground track is depicted in Figure 14.

B. Communication & Surface Visibility

To communicate with Earth, the transfer vehicle needs to have a clear line of sight to Earth. The *Logistics* group have managed to calculate this value which around 2 weeks during whole mission. Surface Communication is a crucial parameter when the transfer vehicle is orbiting the Mars. The time slot of surface visibility is near 10 minutes in each revolution of orbit. The calculations are based on orbital parameters provided by the *Transfer Vehicle* group. Figure 15 provides info on visibility versus elapsed time. The methodology of this section is based on Vallado's book. [9]

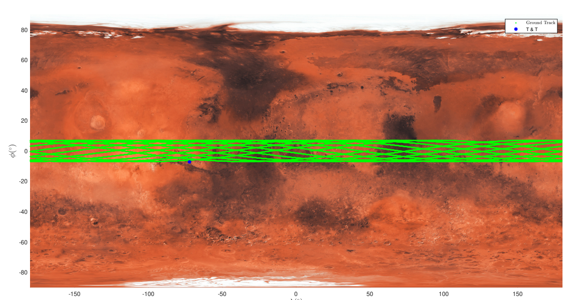

Figure 13. Ground track

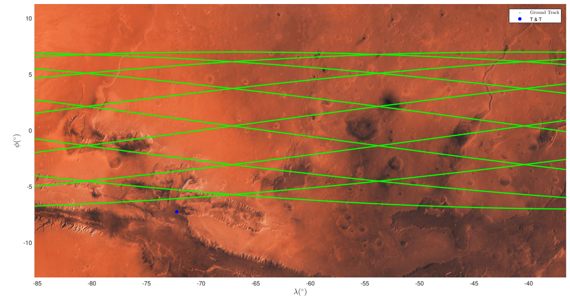

Figure 14. Ground track (zoomed), blue dot is the landing site and green track is orbit over Mars.

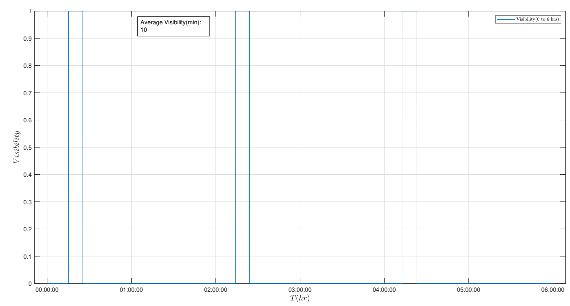

Figure 15. Surface Visibility vs Elapsed Time

VIII. CONCLUSIONS

The mission design for the Pythomspace trip to Mars shows that their goal to perform a round trip mission to Mars within a period of 5 years is possible. The objectives for the mission design were to minimize the mass needed for the mission as well as having a total mission duration below a thousand days.

With these goals in mind the mission was divided into three different phases i.e., departure from Earth/Mars, interplanetary transfer and arrival to Mars/Earth.

For both departure and arrival phases different strategies were analyzed such as *Lunar flyby* or *Low Thrust* for the first phase and *Aerobraking* for the former one. However, after the trade-off analysis both maneuvers will finally be performed using an *Impulsive burn*. Moreover, the associated cost can be seen in Table I.

For the interplanetary transfer phase different methods were analyzed with the aim to reduce the propellant needed for the mission. This methods include: *Lambert's Problem*, *Venus flyby* and *Lagrange points*. As it was explained in the trade-off (see section III) the chosen method was the *Lambert's Problem* with a total cost of $\Delta V = 11.4 km/s$.

Moreover, as it was desired the launch windows studied ranged from 2024 until 2027. However, due to the alignment of the planets there were only two possible launch windows ensuring a ΔV close to 11 km/s. These are for both 2024 and 2026. However, due to the amount of launches needed and the time for assembly is was decided together with the *Logistics team* to chose the launch window corresponding to 2026. The final dates for each phase of the mission can be seen on Table I.

Additionally, to cover for any possible emergencies during the mission an abortion trajectory was studied. Based on the literature research if an emergency is to occur up to a hundred days after the departure from LEO the abortion trajectory can be performed with a cost lower than the nominal trajectory. However, if the emergency occurs later the best option in terms of both ΔV and TOF is to continue with the nominal mission trajectory.

Another crucial step in the design of a mission is to consider the maneuvers needed to ensure the correct path is followed. For this and to account as well for the extra ΔV needed to compensate for the assumptions a literature research was done. The result is an approximate ΔV of 50m/s which can be considered negligible when compared to the total cost of the mission.

Finally, communication, ground track, and surface visibility as one can see in Fig 13, 15 and 14, the orbit around Mars is very close to the landing site and therefor, the landing vehicle does not have to do large corrections of the orbit for landing and docking to the transfer vehicle with a penalty of only 10 minutes of surface communication.

References

[1] E. A. Song, Young Joo, "Potential launch opportunities for a smallsat mission around the moon injected during a lunar flyby en route to mars," *Mathematical Problems in Engineering*, vol. 2019, 09 2019.

[2] A. department Virginia Tech, "Summary of patched conic approximations," Available at dept.aoe.vt.edu/~lutze/AOE4134/patchedconiceqs.pdf (Accessed Feb. 22 2022).

[3] H. D. Curtis, *Orbital Mechanics for Engineering Students*, ser. Elsevier Aerospace Engineering Series. Butterworth-Heinemann, 2010.

[4] Space.com, "Astronauts bound for mars should swing by venus first, scientists say," 2020.

[5] W. Larson, *Human spaceflight : mission analysis and design*. New York: McGraw-Hill, 2000.

[6] R. Eapen and R. Sharma, "Mars interplanetary trajectory design via lagrangian points," *Astrophysics and Space Science*, vol. 353, 09 2014.

[7] J. Horton, T. Kokan, D. Levack, F. Widman, and C. Joyner, "Earth to mars abort analysis for human mars missions," 02 2018.

[8] NASA, "Mars exploration program," Available at https://mars.nasa.gov/mars-exploration/missions/pathfinder/ (Accessed March 12 2022).

[9] D. A. Vallado, *Fundamentals of astrodynamics and applications*. Springer Science & Business Media, 2001, vol. 12.

IX. Appendix

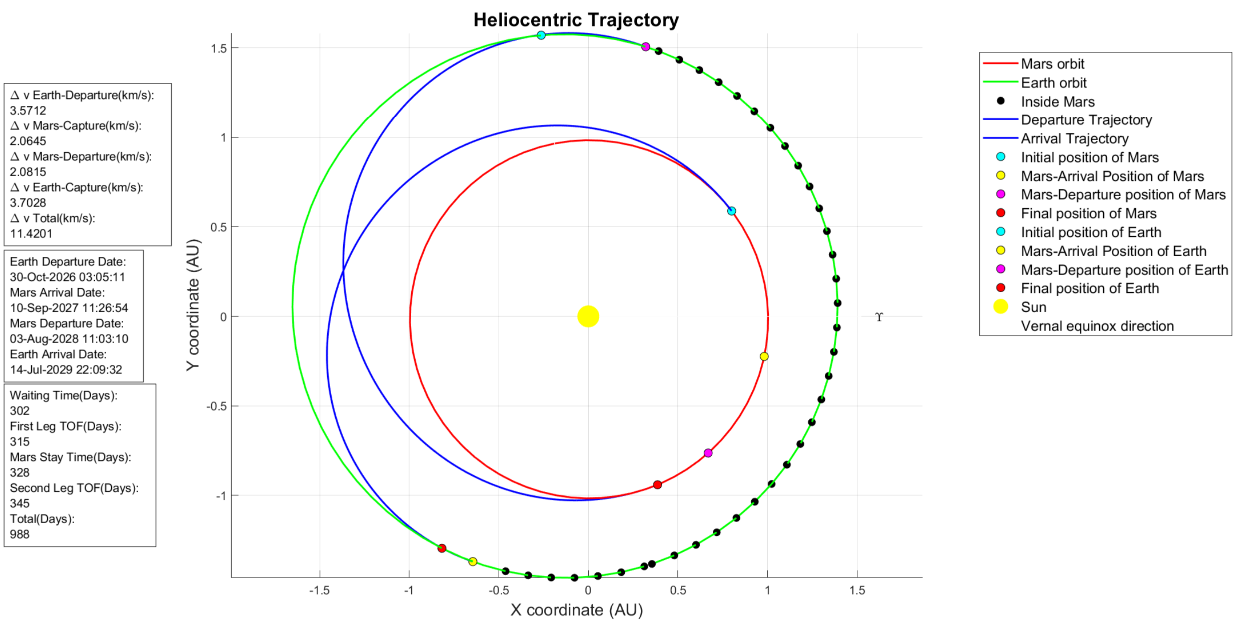

Figure 16. Final mission trajectory