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Benefits of In Situ Resources Utilization (ISRU) for propellant production in a rapid crewed mission to Mars

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Abstract

A way to mitigate the risks of a crewed mission to Mars is reducing the time of transfer to and from the planet, with the downside of a meaningful increase in the mass required to achieve faster travel. In this work it was considered the possibility of producing the propellant needed for the return journey for such a rapid crewed mission on the surface of Mars, taking advantage of In Situ Resources Utilization (ISRU) technology. Although ISRU is typically considered as way to reduce mass when designing a mission to Mars, it was not clear whether this would be the case when the quantity of propellant to produce is very large. Through numerical analysis the Initial Mass in Low Earth Orbit (IMLEO) was computed both for an ISRU fitted spacecraft and a full tank of propellant of comparable size, the numbers were then compared with each other. In particular to estimate the mass of propellant needed for the ISRU spacecraft it was needed to study both ascent and Entry, Descent and Landing (EDL). Moreover for both cases the interplanetary transfer was modeled using a modified rocket equation which takes into account burn losses and jettison of depleted propellant tanks. Five modern and future propulsive systems were considered and for all of them the employment of ISRU resulted in a lower IMLEO, suggesting that even with less optimistic hypothesis this might be a successful approach. On the other hand potential buckling during EDL requires additional work on the lander structure.

Keywords: Fast transfers to Mars, ISRU, Mars exploration, Orbital maneuver, Ascent trajectory

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Nomenclature

Greek symbols

Δv	Change in speed, $\mathrm{m/s}$
γ	Flight path angle , rad
μ	Standard gravitational parameter , $\mathrm{m^3 s^{-2}}$
ϕ	Phase angle between Earth and Mars, rad
π_k	Optimal payload ratio
ρ	Local air density, $\mathrm{kg/m^3}$
ρ_0	Surface density, $\mathrm{kg/m^3}$
ε	Structural ratio of the spacecraft
ε_{pl}	Structural ratio of the ascending payload

Roman symbols

T_{syn}	Synodic period of Earth relative to Mars , day
c_D	Coefficient of drag
D	Drag, N
g	Local acceleration of gravity, $\mathrm{m/s^2}$
g_{0m}	Surface acceleration of gravity on Mars, $\mathrm{m/s^2}$
h_S	Atmospheric scale height for Mars, m
I_{sp}	Specific impulse, s
$k_{lnd\ gear}$	Fraction of landed mass reserved for the landing gear
r	Orbital radius, m
R_m	Mean radius of Mars, m
S_{ref}	Aerodynamic reference surface, $\mathrm{m^2}$

T	Thrust, N
t	time, s
T/W	Thrust over weight ratio

Contents

Chapter 1

Introduction

1.1 Objective and motivation

The objective of this work is to determine whether the usage of In Situ Resources Utilization (ISRU) can be advantageous if used for the production of the propellant needed for the return trip of a rapid crewed mission to Mars. This will be done by calculating the Initial Mass to send from Low Earth Orbit (IMLEO) of the mission if we want to use ISRU and confronting it with that needed if we brought all the propellant from Earth.

It is not easy to understand whether ISRU employment is advantageous, as the case of rapid missions requires a high quantity of propellant and ISRU will require dedicated systems. Moreover, the advantage of not having to bring all mass of propellant for the return trip can eventually be lost by the need for extra dedicated systems able to descend to the surface of Mars, produce a sufficient quantity of propellant and return to orbit. This can be crucial in deciding the final architecture of a rapid mission.

1.2 Context

In this work rapid crewed missions to Mars are considered as main subject, their main characteristic is that the crew is required to spend the lowest time possible in the extra-terrestrial environments. If only impulsive thrust systems are considered, accelerating the crewed spacecraft into a fast trajectory is the way to achieve this. To reduce the mass of propellant needed for these accelerations the crewed segment is reduced to the bare essential and the crew itself can be composed of only two [1]. This would lead to a reduction in the time of exposure to radiations and low gravity environment for the astronauts, other than that it would provide additional safety options in case of medical emergencies during exploration of the planet [1].

The rapid mission of interest follows this timeline [1]: the equipment not necessary for the interplanetary trip of the crew is sent in advance, using a low energy transfer to Mars with two cargo missions; the crewed spacecraft is launched to Low Earth Orbit (LEO) and is accelerated to the fast transfer trajectory; as it arrives to Mars, it performs a propulsive brake and a maneuver of rendezvous and docking with the equipment in orbit. Aerobraking is not considered to isolate the impact of using ISRU. After that the crew lands on the planet and proceeds with the activities on the surface for one month. Finally the crew returns to orbit, docks with the transfer habitat and the spacecraft accelerates to the return trajectory to then reenter on Earth discarding all the spacecraft except a reentry capsule.

The large amount of propellant needed to accelerate to a fast trajectory can represent a problem for the return trip of the crew. As a matter of fact, even if sent in a Hohmann transfer in advance it would itself drive up the IMLEO.

It could be possible to reduce the IMLEO by leveraging ISRU to produce on the surface of Mars the propellant needed for the return, instead of bringing it from Earth. ISRU technology utilizes the resources such as regolith and atmosphere on foreign celestial bodies to produce useful materials.

In general ISRU has the potential to make space exploration more sustainable and more cost effective. Firstly, ISRU can be used to produce life support consumables such as drinking water and oxygen, but also buffer gas to dilute the latter [2]. This means that ISRU may be of help in reducing the mass needed for the life support systems, in particular in the case of missions considering long permanence of the crew on Mars. Moreover, ISRU can provide the propellant for the spacecrafts needed to return to Earth and to transport the crew back to orbit.

In order to achieve the requirements of the mission of interest in an effective manner, the ISRU systems need to be sent in advance in a slow trajectory so that the maneuvers require less propellant to be performed. They would then need to land safely on the surface of Mars and produce a sufficient amount of propellant. Finally the tanks with the propellant needed for the return trip must be able to reach orbit.

1.3 State of the art

Current proposals for missions to Mars involving ISRU aim to use it as means to produce propellant for ascent from the surface of the planet [3], in some other proposals ISRU systems would also produce the propellant needed for the insertion in the return trajectory other than water, oxygen and carbon monoxide for life support during stay on surface [4].

Methods to produce propellant on Mars have been developed by the scientific community. For instance, to produce methane and oxygen, Sabatier reaction and reverse water gas shift may be used, these reaction make use of the carbon dioxide in the atmosphere and Hydrogen brought in the spacecraft [5]. An alternative is the production of oxygen with solid oxide electrolysis cells [6], in this case though the fuel would need to be brought from Earth. An optimistic assumption would be to consider that any fuel used for the propulsion will be able to be produced with ISRU, more realistically to make use of ISRU some mass of either reactants or propellant should be brought to Mars together with the production system itself.

1.4 Original contributions

In order to use ISRU, it is necessary to take it to Mars. A system able to enter the atmosphere, land and then launch in return to orbit with the refueled propellant must be selected, using reasonable assumptions. Once that is designed, it is necessary to redo the transfer trajectories to compare with the non-ISRU option. For that, information about the transfer system must be used [1].

Chapter 2

Problem Definition

2.1 Introduction

2.1.1 Architecture of a rapid crewed mission

As previously mentioned, the rapid mission to Mars which is the basis for this work and that does not consider ISRU for the production of propellant for the return trip is composed of two cargo segments and a crewed segment [1]. The two cargo segments, Cargo I and II, are sent in advance on low energy transfer trajectories to Mars in order to save mass. Cargo I is put in a high eccentricity orbit while Cargo II is sent to the surface of the planet. This architecture includes three spacecrafts which shall be fitted with an Environmental Control and Life Support System (ECLSS), namely a Transfer Habitat (THAB) responsible for the protection of the crew during the transfer to and from Mars, a Surface Habitat (SHAB) and a Mars Ascent Vehicle (MAV) which shall be able to reach orbit from the surface of Mars once all the surface activities have been performed.

When employing ISRU technologies, the propellant needed for the return journey would be produced on Mars, therefore the mission should include landers needed to produce and store the propellant, and then put it in orbit when needed. Other than that the parking orbit of the cargo segments may need to be modified due to the possibility of using propellant produced on the planet to perform maneuvers more cheaply.

2.1.2 Crew travel time optimization

In order to achieve a rapid transfer impulsive maneuvers are employed. To perform them a variety of propulsion systems have been considered [1]. In particular both Thrust over Weight ratios (T/W) and Specific impulses (I_{sp}) of modern and advanced propulsion systems were plotted and the five representative of the performances in Table 2.1 were chosen from a linear regression of this data.

2.2 Components of interest of the mission

If ISRU systems are used to produce the propellant necessary for the return journey, some modification would need to be performed. Instead of sending the tanks with the propellant needed for the return with Cargo I, the

Case	Specific impulse I_{sp} [s]	Engine's Thrust over Weight ratio T/W $[-]$
I	300	190.0
II	1000	75.0
III	3000	32.0
IV	10000	12.6
V	30000	5.4

Table 2.1: Performances of Propulsion systems considered

landers with ISRU technologies would need to be sent on an hyperbolic entry trajectory to Mars, either in one of the two cargo segments already planned or in an additional cargo segment. The propellant produced could then be put on a low altitude circular orbit around Mars (Low Mars Orbit — LMO) and some of it could be used to brake the crewed segment to bring it to this orbit, meaning that also the latter would need to be changed, together with the mission profile which would need to accommodate these new maneuvers.

2.2.1 Performance parameters for mass saved with ISRU

ISRU technologies have the potential to reduce the IMLEO. This potential can concretize if the mass of everything that is necessary to support ISRU is lower than the mass of propellant itself . This mass includes that of the propellant production facilities, of the systems and propellant necessary for landing, and finally that necessary to take off and put the tanks in the target orbit.

If ISRU are not used to produce it, the propellant is sent as part of the Cargo I segment. It starts from a circular orbit around Earth at 500 km of altitude. From there it is put in an escape trajectory in order to achieve an Hohmann transfer to Mars. Finally at the periapsis of the hyperbolic trajectory around Mars, it is slowed down to reach an elliptic orbit with periapsis at 250 km of altitude and apoapsis at 119450 km of altitude [1]. The high eccentricity of the final orbit around Mars makes it so less propellant is needed to reach it from the hyperbolic trajectory or to escape from it when needed.

In order to evaluate the eventual mass savings, a tank of propellant to be produced with ISRU is considered and the mass of all the systems and propellant needed for it to reach orbit around Mars is calculated. Then another tank of propellant is considered, this time starting full from LEO, and the mass needed for it to reach Mars is calculated. Finally the initial masses in LEO for both these cases can be obtained and compared with each other.

Importantly, the tank that starts full from LEO does not actually have the same mass as the tank of propellant produced with ISRU. This is due to the fact that they end up in different orbits around Mars, therefore some of the propellant produced with ISRU is burnt to reach the high eccentricity orbit on the way to the hyperbolic escape trajectory.

2.3 Architecture of the ISRU segment

In the following an ISRU segment will be designed to be implemented in the mission. This ISRU segment of the modified mission must be designed so that the spacecrafts are able to reach Mars, land on the planet, produce enough propellant in the right amount of time and launch the tank into the desired orbit.

2.3.1 Description of components of interest

The ISRU segment of the mission will be composed by several spacecrafts each one capable of producing part of the propellant. This way the technological challenge would be in designing only a small spacecraft, whose dimensions are constrained by the total mass of propellant for the return journey of the crewed segment and by the number of available landing locations.

At this point it should be decided whether to put every tank in the same high eccentricity orbit around Mars that the Cargo I segment uses, or to put the majority of the tanks in a LMO and use some to lower the orbit of the crewed segment and of the remainder of Cargo I. While the first case is appealing because the comparison between masses is one to one, the second appears to be advantageous as the departure burn when the crew leaves Mars would be more efficient. Therefore the second approach was chosen and implemented.

2.3.2 ISRU system design

To proceed with the calculations a rough design of the spacecraft bearing ISRU technology is needed.

In this work it is assumed that all the propulsion systems of Table 2.1 use propellants which can be produced with ISRU. This assumption is considered valid for the lower performing propulsion system whose performances are comparable with modern chemical rocket engines while it becomes more optimistic as better performing thrusters are considered, whose propellant may actually not be available on Mars.

In order to meet mission requirements, the ISRU systems are designed to produce the propellant needed to fill each tank and to put it in orbit in a reasonable amount of time. To achieve this objective the mass of the system is calculated based on the mass of known ISRU systems [7] and normalizing it with their performances.

A possible alternative to producing all the propellant on Mars could be bringing some of it from Earth, this would reduce the mass of the ISRU systems. This alternative appears to not be worthwhile because the reduction in mass of the ISRU systems is minimal. The technological complexity on the other hand would increase considerably due to the necessity of keeping part of the propellant in orbit with the need for additional systems and orbital maneuvers.

2.3.3 Ascent trajectory

In order to bring the propellant to orbit an ascent trajectory was calculated and losses were computed. A two stage to orbit (TSTO) strategy was chosen, alongside a gravity turn trajectory to be integrated from the equations of motion of the rocket. Gravity and atmospheric losses were then evaluated along the trajectory.

To obtain the best performances it was assumed that all the propulsion systems of Table 2.1 could be used for the ascent. This assumption might not be realistic for the better performing engines because of their extravagant propulsive mechanisms. They could for instance not work in an atmosphere or could be subject to early deterioration.

For the ascent configuration both a one stage to orbit and a TSTO were considered. Because of the great size of the payload the TSTO strategy was preferred. Each stage was designed with a different Thrust and staging was not designed to occur at any particular moment during the ascending trajectory.

In order to evaluate gravity and atmospheric losses along the ascent trajectory an iterative approach was used. First the gravity losses were estimated considering them equal to 80% of the vertical velocity needed to reach r_f starting from r_0 in a vacuum [8], therefore using the following equation:

$$\Delta v_{grav\ loss} = 0.8\sqrt{2 g_{0m} R_m^2 \left(\frac{1}{r_0} - \frac{1}{r_f}\right)} \tag{2.1}$$

Where g_{0m} is the surface acceleration of gravity on Mars and R_m is the volumetric mean radius of Mars. In the case of interest $r_0 = R_m$. On the other hand the initial guess for atmospheric losses was set to zero because of the thinness of Martian atmosphere and the fact that the rocket is slower where the atmosphere has higher density. Once the trajectory was determined, the gravity and atmospheric losses were integrated along it using respectively Eq. (2.2) and Eq. (2.3). Having these new guesses the trajectory calculation was performed again and the process was iterated up to convergence.

$$\Delta v_{grav\ loss} = \int_{t_0}^{t_f} g \sin\gamma \, dt \tag{2.2}$$

$$\Delta v_{drag\ loss} = \int_{t_0}^{t_f} \frac{D(t)}{m(t)} \, dt \tag{2.3}$$

Where:

- t_0 is the initial time.
- t_f is the final time.
- g is the local acceleration of gravity.
- γ is the flight path angle, i.e. the acute angle between the velocity and the local horizon.
- $D(t) = c_D \frac{1}{2} \rho v^2 S_{ref}$ is the Drag force at time t.
  With
  - c_D that is the adimensional coefficient of drag.
  - ρ that is the local air density.
  - v^2 that is the square of the speed.
  - S_{ref} that is the reference aerodynamic surface of reference.
- $m(t)$ is the mass of the ascending spacecraft at time t.

In order to obtain the mass of propellant to put tanks in LMO a couple of approaches were considered. A first simple approximation could be obtained by considering an initial vertical segment of the launch up to the target altitude and then an impulsive horizontal thruster firing to circularize the orbit. This approximation was discarded because it was deemed too unrealistic and because the alternative was implementable. The alternative in question is a pure gravity turn trajectory which, although is not the optimal ascent trajectory, is close to it [9].

2.3.4 Entry, descent and landing

EDL are the three phases needed to deliver a spacecraft to the surface of a planet from an interplanetary transfer trajectory or an orbit around the planet [10]. During these phases not only the probe is exposed to the high dynamic pressure and temperature related to the hypersonic regimen but it must also withstand the strong propulsive deceleration needed to land. In the case of interest this is rendered even more difficult by the fact that Mars' atmosphere is very thin and therefore provides limited aerodynamic breaking options. Notably, if the propellant was brought from Earth instead of being produced on the planet, it would not need to undergo EDL, as it could instead stay in orbit around the planet.

The spacecrafts fitted with ISRU technology enter the atmosphere of Mars directly from an hyperbolic trajectory. To decelerate during descent from this trajectory the spacecraft aims at breaking as much as possible with the atmosphere by using an Hypersonic Inflatable Aerodynamic Decelerator (HIAD). This technology would permit the storage of a large breaking surface for use during entry. After the deployment of the HIAD the spacecraft would decelerate to supersonic speed and at that point the inflatable surface would be discarded and propulsive deceleration would bring the platform to a full stop on the surface [11].

2.3.5 Interplanetary transfer

The first phase that the spacecrafts carrying ISRU must undergo from LEO is the insertion in interplanetary transfer. In order to obtain the desired transfer, each spacecraft must receive an acceleration, characterized by the change in speed Δv, such that the resulting escape trajectory results in it.

Some assumptions were made for the interplanetary transfer. The orbits of Earth and Mars were assumed to be circular and coplanar. Under these conditions the optimum energy interplanetary transfer is the Hohmann transfer. Furthermore when calculating the required Δv for the transfer it was not considered any three body effects during the maneuvers.

Other than that because of the great amount of propellant needed to achieve rapid transfer, most of the orbital maneuvers, included those necessary for the Hohmann transfer, were evaluated taking into consideration burn losses and tank discarding. Therefore the fact that the maneuver does not happen instantaneously is taken into account, as well as considering that during the maneuver each tank of propellant was discarded to save mass as soon as it was depleted . In particular for the disposal of the tanks it was considered the limit as the number of tanks tends to infinity [12].

2.3.6 Final Design

In Figure 2.1 a diagram of the spacecraft designed as described until now can be seen. All of its main components are labeled as follows:

- A is the stage needed for insertion in Hohmann transfer, it is discarded as soon as this maneuver is completed. If all the spacecrafts are propelled by a single propulsion system after in orbit assembly, this would represent the part of propellant and structure needed for each of them.
- S is the heat shield that contribute to thermal protection during entry.

- H is a container for the stowed HIAD, which is deployed in high atmosphere and is discarded as the speed reaches Mach 2.02 [11].
- L is the landing gear.
- B is the propulsive breaking and landing stage, it serves as launch platform when the production of propellant is completed. This stage is also fitted with the ISRU system.
- C and D are respectively the first and second stage of the ascending vehicle.
- PL is the payload of the ascending vehicle, i.e. the tank of propellant that should arrive in LMO.

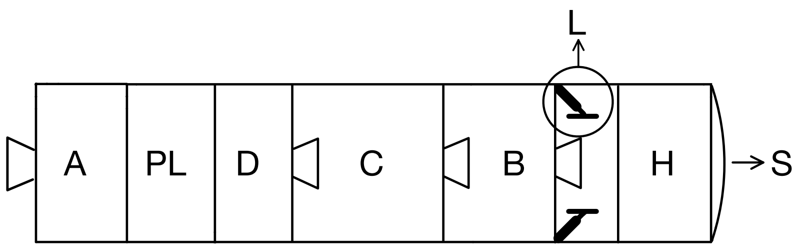

Figure 2.1: Scheme of the ISRU spacecraft as is in LEO

2.4 Problem setup

For the problem to be solved a number of parameters had to be set, other than I_{sp} and T/W which appear in Table 2.1. While some of these are physical quantities, most of them are parameters that identify the problem and to be set require additional assumptions.

The mission considered in this work is set in the future, because of this the parameters representing the performances of the technology used are set at the best values available today. This decision was taken in the hope that when the mission will be developed advancements in technology will make this assumption conservative.

Among the parameters to be set, the structural ratios of the ascending spacecraft and of it's payload can be found. They are considered different from each other because of the fact that the tanks of propellant do not require an engine to fulfill their purpose. The structural ratio of the spacecraft is defined as usual:

$$\varepsilon = \frac{m_{str}}{m_{str} + m_{pr}} \tag{2.4}$$

Where m_{str} is the mass of the structure and m_{pr} is the mass of propellant. This parameter was set to $\varepsilon = 0.05$, as this value "represents the lower limit of the structural mass of a rocket" [9]. On the other hand, in accordance with the mission without ISRU, the structural ratio for the tank is [1]:

$$\varepsilon_{pl} = \frac{m_{str}}{m_{tot}} = \frac{K}{K+1} \tag{2.5}$$

With $K \approx 0.04$ being the ratio of mass of the tank and mass of the propellant it can hold [1]. In this case the denominator is the total mass of the tank, this is because in this case $m_{tot} = m_{str} + m_{pr}$. Moreover the fact that $\varepsilon_{pl} < \varepsilon$ is consistent if it is assumed that the tanks do not include a propulsion system.

Other parameters chosen in order to perform the calculations include the coefficient of drag $c_D = 0.2$, whose value was chosen as is to follow "a general guideline" [8] and so as to simplify the aerodynamic model. Other than that multiple constants of proportionality were defined to obtain first estimates of dimensions which are considered scaling with mass:

- $\left(\frac{S_{ref}}{m_0^{2/3}}\right)^* = \frac{125.66}{32899^{2/3}} \frac{\mathrm{m}^2}{\mathrm{kg}^{2/3}}$ relates the initial mass m_0 to the reference surface S_{ref} for aerodynamic drag during launch. It was estimated considering the ascending spacecraft a cylinder and both propellant and payload a mix of liquid oxygen and liquid hydrogen.
- $\left(\frac{m_{HIAD}}{m_i^{pb}}\right)^* = \frac{11.5}{74.2}$ relates the mass of the HIAD m_{HIAD} to the wet mass of the lander at beginning of propulsive braking m_i^{pb} and was estimated from current proposals of a landing system using this technology [11].
- $\left(\frac{m_{HS}}{m_i^{pb}}\right)^* = \frac{5.707}{74.2}$ relates the mass of the heat shield m_{HS} to m_i^{pb} and was estimated from the same proposal used earlier [11].
- $k_{lnd\ gear} = 0.025$ represents the fraction of landed mass reserved for the landing gear, this value was taken from literature and is in line with that of the Apollo Lunar Entry Module (LEM) [11].
- $m^*_{ISRU} = \frac{231-177}{5110/(600 \cdot 24.67 \cdot 3600)}\mathrm{s}$ is the mass of the ISRU systems normalized with the rate of production of propellant of Modern ISRU systems [13]. As a matter of fact, the system considered has mass of 231 kg

from which 177 kg of mass needed for storage [13] are already accounted for in the structure of the ISRU tank. This system is capable of producing 5110 kg of propellant in 600 sols, i.e. $600 \cdot 24.67 \cdot 3600\mathrm{s}$

Finally, some parameters describing the characteristics of the different celestial bodies involved in the calculations were used [14], [15].

Chapter 3

Implementation of model for ISRU spacecraft on arrival

3.1 Introduction

As mentioned in the previous chapter, the first model to implement in order to get the IMLEO of the spacecraft needs to represent well what happens during launch from Mars. After that it will be possible to evaluate the mass of the entry vehicle with a different model.

3.2 Ascent calculations

In order to achieve the target orbit around Mars the ascending rocket will perform a two stage to orbit gravity turn ascent. This trajectory may be split in three phases: a first vertical acceleration, followed then by a pitch-over maneuver that will change slightly the direction of the rocket and finally the effect of gravity will bend the trajectory more and more until the rocket reaches the altitude of 250 km at the speed needed to maintain the circular orbit.

Assumptions

Some hypothesis were considered to make these calculations simpler while keeping them relevant. In this work the pitch-over maneuver was modeled as an instantaneous change in the direction of motion of the rocket at the end of the vertical ascent. This hypothesis is deemed valid because the attitude variation which kicks off the gravity turn maneuver always results to be of an angle under $5°$, therefore the rocket could reach this angle during the vertical ascent with little change to the total losses.

An estimate of the density of Mars' atmosphere was needed to use Eq. 2.3 to calculate the drag losses along the trajectory. To do so an exponential model was assumed with the following form:

$$\rho(h) = \rho_0 e^{-\frac{h}{h_S}} \tag{3.1}$$

Where:

- $\rho(h)$ is the density at the altitude h
- $\rho_0 = 0.020\mathrm{kg/m^3}$ is the density on the surface of Mars [14]
- $h_S = 11.1\mathrm{km}$ is the scale height for Mars [14]

The mass of the tank to put in orbit around mars was considered as $m_{pl} = 2000\mathrm{kg}$. Having this it will be shown that it is possible to obtain the initial mass of the rocket as well as all the parameters bound to it. A choice for the landing sites was not performed in this work, therefore to remain conservative the rotation of the planet was not considered when evaluating the ascent trajectory.

Finally, the thrust of each stage during ascent was considered constant, however the thrust of the two stages are free to be different with one another. Also, given the thinness of Martian atmosphere, the effect of pressure on thrust was considered negligible.

Ascent stages calculations

With the given assumptions it is now possible to construct a model for the ascent trajectory. It is described the procedure employed to obtain the ascent trajectory.

First of all staging must be addressed, this can be done starting from the total velocity v^* that the propulsion system shall provide to reach orbit. v^* is the sum of the orbital speed and all the losses considered, the first guess values for these losses are obtained by plugging $r_f = R_m + 250000\mathrm{m}$ in Eq. 2.1 and by considering $\Delta v_{drag\ loss} = 0\mathrm{m/s}$. Having the value of v^* and considering the same propulsion and structure technology for all stages the optimal payload ratio π_k turns out to be the same for each stage [16]:

$$\pi_k = \frac{e^{-\beta} - \varepsilon}{1 - \varepsilon} \tag{3.2}$$

Where: $\beta = v^*/(N v_e)$, $N = 2$ is the number of stages, $v_e = I_{sp} g_{0e}$ is the exhaust velocity of the stages [9] and $g_{0e} = 9.82\mathrm{m/s^2}$ is Earth's mean surface gravity [14]. Having set m_{pl} it is then possible to obtain the initial mass $m_{0\ to}$ of the ascent vehicle as $m_{0\ to} = m_{pl}/\pi_k^2$, with the payload ratio squared at the denominator because two stages are considered.

Knowing the value of π_k and the performances of the engines it is possible to calculate the burnout time of the first and second stage, respectively t_{bo1} and t_{bo2}. Considering for instance the first stage the following equation stands:

$$m_{0\ to} - \dot{m}_1 t_{bo1} = m_{str1} + m_{pl1}$$

Where $\dot{m}_1$ is the mass flow rate of the engine of the first stage, m_{str1} is the structural mass of the first stage and m_{pl1} is the mass of the payload of the first stage, i.e. the total mass of the second stage m_{02}. The equation can then be developed by writing the mass of the structure using ε, $m_{str1} = (m_{0\ to} - m_{pl1})\varepsilon$ and dividing both sides by $m_{0\ to}$. The equation becomes then:

$$1 - \frac{\dot{m}_1}{m_{0\ to}} t_{bo1} = (1 - \pi_k)\varepsilon + \pi_k$$

Having substituted $\pi_k = m_{pl1}/m_{0\ to}$.

Having done that $\dot{m}_1$ can be written as a function of the thrust of the first stage T_1 and of the exhaust velocity v_e, $\dot{m}_1 = T_1/v_e$ [9]. The exhaust velocity can then be evaluated using the specific impulse, and the thrust can be written as a multiple of the weight on ground level by defining the thrust parameter T_{p1} and considering the mean surface gravity of Mars as $g_{0m} = 3.73\text{m/s}^2$ [14], $T_1 = T_{p1} g_{0m} m_{0\ to}$, all of this results in the following:

$$1 - \frac{T_{p1} g_{0m}}{I_{sp} g_{0e}} t_{bo1} = (1 - \pi_k)\varepsilon + \pi_k$$

Finally t_{bo1} can be made explicit as a function of the other parameters:

$$t_{bo1} = \frac{1 - (1 - \pi_k)\varepsilon - \pi_k}{T_{p1} g_{0m}} I_{sp} g_{0e} \tag{3.3}$$

By following the same reasoning a similar expression can be obtained for t_{bo2} in which the thrust parameter of the second stage T_{p2} appears instead of T_{p1}.

Considering all the parameters that are already set, the ascending trajectory depends only on T_{p1}, T_{p2}, the time at which the pitch-over maneuver occurs t_{po} and the initial angle γ_0 between the velocity vector after pitch-over and the local horizon. In order to obtain a working trajectory, T_{p2} and t_{po} were set and a numerical solver was used to obtain the values of the other two parameters so that the rocket arrives at the target altitude with an horizontal trajectory. These conditions entail having the speed to sustain circular orbit as the estimates for gravity and drag losses converge with successive iterations. The choice of T_{p2} and t_{po} may be studied in future works to obtain an optimal ascent.

In order to get the ascent trajectory two sets of equations of motion were defined, one for the initial vertical flight and another for the actual gravity turn. During the vertical flight the independent variables are the orbital radius r, the speed v and time t. It is possible to model this vertical ascent as a system of the following two ordinary differential equations representing the derivatives $\frac{dr}{dt}$ and $\frac{dv}{dt}$ as functions of r, v and t:

$$\frac{dr}{dt} = v \tag{3.4}$$

$$\frac{dv}{dt} = \frac{T_1}{m_1(t)} - g(r) - \frac{D(r, v)}{m_1(t)} \tag{3.5}$$

Where it is considered:

- $m_1(t) = m_{0\ to} - \dot{m}_1 t$ for the instantaneous mass.
- $T_1 = T_{p1} g_{0m} m_{0\ to}$ for the thrust of the first stage.
- $g(r) = g_{0m}\left(\frac{R_m^2}{r^2}\right)$, with $R_m = 3389.5\text{km}$ being the volumetric mean radius of Mars [14].
- $D(r, v) = \frac{1}{2}\rho(r - R_m)v^2 S_{ref1} c_D$, considering:
  - Eq. 3.1 for the air density.
  - $S_{ref1} = \left(\frac{S_{ref}}{m_{0\ to}}\right)^* m_{0\ to}^{2/3}$ for the aerodynamic reference surface.

After the vertical flight the pitch-over maneuver occurs and therefore the instantaneous angle between the velocity and the local horizon γ appears in the new equations as an independent variable. While Eq. 3.4 remains the same, Eq. 3.5 has to be modified in Eq.s 3.6 and 3.7 is added to model $\frac{d\gamma}{dt}$ [9]:

$$\frac{dv}{dt} = \frac{T_1}{m_1(t)} - g(r)\sin\gamma - \frac{D(r,v)}{m_1(t)} \tag{3.6}$$

$$\frac{d\gamma}{dt} = -\left(\frac{g(r)}{v} - \frac{v}{r}\right)\cos\gamma \tag{3.7}$$

The complete ascent trajectory is obtained by numerical integration of the equations above. Staging happens along the trajectory and affects T_1, $m_1(t)$ and S_{ref1} that get substituted respectively by $T_2 = T_{p2} g_{0m} m_{0\ to}$, $m_2(t) = m_{02} - \dot{m}_2(t - t_{bo1})$ and $S_{ref2} = \left(\frac{S_{ref}}{m_{0\ to}}\right)^* m_{02}^{2/3}$, in which $m_{02} = m_{0\ to}\pi_k$ and $\dot{m}_2 = T_2/v_e$.

As it can be seen from Eq. 3.7 as long as the rocket stays vertical, i.e. $\gamma = \pi/2$, there is no gravity turn effect on γ. This is the reason why an initial imposed change in attitude is needed to start the gravity turn trajectory. Moreover the trajectory needs an initial vertical ascent to accelerate as for v that tends to 0 the right hand side of Eq. 3.7 tends to infinity.

As mentioned in Chapter 2, the first trajectory is calculated considering no drag losses and estimating the gravity losses with the potential model of Eq. 2.1. Therefore an initial guess for v^* is available to be used in the search for the values of T_{p1} and γ_0.

Once these values are obtained it is possible to integrate the trajectory and as a result the values of r, v and γ become available for each instant of the integration. With this data it is possible to use Eq.s 2.2 and 2.3 to obtain the exact gravity and drag losses on the first trajectory.

The new estimates for losses along the trajectory can be used to obtain a new value for v^* and a new cycle can begin. This process is iterated until the relative difference between two successive values of the losses is considered small enough, in this work this condition was met when both normalized differences fell below 10^{-6}.

The output of this iterative process needed for the calculation of the IMLEO is the best estimate for the initial mass of the spacecraft that will take off $m_{0\ to}$. In table 3.1 the value of $m_{0\ to}$ as well as those of $\Delta v_{grav\ loss}$ and $\Delta v_{drag\ loss}$ for all the propulsion cases considered.

Case	$\Delta v_{grav\ loss}$ [m/s]	$\Delta v_{drag\ loss}$ [m/s]	$m_{0\ to}$ [kg]
I	1036.27	53.30	10373.85
II	1035.21	62.06	3244.71
III	1033.77	65.76	2348.37
IV	1032.80	67.23	2098.56
V	1019.45	69.12	2032.31

Table 3.1: Results of the iterative integration of the ascent trajectory

These results are in line with the value of gravity losses predicted by the potential model equal for all cases:

$$\Delta v_{grav\ loss}^{potential} = 1054.32\text{m/s}$$

In the following figures 3.1 to 3.5 the final ascent trajectories that were found for every propulsion system of table 2.1 are shown. In red and in black are shown the sections in which respectively the first and second stage are fired.

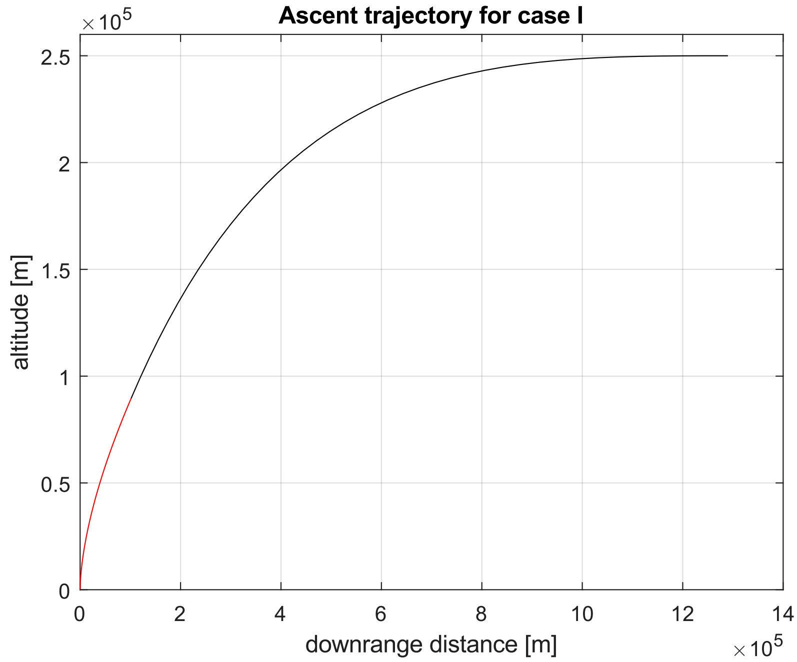

Figure 3.1: Ascent trajectory for case I

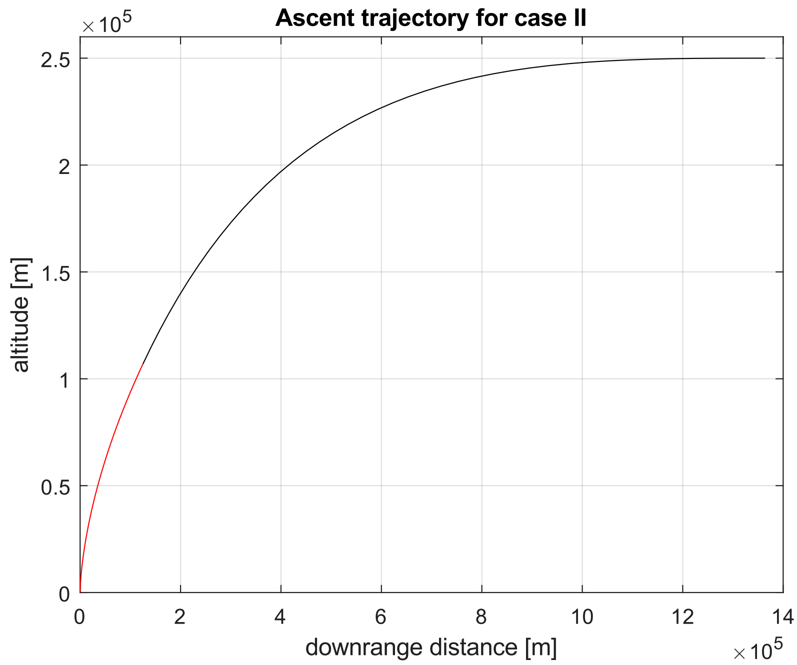

Figure 3.2: Ascent trajectory for case II

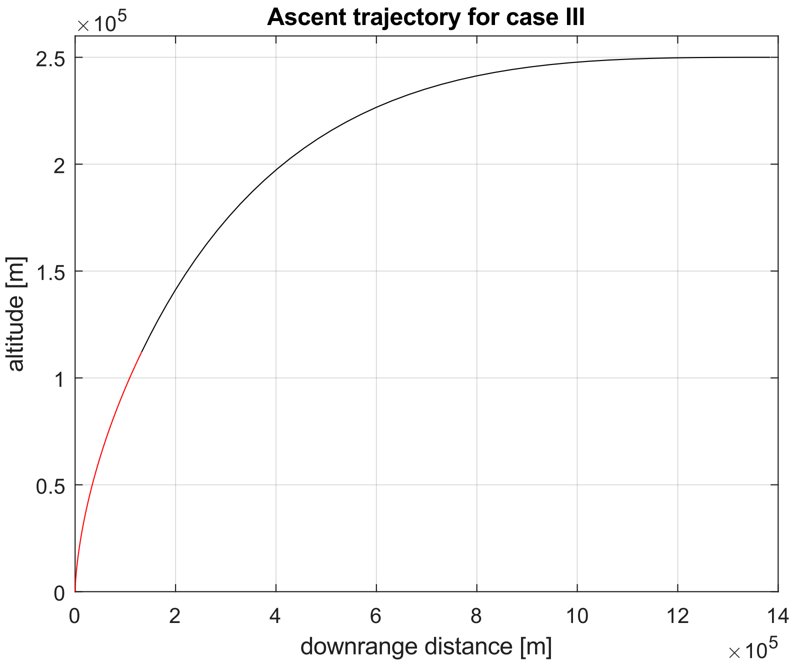

Figure 3.3: Ascent trajectory for case III

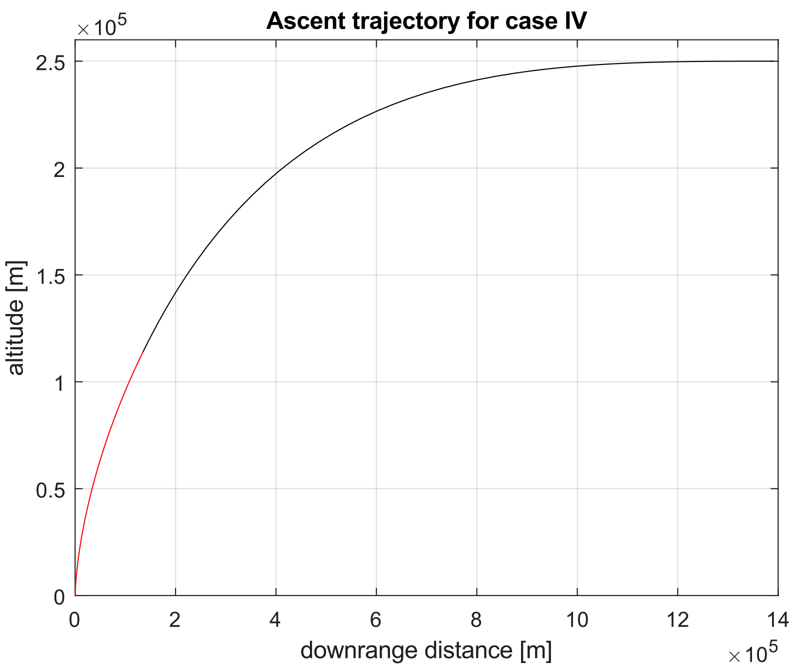

Figure 3.4: Ascent trajectory for case IV

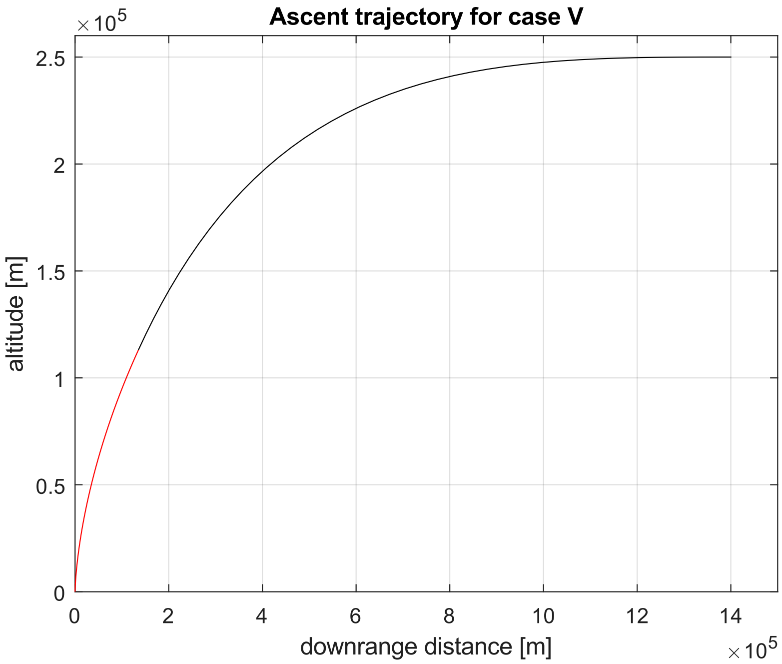

Figure 3.5: Ascent trajectory for case V

3.3 Entry, descent and landing calculations

Once the calculations concerning the ascent trajectory are performed the next phase to study is EDL. With the total amount of propellant needed during ascent the mass of the systems that need to land can be obtained. To make these systems reach the ground the strategy described in Section 2.3.4 is employed. In particular the jettison of the HIAD happens as the entry vehicle reaches a speed of Mach 2.02, and the propulsive burn for landing needs $\Delta v_{br} = 601\mathrm{m/s}$ [11].

3.3.1 Assumptions

In this case as well before proceeding with the calculations some hypothesis were considered. Firstly it was considered that the engines described by Table 2.1 could be used for propulsive landing as well. Moreover the same structural ratio ε used for the ascending stages was used for the propulsive brake system.

Secondly it was considered that the time available for the production of propellant is that between the moment of arrival of the landers on Mars and the next moment in which Earth is in phase with Mars for a Hohmann transfer to depart from Earth orbit. This assumption is conservative as the crewed segment will employ a fast transfer that requires a smaller phase angle between Earth and Mars and because in any case the time of transfer of the crew to Mars would be available for propellant production.

Finally during propulsive braking the heat shield was assumed to remain attached during the entirety of the maneuver in order to simplify calculations while remaining conservative.

3.3.2 Computational Analysis

The mass of the entry vehicle m_{entry} can be modeled as the following sum:

$$m_{entry} = m_{str\ to} + m_{ISRU} + m_{prop\ lnd} + m_{HS} + m_{lnd\ gear} + m_{HIAD} \tag{3.8}$$

Where:

- $m_{str\ to} = (m_{0\ to} - m_{pl})\varepsilon + m_{pl}\varepsilon_{pl}$ is the mass of the structure needed for take off, as all the propellant for ascent and needed as payload is produced on Mars.
- m_{ISRU} is the mass of the ISRU systems.
- $m_{prop\ lnd}$ is the mass needed for propulsive braking and includes the mass of propellant.
- m_{HS} is the mass of the heat shield.
- $m_{lnd\ gear}$ is the mass of the landing gear.
- m_{HIAD} is the mass of the HIAD.

Therefore to obtain m_{entry} it will be necessary to first evaluate m_{ISRU}, then

$$m_i^{pb} = m_{prop\ lnd} + m_{HS} + m_{lnd\ gear} + m_{ISRU} + m_{str\ to}$$

will be evaluated as the propulsive brake is studied. And finally m_{HIAD} will be estimated using $\left(\frac{m_{HIAD}}{m_i^{pb}}\right)^*$ and m_i^{pb}.

In order to evaluate m_{ISRU} the following relation was employed that represents direct proportionality with the mass of propellant to produce m_{pp} and inverse proportionality with the time available to produce it t_{pp}:

$$m_{ISRU} = m^*_{ISRU} \frac{m_{pp}}{t_{pp}} \tag{3.9}$$

On one hand m_{pp} can be simply calculated after obtaining the ascent trajectory with the relation:

$$m_{pp} = (m_{0\ to} - m_{pl})(1 - \varepsilon) + m_{pl}(1 - \varepsilon_{pl})$$

On the other hand calculating t_{pp} can prove to be harder. To do so the methods described in [17] were employed to study the Hohmann transfer from Earth to Mars under the assumption of coplanar circular orbits, using the orbital data for the celestial bodies involved available in [14] and [15]. The following results were obtained:

- $\phi_i = 0.7648\text{rad}$ for the initial phase angle between Earth and Mars to perform the Hohmann transfer.
- $\text{T}_{syn} = 779.93\text{day}$ for the synodic period of Earth relative to Mars, i.e. the time needed for Earth to span a 2π rad angle in a reference frame fixed with Mars.
- $t_{12} = 2.237 \times 10^7\text{s} = 258.9\text{days}$ for the time of flight to Mars along the Hohmann transfer
- $\phi_f = -1.312\text{rad}$ for the phase angle between the planets at the time of arrival to Mars. With the negative sign representing the fact that Earth is preceding Mars at this time. Note that this particular makes sense also because the transfer takes more than 6 months and at the end of it Mars must be π rad away from where Earth was in the beginning.

With this data the following expression can be used to obtain t_{pp}:

$$t_{pp} = \frac{\text{T}_{syn}}{2\pi}(2\pi - \phi_i + \phi_f) = 522.17\text{day} = 4.512 \times 10^7\text{s} \tag{3.10}$$

Knowing m_{pp} and t_{pp} it is then possible to obtain m_{ISRU} using Eq. 3.9.

The next step to get m_{entry} is calculating the value of m_i^{pb}. To do so let's consider the lander during propulsive braking, i.e. when the HIAD has already been jettisoned. In this case the system of the following equations is valid, with the unknowns being the mass at the beginning of engine firing m_i^{pb} and that at burnout m_f^{pb}:

$$m_f^{pb} = m_{str\ to} + m_{ISRU} + m_{str\ pb} + m_{lnd\ gear} + m_{HS} \tag{3.11}$$

$$m_i^{pb} = m_f^{pb} e^{\frac{\Delta v_{br}}{g_{0e} I_{sp}}} \tag{3.12}$$

Where $m_{str\ pb}$ is the structure needed for propulsive braking and the second equation is the Tsiolkovsky rocket equation [9].

In the first equation it is possible to identify the mass of the landed system at touch down

$$m_{td} = m_{str\ to} + m_{ISRU} + m_{str\ pb} + m_{lnd\ gear},$$

considering that

$$m_{lnd\ gear} = k_{lnd\ gear} m_{td},$$

m_{td} can be rewritten as

$$m_{td} = (m_{str\ to} + m_{ISRU} + m_{str\ pb}) \frac{1}{1 - k_{lnd\ gear}},$$

to account for $m_{lnd\ gear}$. It is then possible to substitute in Eq. 3.11 the expression just found for m_{td} with

$$m_{str\ to} = (m_{0\ to} - m_{pl})\varepsilon + m_{pl}\varepsilon_{pl},$$

$$m_{str\ pb} = (m_i^{pb} - m_f^{pb}) \frac{\varepsilon}{1 - \varepsilon}$$

and substituting also

$$m_{HS} = \left(\frac{m_{HS}}{m_i^{pb}} \right)^* m_i^{pb},$$

Eq. 3.11 can be written as

$$m_f^{pb} = (m_f^{pb})_0 + k_m m_i^{pb} \tag{3.13}$$

with

$$(m_f^{pb})_0 = \frac{(m_{0\ to} - m_{pl})\varepsilon + m_{pl}\varepsilon_{pl} + m_{ISRU}}{1 - k_{lnd\ gear} + \frac{\varepsilon}{1-\varepsilon}}$$

and

$$k_m = \frac{(1 - k_{lnd\ gear})k_{HS} + \frac{\varepsilon}{1-\varepsilon}}{1 - k_{lnd\ gear} + \frac{\varepsilon}{1-\varepsilon}}$$

Substituting Eq. 3.13 in Eq. 3.12 it is possible to obtain

$$m_i^{pb} = \frac{(m_f^{pb})_0 e^{\frac{\Delta v_{br}}{g_{0e} I_{sp}}}}{1 - k_m e^{\frac{\Delta v_{br}}{g_{0e} I_{sp}}}} \tag{3.14}$$

Once the value of m_i is known, the following relation can be used to evaluate m_{HIAD}:

$$m_{HIAD} = \left(\frac{m_{HIAD}}{m_{lnd}} \right)^* m_i^{pb}$$

In table 3.2 the values of m_{entry} obtained with Eq. 3.8 for every case considered can be seen.

Case	m_{entry} [kg]
I	1006.27
II	244.92
III	160.88
IV	138.30
V	132.39

Table 3.2: Results of the EDL analysis

This would conclude the evaluation of m_{entry}. However a study of the resistance to buckling of the empty tanks during descent was performed to check whether they could resist the peak entry deceleration.

The maximum contingency factor considered for deceleration was considered $n_{max} = 3\, g_{0e}$, i.e. the maximum considered in [11] for crew entry. This value is therefore optimistic as the absence of crew would remove the constraints on deceleration necessary for life support.

Both a buckling analysis on cylindrical shells [18] and on stiffened cylindrical shells [19] were performed with the intent of finding a structure that could resist and that could respect the structural ratio. All attempts at finding such a structure failed, considering the materials of tanks to be Aluminum, Steel or Titanium, and making the structural ratio vary to permit the usage of thicker walls.

Further studies on structural analysis are therefore needed to permit the usage of conventional tanks for produced propellant storage. A possible solution that needs deeper study as well is the usage of inflatable tanks for fuel storage, as this would possibly remove the problem of buckling. From this point onward it was assumed that it will be possible to use this kind of tank with the chosen structural ratio.

Chapter 4

Comparison between the IMLEO of ISRU spacecraft and full tank

4.1 Introduction

With the mass of the entry vehicle available it is possible to implement the model mentioned in Section 2.3.5 to obtain the IMLEO for both the case of the spacecraft with ISRU, m_{LEO}^{ISRU}, and the case of a tank of propellant of comparable size sent already full from LEO, $m_{LEO}^{full\ tank}$.

As the evaluations of these masses are performed a comparison can be done between them. In particular a ratio of them can be obtained for each of the five cases of table 2.1 and therefore it can be seen how this value changes as the engines become more efficient.

4.2 Implementation of model for interplanetary transfer

To implement the interplanetary transfer model first the value of m_{LEO}^{ISRU} was obtained, then the mass of the tanks that would depart full from LEO and arrive in the high eccentricity orbit around Mars (HMO - High Mars Orbit) was determined so as to make it comparable with m_{pl}, which is instead put in LMO. Finally having this mass it was possible to obtain the value of $m_{LEO}^{full\ tank}$.

4.2.1 Assumptions

As mentioned in Section 2.3.5 the orbital maneuvers were calculated under the assumption of coplanar circular Earth and Mars orbits and without considering third body effects.

Moreover direct entry on Mars from hyperbolic trajectory was assumed for the ISRU spacecraft, this makes it so this spacecraft only needs a maneuver of insertion in Hohmann transfer from LEO. In contrast the full tanks are assumed to employ a propulsive capture maneuver at their arrival to Mars as well.

The full tank that arrives in HMO from Earth should have a smaller mass than the tank filled with propellant produced with ISRU. This can be explained by considering that when the latter is used to leave Mars, it needs first to accelerate to the HMO on the way to the hyperbolic trajectory of insertion to the fast transfer to Earth.

4.2.2 Computational analysis

In the following calculations the rocket equation Eq. 4.1 that accounts for burn losses and tank disposal in the limit as the number of tanks tends to infinity [12] was used, in line with what was previously mentioned, for the maneuvers related to interplanetary transfer.

$$m_p = m_*(1-\varepsilon_t)\left(\exp\left[\frac{1}{1-\varepsilon_t}\frac{\Delta v_{imp}}{g_{0e}I_{sp}}\left(1+\frac{1}{24}\frac{\mu}{r^3}\frac{g_{0e}^2 I_{sp}^2}{T^2}m_p^2\right)\right]-1\right) \tag{4.1}$$

In this equation:

- m_p is the mass of propellant necessary for the maneuver.
- m_* is the mass at the end of the maneuver.
- ε_t is the structural ratio of the tanks expelled during the maneuver, in the case of interest $\varepsilon_t = \varepsilon_{pl}$.
- Δv_{imp} is the Δv that would be needed for the maneuver if it was impulsive.
- μ is the standard gravitational parameter of the planet around which the maneuver happens.
- r is the orbital radius of the spacecraft when it is executing the maneuver.
- T is the thrust used during the maneuver.

It should be noted that m_* includes the mass of the propulsion system needed for the maneuver, other than the mass of the effective payload that needed to change trajectory m_{*eff}. For that reason the following equation stands [12]:

$$m_* = m_{*eff} + m_{engines} + m_{other} \tag{4.2}$$

In which the mass of the propulsion system was divided into the mass of engines $m_{engines}$ and the mass of all the other structure necessary m_{other}. Both of these masses were considered scaling with the thrust T using two thrust to weight ratios. For $m_{engines}$ it was used $(T/W)_{engines}$ with the values of T/W from Table 2.1. On the other hand $(T/W)_{other}$ for m_{other} was estimated with ”Fregat's upper stage data” [12] from [20]. We can therefore substitute the following expressions in Eq. 4.2:

$$m_{engines} = \frac{1}{g_{0e}(T/W)_{engines}} T$$

$$m_{other} = \frac{1}{g_{0e}(T/W)_{other}} T$$

Having done that and considering the value of Δv_{imp} necessary for the maneuver under exam it is possible to plug Eq. 4.2 into Eq. 4.1 and therefore obtain a relation between m_p and T. This relation can then be used to evaluate the mass of propellant needed for a maneuver by searching for the value of T that minimizes m_p.

It is now shown how Δv_{imp} was calculated for the Hohmann transfer. For the maneuver from LEO to Hohmann a Δv for ejection from Earth orbit Δv_{eject} is needed. It's value is calculated with the patched conics method [17] by considering the excess speed $(v_\infty)_{eject}$ needed as the spacecrafts leave the region of space in which Earth's gravity is the most important, i.e. Earth's sphere of influence (SOI), such that summed with the planet's orbital speed the spacecraft ends at the periapsis of the Hohmann transfer. It is then possible to obtain Δv_{eject} as the speed to add to the orbital velocity of the spacecraft parked in the circular LEO at an altitude of $h_{LEO} = 500\text{km}$ so that it is put in the hyperbolic trajectory with asymptotic speed $(v_\infty)_{eject}$.

The tanks full of propellant sent from Earth need to perform an injection burn as well characterized by Δv_{inject} and that puts them in the HMO described in Section 2.2.1. In order to obtain this speed change the procedure is similar to the one employed earlier with the difference that Δv_{inject} needs to reduce the speed of the spacecraft to move it from the hyperbolic trajectory around Mars to the HMO.

These calculations resulted in $\Delta v_{eject} = 3556\text{m/s}$ and $\Delta v_{inject} = 812.18\text{m/s}$.

Now it is possible to apply Eq. 4.1 with $m_{*eff} = m_{entry}$ and $\Delta v_{imp} = \Delta v_{eject}$ to obtain the mass of propellant $m_{p\ optimum}^{eject\ ISRU}$ needed for the injection into Hohmann transfer from LEO of the spacecraft bearing ISRU technology. The optimal value of thrust $T_{optimum}^{eject\ ISRU}$ that minimizes m_p is estimated numerically and the value of m_{LEO}^{ISRU} can finally be obtained as the sum of m_{entry}, the mass of the propulsion system and the mass of the tanks of propellant necessary for the maneuver:

$$m_{LEO}^{ISRU} = m_{entry} + \left(\frac{1}{g_{0e}(T/W)_{engines}} + \frac{1}{g_{0e}(T/W)_{other}} \right) T_{optimum}^{eject\ ISRU} + m_{p\ optimum}^{eject\ LEO}(1+k) \tag{4.3}$$

Next it is necessary to calculate the mass of a tank full of propellant $m_{full\ tank}$ that is comparable with m_{pl}. This is done by evaluating the mass of propellant consumed from m_{pl} to reach HMO from LMO. In particular the methodology is similar to that employed to obtain $m_{p\ optimum}^{eject\ ISRU}$ if not for some key changes.

Firstly the rocket equation employed in this case, Eq. 4.4, only takes into account burn losses [12].

$$m_p = m_* \left(\exp\left[\frac{\Delta v_{imp}}{g_{0e} I_{sp}} \left(1 + \frac{1}{24} \frac{\mu}{r^3} \frac{g_{0e}^2 I_{sp}^2}{T^2} m_p^2 \right) \right] - 1 \right) \tag{4.4}$$

Secondly m_{*eff} can be written as $m_{*eff} = m_{pl} - m_p$, therefore m_* becomes a function of m_p as well. Finally Δv_{imp} is now the increase in velocity needed to set the spacecraft in HMO with a value of $\Delta v_{LMO\ to\ HMO} = 1350.604\text{m/s}$.

Again the optimization problem for T was solved numerically obtaining the values of $T_{optimum}^{LMO\ to\ HMO}$ and

$m_{p\ optimum}^{LMO\ to\ HMO}$ for the thrust and mass of propellant respectively. The mass of the full tank to send from Earth is then obtained by subtracting $m_{p\ optimum}^{LMO\ to\ HMO}$ from the mass of propellant in m_{pl} and adding the tank structure needed to store it:

$$m_{full\ tank} = \left(\frac{m_{pl}}{k+1} - m_{p\ optimum}^{LMO\ to\ HMO} \right) (k+1) \tag{4.5}$$

This mass is then used as m_{*eff} in Eq. 4.1 along with Δv_{inject} following the same approach that resulted in Eq. 4.3 to obtain the mass $m_{full\ tank}^{Hohmann}$ that needs to be put in Hohmann transfer. And finally the optimization problem was solved one last time using Eq. 4.1 with Δv_{eject} and $m_{*eff} = m_{full\ tank}^{Hohmann}$ to obtain the final mass $m_{LEO}^{full\ tank}$.

4.3 Comparison of the two approaches

The results for both the ISRU spacecraft and the full tank for each case are shown in Table 4.1. It can be seen how the IMLEO when using ISRU becomes much lower than what it is when not using it, especially as the efficiency of the engine, i.e. its I_{sp}, increases.

Case	$m_{LEO}^{full\ tank}$ [kg]	m_{LEO}^{ISRU} [kg]	$m_{LEO}^{ISRU}/m_{LEO}^{full\ tank}$ [%]
I	4881.05	8046.30	60.66
II	420.34	3332.86	12.61
III	208.72	2636.07	7.918
IV	164.66	2465.71	6.678
V	157.46	2493.95	6.314

Table 4.1: Results of the evaluations of IMLEO for every engine considered

On the other hand the overall time that the mission takes changes significantly if the employment of ISRU systems is considered. As a matter of fact ISRU systems need to reach Mars to start producing propellant for the mission, therefore the launch of the crewed spacecraft takes place no earlier than after $T_{syn} = 779.93$days from the launch of the cargo spacecrafts. On the other hand if the propellant travels with Cargo I the first launch window available for the crewed spacecraft is the one that results in a contemporary arrival of crew and cargo around Mars orbit.

Moreover the usage of ISRU systems greatly increases the technological complexity of the mission, as it requires additional landings on Mars and systems for the actual production that are not yet fully proved to be working. This is true in particular for the cases II to V that in order to achieve better performances may require unconventional propellants.

4.4 Discussion of results

Under the assumptions of the current work, the employment of ISRU systems seems to result in mass savings so important that even if the complexity of the mission increases significantly it should be considered seriously when designing the mission. Although the significant increase in the overall time contributes to make the decision to use these technologies not simple.

Chapter 5

Conclusions

5.1 Achievements and future work

This work had the goal of evaluating the possible advantages of employing ISRU technologies for the production of the propellant for the return trip of a rapid crewed mission to Mars. Five propulsion systems were considered and a two stage to orbit gravity turn ascent trajectory was integrated for each of them to know the mass that would need to land on Mars. Then an EDL phase that takes advantage of a HIAD was applied to the case in study. After that two modified rocket equations that take into account burn losses and tank disposal were used to study the orbital maneuvers, other than to obtain a mass of full propellant tank comparable with the ISRU payload. This way the IMLEO for both options was calculated and from their comparison the employment of ISRU appeared more advantageous, although it should be implemented carefully in the mission design as it may increase the duration of the mission and its complexity.

During this study some topics were encountered that could benefit from further analysis. Namely from a preliminary structural analysis of resistance to buckling a cylindrical shell design both with and without stiffeners for the empty tanks appeared to not be able to sustain the EDL stresses. Structures capable of resisting buckling or inflatable tanks should be studied in the future to tackle this problem. Furthermore the ascent trajectory was not optimized, therefore an optimization, possibly with a Monte Carlo method to find the best initial conditions may be implemented to further reduce the IMLEO of the ISRU spacecraft.

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