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Modular ISRU Systems as a Building Block for Sustainable Space Exploration

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Abstract

Leading space agencies and companies identified the utilization of local resources as foundation for the continuous long-term exploration of the solar system within the Global Exploration Roadmap. In-Situ Resource Utilization (ISRU) will be one of the key technologies for sustainable space exploration and habitation missions to Moon, Mars and beyond. Defined production systems based on hydrogen value chains are utilized as essential building blocks to transform the available resources into valuable elements for energy, propulsion, life support or production systems.

The current research at TU Dresden addresses the engineering of ISRU systems using a modular plant concept of standardized, interchangeable and reusable modules. The development, production and automation processes are defined based on the existing standards on modular process plants (VDI 2776) and modular automation (VDI/VDE/NAMUR 2658) for terrestrial application. Therein, a module called Process Equipment Assembly (PEA) realizes unit operations which are interconnected to form different plant configurations that enable flexible production paths. These standardized architectures and concepts are adapted to the requirements of space utilization. Compared to currently developed highly specialized single use systems, a modular plant approach according to these standards allows the flexible application to different mission requirements and environmental conditions. In advance, modular plants minimize the development effort and risk, lead to a lowered market entry threshold for small and medium-sized enterprises from diverse market sectors and encourage commercial competition between manufacturers.

This paper includes initial research work as a foundation for the future development process of modular ISRU process plants. A major input for the holistic development of modular ISRU plants are top-level mission characteristics and derived sub-level requirements. They represent a valuable input for the development and utilization process of modular ISRU plants in space applications. In order to compile these requirements, representative return missions to Moon and Mars shall be compared via a TU Dresden in-house calculation toolbox. To establish this toolbox, the SpaceX manned return mission to Mars is selected as a first reference mission and is used to develop an analytic calculation logic based on Hohmann Transfer, Patched Conic Approximation, aerocapture and aerobraking. The required level of detail as well as necessary input and output parameters are examined. Seven mission variants with different levels of detail are developed and compared regarding their feasibility and the mission parameters velocity, Δv, propellant mass and payload mass.

The investigation results in a feasible analytical calculation logic which is used as foundation for the establishment of the calculation toolbox. A first overview of requirements for the production goods, capacity and time and cycle of production is derived based on the analyzed reference mission. The results are transferred to the development process of the modular ISRU plant and build the basis for further research.

Keywords: In Situ Resource Utilization, Manned Mars Return Mission, SpaceX Starship, Hohmann Transfer, Patched Conic Approximation, Aerocapture

Nomenclature

μ	= gravity constant
a	= semi major axis
e	= eccentricity
g	= gravity
g_0	= natural gravity constant
h	= height
Isp	= specific impulse
m	= mass
n	= number
r	= radius
t	= time of one resolution around rotation axis
T	= time of orbit period
v	= velocity
Δv	= difference in velocity

Indices

∞	= infinity
Ar	= Arrival
Astro	= Astronaut
Atmo	= Atmosphere
Cons	= Consumables
Dep	= Departure
Dry	= dry, without propellant
E	= Earth
Ell	= Ellipse
Empty	= empty, without propellant and payload
Esc	= Escape
FR	= Free Return
g	= Gravity
Hyp	= Hyperbel
IB	= Inbound
Incl	= Inclination
Land	= Landing
Launch	= Launch
M	= Mars
max	= Maximum
min	= Minimum
OB	= Outbound
P	= Perigee
PL	= Payload
Prop	= Propellant
Rest	= Leftover
Rot	= Rotation
S	= Sun
Safety	= Safety, including margin
SBS	= Step-by-Step
S-E	= Sun to Earth
S-M	= Sun to Mars
Wet	= wet, with propellant

Acronyms/Abbreviations

ISRU	= In Situ Resource Utilization
PEA	= Process Equipment Assembly
LEO	= Low Earth Orbit
LMO	= Low Mars Orbit
TMI	= Trans Mars Injection
MOI	= Mars Orbit Insertion
TEI	= Trans Earth Injection
EOI	= Earth Orbit Insertion
SOI	= Sphere of Influence
EVA	= Extra Vehicular Activity
DRM	= Design Reference Mission

1. Introduction

In-Situ Resource Utilization (ISRU) will be one of the key technologies for sustainable space exploration and habitation missions to Moon, Mars and beyond. Defined production systems based on hydrogen value chains are utilized as essential building blocks to transform the available resources into valuable elements for energy, propulsion, life support or production systems.

The current research at TU Dresden addresses the engineering of ISRU systems using a modular plant concept of standardized, interchangeable and reusable modules. The development, production and automation processes are defined based on the existing standards on modular process plants (VDI 2776) and modular automation (VDI/VDE/NAMUR 2658) for terrestrial application. Therein, a module called Process Equipment Assembly (PEA) realizes unit operations which are interconnected to form different plant configurations that enable flexible production paths. These standardized architectures and concepts are adapted to the requirements of space utilization. Compared to currently developed highly specialized single use systems, a modular plant approach according to these standards allows the flexible application to different mission requirements and environmental conditions. In advance, modular plants minimize the development effort and risk, lead to a lowered market entry threshold for small and medium-sized enterprises from diverse market sectors and encourage commercial competition between manufacturers.

For the development of modular ISRU process plants, realistic requirements for production goods, capacity and time and cycle of the production process need to be defined. They can be derived by comparing the velocities, Δv, propellant mass and payload mass of existing mission concepts.

A TU Dresden in-house calculation toolbox is developed to compare the mission parameters and requirements of different representative return missions to Moon and Mars. The toolbox shall be able to compute the velocities, Δv, payload mass and propellant mass required for each mission concept with sufficient accuracy based on a limited amount of input parameters. Therefore, the toolbox uses an analytic calculation logic based on Hohmann Transfer, Patched Conic Approximation, Aerocapture and Aerobraking. The desirable amount of propellant and consumable goods to be provided by an ISRU plant as well as the available production time are derived from the computed mission parameters. The generated set of data will be transferred to the development process of modular ISRU process plants that present an efficient solution for a variety of mission concepts.

Different existing concepts for manned Mars return missions are examined and compared. The SpaceX manned Mars return mission is selected as reference mission for demonstrating the toolbox capabilities to derive preliminary requirements for modular ISRU production plants. Compared to the NASA Design Reference Mission (DRM) 5.0, that uses a variety of vehicles and rendezvous-maneuvers, SpaceX plans to use a direct transfer trajectory with one constant spacecraft configuration. This mission layout is less complex and therefore reduces potential sources for inaccuracies and deviations in the calculation process. It is found to be a suitable reference mission in the initial development process of the calculation toolbox.

Seven calculation variants are developed to optimize and validate the calculation logic and to select the variant with suitable level of detail. In parallel, the feasibility of the proposed SpaceX mission is examined.

Chapter 2 gives a detailed overview of the selected reference mission before Chapter 3 presents the general calculation approach, the calculation variants and their results. The generated results and their impact on the requirements for the ISRU plant development process are discussed in chapter 4. Chapter 5 presents an outlook on the future research work and Chapter 6 is used to conclude the overall work and achievements.

2. Reference Mission

In the process of establishing long term missions to Mars and beyond, SpaceX appears to offer a promising concept of realizing manned Mars missions before 2050 [1] and presents a suitable reference mission with its spacecraft 'Starship'. Only a limited amount of data is publicly available, therefore missing information is generated through profound assumptions.

The selected reference mission is calculated as direct transfer on a long-stay conjunction type trajectory on outbound and inbound [2][3][4]. The following mission layout is taken as reference [1]:

1. Launch with Falcon Heavy to Low Earth Orbit (LEO)
2. In orbit refueling via Starship cargo missions
3. Trans Mars Injection (TMI) to transfer ellipse
4. Cruise phase to mars
5. Mars Orbit Insertion (MOI) to Low Mars Orbit (LMO)
6. Descent and landing on mars
7. Surface mission
8. Refilling via ISRU
9. Launch to LMO
10. Trans Earth Injection (TEI) to transfer ellipse
11. Cruise phase to Earth
12. Earth Orbit Insertion (EOI), descent and landing on Earth

The calculation logic focuses on the independent travel to and return from Mars and excludes the launch from Earth to LEO and the cargo missions as well as station-keeping processes during the refueling process. Further simplifications include the assumptions that Earth and Mars are travelling on circular coplanar orbits in the ecliptic and are located in periapsis and apoapsis of the transfer ellipse for the execution of maneuvers.

It is assumed that the launch from Earth will be executed from Cape Canaveral in eastern direction which delivers a minimum inclination of the LEO parking orbit of 28° [5]. As Earth's rotation axis is tilted by 23.5° [6], a difference of $\Delta\Theta$=4.5° remains relative to the ecliptic. The correlations are depicted in Figure 1 and Figure 2.

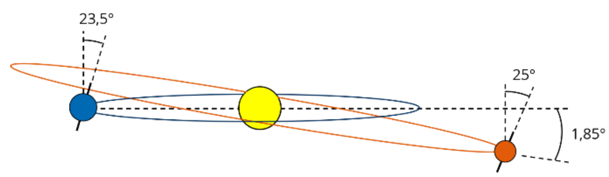

Figure 1. Earth and Mars orbit in reference to ecliptic

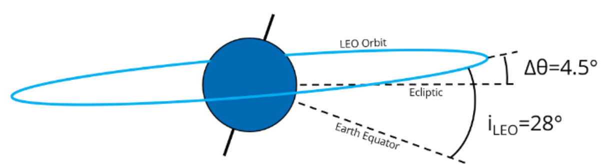

Figure 2. LEO Orbit in Reference to Ecliptic

Resulting from a review of possible landing and launch sites on Mars [7][8][9], a location in the Amazonis Planitia (195°W,25°N) is taken as reference.

Based on a literature review, the orbit heights of LEO (h_{LEO}) and LMO (h_{LMO}) displayed in Table 1 are taken as reference [10][11][12]. As the refilling process in LEO requires a longer stay in the parking orbit, a higher altitude is selected to reduce the station-keeping effort.

Table 1. Selected circular orbit altitudes

Parameter	Unit	Value
h_{LEO}	[m]	$500 \cdot 10^3$
h_{LMO}	[m]	$250 \cdot 10^3$

For Starship, the spacecraft data presented in Table 2 is taken as reference [1].

Table 2. Starship spacecraft data

Parameter	Unit	Value
Number of Astronauts	[-]	12 - 100
Empty mass	[kg]	$85 \cdot 10^3$
Propellant mass	[kg]	$1100 \cdot 10^3$
Payload Outbound	[kg]	$>100 \cdot 10^3$
Payload Inbound	[kg]	$< 50 \cdot 10^3$
Specific Impulse (Vacuum)	[s]	375

A large portion of the available payload mass consists of the consumable mass required for the astronaut's life support system. As this mass is directly dependent on the number of astronauts and the mission duration, a more detailed examination is fulfilled in order to derive the required mass per astronaut per day and establish a detailed mass model of the payload masses [Appendix A]. The results are displayed in Table 3. It is assumed that a closed loop system is implemented which is able to recycle 90% of potable water, 80% of gases and 50% of hygienics (water portion).

Table 3. Mass model of payload masses

Parameter	Unit	Value
Astronaut	[kg]	80
Spacesuit Inside	[kg]	30
Spacesuit EVA	[kg]	120
Crew personal equipment	[kg]	50
Mass /Astronaut*	[kg]	280
Scientific Payload	[kg]	100
Water	[kg/pers/day]	0.25
Food	[kg/pers/day]	2.4
Gases	[kg/pers/day]	0.4
Hygienics	[kg/pers/day]	1.5
Waste	[kg/pers/day]	2
Margin	[%]	30
Mass/Astronaut/Day*	[kg]	8.515

* Sum of the previous values

Depending on the relative motion of Earth and Mars, suitable launch windows for Mars missions occur every 26 months [13]. Referred to the current status of technical and operational mission developments, the 2037 option is selected as reference for the calculations. Based on a literature review, the mission duration displayed in Table 4 is used as reference [Appendix B].

Table 4. Selected mission duration 2037

Parameter	Unit	Value
Outbound	[d]	230
Surface	[d]	500
Inbound	[d]	300
Total	[d]	1030
3-year free return	[d]	1095

The 2037 mission includes the opportunity to enter a free return trajectory. In case of emergencies, the spacecraft can stay on this trajectory and return to Earth without the need for large propulsive maneuvers. This return trip can be realized within a maximum timespan of three years (1095 days) from launch to landing [14]. For early Mars missions, it is assumed that the spacecraft is prepared for the worst-case scenario and is capable to support the astronauts for this duration.

For all calculations, the planetary data of the radius r, the mass m, the gravity constant μ, the gravity g, the distance to the Sun a and the time used for one resolution around the rotation axis t presented in Table 5 is taken as foundation [Appendix C].

Table 5. Planetary Data of Earth, Mars and Sun

	Unit	Earth	Mars	Sun
r	[m]	$6378.14 \cdot 10^3$	$3396.2 \cdot 10^3$	
m	[kg]	$5.974 \cdot 10^{24}$	$0.642 \cdot 10^{24}$	
μ	[m^3/s^2]	$3.986 \cdot 10^{14}$	$4.283 \cdot 10^{13}$	$1.327 \cdot 10^{20}$
g	[m/s^2]	9.81*	3.69	
a	[m]	$1.496 \cdot 10^{11}$	$2.279 \cdot 10^{11}$	
t	[s]	24.6•3600	23.9•3600	

* $g_E = g_0$

From the presented parameters, a nominal spacecraft configuration can be derived. Starship contains an empty mass of m_{empty}=85000 kg, a propellant mass of $m_{prop,Starship}=1100 \cdot 10^3$ kg and a vacuum specific impulse of Isp=375 s. The spacecraft is capable to transport and service 12 astronauts on a 3-year free return trajectory of maximum 1095 days whereas each astronaut is represented by a mass of 280 kg and uses additional 8.515 kg of consumables per day. The nominal mission duration is divided in 300 days outbound flight, 500 days surface stay and 230 days inbound flight. This information can be used to derive the payload masses, maximum accomplishable Δv and required propellant masses for different scenarios as presented in chapter 3.

3. Calculation and Results

Basic transfers between two orbits can be solved analytically using Hohmann Transfer. An alternative is offered by solving Lambert's problem. The solutions are more detailed and complex and are most often generated through a numerical approach. For the developed calculation logic, an analytical two body calculation approach based on Hohmann Transfer is selected as it requires less input parameters and offers full transparency of the calculation process.

The following seven calculation variants examine the given reference mission with increasing level of detail. The results are evaluated based on their feasibility regarding the mission success. Accordingly, the calculation logic is adapted from variant to variant through the implementation of additional strategies used in orbital mechanics.

3.1 Variant 1 – Hohmann Transfer

The first calculation variant is based on a simple Hohmann Transfer. The schematic trajectory is depicted in Figure 3. It is assumed that the spacecraft travels with the planets on a circular orbit around the Sun and completes the interplanetary transfer on a transfer ellipse with a semi major axis a (Equation (1)) [15].

$$a_{Transfer} = \frac{a_{S-M} + a_{S-E}}{2} \quad (1)$$

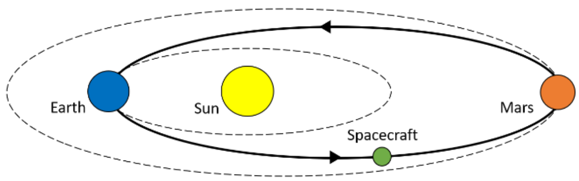

Figure 3. Schematic Hohmann Transfer Trajectory

The velocities of Earth (v_E) and Mars (v_M) on circular orbits around the Sun and of the spacecraft in circular LEO (v_{LEO}) and LMO (v_{LMO}) and the velocities at the time of the maneuvers TMI (v_{TMI}), MOI (v_{MOI}), TEI (v_{TEI}) and EOI (v_{EOI}) in the apogee and perigee of the elliptical transfer orbit are computed in equation (2) to (7) using the Vis Viva equation [15]. The values for the gravitational constant μ of the planets, the distance a between the planets and the orbit altitudes h of LEO and LMO can be found in Table 1 and Table 5.

$$v_E = \sqrt{\frac{\mu_E}{a_{S-E}}} \quad (2)$$

$$v_M = \sqrt{\frac{\mu_M}{a_{S-M}}} \quad (3)$$

$$v_{LEO} = \sqrt{\frac{\mu_E}{r_E + h_{LEO}}} \quad (4)$$

$$v_{LMO} = \sqrt{\frac{\mu_M}{r_M + h_{LMO}}} \quad (5)$$

$$v_{TMI} = v_{EOI} = \sqrt{\frac{2\mu_S}{a_{S-E}} - \frac{\mu_S}{a}} \quad (6)$$

$$v_{MOI} = v_{TEI} = \sqrt{\frac{2\mu_S}{a_{S-M}} - \frac{\mu_S}{a}} \quad (7)$$

The Δv of the different maneuvers TMI (Δv_{TMI}), MOI (Δv_{MOI}), TEI (Δv_{TEI}) and EOI (Δv_{EOI}) is calculated according to equation (8) to (11) as difference between the initial and the target velocity [15]. It is assumed that all maneuvers are executed tangentially in reference to their orbit position.

$$\Delta v_{TMI} = v_{TMI} - v_E \quad (8)$$

$$\Delta v_{MOI} = v_M - v_{MOI} \quad (9)$$

$$\Delta v_{TEI} = v_{TEI} - v_M \quad (10)$$

$$\Delta v_{EOI} = v_E - v_{EOI} \quad (11)$$

For the launch from a planet, atmospheric, gravitational and rotational effects have to be considered. In addition to the Δv required to reach the velocity of the target orbit (Δv_{LEO}, Δv_{LMOI}), respective Δv to overcome the existing gravitational forces of Earth ($\Delta v_{g,E}$) and Mars ($\Delta v_{g,M}$) has to be included according to equations (13) and (17). To simplify the fact that the density of the atmosphere changes with the height above the surface, the atmospheric influence at Earth and Mars is represented by constant values $\Delta v_{Atmo,E}$ and $\Delta v_{Atmo,M}$ depicted in equations (12) and (16) [16][17][18]. The values respect the fact that the Martian atmosphere has a density of approximately 1% of the density of Earth's atmosphere [19]. For a launch from Earth or Mars in eastern direction, the rotation of the planet leaves the spacecraft with an initial velocity in the right direction, that can be subtracted from the required Δv [16][15]. These values $\Delta v_{rot,E}$ and $\Delta v_{rot,M}$ are computed in equations (14) and (18). The simplified assumption that launch and landing maneuvers require the same amount of Δv is included in equations (15) and (19) [16][20].

$$\Delta v_{Atmo,E} = 150 \frac{m}{s} \quad (12)$$

$$\Delta v_{g,E} = \sqrt{2 * g_E * h_{LEO}} \quad (13)$$

$$\Delta v_{rot,E=} \frac{2 * \Pi * r_E}{t_E} \quad (14)$$

$$\begin{aligned} \Delta v_{Land,E} &= \Delta v_{Launch,E} \\ &= \Delta v_{LEO} + \Delta v_{Atmo,E} + \Delta v_{g,E} - \Delta v_{rot,E} \end{aligned} \quad (15)$$

$$\Delta v_{Atmo,M} = 0.01 * v_{Atmo,E} \quad (16)$$

$$\Delta v_{g,M} = \sqrt{2 * g_M * h_{LMO}} \quad (17)$$

$$\Delta v_{rot,M=} \frac{2 * \Pi * r_M}{t_M} \quad (18)$$

$$\begin{aligned} \Delta v_{Land,M} &= \Delta v_{Launch,M} \\ &= \Delta v_{LMO} + \Delta v_{Atmo,M} + \Delta v_{g,M} - \Delta v_{rot,M} \end{aligned} \quad (19)$$

The total Δv required for the outbound (Δv_{OB}) and inbound trip (Δv_{IB}) consists of the sum of the Δv required for the maneuvers as shown in equations (20) and (21).

$$\Delta v_{OB} = \Delta v_{TMI} + \Delta v_{MOI} + \Delta v_{Land,M} \quad (20)$$

$$\begin{aligned} \Delta v_{IB} = \Delta v_{Launch,M} + \Delta v_{TEI} + \Delta v_{EOI} \\ + \Delta v_{Land,E} \end{aligned} \quad (21)$$

While the values of velocity and Δv depend on the mission layout, the required propellant mass is furthermore influenced by the spacecraft's characteristics including dry mass (m_{dry}), propellant mass (m_{prop}) and specific impulse (Isp). The minimum required propellant mass $m_{prop,min}$ for the calculated Δv on outbound and inbound is calculated using equations (22) to (25). In addition, the maximum achievable Δv on outbound and inbound Δv_{max} for the total available propellant mass $m_{prop,Starship}=1100 \cdot 10^6$ kg is calculated using equation (26). Both approaches take the Tsiolkovsky equation as foundation [15]. Finally, the left-over amount of Δv and propellant mass (Δv_{Rest}, $m_{prop,Rest}$) are calculated using equation (27) and (28). The number of astronauts n_{Astro}, number of Days n_{Days}, mass per astronaut m_{Astro}, mass of consumables m_{Cons}, mass of Science Payload $m_{Science}$ and specific Impulse Isp can be found in Table 2, Table 3 and Table 4.

$$m_{PL} = n_{Astro} * m_{Astro} + n_{Astro} * n_{Days} * m_{Cons} + m_{Science} \tag{22}$$

$$m_{dry} = m_{empty} + m_{PL} \tag{23}$$

$$m_{wet} = m_{dry} * e^{\frac{\Delta v_{Total}}{Isp * g_0}} \tag{24}$$

$$m_{prop,min} = m_{wet} - m_{dry} \tag{25}$$

$$\Delta v_{max} = (Isp * g_0) * \log(\frac{m_{wet}}{m_{dry}}) \tag{26}$$

$$\Delta v_{Rest} = \Delta v_{max} - \Delta v \tag{27}$$

$$m_{prop,Rest} = m_{prop,Starship} - m_{prop,min} \tag{28}$$

The calculation results for variant one are summarized in Table 6.

Table 6. Results Calculation Variant 1

Outbound			Inbound		
v_E	[m/s]	29783.3	v_M	[m/s]	24078.4
v_{LEO}	[m/s]	7612.6	v_{LMO}	[m/s]	3427.3
v_{TMI}	[m/s]	32754.7	v_{TEI}	[m/s]	21408.3
v_{MOI}	[m/s]	21408.3	v_{EOI}	[m/s]	32754.7
Δv_{TMI}	[m/s]	2971.4	$\Delta v_{Launch,M}$	[m/s]	4546.2
Δv_{MOI}	[m/s]	2670.1	Δv_{TEI}	[m/s]	-2670.1
$\Delta v_{Land,M}$	[m/s]	4546.2	Δv_{EOI}	[m/s]	-2971.4
			$\Delta v_{Land,E}$	[m/s]	10430
Δv_{OB}	[m/s]	10188	Δv_{IB}	[m/s]	20617
$\Delta v_{OB,max}$	[m/s]	6880.5	$\Delta v_{IB,max}$	[m/s]	8556
$\Delta v_{OB,Rest}$	[m/s]	-3307.1	$\Delta v_{IB,Rest}$	[m/s]	-12061
$m_{PL,OB}$	[kg]	115347	$m_{PL,IB}$	[kg]	34114
$m_{prop,OB,min}$	[kg]	$2.995 \cdot 10^6$	$m_{prop,IB,min}$	[kg]	$32.235 \cdot 10^6$
$m_{prop,OB,Rest}$	[kg]	$-1.895 \cdot 10^6$	$m_{prop,IB,Rest}$	[kg]	$-31.135 \cdot 10^6$

From the results, it is visible that the maximum achievable Δv on outbound and inbound ($\Delta v_{OB,max}$, $\Delta v_{IB,max}$) is lower than the required amount of Δv (Δv_{OB} , Δv_{IB}) for the trips. Respectively, the available amount of propellant in the tanks m_{prop} is lower than the minimum required amount of propellant $m_{prop,OB,min}$ and $m_{prop,IB,min}$. Both leftover values Δv_{Rest} and $m_{prop,Rest}$ are of negative value. The available propellant capacity is not sufficient to achieve the computed amount of Δv. Starship in its referenced configuration is by calculation not capable to accomplish the interplanetary mission to mars according to the simplified model based on Hohmann Transfer described above.

3.2 Variant 2 – Patched Conic Approximation

Following the results of the first calculation variant, a more accurate model has to be established. Therefore, the impact of the planets and the influence of LEO and LMO are included in the calculation logic by applying the principle of Patched Conic Approximation as depicted in Figure 4. In this approach, the four-body problem between Sun, Earth, Mars and the spacecraft is divided resulting in three connected two-body-problems that can each be solved analytically [21].

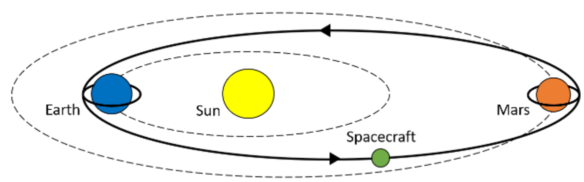

Figure 4. Schematic Patched Conic Approximation Trajectory

The first two-body problem consists of the transfer of the spacecraft from the edge of Earth's Sphere of Influence (SOI) to the edge of Mars' SOI relative to the Sun as depicted in Figure 5. In this approach the SOI of Earth and Mars can be considered to be punctiform. The relevant velocities v_{TMI}, v_{MOI}, v_{TEI} and v_{EOI} are calculated analogue to the simple Hohmann Transfer in equation (6) and (7) and are taken as foundation to solve the remaining two problems.

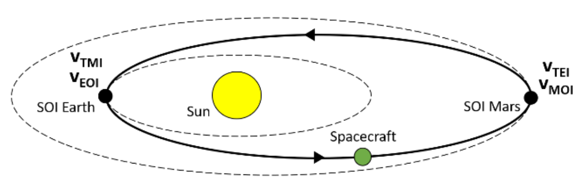

Figure 5. Schematic Two-Body Problem 1

The second two-body problem includes the spacecraft travelling in LEO around Earth in its SOI as depicted in Figure 6. The object of the TMI is to accelerate the spacecraft in the perigee of a hyperbolic trajectory to leave Earth's SOI at $r \rightarrow \infty$ with the targeted transfer velocity v_{TMI} relative to Sun. The departure speed $v_{\infty,Dep,OB}$ at $r \rightarrow \infty$ relative to Earth takes into account the velocity of Earth v_E (Equation (2)) on its orbit around the sun and is depicted in equation (29).

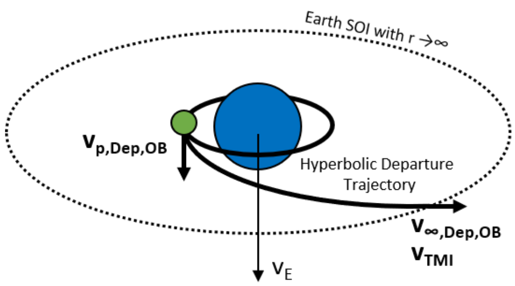

Figure 6. Schematic Two-Body Problem 2

Assuming that the spacecraft is accelerated to an extend where it reaches the Earth escape velocity $v_{esc,E}$ (Equation (30)) in the perigee, its velocity at $r \rightarrow \infty$ would be zero. In order to reach $v_{\infty,Dep,OB}$ at $r \rightarrow \infty$, the velocity in perigee must be by this amount higher than the escape velocity. The velocity $v_{p,Dep,OB}$ in equation (31) results as target velocity in perigee. As the spacecraft is already travelling around Earth with v_{LEO} (Equation (4)), the Δv required to be implied through TMI Δv_{TMI} results from Equation (32).

$$v_{\infty,Dep,OB} = v_{TMI} - v_{Earth} \tag{29}$$

$$v_{esc,E} = \sqrt{2} * v_{LEO} \tag{30}$$

$$v_{p,Dep,OB} = \sqrt{v_{\infty,Dep,OB}{}^2 + v_{esc,E}{}^2} \tag{31}$$

$$\Delta v_{TMI} = v_{p,Dep,OB} - v_{LEO} \tag{32}$$

The third two body problem includes the spacecraft entering LMO around Mars in its SOI as depicted in Figure 7. According to the assumption that the Martian SOI is punctiform in reference to the sun, the location of the spacecraft on its arrival trajectory relative to Mars can be freely selected for the purpose of calculation. As the spacecraft arrives at Mars with a velocity v_{MOI} lower than the planet's velocity v_M (Equation (3)) it will enter the SOI on a hyperbolic trajectory starting on the front side with reference to the direction of the Martian motion.

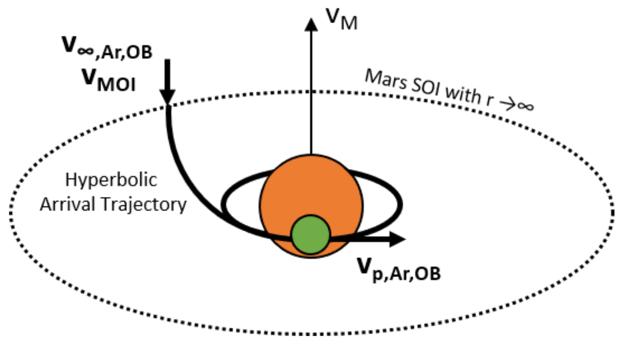

Figure 7. Schematic Two-Body Problem 3

The arrival velocity $v_{\infty,Ar,OB}$ relative to Mars and the escape velocity $v_{esc,M}$ are depicted in equation (33) and (34). The respective velocity in the perigee of the hyperbolic trajectory $v_{p,Ar,OB,Hyp}$ is derived analogue to equation (31) in equation (35). If the spacecraft remains at the same velocity, it will fulfil a swing-by maneuver and leave the Martian SOI. The object of MOI is to decelerate the spacecraft at the perigee of the hyperbolic trajectory to a velocity that allows it to enter the circular LMO. To accomplish this, the current velocity $v_{p,Ar,OB,Hyp}$ in the perigee of the hyperbolic trajectory has to be reduced to equal the velocity $v_{p,AR,OB,Ellipse}$ (Equation (36)) of the targeted circular LMO with an eccentricity of $e_{M,Ell} = 0$. The Δv_{MOI} which has to be implied to realize the reduction of speed is displayed in equation (37).

$$v_{\infty,Ar,OB} = v_{Mars} - v_{MOI} \tag{33}$$

$$v_{esc,M} = \sqrt{2} * v_{LMO} \tag{34}$$

$$v_{p,Ar,OB,Hyp} = \sqrt{v_{\infty,Ar,OB}{}^2 + v_{esc,M}{}^2} \tag{35}$$

$$v_{p,Ar,OB.Ell} = \mu_M * \frac{(1 + e_{M,Ell})}{r_M + h_{LMO}} \tag{36}$$

$$\Delta v_{MOI} = v_{p,Ar,OB,Hyp} - v_{p,Ar,OB.Ell} \tag{37}$$

The velocities and Δv for the departure and arrival on the inbound trip are calculated based on the same logic. The process is depicted in detail in equation (38) to (44):

$$v_{\infty,Dep,IB} = v_{TEI} - v_{Mars} \tag{38}$$

$$v_{p,Dep,IB} = \sqrt{v_{\infty,Dep,IB}{}^2 + v_{esc,M}{}^2} \tag{39}$$

$$\Delta v_{TEI} = v_{p,Dep,IB} - v_{LMO} \tag{40}$$

$$v_{\infty,Ar,IB} = v_{Earth} - v_{EOI} \tag{41}$$

$$v_{p,Ar,IB,Hyp} = \sqrt{v_{\infty,Ar,IB}{}^2 + v_{esc,E}{}^2} \tag{42}$$

$$v_{p,Ar,IB.Ell} = \mu_M * \frac{(1 + e_{E,Ell})}{r_E + h_{LEO}} \quad (43)$$

$$\Delta v_{EOI} = v_{p,Ar,IB,Hyp} - v_{p,Ar,IB.Ell} \quad (44)$$

The results of calculation variant 2, following the Patched Conic Approximation, are summarized in Table 7.

Table 7. Results Calculation Variant 2

Outbound			Inbound		
$v_{\infty,Dep,OB}$	[m/s]	2971.37	$v_{\infty,Dep,IB}$	[m/s]	-2670.09
$v_{p,Dep,OB}$	[m/s]	11168.37	$v_{p,Dep,IB}$	[m/s]	5685.25
$v_{\infty,Ar,OB}$	[m/s]	2670.09	$v_{\infty,Ar,IB}$	[m/s]	-2971.37
$v_{p,Ar,OB,hyp}$	[m/s]	5533.74	$v_{p,Ar,IB,Hyp}$	[m/s]	11168.37
$v_{p,Ar,OB,El}$	[m/s]	3427.31	$v_{pAr,IB,El}$	[m/s]	7612.60
$e_{M,Ell}$	[-]	0	$e_{E,Ell}$	[-]	0
Δv_{TMI}	[m/s]	3555.8	$\Delta v_{Launch,M}$	[m/s]	4546.2
Δv_{MOI}	[m/s]	2106.4	Δv_{TEI}	[m/s]	2257.9
$\Delta v_{Land,M}$	[m/s]	4546.2	Δv_{EOI}	[m/s]	3555.8
			$\Delta v_{Land,E}$	[m/s]	10430
Δv_{OB}	[m/s]	10208	Δv_{IB}	[m/s]	20789
$\Delta v_{OB,max}$	[m/s]	6880.5	$\Delta v_{IB,max}$	[m/s]	8556
$\Delta v_{OB,Rest}$	[m/s]	-3327.9	$\Delta v_{IB,Rest}$	[m/s]	-12233
$m_{PL,OB}$	[kg]	115347	$m_{PL,IB}$	[kg]	34114
$m_{prop,OB,min}$	[kg]	$3.013 \cdot 10^6$	$m_{prop,IB,min}$	[kg]	$33.786 \cdot 10^6$
$m_{prop,OB,Rest}$	[kg]	$-1.913 \cdot 10^6$	$m_{prop,IB,Rest}$	[kg]	$-32.686 \cdot 10^6$

From the results, it becomes visible that the implementation of the Patched Conic Approximation increases the required Δv (Δv_{OB}, Δv_{IB}) as well as the required propellant mass ($m_{prop,OB,min}$, $m_{prop,IB,min}$) on outbound and inbound. Both leftover values Δv_{rest} and $m_{prop,Rest}$ are of negative value and lead to the conclusion that Starship in its referenced configuration is by calculation not capable to accomplish the presented reference mission to Mars.

3.3 Variant 3 – Inclination and Safety Margin

The previous calculation variants are highly theoretical calculations. To adapt the calculation towards a more realistic approach, variant three considers an inclination maneuver on the outbound leg. A $\Delta v_{Incl,OB}$ is applied via equation (45) to overcome an inclination of ΔΘ=4.5° (Chapter 2) and place the LEO parking orbit in line with the ecliptic prior to TMI. The latitude of the selected Martian landing site (Chapter 2) and the absence of infrastructural limitations for the direction of launch from Mars result in a LMO with a minimum inclination of 25°. Combined with the Martian rotation axis which is tilted around 25° [22], it is possible to place the spacecraft in a LMO in line with the ecliptic without an additional inclination maneuver on the inbound as depicted in Figure 8.

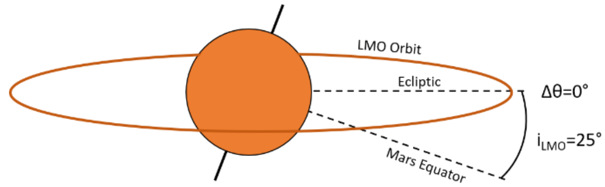

Figure 8. LMO Orbit in Reference to Ecliptic

In addition, a safety factor $S = 1.1$ representing a safety margin of 10% is added to the total calculated Δv for outbound and inbound in equation (46) to cover course correction, additional unplanned maneuvers and further inaccuracies. The feasibility of the height of the safety factor will be analyzed and validated within future research work.

$$\Delta v_{Incl,OB} = 2 * v_{LEO} * \sin\left(\frac{\Delta\theta}{2}\right) \quad (45)$$

$$\Delta v_{Safety} = S * \Delta v \quad (46)$$

The changed values for the mission Δv and the mass model are summarized in Table 8.

Table 8. Results Calculation Variant 3

Outbound			Inbound		
$\Delta v_{Incl,OB}$	[m/s]	1194.6	$\Delta v_{Launch,M}$	[m/s]	4546.2
Δv_{TMI}	[m/s]	3555.8	Δv_{TEI}	[m/s]	2257.9
Δv_{MOI}	[m/s]	2106.4	Δv_{EOI}	[m/s]	3555.8
$\Delta v_{Land,M}$	[m/s]	4546.2	$\Delta v_{Land,E}$	[m/s]	10430
Δv_{OB}	[m/s]	11403	Δv_{IB}	[m/s]	20789
$\Delta v_{OB, Safety}$	[m/s]	12543	$\Delta v_{IB,Safety}$	[m/s]	22868
$\Delta v_{OB,max}$	[m/s]	6880.5	$\Delta v_{IB,max}$	[m/s]	8556
$\Delta v_{OB,rest}$	[m/s]	-5662.7	$\Delta v_{IB,rest}$	[m/s]	-14313
$m_{PL,OB}$	[kg]	115347	$m_{PL,IB}$	[kg]	34114
$m_{prop,OB,min}$	[kg]	$5.861 \cdot 10^6$	$m_{prop,IB,min}$	[kg]	$59.542 \cdot 10^6$
$m_{prop,OB,Rest}$	[kg]	$-4.761 \cdot 10^6$	$m_{prop,IB,Rest}$	[kg]	$-58.442 \cdot 10^6$

The results of calculation variant 3, which include the consideration of both the change of inclination and a safety margin, display the highest number of required Δv ($\Delta v_{OB, Safety}$, $\Delta v_{IB, Safety}$) and propellant mass ($m_{prop,OB,min}$, $m_{prop,IB,min}$). Equivalent to variant one and two, this required number exceeds the capacity of the Starship spacecraft ($\Delta v_{OB,max}$, $\Delta V_{IB,max}$, m_{prop}) and leaves negative values for the left over values (Δv_{rest}, $m_{prop,Rest}$).

Examining a manned Mars return mission only using Hohmann Transfer and Patched Conic Approximation delivers results with large deviations from realistic values. It is assumed that Starship's current configuration was developed by SpaceX based on its targeted manned mars return mission and that its feasibility was theoretically proven by calculation. Further optimization strategies have to be considered to develop a feasible and comparable analytical calculation logic.

3.4 Variant 4 – Aerocapture and Aerobraking

To further reduce the required Δv of the reference mission, the concept of aerocapture and aerobraking can be applied. If a spacecraft makes use of an aerocapture maneuver, it is placed from the hyperbolic arrival trajectory of the Patched Conic Approximation on an elliptical orbit around the target planet with a perigee at the height of the circular target orbit. As the elliptical orbit has a higher velocity in the perigee (Equation (39), (43)) the required Δv for MOI and EOI can be reduced compared to directly entering LMO or LEO. The velocity of the elliptical orbit depends on the eccentricity e of the ellipse. As the spacecraft passes through the planet's atmosphere in every revolution, the apogee of the ellipse is continuously reduced until the circular target orbit is reached as depicted in Figure 9. While this maneuver increases the mission duration and respectively the required consumable mass, it can significantly reduce the Δv for the mission.

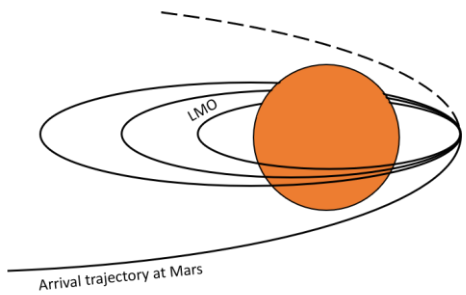

Figure 9. Schematic Aerocapture Trajectory

Based on equations (47) and (48) for the semi major axis of the ellipse [21], equation (49) can be derived, offering to compute the eccentricity e of the ellipse from a defined orbit period T.

$$a = \frac{r_p}{1-e} \tag{47}$$

$$a = \left(\frac{T * \sqrt{\mu}}{2\pi}\right)^{\frac{2}{3}} \tag{48}$$

$$e = 1 - \frac{r_p}{\left(\frac{T * \sqrt{\mu}}{2\pi}\right)^{\frac{2}{3}}} \tag{49}$$

For this calculation logic an orbit period of 48 hours at Earth and Mars is taken as reference neglecting the detailed effects it has on the mission duration. It is assumed that small additional maneuvers in the elliptical orbit covered by the included safety margin can be used to keep the mission duration within an effective scope. A detailed validation of these assumptions has to be included in future research work.

The effect of the atmosphere present at Mars and Earth can also be used for descent and landing. While this approach is currently under development and is not backed by a large number of publicly available information, reviewed literature claims that most of the Δv required for landing can be compensated by aerobraking [20][23][24]. SpaceX plans to use the Belly Flop Maneuver [24][25] to maximize the surface of attack used for aerobraking. It is assumed, that even though a heavy ablative heat shield is used to handle the thermal heat resulting from friction effects in the atmosphere, this approach leaves a required Δv of only 250 m/s to successfully land on Mars [25]. Based on this assumption and the fact that additional detailed sources on the topic of aerobraking are rare to date, a Δv of 250 m/s is used as reference value for landing at Mars and Earth. Adapting and validating these values will be part of future research work.

The results of calculation variant four are summarized in Table 9.

Table 9. Results Calculation Variant 4

Outbound			Inbound		
$v_{\infty,Dep,OB}$	[m/s]	2971.37	$v_{\infty,Dep,IB}$	[m/s]	-2670.09
$v_{p,Dep,OB}$	[m/s]	11168.37	$v_{p,Dep,IB}$	[m/s]	5685.25
$v_{\infty,Ar,OB}$	[m/s]	2670.09	$v_{\infty,Ar,IB}$	[m/s]	-2971.37
$v_{p,Ar,OB,Hyp}$	[m/s]	5533.74	$v_{p,Ar,IB,Hyp}$	[m/s]	11168.37
$v_{p,Ar,OB,El}$	[m/s]	4706.31	$v_{pAr,IB,El}$	[m/s]	1048.61
$T_{M,Ell}$	[s]	48•3600	$T_{E,Ell}$	[s]	48•3600
$e_{M,Ell}$	[-]	0.8856	$e_{E,Ell}$	[-]	0.8974
$\Delta v_{Incl,OB}$	[m/s]	1194.6	$\Delta v_{Launch,M}$	[m/s]	4546.2
Δv_{TMI}	[m/s]	3555.8	Δv_{TEI}	[m/s]	2106.4
Δv_{MOI}	[m/s]	827.43	Δv_{EOI}	[m/s]	682.24
$\Delta v_{Land,M}$	[m/s]	250	$\Delta v_{Land,E}$	[m/s]	250
Δv_{OB}	[m/s]	5827.8	Δv_{IB}	[m/s]	7584.8
$\Delta v_{OB, Safety}$	[m/s]	6410.5	$\Delta v_{IB,Safety}$	[m/s]	8343.3
$\Delta v_{OB,max}$	[m/s]	6880.5	$\Delta v_{IB,max}$	[m/s]	8556
$\Delta v_{OB,Rest}$	[m/s]	469.97	$\Delta v_{IB,Rest}$	[m/s]	212.71
$m_{PL,OB}$	[kg]	115347	$m_{PL,IB}$	[kg]	34114
$m_{prop,OB,min}$	[kg]	0.944•10^6	$m_{prop,IB,min}$	[kg]	1.032•10^6
$m_{prop,OB,Rest}$	[kg]	0.156•10^6	$m_{prop,IB,Rest}$	[kg]	0.068•10^6

From the results of calculation variant 4, it can be seen, that the implementation of the aerocapture and aerobraking approach leads to a more realistic mission layout. The required Δv ($\Delta v_{OB,\ Safety}$, $\Delta v_{IB,\ Safety}$) and minimum propellant mass ($m_{prop,OB,min}$, $m_{prop,IB,min}$) is reduced to a value lower than the performance limits of Starship ($\Delta v_{OB,max}$, $\Delta V_{IB,max}$, m_{prop}). The optimized calculation variant four leaves positive values for the leftover values (Δv_{rest}, $m_{prop,Rest}$). If aerocapture and aerobraking maneuvers are included to optimize the required Δv for a mission, the capabilities of Starship in its current configuration are sufficient to accomplish the outbound and inbound trip of the presented reference mission.

The previous calculation delivers the minimum required propellant mass referring to a constant spacecraft mass and total Δv. During a real mission, the spacecraft mass is not constant but is reduced by the propellant mass used for the executed maneuvers. Applying a step-by-step calculation offers increasingly realistic results. This approach uses the Tsiolkovsky equation to compute the propellant mass m_{prop} required for each maneuver. Therefore, it considers the spacecraft mass prior to the maneuver $m_{initial}$ and the Δv which was previously determined for the maneuver multiplied by the implemented safety factor $S = 1.1$ (Equation (50), (51)). The following maneuver is computed according to the same approach shown in equations (52) and (53). The new initial mass is determined by reducing the initial spacecraft mass by the propellant mass used during the previous maneuvers. The calculations are executed assuming that Starship is launched with full tanks.

$$m_{initial,1} = m_{empty} + m_{prop} + m_{PL} \tag{50}$$

$$m_{prop,1} = m_{initial,1} * \left(1 - e^{-\frac{S*\Delta v_1}{Isp*g_0}}\right) \tag{51}$$

$$m_{initial,2} = m_{initial,1} - m_{prop,1} \tag{52}$$

$$m_{prop,2} = m_{initial,2} * \left(1 - e^{-\frac{S*\Delta v_2}{Isp*g_0}}\right) \tag{53}$$

The results of the step-by-step calculation are presented in Table 10.

Table 10. Results Step-By-Step Calculation Variant 4

Outbound			Inbound		
$m_{prop,Incl,OB}$	[kg]	$0.391 \cdot 10^6$	$m_{prop,Launch,M}$	[kg]	$0.906 \cdot 10^6$
$m_{prop,TMI}$	[kg]	$0.596 \cdot 10^6$	$m_{prop,TEI}$	[kg]	0.146•106
$m_{prop,MOI}$	[kg]	$0.069 \cdot 10^6$	$m_{prop,EOI}$	[kg]	$0.031 \cdot 10^6$
$m_{prop,Land,M}$	[kg]	$0.018 \cdot 10^6$	$m_{prop,Land,E}$	[kg]	$0.010 \cdot 10^6$
$m_{prop,OB}$	[kg]	$1.073 \cdot 10^6$	$m_{prop,IB}$	[kg]	$1.093 \cdot 10^6$
$m_{prop,OB,Rest}$	[kg]	$0.027 \cdot 10^6$	$m_{prop,IB,Rest}$	[kg]	$0.007 \cdot 10^6$

The results show that the applied step-by-step calculation approach results in higher propellant masses for outbound ($m_{prop,OB}$) and inbound ($m_{prop,IB}$) than the previously calculated minimum propellant masses $m_{prop,OB,min}$ and $m_{prop,IB,min}$ displayed in Table 9. The numbers remain closely below Starships maximum available propellant mass of $m_{prop,Starship} = 1100 * 10^6 kg$. Therefore, it is concluded that Starship is by calculation still able to accomplish the outbound and inbound trip to of the presented return mission to Mars.

3.5 Variant 5 – Free Return

Based on the established calculation logic, calculation variant five analyzes if Starship in its referenced configuration is by calculation able to fulfil a three-year free return trajectory in case of emergency. It is assumed that an emergency situation occurs after TMI during the cruise phase from Earth to Mars, which leaves the spacecraft with a functioning propulsion system but imposes the urge to return to Earth. The spacecraft stays on the elliptical transfer orbit and travels back to Earth, as depicted in Figure 10, where it executes EOI, descent and landing.

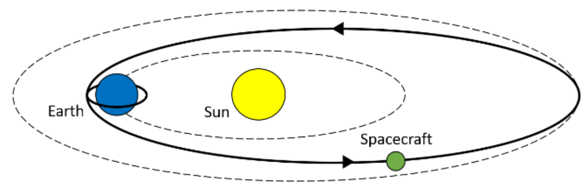

Figure 10. Schematic Free Return Trajectory

The calculation is executed analogue to the previously introduced calculation strategy. Based on the adapted mission layout, the considered maneuvers are reduced to the inclination maneuver in LEO, TMI, EOI and landing on Earth. The results of calculation variant 5, considering a free return trajectory, are summarized in Table 11.

Table 11. Results Calculation Variant 5

	Free Return	
$\Delta v_{Incl,OB}$	[m/s]	1194.6
Δv_{TMI}	[m/s]	3555.8
Δv_{EOI}	[m/s]	682.24
$\Delta v_{Land,E}$	[m/s]	250
Δv_{FR}	[m/s]	5682.6
$\Delta v_{FR,Safety}$	[m/s]	6250.8
$\Delta v_{FR,max}$	[m/s]	6880.5
$\Delta v_{FR,Rest}$	[m/s]	629.68
$m_{PL,FR}$	[kg]	115347
$m_{prop,FR,min}$	[kg]	$0.8954 \cdot 10^6$
$m_{prop,FR,Rest}$	[kg]	$0.2046 \cdot 10^6$
$m_{prop,FR,SBS}$	[kg]	$1.063 \cdot 10^6$
$m_{prop,FR,SBS,Rest}$	[kg]	$0.037 \cdot 10^6$

The results show that the Δv and propellant mass required to accomplish a free return trajectory ($\Delta v_{FR,Safety}$, $m_{prop,FR,min}$, $m_{prop,FR,SBS}$) are lower than the required Δv ($\Delta v_{FR,max}$) and the available propellant mass (m_{prop}) for a nominal outbound and inbound trip. The left over values ($\Delta v_{FR,Rest}$, $m_{prop,FR,Rest}$, $m_{prop,FR,SBS,Rest}$) remain positive. Starship in its referenced configuration is by calculation able to accomplish this emergency scenario. While the realization of free return trajectories with different spacecraft conditions is a highly complex topic, a more detailed examination is excluded from the current research work.

3.6 Variant 6 – Reduced Consumable Mass

While early Mars missions require a high level of safety and redundancy standards, long term Mars missions offer the opportunity to reduce these precautions. An increased level of experience and expertise in technology as well as operations supports the implementation of different strategies. One option is to use consumables produced on or delivered to the Martian surface to operate the life support systems during the surface mission and on the inbound trip. This approach reduces the consumable mass carried on the outbound trip $m_{PL,OB}$ from the worst-case assumption of 1095 days to the nominal trip duration for the outbound leg of 230 days.

Table 12. Results Calculation Variant 6

Parameter	Unit	Value
Δv_{OB}	[m/s]	5827.8
$\Delta v_{OB,\ Safety}$	[m/s]	6410.5
$\Delta v_{OB,max}$	[m/s]	8762.2
$\Delta v_{OB,rest}$	[m/s]	2351.7
$m_{PL,OB}$	[kg]	26961.4
$m_{prop,OB,min}$	[kg]	$0.528 \cdot 10^6$
$m_{prop,OB,Rest}$	[kg]	$0.572 \cdot 10^6$
$m_{prop,OB,SBS}$	[kg]	$1.000 \cdot 10^6$
$m_{prop,OB,SBS,Rest}$	[kg]	$0.100 \cdot 10^6$

As shown by the results in Table 12, this approach significantly reduces the required payload mass on the outbound $m_{PL,OB}$ by 76.6 % from 115347 kg (Table 9) to 26961.4 kg. This reduced payload mass results in a lower amount of required fuel on the outbound ($m_{prop,OB,min}$, $m_{prop,OB,SBS}$). The lower payload mass can be used to accomplish a higher $\Delta v_{OB,max}$ of 8762.2 m/s compared to 6880.5 m/s in variant four (Table 9) or allows to use the saved mass to include additional types of payloads on the trip to mars. As the values for the inbound trip are not adapted the results remain unchanged.

3.7 Variant 7 – 100 Astronauts

For the long-term human colonization of Mars SpaceX states that Starship will be capable to transport 100 humans on an interplanetary Martian mission [26][27]. In calculation variant 7, it is examined if Starship, in its current configuration, is able to meet this requirement.

It is assumed that the gained expertise through earlier Mars missions and increased safety standards allow to reduce the necessary mass per person. The assumptions include that not every person aboard Starship carries a heavy suit for Extra Vehicular Activities (EVA) and that required consumable masses are only included for the time of outbound or inbound as infrastructure to provide further life support is available at the Martian surface. These assumptions reduce the mass per person on outbound and inbound to 130 kg and the number of days on the outbound to 230.

In Table 13, the results of calculation variant seven are summarized.

Table 13. Results Calculation Variant 7

Outbound			Inbound		
$m_{PL,OB}$	[kg]	208945	$m_{PL,IB}$	[kg]	268550
$m_{prop,OB,min}$	[kg]	$1.385 \cdot 10^6$	$m_{prop,IB,min}$	[kg]	$3.062 \cdot 10^6$
$m_{prop,OB,Rest}$	[kg]	$-0.285 \cdot 10^6$	$m_{prop,IB,Rest}$	[kg]	$-1.962 \cdot 10^6$

It is visible that the increased number of astronauts leads to an increased payload mass m_{PL} compared to the previous calculation variants with 12 astronauts (Table 9), despite the fact that the mass of each person and the mass per person per day was reduced. The minimum required propellant mass $m_{prop,min}$ is higher than the maximum available propellant mass m_{prop} of the installed propellant tanks. This concludes that Starship in its current configuration and under the presented assumptions is not able to transport 100 persons to and from Mars, according to the presented calculation approach.

To create a value of reference for further examinations, the maximum available mass per person per day of the given scenario is derived from Starships maximum payload mass. The maximum payload mass $m_{PL,max}$ for the given mission Δv and the propellant mass m_{prop} for full tanks is computed using a converted variant of the Tsiolkovsky equation (Equation (54)):

$$m_{PL,max} = \frac{m_{empty} + m_{prop} - m_{empty} * e^{\frac{\Delta v}{Isp*g_0}}}{e^{\frac{\Delta v}{Isp*g_0}} - 1} \quad (54)$$

The maximum available mass per person per day is derived using equation (55).

$$m_{Cons,max} = \frac{m_{PL,max} - (n_{Astro} * m_{Astro})}{n_{Astro} * n_{Days}} \quad (55)$$

The results displayed in Table 14 show, that the maximum available mass $m_{cons,OB,max}$ on the outbound is higher than the maximum available mass $m_{cons,IB,max}$ on the inbound.

Table 14. Results Maximum Payload Mass

Outbound			Inbound		
$m_{PL,OB,max}$	[kg*]	148443	$m_{PL,IB,max}$	[kg*]	42011
$m_{Cons,OB,max}$	[kg*]	5.89	$m_{Cons,IB,max}$	[kg*]	0.97

* kg per person per day

While the number of 5.89 kg per person per day computed for the outbound leg is feasible to be reached through technical developments such as the optimization of closed loop systems, a value of 0.97 kg per person per day seems unrealistic to be realized. These results impose the conclusion that, according to the presented calculation approach, Starship is able to transport 100 persons to Mars but is not able to return them. For an inbound trip of 100 persons, the mission layout could be adapted, e.g. through the inclusion of a refueling maneuver in LMO or the realization of stopover strategies.

3.8 Capacity ISRU

From the presented calculation variants, initial requirements for the capacity of an ISRU plant can be derived. Regarding the propellant capacity, the calculation variants show that Starship uses most of the available propellant mass on outbound and inbound and will therefore most likely be completely refueled for the inbound trip. The developed ISRU plant should contain the capacity to produce and store a minimum amount of $1100 \cdot 10^6$ kg of propellant consisting of 240000 kg liquid methane and 860000 liquid oxygen [1] and additional safety and redundancy margins.

Considering the consumable goods to be provided for the life support systems, the minimum desired masses $m_{good,total}$ can be derived from the required amount of each good per astronaut per day m_{good} (Table 3), the number of astronauts n_{Astro} and the number of days n_{Days} following Equation (56).

$$m_{good,total} = m_{good} * n_{Astro} * n_{Days} \tag{56}$$

For the requirements of this reference mission, it is assumed that the ISRU plant shall be able to provide consumables for 12 astronauts during a surface stay of 500 days and an inbound trip of 300 days. The results are presented in Table 15.

Table 15. Capacity ISRU Consumable Masses

Parameter	Unit	Value
$m_{water,total}$	[kg]	2400
$m_{oxygen,total}$	[kg]	3840
$m_{food,total}$	[kg]	23040
$m_{hygienics,total}$	[kg]	14400

4. Discussion

The results in chapter 3 show that it is possible to develop a feasible analytical calculation logic for a manned return mission to mars. As the simple Hohmann Transfer combined with the Patched Conic Approximation show significant deviations from realistic values, further optimization strategies to reduce the mission Δv such as aerocapture and aerobraking have to be considered. It can be concluded that the development and execution of interplanetary missions to Mars, such as SpaceX Starship mission, take strongly optimized mission profiles and technologies as foundation, which need to be modelled in sufficient detail.

The preparation for Mars missions provides different opportunities for the development of technologies that further improve and optimize the spacecraft's performance. These include among others high performance propulsion systems, lightweight structures, high efficiency closed loop life support systems and ISRU production plants.

The critical character of ISRU production plants for Mars missions is verified within calculation variant 4. The computed results displayed in Table 9 show that Starship in its referenced configuration does not have the capability to carry the combined amount of fuel required for the outbound and inbound trip of the reference mission. A refueling process on the Martian surface is an inevitable mission component. While this refueling process can be realized by additional cargo missions, the propellant production through ISRU plants could provide a competitive and sustainable alternative.

For long-stay mars missions the production of goods for the life support system can further improve the efficiency of a mission. Martian soil and regolith in combination with the existing atmosphere can be used to produce different goods using strategies such as the Sabatier reaction [23][28]. Establishing a production system which reliably provides consumables to use during the surface stay and for the inbound trip can significantly reduce the payload mass that has to be considered for the outbound trip. As calculation variant six displays in Table 12, a reduction of the payload mass of up to 76.6% could be reached allowing for a higher outbound Δv or a higher payload of non-consumable goods.

While autonomous production systems can begin the installation and production processes prior to human arrival at mars, systems that require human interaction have to reach the required product capacity within a limited operation time depending to the surface time of the mission. The advantages and disadvantages of each variant are examined during the further development process of the ISRU production plants in order to establish an optimized modular solution with maximized efficiency and flexibility.

Referring to the SpaceX reference mission, the developed ISRU plant should contain the capacity to produce and store a minimum amount of $1100 \cdot 10^6$ kg of propellant consisting of 240000 kg liquid methane and 860000 kg liquid oxygen [1] and additional safety and redundancy margins to completely refuel Starships propellant tanks. In addition, the possibility to produce 2400 kg potable water and 3840 kg breathable oxygen as well as goods for the creation of 23040 kg foods and 14400 kg hygienics plus margin to service 12 astronauts for 800 days should be considered. Depending on the selected operation mode of the ISRU plant, the operation time might be limited to the duration of the surface stay of 500 days. The ISRU plant should be able to produce all required goods within this duration. Furthermore, the chances and challenges of producing elements to be used for other applications such as construction purposes should be further examined.

Following the potential development towards regularly realized manned mars missions, the production capacity should be scalable to service multiple spacecrafts and a larger number of astronauts for a longer duration. This adaptation of capacities could be realized through the development of a flexible modular ISRU system approach.

5. Future Work

To extend the validation of the developed analytical calculation approach, available calculation programs will be reviewed and used to numerically compute analogue results for the selected SpaceX reference mission. The focus is set to the examination of necessary input parameters as well as the impact of different simplifications on the calculation results. The results of both calculations will be compared and assessed towards their level of deviation. The findings of the comparison will be used to decide which level of detail is desirable for an effective calculation toolbox and which simplifications can be tolerated. A set of requirements considering among others handling, input parameters, calculation strategy, transparency, level of detail, output parameters and costs will be established. Based on this set of requirements, an available program will be selected or an individual toolbox will be developed.

The toolbox will then be used to analyze different mission concepts to Moon and Mars such as the NASA Design Reference Mission 5.0. The gained results will be used to derive a qualitative and quantitative set of requirements for the capacity, production goods and time and cycle of production as input for the continuous design and development process of modular ISRU production plants.

The potential of modular ISRU production plants to present an efficient and sustainable alternative to cargo supply missions will be closely analyzed.

6. Conclusion

In this contribution, a calculation toolbox developed by TUD as a tool to determine the necessary production capabilities of a modular ISRU system is applied to a manned Mars return mission. The model uses Hohmann Transfer, Patched Conic Approximation, aerocapture and aerobraking as part of its calculation logic to estimate the velocities, Δv, propellant mass and payload mass of the selected mission. Seven calculation variants are established with increasing level of detail. The available data of the SpaceX Starship manned Mars return mission and profound assumptions build a reference mission to validate and optimize the established calculation logic. The results show that the implementation of Hohmann Transfer, Conic Approximation, Aerocapture and Aerobraking is required to provide a sufficient level of detail.

It can be concluded that the development and execution of interplanetary missions to Mars require a high level of detail and take strongly optimized mission profiles and technologies as foundation. According to the developed calculation approach, Starship is in its current configuration is able to accomplish the outbound and inbound trip of a long-stay conjunction type trajectory manned mars return mission as well as a 3-year free return trajectory. In order to support the transport of 100 astronauts during long-term missions, the adaptation of Starships capacity seems to be desirable.

The calculated results prove that ISRU systems present necessary building block for the production of propellant and consumable goods for manned Mars return missions. A first set of requirements regarding production goods, propellant and consumable capacity as well as time and cycle of production was derived and is transferred to the development process of the modular ISRU plant. These include 240000 kg liquid methane and 860000 kg liquid oxygen for the spacecraft refueling on martian surface. Additional 2400 kg potable water, 3840 kg breathable oxygen and goods for the creation of 23040 kg foods and 14400 kg hygienics plus additional margins are desirable to service 12 astronauts for a duration of 800 days. Depending on the need for human interaction, the operating time of the production plant may be limited to the duration of the surface stay in the order of 500 days.

The developed calculation logic will be extended to different types of crewed and cargo missions to Moon and Mars. Comparing different mission concepts will provide a comprehensive data record representing valuable input data for the development of modular ISRU production plants.

Appendix A (Comparison of Consumable Masses)

Parameter	Unit	[23]	[29]	[30]	[31]	[32]	[33]	[34]
Astronaut	[kg]	-	-	-	-	-	-	82
Water	[kg/astronaut/day]	2	0.7	2.9	2.4	2.1	2.38	2.5
Food	[kg/astronaut/day]	1.5	3	2.5	2.3	0.5	-	2.39
Gases	[kg/astronaut/day]	4	2	-	-	1.5	1	0.895
Hygienics	[kg/astronaut/day]	25.6	23	0.5	20.5	0.5	1.7	0.7
Waste	[kg/astronaut/day]	0.5	1.7	0.5	-	0.2	5.57	4.6
Recycling	[kg]	-	-	70	-	37	80 / 99	-

Appendix B (Comparison of Mission Duration)

Parameter	Unit	[14]	[35]	[36]	[13]	[37]	[38]	[39]	[11]
Outbound	[d]	-	257	228	200-230	180	230	347,5	174
Surface	[d]	530	500	450	-	543	460	358,1	539
Inbound	[d]	-	241	301	-	180	300	282,9	201
Total	[d]	-	998	980	-	903	973,1	988,5	914

Appendix C (Comparison of Planetary Data)

Parameter	Unit	[23]	[40]	[15]	[37]
AU	[m]	$1,49597870691 \cdot 10^{11}$	$1,49597870691 \cdot 10^{11}$	$1,49598 \cdot 10^{11}$	-
r_E^*	[m]	$6378,14 \cdot 10^3$	$6378 \cdot 10^3$	$6,378 \cdot 10^6$	$6378,14 \cdot 10^3$
m_E	[kg]	-	$5,9736 \cdot 10^{24}$	$5,974 \cdot 10^{24}$	-
μ_E	[m^3/s^2]	$3,9860044 \cdot 10^{14}$	$3,986 \cdot 10^{14}$	$3,986 \cdot 10^{14}$	$3,9860044 \cdot 10^{14}$
g_E	[m/s^2]	-	9,78	9,81	-
a_{S-E}	[m]	$149,5982674 \cdot 10^9$	$149,6 \cdot 10^9$	-	-
r_M^*	[m]	$3396,2 \cdot 10^3$	$3397 \cdot 10^3$	-	3397•103
m_M	[kg]	-	$0,64 \cdot 10^{24}$	-	-
μ_M	[m^3/s^2]	$42828,3 \cdot 10^9$	$0,04283 \cdot 10^{15}$	-	$42828,3 \cdot 10^9$
g_M	[m/s^2]	-	3,69	-	-
a_{S-M}	[m]	$227,9438319 \cdot 10^9$	$227,9 \cdot 10^9$	-	-
μ_S	[m^3/s^2]	$1,32712440018 \cdot 10^{20}$	-	$1,327 \cdot 10^{20}$	-

*Equatorial radius

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