The Blue Mars Mission
Report of group 2

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Abstract

The main work of Group 2 described in this report consists of the overall trip design, from the surface of the Earth to the surface of Mars, and back. Particular emphasis is given to the design of the interplanetary trajectory for the crew vehicle, to the design of the rockets needed for the payloads, to the propulsion system, to the Δv requirements and finally to the sequence of launches from the Earth and to the sequence of events and vehicles needed for landing on the Red Planet.

For the trajectory design the NASA Mission Design Center's Trajectory Browser, is used. Accordingly, the Δv are obtained. The design of the rockets is based on the performances of the propellants here considered and on the payload dimensions. Finally, on the basis of the previous results of this project, the number and sequence of launches from the Earth to LEO is calculated. The design of the events and vehicle needed to land on Mars is based on the description of reference missions, on the amount of payload that has to land on Mars and on the characteristics of Mars and its atmosphere.

The chosen mission has a duration of 577 days with 112 days on Mars. The outbound trip i 304 days and include a Venus swing-by. The return trip is 161 days with crew direct entry to Earth. Total mission Δv is 10 km/s. The vehicles sent to Mars are one crew rocket and two cargo rockets. The Mars descent/ascent vehicle, surface habitat and equipment is pre-deployed prior to crew departure. The total mass delivered to LEO for this mission is 1580 tonnes dived into 17 launches during a total launch campaign of 6 years duration. Mass saving have been accomplished using pre-deployment of Earth return stage and also using In-situ-resource-utilization for life support and Mars ascent propellant. The total launch cost is estimated to 7370 M$ and this is achieved by using commercial launch systems from SpaceX in a large extent.

Contents

Introduction

Spinning around the sun at a 1.5 times the distance our own planet, the red soil of Mars has recently ventured away from the realm of dreams and impossibilities and into the realm of strict feasibility. This report, authored by group two of the Blue Mars mission, explains parts of the big endeavour that is putting man on Mars and ensuring a safe return.

The overlaying goal of group 2 started out as the propulsion group responsible of "To and from the surface of planets", but quickly evolved into more. Outlined in this report are the overall mission planning (excluding development and testing phases), the full trajectories for both human and cargo flights, the transport into and assembly in LEO, the descent and ascent on Mars and the overall Mars base layout. The Transhab and total ground equipment weight has been treated as payloads and the new overall objective became to design a mission including of the different gives payloads.

There has been a lot of research into the subject and there have even been complete mission proposals. NASA has published several reference missions and the latest Design Reference Architecture 5 [1] (DRA 5) together with its austere implementation [2] has been important parts in setting the upper boundary for feasibility, assisted by several other studies which referenced where applicable.

Due to the open nature of this project, in order to get anywhere one has to set up design goals and constraints, which resulted in the following:

- The mission should be feasible and based on existing technologies.
- The main launches should start in roughly 20 years.
- Total mission time (excluding testing and development) should be as low as possible.

Additionally, although it was not a direct constraint, the economical aspect was weighted in where possible. Finally, it should be noted that a big project like this is an iterative process, but this report would be far too long if the entire process were to be documented.

To get a full overview of the mission proposed in this report, please see chapter "Mission Summary" further into this report.

Trajectory analysis

The content of this chapter analyzes the trajectories designed for the mission: the outbound trip to Mars, the orbits around the Red Planet and finally the trip from Mars to the Earth.

A trip to Mars poses a different level of challenges with respect to a trip to orbits around our planet, or to the Moon. The first aspect that must be taken into account is the distance between the Earth and Mars: according to the relative positions of the two planets, they are separated by a distance that varies between 55 and 400 million km. Also, when planning a trip to Mars, it is not possible to launch at any time, any year. It is necessary to consider well defined launch windows that guarantee that the spacecraft can reach the orbit of Mars (and Earth on the way back) in the very moment when the planet is in that position on its orbit.

On the basis of these considerations, different options were analyzed in the early phase of this work. The final trajectory design is the result of compromises between the pros and cons of the considered options.

Before presenting the trajectory options, it is necessary to give a brief theoretical guideline about the main actors of the orbital mechanics that rule such a space flight. There are different interplanetary transfer maneuvers each of which featuring pros and cons. The main tasks that must be taken into account when design a transfer orbit are two: The time of flight and the cost of the maneuver, in terms of difference of velocity necessary to the maneuver, the so-called Δv. Usually the optimum is a compromise between these two aspects since a fast trip requires a higher Δv and a "cheap" trajectory implies a long transfer time.

Starting with the low-Δv options, the first analytical solution to go to Mars would be the Bi-elliptic transfer maneuver. This maneuver consists of two elliptical trajectories that link the initial and the final orbit. This option is the most efficient in terms of Δv but implies a long transfer time. Another relevant obstacle is the fact that this maneuver is only efficient when it links two circular orbits and, most important, two *coplanar* orbits. This is not the case of Mars and Earth that orbit the Sun on different orbital planes. Figure 1 shows the bi-ellpitic tranfer.

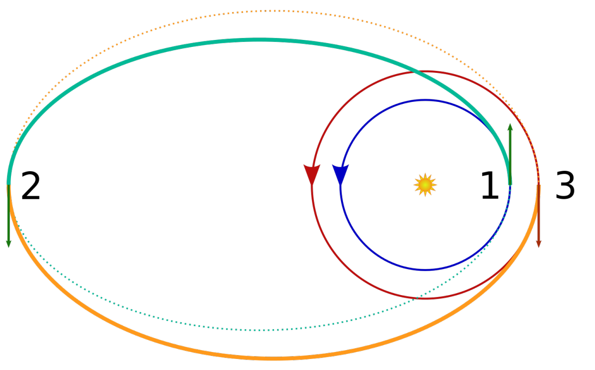

Figure 1: Bi-elliptic transfer. Picture courtesy of Wikipedia.

The second option is the Hohmann transfer. This maneuver is similat to the bi-elliptic transfer but it includes a single elliptic transfer orbit. As for the bi-elliptic transfer, the Hohmann maneuvers links two circular orbits and again, these must be coplanar. Figure 2 shows the Hohmann tranfer.

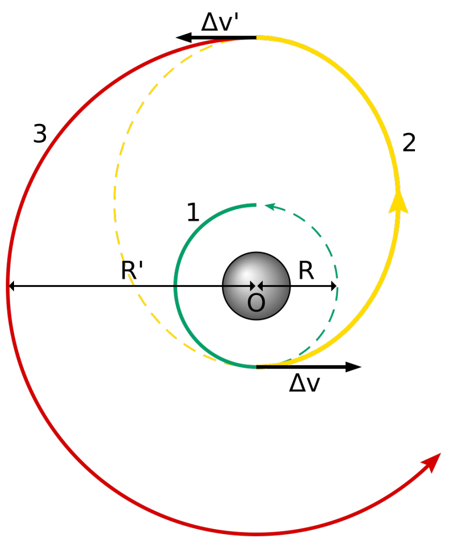

Figure 2: Hohmann transfer. Picture courtesy of Wikipedia.

The third possibility is a Lambert's problem solution. This problem relies on Lambert's theorem, so formulated: *"The transfer time of a body moving between two points on a conic trajectory* [the transfer trajectory] *is a function only of the sum of the distances of the two points from the origin of the force, the linear distance between the points and the semimajor axis of the conic."* [3]

Accordingly, the Lambert's problem is the boundary value problem of the differential equation seen in equation (1), that represents Kepler's two body problem. The general solution of the equation is the keplerian transfer orbit, a *ballistic* orbit.

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

This problem is implemented in this project through the NASA Mission Design Center's Trajectory Browser [4]. This tool determines the possible trajectories that link two celestial bodies using Lambert's problem. The software receives as inputs the celestial bodies to be linked and a set of specific constraints set by the user, like the time window of the launch, the estimated duration of the mission and the maximum Δv. The browser does not provide the result when queried but pulls desired solutions from a database of pre-calculated trajectories.

There are several approximations applied to the Trajectory Browser but the most relevant one, that must be mentioned here, is the "Patched two-body approximation" used by this Lambert's solver. The approximation lies in the fact that the transfer orbit is a heliocentric ballistic (Keplerian orbit) transfer orbit and is calculated without taking into account the "third body perturbation", namely the gravitational perturbation caused by planetary bodies and Sun, the "Solar pressure perturbation", and other minor perturbations.

Therefore Δv and the velocities local to a planet are evaluated with two body dynamics and, in particular, the dynamics of the spacecraft, during its transfer trip, it's only defined by the action of the Sun. It is now possible to describe how the Tajectory Browser is used in this work.

Launch options

The planned launch windows cover the decade 2030-2040, so this is the first input constraint given to the Trajectory Browser. The second constraint is about the Δv: The browser should provide the missions that require the smallest Δv. Once these parameters are fixed, the choice between the different solutions of the browser is based on two criteria: A permanence time on Mars that can guarantee ground operations and the shortest transfer time between the planets.

The first option is the so-called Long Stay mission. For what concerns the crew vehicle only, this option features a 500 days stay on Mars, an outbound trip of 180 days and an inbound trip of 180 days, for an overall mission duration of 860 days. A representation of the Long Stay mission is given in figure 3.

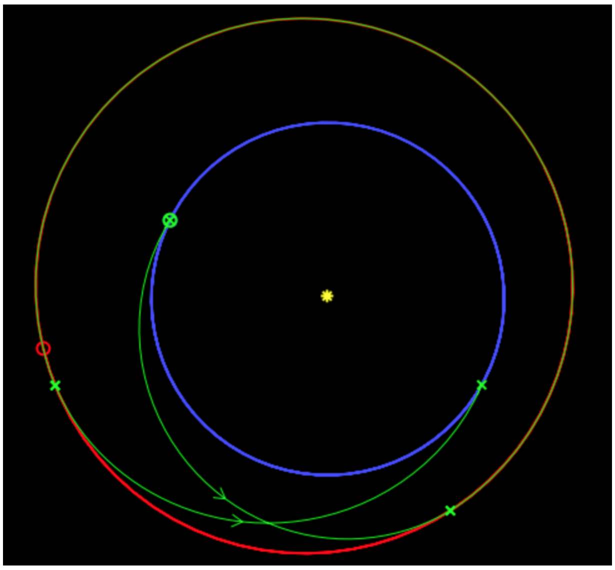

Figure 3: Long stay mission. Picture courtesy of NASA Trajectory Browser.

For this mission the departure of the crew vehicle from the Earth occurs on February the 23th, 2031. The arrival on Mars is on September the 19th, 2031. The departure from Mars would be on January the 27th, 2033 and the Earth reentry occurs on August the 23th, 2033. The trajectories followed by the crew spacecraft are two keplerian ballistic trajectories that directly link the orbits of the Earth and Mars. The required Δv, without considering the launch from Earth and the landing on Mars, is 5.6 km/s.

The second option is the so-called Short Stay mission. In this case the permanence on Mars is of 30 days. The outbound trip takes 150 days and the inbound trip takes 250 days. The total duration of the mission is 430 days. Figure 4 shows a representation of the Short Stay mission.

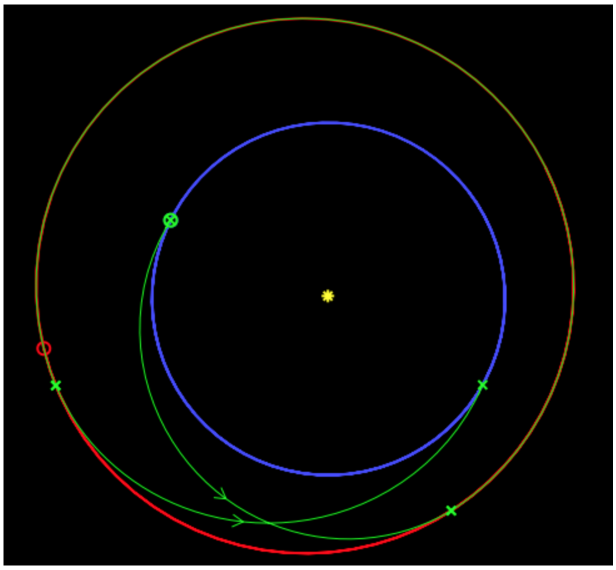

Figure 4: Short stay mission. Picture courtesy of NASA Trajectory Browser.

The key dates of this mission are: departure from Earth on September the 18th, 2037; arrival on Mars on February the 5th, 2038; Mars departure on March the 7th, 2038; Earth reentry, October the 28th, 2038. As for the Long Stay mission, the trajectories followed by the crew vehicle are two ballistic heliocentric trajectories that link the orbits of the two planets. The overall Δv required for the mission is 11 km/s.

These two options were considered and discarded. If a reasonably long stay on Mars is desirable, it is therefore true that 500 days on the surface of the Red Planet pose serious challenges for the equipment and supply needed for the mission. This mission was discarded for its too long duration. The second option, on the other hand, features a short stay time on Mars that ensures limited issues related to the supplies but the required Δv for the crew vehicle on earth return was simply too high and too costly to counter. The trajectory chosen for this project is a compromize between the two just presented.

Proposed launch window

The trajectory design chosen for the crew vehicle in this project features a Lambert transfer from Earth to Venus, a fly-by around the latter to gain speed by means of a gravity-assist maneuver, a Lambert transfer from Venus to Mars and finally a direct Lambert transfer from Mars to the Earth. During the outbound trip, in proximity of Venus it is also possible to enter a free-reentry trajectory to the Earth in case of mission abort.

These are the details of the different phases of the mission: Earth departure on June the 12th, 2036: the insertion maneuver into the transplanetary orbit requires a Δv of 4.24 km/s; Venus fly-by on November the 19th, 2036, that requires a Δv of 60 m/s; Mars arrival on April the 12th, 2037: the insertion into the system of Mars requires a Δv of 3.98 km/s; the departure from Mars occurs on August the 2nd, 2037 and requires a Δv of 1.62 km/s; the Earth reentry is on January the 9th, 2038.

The permanence on Mars is of 112 days, the outbound trip requires 304 days and the inbound trip lasts 160 days for an overall duration of 586 days and a total Δv of 10 km/s. the initial parking orbit around the Earth is a Low Earth circular Orbit with an altitude of 407 km and the parking orbit around Mars features an eccentricity of 0.8 and a perigee altitude of 250 km. Figure 5 shows the trajectory plan for the mission.

The choice of the actual trajectory plan is based on a pros and cons analysis. This plan features a good compromise between 500 days and 30 days on the surface of Mars: 112 days are long enough for scientific research but still pose issues for the supplies. Also, there is the possibility of a free reentry from Venus but the passage around this planet implies a close passage to the Sun that implies additional protections for the crew and cause problems for the communications. Also the outbound trip is long. Finally, for what concerns the Δv, this mission includes a high Δv of 10 km/s but the reentry speed on the Earth is limited to 12 km/s, a desirable speed value.

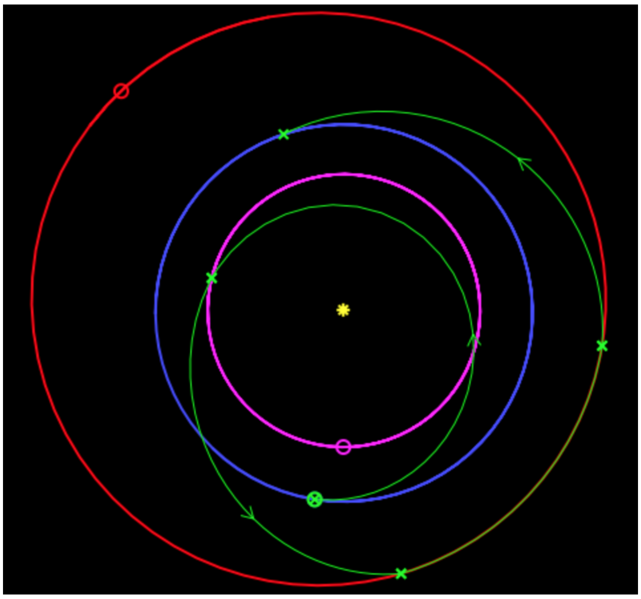

Figure 5: Trajectory of the proposed Blue Mars mission, crew vehicle. Picture courtesy of NASA Trajectory Browser.

The analysis here presented is only valid for the crew vehicle. The approach of this work is infact to design the most feasible trajectory for the crew, in terms of costs and time of flight. The time of flight, in particular, poses constraints according to the fact that the crew should experience the shortest time of flight possible. As such, the trajectories of the cargo rockets are of secondary concern and follow the cheapest possible trajectories that can be found under such constraints.

Mission specifics

In this section the important specifics of the proposed mission are explained. Since all specifics influence each other, their order in this report is mainly structured for ease of reading rather than importance.

Payloads to transport

The main components to transport are:

- Crew Transit Habitat + Earth ascent/descent capsule
- Mars Surface Habitat
- Mars Power/ISRU and Ascent vehicle
- Mars Crew Descent vehicle

Delta-V requirements

With given crew and cargo trajectories Table 1 shows the Δv requirements for the different manoeuvres. All values except MOI have been obtained using the NASA Trajectory Browser [4].

Manouvre	**ΔV [m/s]**	**Comment**
TMI	4300	From LEO (407 km) incl. Venus swingby
Capture	3980	Insertion to mininum capture.
MOI	550	To go into a 250 x 33000 km orbit
MOI inv	550	To go from a 250 x 33000 km orbit
TEI	1620	From min. capture towards earth
Mars ascent	4500	Surface to LMO
Earth entry	12000	Direct re-entry (no LEO rendevouz)

Table 1: Crew mission Δv requirements.

Additional ΔV includes LEO/LMO station keeping, attitude control and also course corrections. In this analysis only the LEO station-keeping has been included since this involves keeping large masses in place during assembly. A value of 161 m/s per year [5] has been used in sizing the LEO booster modules. In order to use the above numbers it has been assumed that all propulsion stages have enough thrust such that the burns can be considered to be impulsive.

Entry, Descent and Landing

Landing payloads on Mars is considered one of the large obstacles to overcome in order to realize an Exploration Class mission. Martian atmosphere is much thinner and has a lower density than earth. Figure 6 shows a comparison between Earth and Mars Atmosphere.

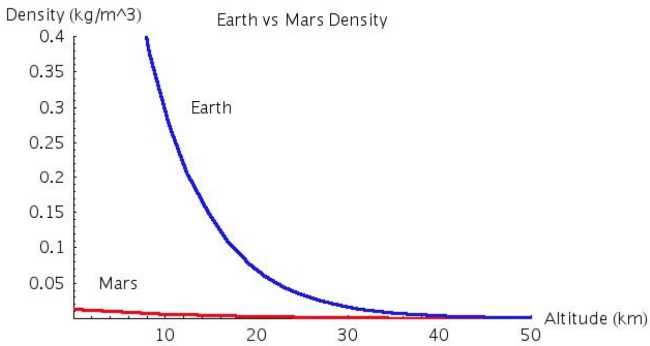

Figure 6: Density comparison of Mars and Earth atmosphere.

Previous robotic missions to Mars has built upon the legacy of the Viking lander technology. They use a heat shield for hypersonic deceleration together with a supersonic parachute in order to slow down the payload below supersonic speeds. The last subsonic phase of the landing has been carried out with a combination of propulsion together with either subsonic parachute, inflatables (as with Spirit/Opportunity) or more recently the "Sky-crane Manoeuvre" used for the Mars Science Laboratory. The ballistic coefficient β of an object flying through the atmosphere is the ratio between its mass m, its drag coefficient C_D and its aerodynamic area A, as seen in equation (2).

$$\beta = \frac{m}{C_D A} \tag{2}$$

What these previous payloads have in common is that their mass is quite low (244-775 kg usable payload) and that they use the same aerodynamic 70 degree cone shell shape which provides the largest hypersonic drag coefficient. If this shape is scaled up (assuming the same payload density) the mass increases with the cube while the drag area only increases with the square. This means that the ballistic coefficient increases. Looking at figure 7 we see that a high ballistic coefficient combined with the martian atmosphere means that at a certain mass there will not be enough time (or altitude) to slow down the payload enough before it hits the ground. To solve this one has

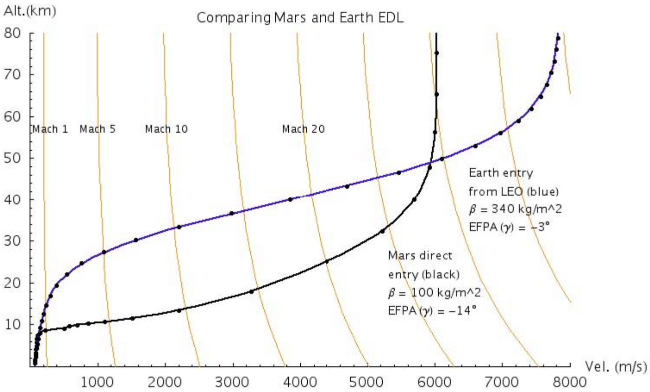

Figure 7: Velocity/Altitude profile comparison between Mars and Earth EDL.

to increase the hypersonic and supersonic drag while staying within geometrical limits of Heavy Launcher capabilities. The use of parachutes for supersonic deceleration have been doomed infeasible [6] since they become too large and also because of material strength and thermal limits.

Current NASA studies [7] are investigating the most promising concepts with inflatable shells HIAD (Hypersonic inflatable aerodynamic device) and supersonic retro propulsion. Table 2 shows a summary of the estimated masses of the EDL components. These numbers are based on simulation results for landing a 40 tonne payload. The arrival mass for this type approach is around 110 tonnes depending on exact configuration. The simulation performed in the study include constraints on maximum dynamic pressure, thermal heat load and also acceleration (max 4g axial). The estimated mass of the aeroshell also includes enough thermal protection to provide the aerocapture part of the mission.

Component	**m [ton]**	**Comment**
Aeroshell	25	For aerocapture and hypersonic de-cel.
Other	4.3	Structure, avionics, separation eq. etc.
Entry RCS	10	Wet mass of entry propulsion(s).
HIAD	6	Hypersonic Inflatable shell w. thermal prot.
Prop.desc	22.8	Wet mass of subsonic descent stage
Total	68.1	

Table 2: Mass of EDL components.

The values in Table 2 have been used as fixed discarded masses for calculations with only the descent propellant being calculated as a function of payload mass. Some additional assumptions and requirements identified are:

- Successful orbit insertion/aerocapture has been performed.
- Sufficient navigation capabilities (satellite/star tracker/beacons).
- Good statistical data on Mars Atmosphere.
- The terrain is known in detail (map) or the lander can utilize real-time hazard avoidance.
- Final descent dV is 700 m/s (Retro propulsion is initiated at Mach 2.5).

Propulsion

Looking at the objectives ,the trajectory and the large masses involved with this type of mission the following is observed:

- A high Isp is needed for crew Trans-Hab vehicle in order to keep propellant mass within reasonable limits.
- Ability for TEI propulsion system to loiter for long time before being fired is needed.
- Use of storable and reliable propellants for crew Mars descent vehicle.
- Storable or manufacturable propellants for crew Mars ascent vehicle.

Liquid Hydrogen vs. Methane

Restricting oneself to chemical propulsion there are two feasible alternatives for TMI/MOI/TEI stages. These are Liquid Oxygen/Liquid Hydrogen or Liquid Oxygen/Liquid Methane. As can be seen in Table 3 the Methane alternative has lower Isp but higher storage temperature and a higher density which is very attractive. No LOX/CH4 engine has ever been used in practice but there are several R & D projects in progress[1].

Propellant	Isp [s]	Density [kg/m^2]	Mix ratio
LOX/LH2	450	0.3	5
LOX/LCH4	360	0.8	3.45

Table 3: Propellant comparison. For both fuels, cryocooling is needed; 20/90 K for LOX/LH2 and 90/90 K for LOX/LCH4.

Although Methane seems beneficial the mass penalty is too large. For this mission LOX/LH2 was choosen for the TMI/MOI/TEI stages with a resulting reduction in LEO mass of about 500 tonnes.

Mars Descent/Ascent Vehicle

The choice of propellant for the descent vehicle was driven by reliability. A common choice is MMH/NTO which is a hypergolic bi-propellant with a specific impulse of about 340 s. One issue might be contamination of landing site and pre-deployed habitat but since the crew is living in a closed system and using protective gear when performing EVA this was not considered a big problem. The ascent vehicle was estimated to be 6 tonnes (extrapolated lunar module) and the dV-requirement was 4.5 km/s. For the ascent vehicle the choice was LOX/CH4 since this can be made on-site with a positive mass trade-off. The total required ascent propellant was higher than the ISRU system mass including the hydrogen needed for production (more about this in section about ISRU). Producing ascent propellant with ISRU reduces the mass to LMO with 20 tonnes. A fallback would be MMH/NTO with reduced complexity and mass penalty.

[1] http://en.wikipedia.org/wiki/Raptor_(rocket_engine) http://science1.nasa.gov/science-news/science-at-nasa/2007/04may_methaneblast/

Structural factors

The structural factor $\varepsilon = M_c / (M_c + M_p)$ relating dry construction mass M_c to propellant mass M_p for the rocket stages used have been taken mainly from [1] and [5]. Common for all large TMI/MOI/TEI modules is the need for their own power ,avionics and cryogenics. This results in every stage being almost a full feathered spacecraft with all necessary systems. The assumptions for these were 15 %. The mars descent stage also has a high structural factor since it has to have a landing structure capable of supporting the 30-40 tonnes of payload. This was set to 50 % which also corresponds to results from EDL studies [7].

TEI stage pre-deployment

Due to the high Δv associated with the crew trajectory the possibility of sending the Transhab return stage in advance as payload was considered. This would reduce mass to LEO for the cost of more complexity and risk. The analysis showed doing this would save 475 tonnes in total mass to LEO. Pre-deploying the TEI stage would require it to loiter in LMO for 2.5 years before being fired. The choice was to use this approach despite the risk and complexity since the mass (and thus the cost) saving is so big.

In-Situ Resource Utilization

In order to reduce the total mass carried to Mars it becomes a necessity to bring as little as possible of everything. One way to reduce this mass is to produce as much as possible using elements already existing on Mars. ISRU stands for In-Situ-Resource-Utilization. If we look at the atmosphere of Mars it consists of over 95 % carbon dioxide. Using the Sabatier reaction makes it possible to produce methane and water from hydrogen and carbon dioxide. The reaction is seen in equation (3). If combined with the reverse water-gas shift reaction, seen in equation (4) the full equation can be seen in equation (5).

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O + energy \quad (3)$$
$$CO + H_2O \leftrightarrow CO_2 + H_2 \quad (4)$$
$$2H_2 + 3CO_2 \rightarrow CH_4 + 2O_2 + 2CO \quad (5)$$

This reaction has a mass ratio of 20:1, meaning bring 1 kg of hydrogen, get 20 kg of methane and oxygen. Specifically 16 kg of oxygen and 4 kg of methane.Through this reaction it is possible to calculate how much hydrogen it would be necessary to bring to mars to produce. The leftover oxygen not needed for the rocket fuel can be used for life support.

The reaction is not without it's complications, since it requires keeping almost 2 ton of hydrogen below a temperature of 20.23 K for a long duration.Testing on this has been done before, in 2011 hydrogen and carbon dioxide were used in this way to create oxygen and methane, the conversion rate was 98 % and energy required for one ton of fuel was 17 MWh [8]. Scaled to this mission, ISRU unit and nuclear power generator together equates to a mass of roughly 10 ton and would be sent well in advance of crew arrival.

Mars base organizaion

Having a picture of the rough requirements and limitations of the descent and its implications on the whole the overall picture of the equipment put on ground starts to take form. A total of two modules will reach the soil of Mars, with a ground weight of 20 ton and 26 ton for the PWR/ISRU and habitat modules, respectively.

They will arrive on the slower cargo trajectory on separate crafts and land in a designated area as close as can be considered safe with respect to the exhaust gases and their effects close to ground. The assumed distance is somewhere between 20 to 100 meters.

After the two modules are safely in suitable positions, 497 days before the crew arrive, and are diagnosed as still intact, a hatch in the utility module will open up, letting an unmanned rover out on the ground and then close. This rover is an integral part of the mission setup and has several functions. This is because the rover is basically a power plant on wheels. With an estimated minimum continuous output of 40 kW, the rover will first drive to the crew habitat and connect a special cable from the utility module to the habitat module. This connection will include electrical power, water and oxygen and convert the two modules into one single system.

With one cable connected, the rover will slowly make its way away from the modules, until placed at a comfortable distance away, preferably behind some terrain, to minimize crew radiation exposure. Current weight distributions allot for up to 1 km of electrical cable connecting the rover and the utility module. The total setup put in place by group 2 can be seen in figure 8. It should be noted that this is not the full setup – as for the workings of the habitat module, please refer to report from group 4. As for the full weight distribution, please see appendix C.

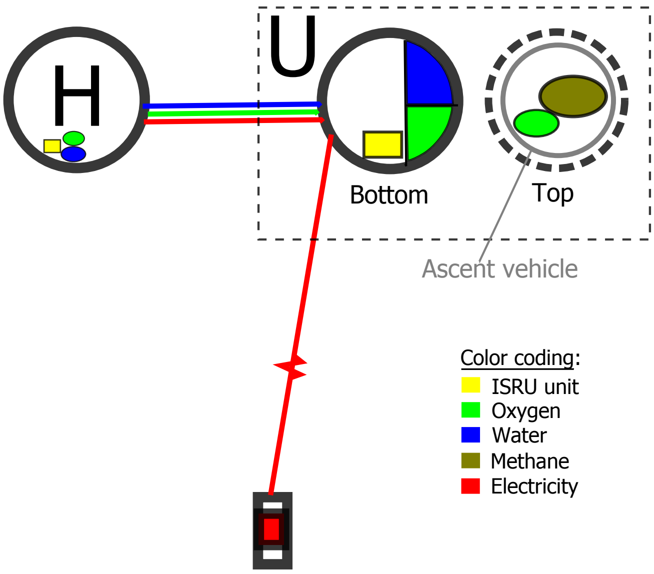

Figure 8: Blue mission mars setup, excluding habitat and science contributions from group 4. Ascent vehicle and life support water and oxygen filled up at the time of crew launch. H is habitat and U utility.

Together with the ISRU technologies mentioned in the previous section, this setup will ensure a radiation free[2] mars base with fully fuelled return vehicle, water and oxygen tanks well in time for the crew launch from earth.

Redundancy measures includes a smaller 3 tonnes ISRU and power rover in the habitat module. This can be used in case the main system fails when crew is en route or during the surface mission.

Earth to LEO

With Δv's and payload masses known, the rockets can be designed. With the choice of conventional over nuclear propulsion, the total assembled LEO masses are heavy. See table 4 for a summary. As a comparison, the total weight of the ISS is around 450 ton.

Vehicle	Mass [ton]	Cost/ton [M USD]	No. of Launches
Crew	802	7.86	9
Cargo A	238	1.04	2
Cargo B	252	1.72	3
Cargo C	274	6.39	3

Table 4: Assembled LEO weight of the misison components. The launch cost are higher for the crew and Cargo C vehicle due to the use of the SLS.

With all LEO vehicles being several times larger than the current feasible maximum payload, it is needed to assemble the rockets in LEO. And from a cost and redundancy perspective, the stages should also be modular and made up of as much standard parts as possible.

As for the size and weight of the individual parts, they have to be designed with current launcher capabilities in mind. For this project, two different partners are chosen, both for practical and political reasons. The first one is SpaceX, with an estimated fleet around the time of launch seen in figure 9. The second partner is NASA, which due to cost of their new SLS system will be used sparsely.

To limit complexity and cost, the LEO launchers proposed in this project can be seen in

[2]With respect to nuclear power generation.

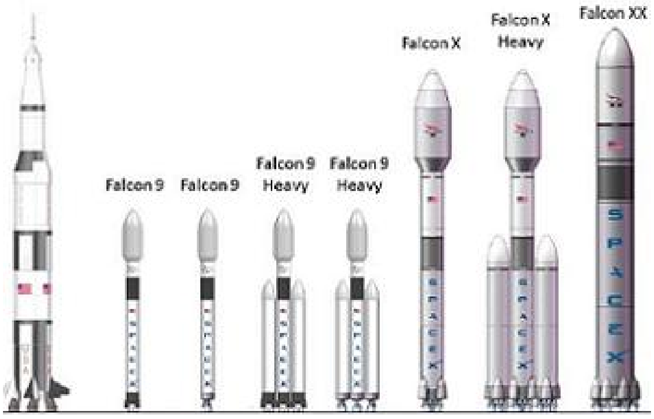

Figure 9: Estimated SpaceX launcher family at the time of mission start. To the left is the Saturn V for comparison.

table 5, which makes for the summarized values seen in table 6. For the total launch setup, please see appendix A and B. In addition to all this there is one final launch with the crew that can be sent with any launch vehicle that politics or economics demand.

Vehicle	Payload [ton]	Cost [M USD]	Times used
Falcon Heavy	53	135	2
Falcon XX	140	300	13
SLS Block 1a	105	1600	2

Table 5: Summary of the different launch systems used during the proposed mission.

Launch figure	Value	[Unit]
Total launches	17	
Total LEO mass	1566	ton
Total launch cost	7370	M $
Average cost/launch	433.5	M $
Average cost/ton	4.71	M $

Table 6: Summarized earth to LEO launch figures. For full values, please see appendix B.

Mission summary

An overview of the mission profile can be seen in figure 10. Surface equipment, return stage and crew lander is assembled in LEO starting 2030 and then pre-deployed about 3.5 years before crew departure. Upon arrival to Mars the surface habitat and power/ascent module performs aerocapture and lands on the surface. The crew lander and return stage loiters in LMO. Crew vehicle begins LEO assembly 2 years prior to departure. Once the assets on and around Mars are verified the crew launch to rendezvous with the Transhab before departing Earth on June 12 2036.

After a 304 day outbound trip including a Venus swing-by the crew arrives at Mars for a propulsive orbit insertion using the MOI stage. They then perform LMO rendezvous with the pre-deployed TEI stage and the crew descent vehicle. The crew descents to the surface, performs the 112 day mission and then goes back up to the Transhab using the pre-deployed ascent vehicle powered by the in-situ manufactured propellant. The ascent vehicle is discarded before departing Mars. When approaching Earth after a 161 day return trip the crew transfer over back into the same vehicle used for leaving Earth Surface. The Transhab is then discarded and the crew enters earth atmosphere directly at a speed of 12 km/s.

Failure scenarios

There is always a risk of a failure when you launch something into space, and with the many launches is required to go to mars the odds of failing at least one launch is not negligible. Looking at statistics from 2012's launches, there were 78 launches with 6 of them failing [9], meaning a success rate of 92.4 %. When having so many launches during a short period of time this becomes a concern.

Assuming zero failures, getting everything into LEO will require a total of 17 launches. If calculated with 92.4 % chance of success the odds of one failing is 74 %. This means there must be flexibility in the launch dates, back-up material and vehicles available. If a rocket was to fail, it is very important that both payload and launch vehicle can be replaced quickly.

The plan is to launch and assemble the crew vehicle in 2034-2036. The crew is to be launched on June 12, 2036. This is a date which cannot be switched around much, we must hit this window to make sure launch from LEO to mars uses the correct amount of fuel and life support. This means an 30 month launch window to do at least 9 launches. Averaging this out it becomes 1 launch per 3.3 months.

Currently, SpaceX launches occur more frequently than this so it shouldn't be a problem [10]. The problem would lie in the companies' abilities to produce enough of the launchers in time, and having launch sites able to launch the specific needed vehicles. Ordering the launchers far in advance can counteract one of these problems. The launch timetable must also be able to afford loss of launchers.

The cargo going to Mars first will be launched from 2030-2032 with 8 launches. These launches also need to work properly but have a less hectic schedule, if we are able to handle the crew vehicle situation, these launches should not be a problem.

Should something go wrong after the Venus swingy there is no option but to go to Mars, get into Mars orbit and then wait for the return window. There is enough supplies on the vehicle to get the crew both there and back but also to loiter in LMO for 112 days in case landing is not possible. Other possible failures are docking not happening correctly during LEO docking for the crew, in this case the crew can simply return to earth in their module. Should this happen the whole mission is compromised as there is a launch window we must hit. Should something go wrong with the vehicles sent in advance to Mars, the mission can be aborted without the risk of losing human lives.

Overall the mission is very expensive and complex. The risk of loosing the crew is is quite high given the previous success rate of Mars mission together with the increased complexity associated with Human Space flight systems and requirements.

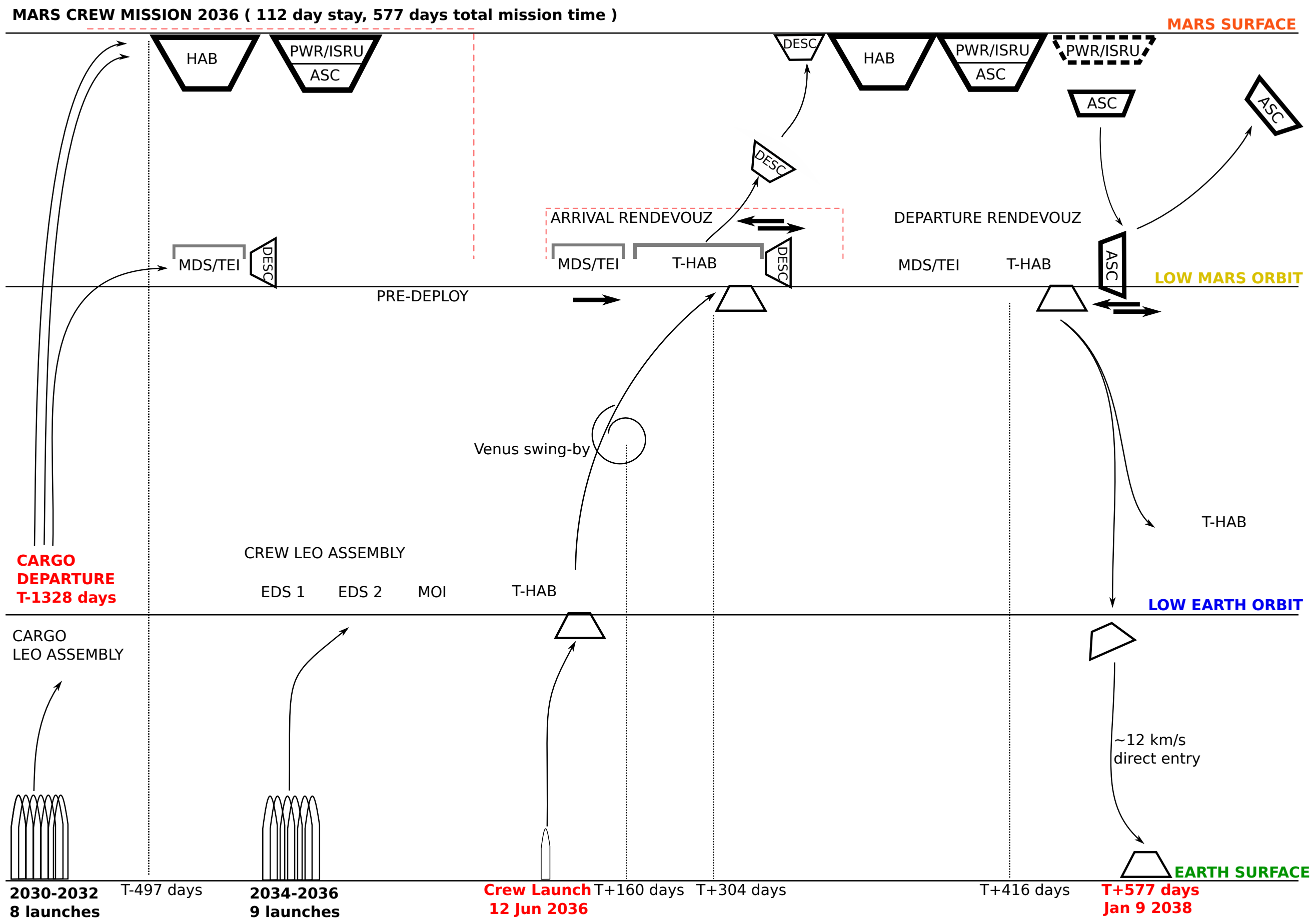

Figure 10: Timeline of the proposed mission with a 2036 crew launch.

Final thoughts

Going back to the constraints posed in the beginning of this project, it becomes clear that this is an ambitious but still very much feasible project. Except for the hypersonic retro-propulsion used during the descent on Mars, all technologies are or will be ready at the time they are needed. The development and testing schedule presented by group 1 also point towards this.

The total mission time could have been shorter at the cost of a very big Earth Injection Stage that would have to travel to Mars, but there's no proof that would reduce the overall mission risk. A risk that is already high due to the sheer complexity of the project. And if something happened during the first, it might take a very long time before another Mars mission gets funded.

Taking a look at the earth to LEO launch division in appendix A and B, one can see that the usage of SLS comes with a significant price tag. If a mission the size of the Blue Mars Mission were to be realized, it would probably be ran as an international corporation, to whom NASA would have to sell their launch systems at a discount. But since costs never just disappear, it would mean the burden would instead be shifted to American tax payers.

The heavy reliance on U.S. technology can pose a problem since other contributors would want something in return for their contributions. It might actually be such that the biggest single challenge in the Blue Mars Mission is not the technical aspects or finding suitable astronauts, but in the politics of getting the project started.

Finally, the total price tag of this project might have been reduced if nuclear propulsion had been selected for the interplanetary stages. Preliminary calculations indicated an almost 50 % reduction in LEO mass, meaning significantly lower launch costs, for the price of proper nuclear technology development. For missions further into the future, this might be a good choice assuming enough development and testing time. For that however, one needs to mind the political landscape as the word "nuclear" is associated with a plethora of other issues.

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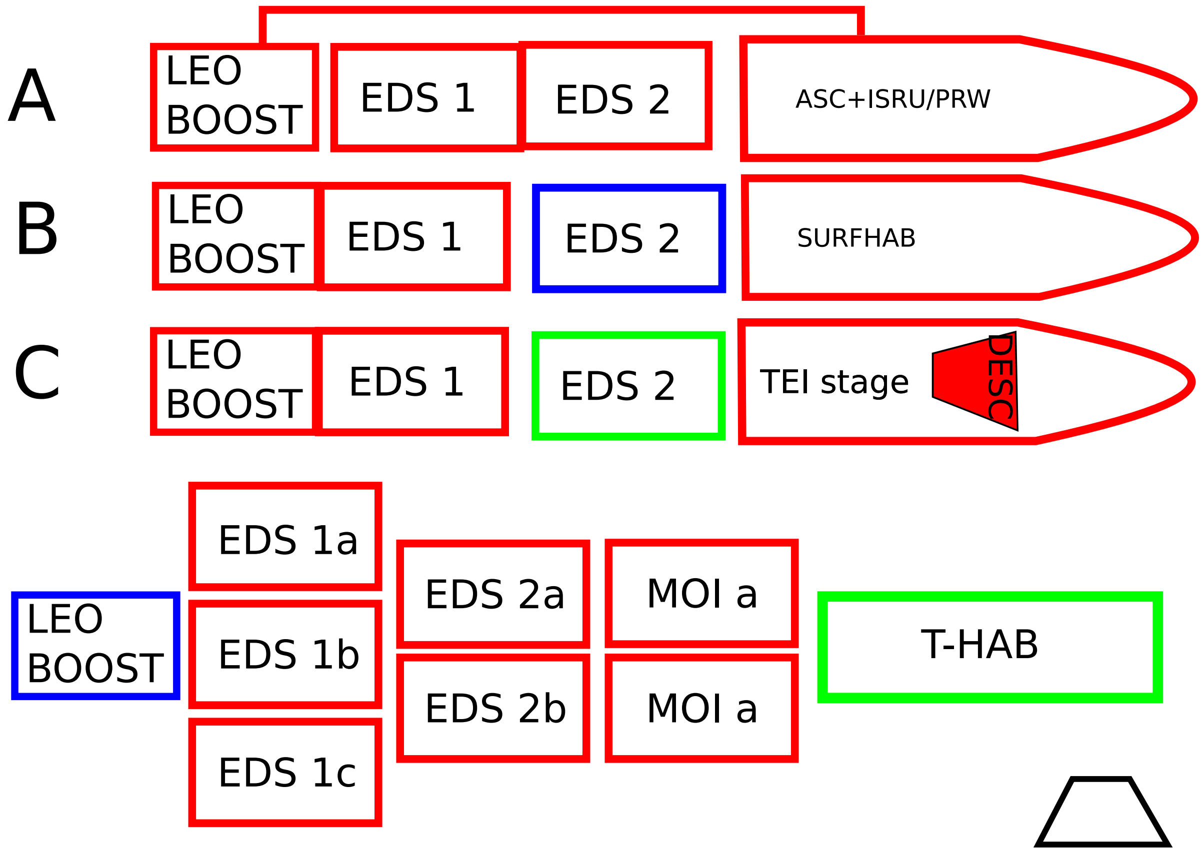
Appendix A - Earth to LEO launch diagram



A, B and C stands for the different cargo rockets being assembled LEO to be sent towards Mars. EDS stands for Earth Departure Stage and MOI for Mars Orbit Injection stage.

Regarding the coloring: Items colored red are scheduled to be sent up with the Falcon XX. In some cases, some of the red parts (but not all) can be sent up with the Falcon X Heavy if it is cheaper than the Falcon XX and available at the time of launch. Items colored blue are to be sent up with the Falcon Heavy and the green colored items with NASA's SLS.

Do note that SLS gets the "honor" of sending up arguably the most prestigious part, namely the Trans-hab. This is a preliminary decision taken due to politics. Both green and some of the red colored items could just as well be sent up using for example the proposed Chinese launcher Long March 9A, if that turns out to be a better choice.

The grey symbol in the right corner is the earth re-entry vehicle which the astronauts will go from earth to the crew vehicle in LEO in. It will then be attached to the Trans-Hab the entire journey, serving as a means for additional space. Once the crew are getting close back to earth, they will once again enter it and disconnect from the Trans-Hab, which will most likely burn up in earth's atmosphere. The re-entry vehicle will be sent up by a separate launch once everything is already assembled and ready.

The LEO boost might also need an explanation. They are attached to each rocket and provide stability and movement for the duration of the assembly. Their main tasks are helping out during assembly, altitude control and collision avoidance whilst in LEO. All LEO boost stages will disconnect before ignition.

Appendix B - Earth to LEO spreadsheet

Chemical crew vehicle

Stage	Weight [t]	Launch method	Launches	Launch cost [M $]	Tot cost [M $]
EDS 1	333	Falcon XX	3	300	900
EDS 2	185	Falcon XX	2	300	600
LEO Boost	55	Falcon Heavy	1	135	135
MOI	171	Falcon XX	2	300	600
T-HAB	58,4	SLS Block 1	1	1600	1600
TOTAL:	**802,4**		**9**		**3835**
					Average cost:

Cargo Vehicle A (ASC/DESC VEHICLE - INCL SURFACE POWER AND ISRU)

Stage	Weight [t]	Launch method	Launches	Launch cost [M $]	Tot cost [M $]
EDS 1 + 2	140	Falcon XX	1	300	300
LEOB + PL	98	Falcon XX	1	300	300
TOTAL:	**238**		**2**		**600**
					Average cost:

Cargo Vehicle B (SURFACE HABITAT - EXCL PWR/ISRU)

Stage	Weight [t]	Launch method	Launches	Launch cost [M $]	Tot cost [M $]
EDS1+ LEOB	109	Falcon XX	1	300	300
EDS 2	56	Falcon Heavy	1	135	135
Payload	87	Falcon XX	1	300	300
TOTAL:	**252**		**3**		**735**
					Average cost:

Cargo Vehicle C (TEI + CREW DESCENT)

Stage	Weight [t]	Launch method	Launches	Launch cost [M $]	Tot cost [M $]
EDS1+LEOB	118	Falcon XX	1	300	300
EDS 2	61	SLS Block 1	1	1600	1600
Payload	95	Falcon XX	1	300	300
TOTAL:	**274**		**3**		**2200**
					Average cost:

TOTAL FIGURES FOR ALL LEO LAUNCHES BELOW

Total launches:	**17**	
Total cost:	**7370**	*In Million USD*
Average cost/launch:	**433,53**	*In Million USD*
Total mass in LEO:	**1566**	*tons*
Average cost/ton:	**4,71**	*In Million USD*

APPENDIX C

g0 9.81 m/s

Mars surface payload details				
Earth return sample	0.025	tonnes	rocks/martian material	G4
Surface habitat mass	23	tonnes	includes everything except water/air (supplied by ISRU)	G4

Trajectory details				
Outbound travel days	304	days	jun12 2036 - april12 2037(incl venus swingby)	nasa traj browser
Stay days	112	days	april12 2037 - aug02 2037	nasa traj browser
Return travel days	160	days	aug2 2037 - jan09 2038	nasa traj browser
TOTAL	576			

Transhab details				
Transhabl total outbound mass	48.413	tonnes	including all consumables	Transhab team
Transhab total return mass	43.731	tonnes	including return consumables	Transhab team
Earth reentry mass	10	tonnes	"Orion size" Vechicle	DRA 5.0

Required dV crew			valid for 12 jun 2036 trajectory (112 day stay)	**SOURCE**
Launch	9400	m/s		
TMI(incl venus flyby)	4300	m/s		nasa traj browser
Capture	3980	m/s		nasa traj browser
MOI	550	m/s	to go from capture orbit to LMO	own calculation
MOI inverse	550	m/s	to go from LMO to capture orbit	own calculation
TEI	1620	m/s		nasa traj browser
Earth entry	12000	m/s		nasa traj browser

Required dV cargo			valid for cargo launch Oct 23 2032 and arrival Feb18 2035	**SOURCE**
Launch	9400	m/s		nasa traj browser
TMI	3650	m/s		nasa traj browser
Capture	1000	m/s		nasa traj browser

Additional dV				**SOURCE**
LEO stationkeeping	160	m/s	valid for one year of LEO stationkeeping+rendevouz	DRA 5.0
Course corrections	75	m/s	not included	DRA 5.0
LMO stationkeeping	100	m/s	not incuded	DRA 5.0
EDL Descent dV	700	m/s	assumed retroproulsion initiated around Mach 2.5 for about 40t payload	Overview NASA EDL systems report 2010
EDL Ascent dV	4500	m/s		Wikipedia

Engine and propellants				
Stage	**Struct.ratio**	**Isp [s]**	**Propellant**	**Notes**
TMI	0.15	460	LOX/LH2 Engine: RL10-B2	incl cryocooler + solar pwr
MOI	0.15	460	LOX/LH2 Engine: RL10-B2	incl cryocooler + solar pwr
Martian Descent	0.5	340	MHH/NTO	incl landing leg structure
Martian Ascent	0.1	350	LOX/CH4	relly rough estimate struct factor!
TEI	0.15	460	LOX/LH2	incl cryocooler+ solar pwr
LEO reboost	0.25	340	MMH/NTO	incl solar pwr
NUCLEAR TMI/MOI/TEI	0.15	850	LH2 + nuclear heasting	incl cryocooler + reactor

Entry, Descent and landing				
Component	**Mass**		**Note**	**SOURCE**
ASC module	6	tonnes	Ascent payload mass	Austere (2*8 lunar module)
DESC module	10	tonnes	Descent payload mass	Orion size vehicle
Crew	0.32	tonnes	4*80 kg people	Own assumption
Martian material	0.025	tonnes		Own choice G4
Total	**0.345**	tonnes	mass to bring from mars for ascent vehicle	
Aeroshell mass	25	tonnes	Hypersonic shell (rough estimate on 40t payload)	Overview NASA EDL systems report 2010
HIAD mass	6	tonnes	Hypersonic chute (rough estimate on 40t payload)	Overview NASA EDL systems report 2010
RCS entry propellant	10	tonnes	Guided entrance propellant (rough estimate on 40t payload)	Overview NASA EDL systems report 2010
Separation structure + avionics	4.3	tonnes		Overview NASA EDL systems report 2010
Total mass lost after entry	**20.3**	tonnes		

ISRU				
ISRU/PWR secondary	3	tonnes	Needed to create oxygen and water for life support (backup)	DRA3.0 and DRA5.0
ISRU/PWR primary	10	tonnes	Needed to create LOX/CH4 using Hydrogen , incl cooler and wheels (movable)	DRA3.0 and DRA5.0
Mixture ratio LOX/CH4	3.21	tonnes	Used in the sabatier process: H2 + CO2 --> CH4 + H2O	Wikipedia "Liquid propellants"

Appendix C: Cargo Rocket dV calculations

CARGO ROCKET A: ASC/DESC VEHICLE (INCL SURF. PWR/ISRU)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre and discarding spent propulsion stage)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
LEO	237.66	TMI	3650	460	0.15	2.25	155.08	23.26	mars	82.59			82.59	2 stage cryogenic
	82.59					1.00	0.00	0.00		82.59			82.59	Aerocapture
	82.59					1.00	0.00	0.00		82.59			82.59	

ASC/DESC + AC shield

CARGO ROCKET B: SURFACE HABITAT (EXCL PWR/ISRU)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre and discarding spent propulsion stage)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
LEO	250.77	TMI	3650	460	0.15	2.25	163.63	24.54	mars	87.14			87.14	2 stage cryogenic
	87.14					1.00	0.00	0.00		87.14			87.14	Aerocapture
	87.14					1.00	0.00	0.00		87.14			87.14	

HAB lander + AC shield

CARGO ROCKET C: TEI + CREW DESCENT

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre		Reconfiguration (before next			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
LEO	273.10	TMI	3650	460	0.15	2.25	178.20	26.73	TMI	94.90			94.90	2 stage cryogenic
	94.90					1.00	0.00	0.00		94.90			94.90	Aerocapture
	94.90					1.00	0.00	0.00		94.90			94.90	

TEI stage + Crew descent + AC sheild

CARGO ROCKET C2: CREW DESCENT only (valid for "incl return")

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre		Reconfiguration (before next			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
LEO	147.03	TMI	3650	460	0.15	2.25	95.94	14.39	TMI	51.09			51.09	2 stage cryogenic
	51.09					1.00	0.00	0.00		51.09			51.09	Aerocapture
	51.09					1.00	0.00	0.00		51.09			51.09	

Crew descent module + AC shield

CARGO ROCKET B2: NOT USED!

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre and discarding spent propulsion stage)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
LEO	0.00	TMI	3650	460	0.15	2.25	0.00	0.00	mars	0.00			0.00	2 stage cryogenic
	0.00					1.00	0.00	0.00		0.00			0.00	Aerocapture
	0.00					1.00	0.00	0.00		0.00			0.00	

Appendix C: Crew Rocket dV calculations

CREW ROCKET (INCL TEI STAGE)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre and discarding spent propulsion stage)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
Ground	16436.99	Launch	9400	460	0.04	8.03	14989.35	599.57	LEO	1447.64	0.00	0.00	1447.64	3 stage launcher
Low Earth Orbit	1447.64	TMI(incl venus flyby)	4300	460	0.15	2.59	1046.34	156.95	Mars Transfer	401.29	0.00	0.00	401.29	2 EDS stage cryo
Mars Transfer	401.29	Capture	4530	460	0.15	2.73	299.10	44.86	Marc C3=0	102.19	0.00	0.00	102.19	1 Capture/MOI stage
Mars C3 = 0	102.19	MOI				1.00	0.00	0.00	LMO	102.19	4.68	0.03	97.54	
LMO	97.54	-				1.00	0.00	0.00	Marc C3=0	97.54	0.00	0.00	97.54	1 TEI return stage
Mars C3 = 0	97.54	TEI	2170	460	0.15	1.62	43.81	6.57	Earth Transfer	53.73	0.00	0.00	53.73	
Earth Transfer	53.73	EOI				1.00	0.00	0.00	LEO	53.73	43.73	0.00	10.00	
LEO	10.00	Earth entry	12000			1.00	0.00	0.00	Ground	10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	bring back 10t "orion vechicle"
		Tot dV	**20400**											

CREW ROCKET (EXCL TEI STAGE)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre		Reconfiguration (before next			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
Ground	9395.12	Launch	9400	460	0.04	8.03	8567.67	342.71	LEO	827.45	0.00	0.00	827.45	3 stage launcher
Low Earth Orbit	827.45	TMI(incl venus flyby)	4300	460	0.15	2.59	598.07	89.71	Mars Transfer	229.37	0.00	0.00	229.37	2 EDS stage cryo
Mars Transfer	229.37	Capture	4530	460	0.15	2.73	170.96	25.64	Marc C3=0	58.41	0.00	0.00	58.41	1 Capture/MOI stage
Mars C3 = 0	58.41	MOI				1.00	0.00	0.00	LMO	58.41	0.00	0.00	58.41	
	58.41					1.00	0.00	0.00		58.41	0.00	0.00	58.41	
	58.41					1.00	0.00	0.00		58.41	0.00	0.00	58.41	
	58.41					1.00	0.00	0.00		58.41	0.00	0.00	58.41	
	58.41					1.00	0.00	0.00		58.41	0.00	0.00	58.41	
	58.41					1.00	0.00	0.00		58.41	0.00	0.00	58.41	
	58.41					1.00	0.00	0.00		58.41	0.00	0.00	58.41	transhab outbound mass + "orion"
		Tot dV	**18230**											

CREW ROCKET NUCLEAR

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre		Reconfiguration (before next			Comment
Initial Orbit/Trajectory	Initial Mass (tons)	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
Ground	2434.39	Launch	9400	460	0.04	8.03	2219.99	88.80	LEO	214.40	0.00	0.00	214.40	3 stage launcher
Low Earth Orbit	214.40	TMI(incl venus flyby)	4300	850		1.67	86.38	0.00	Mars Transfer	128.02	0.00	0.00	128.02	1 nuclear stage
Mars Transfer	128.02	Capture	4530	850		1.72	53.66	0.00	Marc C3=0	74.36	0.00	0.00	74.36	
Mars C3 = 0	74.36	MOI				1.00	0.00	0.00	LMO	74.36	4.68	0.03	69.70	
LMO	69.70	-				1.00	0.00	0.00	Marc C3=0	69.70	0.00	0.00	69.70	
Mars C3 = 0	69.70	TEI	2170	850		1.30	15.97	0.00	Earth Transfer	53.73	0.00	0.00	53.73	
Earth Transfer	53.73	EOI				1.00	0.00	0.00	LEO	53.73	43.73	0.00	10.00	
LEO	10.00	Earth entry	12000			1.00	0.00	0.00	Ground	10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	bring back 10t orion

Appendix C: LEO Reboost modules

REBOOST MODULE FOR : CARGO ROCKET A: ASC/DESC VEHICLE (INCL SURF. PWR/ISRU)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	253.49	Launch	160	340	0.25	1.05	15.83	3.96	Launch	237.66			237.66
	237.66					1.00	0.00	0.00		237.66			237.66

mass to keep in LEO

REBOOST MODULE FOR : CARGO ROCKET B: SURFACE HABITAT (EXCL PWR/ISRU)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	267.48	Launch	160	340	0.25	1.05	16.70	4.18	Launch	250.77			250.77
	250.77					1.00	0.00	0.00		250.77			250.77

mass to keep in LEO

REBOOST MODULE FOR : CARGO ROCKET B2: NOT USED!

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	0.00	Launch	160	340	0.25	1.05	0.00	0.00	Launch	0.00			0.00
	0.00					1.00	0.00	0.00		0.00			0.00

mass to keep in LEO

REBOOST MODULE FOR : CARGO ROCKET C: TEI + CREW DESCENT

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	291.29	Launch	160	340	0.25	1.05	18.19	4.55	Launch	273.10	0.00	0.00	273.10
	273.10		0	0	0.00	1.00	0.00	0.00		273.10	0.00	0.00	273.10

mass to keep in LEO

REBOOST MODULE FOR : CREW ROCKET (INCL TEI STAGE)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	1544.06	Launch	160	340	0.25	1.05	96.43	24.11	Launch	1447.64	0.00	0.00	1447.64
	1447.64		0	0	0.00	1.00	0.00	0.00		1447.64	0.00	0.00	1447.64

mass to keep in LEO

REBOOST MODULE FOR : CREW ROCKET (EXCL TEI STAGE)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	882.56	Launch	160	340	0.25	1.05	55.12	13.78	Launch	827.45	0.00	0.00	827.45
	827.45		0	0	0.00	1.00	0.00	0.00		827.45	0.00	0.00	827.45

mass to keep in LEO

REBOOST MODULE FOR : CREW ROCKET NUCLEAR

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	228.68	Launch	160	340	0.25	1.05	14.28	3.57	Launch	214.40	0.00	0.00	214.40
	214.40		0	0	0.00	1.00	0.00	0.00		214.40	0.00	0.00	214.40

mass to keep in LEO

REBOOST MODULE FOR : CARGO ROCKET C2: CREW DESCENT only (valid for "incl return")

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)		
Initial state	Initial Mass (tons)	Orbit/trjector	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)
Ground	156.83	Launch	160	340	0.25	1.05	9.79	2.45	Launch	147.03	0.00	0.00	147.03
	147.03		0	0	0.00	1.00	0.00	0.00		147.03	0.00	0.00	147.03

mass to keep in LEO

Appendix C: Surface payloads dV calculations

Descent / Ascent vehicle (INCLUDING ISRU and PWR unit)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (ton	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
APPROCACH	77.68	Aerocapture				1.00	0.00	0.00	MLO	77.68	25.00	0.00	52.68	Drop Aeroshell
MLO	52.68	Aerobreaking				1.00	0.00	0.00	ATMO	52.68	20.30	0.00	32.38	Drop HIAD+RCS propellant + avionics
ATMO	32.38	Retro proulsion	700	340	0.50	1.23	12.26	6.13	SURFACE	20.12	0.00	0.00	20.12	
	20.12					1.00	0.00	0.00		20.12	0.00	0.35	20.46	Add crew + material
	20.46					1.00	0.00	0.00		20.46	1.88	0.00	18.58	Leave brought H2
	18.58					1.00	0.00	0.00		18.58	10.00	0.00	8.58	Leave ISRU+ PWR unit
	8.58					1.00	0.00	0.00		8.58	0.00	23.25	31.83	Add manufactured O2 and CH4
	31.83					1.00	0.00	0.00		31.83	0.00	0.00	31.83	
SURFACE	31.83	Ascent	4500	350	0.10	3.71	25.83	2.58	MLO	6.00	0.00	0.00	6.00	
	6.00					1.00	0.00	0.00		6.00	0.00	0.00	6.00	ASC module dry mass

Descent / Ascent vehicle (excl ISRU, all propellant brought) NOT USED!

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (ton	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
APPROACH	95.97	Aerocapture				1.00	0.00	0.00	MLO	95.97	25.00	0.00	70.97	Drop Aeroshell
MLO	70.97	Aerobreaking				1.00	0.00	0.00	ATMO	70.97	20.30	0.00	50.67	Drop HIAD+RCS propellant + avionics
ATMO	50.67	Retro proulsion	700	340	0.50	1.23	19.18	9.59	SURFACE	31.48	0.00	0.00	31.48	
	31.48					1.00	0.00	0.00		31.48	0.00	0.35	31.83	Add crew + material
	31.83					1.00	0.00	0.00		31.83	0.00	0.00	31.83	
	31.83					1.00	0.00	0.00		31.83	0.00	0.00	31.83	
	31.83					1.00	0.00	0.00		31.83	0.00	0.00	31.83	
	31.83					1.00	0.00	0.00		31.83	0.00	0.00	31.83	
SURFACE	31.83	Ascent	4500	350	0.10	3.71	25.83	2.58	MLO	6.00	0.00	0.00	6.00	
	6.00					1.00	0.00	0.00		6.00	0.00	0.00	6.00	ASC module dry mass

Needed propellant	23.247	LOX/CH4 for ascent
CH4 produce	5.5	mass of methane needed for ascent propulsion
O2 produce	17.7	mass of oxygen needed for ascent propulsion
LH2 to bring	1.4	mass of LH2 needed (brought)
O2 spare	4.4	mass of produced spare oxygen, can be used for life support
LH2 water	0.5	mass of LH2 needed for life support (brought)

Surface habitat (incl all consumbles and excl PWR)

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (ton	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
APPROACH	87.14	Aerocapture	0	0		1.00	0.00	0.00	MLO	87.14	25.00	0.00	62.14	Drop Aeroshell
MLO	62.14	Aerobreaking	0	0		1.00	0.00	0.00	ATMO	62.14	20.30	0.00	41.84	Drop HIAD+RCS propellant + avionics
ATMO	41.84	Retro proulsion	700	340	0.50	1.23	15.84	7.92	SURFACE	26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	
	26.00					1.00	0.00	0.00		26.00	0.00	0.00	26.00	Surf HAB

Crew entry vehicle

Initial state (Before maneuvre)		Delta V Maneouvre							Final State (after maneuvre)		Reconfiguration (before next maneuvre)			Comment
Initial Orbit/Trajectory	Initial Mass (ton	Orbit/trjectory maneuvre	Reactive Delta V (m/s)	Isp (s)	Structure Factor	Propellant mass multiplier	Propulsion stage wet mass	Propulsion stage dry mass	Final Orbit/trajectory	Final mass (tons)	Discarded payload (tons)	Added payload (tons)	Resulting Mass (tons)	
APPROACH	26.09	Aerocapture	0	0		1.00	0.00	0.00	MLO	26.09	0.00	0.00	26.09	
MLO	26.09	Aerobreaking	0	0		1.00	0.00	0.00	ATMO	26.09	10.00	0.00	16.09	Drop HIAD+RCS propellant + avionics
ATMO	16.09	Retro proulsion	700	340	0.50	1.23	6.09	3.05	SURFACE	10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	
	10.00					1.00	0.00	0.00		10.00	0.00	0.00	10.00	Desc module dry mass

Appendix C: Total LEO mass summary

	Mass to LEO N=1	Input from dV bugdget sheets: TMI mass	Inj. Isp	Inj. epsi	Inj. dV	N	Payload ratio	Mass to LEO (N stage)
Crew rockets								
CREW ROCKET (INCL TEI STAGE)	1448	401.3	460	0.15	4300	2	0.31	1307
CREW ROCKET (EXCL TEI STAGE)	827	229.4	460	0.15	4300	2	0.31	747
CREW ROCKET NUCLEAR	214							214
Cargo rockets (applicable for all)								
CARGO ROCKET A: ASC/DESC VEHICLE (INCL SURF. PWR/ISRU)	238	87.1	460	0.15	3650	2	0.37	235
CARGO ROCKET B: SURFACE HABITAT (EXCL PWR/ISRU)	251	87.1	460	0.15	3650	2	0.37	235
CARGO ROCKET B2: NOT USED!	0	0.0	460	0.15	3650	2	0.37	0
CARGO ROCKET C: TEI + CREW DESCENT	273	95	460	0.15	3650	2	0.37	256
CARGO ROCKET C2: CREW DESCENT only (valid for "incl return")	147	51	460	0.15	3650	2	0.37	138
Reboost LEO modules								
CARGO ROCKET A: ASC/DESC VEHICLE (INCL SURF. PWR/ISRU)	16	-	-	-	-	-	-	16
CARGO ROCKET B: SURFACE HABITAT (EXCL PWR/ISRU)	17	-	-	-	-	-	-	17
CARGO ROCKET B2: NOT USED!	0	-	-	-	-	-	-	0
CARGO ROCKET C: TEI + CREW DESCENT	18	-	-	-	-	-	-	18
CARGO ROCKET C2: CREW DESCENT only (valid for "incl return")	10							10
CREW ROCKET (INCL TEI STAGE)	96							96
CREW ROCKET (EXCL TEI STAGE)	55							55
CREW ROCKET NUCLEAR	14							14
TOTAL MASS TO LEO [tonnes]	**2222**	**Total (incl return)**						**2054**
	1695	**Total(separate return)**						**1580**
	897	**Total(incl return, NUCLEAR)**						**870**
	INITAL							**OPTIMAL N**

Appendix: TMI/MOI injection stage mass breakdown

			Stage 1				Stage 2			
Mass Breakdown	ROCKET	Propellant fraction	Propellant mass	Construction mass	Stage mass	M02	Propellant mass	Construction mass	Stage mass	TOT
	CREW ROCKET (INCL TEI STAGE)	0.38	495	87	583	724	274	48	323	**1307**
	CREW ROCKET (EXCL TEI STAGE)	0.38	283	50	333	414	157	28	185	**747**
	CREW ROCKET NUCLEAR							0		
								0		
								0		
	CARGO ROCKET A: ASC/DESC VEHICLE (INCL SURF. PWR/ISRU)	0.3	78	14	92	143	48	8	56	**235**
	CARGO ROCKET B: SURFACE HABITAT (EXCL PWR/ISRU)	0.3	78	14	92	143	48	8	56	**235**
	CARGO ROCKET B2: NOT USED!	0.3	0	0	0	0	0	0	0	**0**
	CARGO ROCKET C: TEI + CREW DESCENT	0.3	85	15	100	156	52	9	61	**256**