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Mission design options for human Mars missions

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Abstract

Background: Interplanetary trajectory selection will be a significant driver for the design of human Mars missions, impacting propulsive, habitation, and atmospheric entry system requirements. Conjunction-class interplanetary trajectories are the leading candidate for human Mars missions, due to their short in-space durations, long surface stays, and lower propulsion requirements, in contrast to the long in-space durations, short surface stays, and high propulsion requirements characteristic of opposition-class missions. Within conjunction-class mission trajectories, considerations for providing an abort option to return the crew to Earth without stopping at Mars are also worthwhile.

Approach: This paper presents Earth-Mars and Mars-Earth conjunction trajectories across a series of mission opportunities and transfer times in order to provide human Mars mission designers with an option space of possible crew and cargo transfer trajectories. For the specific case of crew transfer from Earth to Mars, the potential for aborting the mission without capture into Mars orbit is also examined. Two additional sub-classes of trajectories are thus presented: free return trajectories, where the outbound trajectory would return the crew to Earth after a fixed period of time without major propulsive maneuvers; and propulsive-abort trajectories, where the propulsive capability of the transfer vehicle is used to modify the trajectory during a Mars swing-by. Beyond propulsive requirements, trajectory selection also has a significant impact on the entry velocity and therefore the aeroassist system requirements, which are also examined in this paper.

Results: Conjunction-class interplanetary trajectory data across a range of mission and architecture options is provided for use by designers of human Mars missions. Our investigation suggests potential constraints for entry velocities at Earth and Mars due to aeroassist considerations and describes feasible trajectories within these constraints. Based upon Mars entry velocity, the 2-year period free return abort trajectory is found to be less desirable for many mission opportunities than previously considered.

Introduction

This paper provides an overview of interplanetary trajectory options for human exploration of Mars, taking into account propulsive, crew support, and aeroassist requirements, with the intent of providing data useful for planning human missions to Mars. In order to provide a full range of information for mission planners, the trajectory options are examined across the 8 mission opportunities between 2020 and 2037 (covering the complete planetary cycle plus one opportunity), inclusively. Since the complete period of the relative positions of the Earth and Mars in their orbits is approximately 15 years, the data can thus be employed for conceptual design of Mars missions for any mission opportunity. The primary metrics computed for the trajectory options include the propulsive delta-v required to escape the departure planet and reach the desired trajectory, the in-space transfer duration, and the entry velocity at the arrival planet. Each of these parameters is important for the design of spacecraft in terms of propulsion, crew habitation, and aeroassist systems. In addition, the choice of trajectories dictates the available stay time at Mars, which is also provided.

There have been numerous studies associated with interplanetary trajectory selection for human Mars exploration (Strepe et al. 1993; Strepe et al. 1993; Patel et al.

1998; George and Kos 1998; Landau and Longuski 2006; Landau and Longuski 2006). Average performance of conjunction-class, free return and cycler trajectories for high-thrust systems across a range of mission opportunities is provided in (Landau and Longuski 2006). Low-thrust transfer performance is summarized in (Landau and Longuski 2006). Departure energies and arrival energies for conjunction-class and free return trajectories for mission opportunities from 2009 to 2024, along with an assessment of the impact of thrust to weight on finite burn losses, are provided in (George and Kos 1998). Building on these past studies, this investigation couples entry system considerations as well as potential free return and propulsive abort scenarios with interplanetary trajectory selection, providing a complete set of high-thrust mission design information. Data specific to each mission opportunity across the complete planetary cycle (in the 2020-2037 timeframe) are provided.

Interplanetary trajectory modeling

Two major types of human Mars missions have been proposed based upon their interplanetary trajectories and associated Mars stay times, known as conjunction-class (or long-stay) and opposition-class (or short-stay) missions (Walberg 1993). Conjunction-class missions are characterized by long stay times on Mars (order of 400 to 600 days), short in-space durations (approximately one year total for the Earth-Mars and Mars-Earth legs), and relatively small propulsive requirements. Opposition-class missions have significantly shorter Mars stay times (order of 30 to 90 days), long in-space durations (approximately 1.5 years total for the Earth-Mars and Mars-Earth legs), and relatively large propulsive requirements. In addition, opposition-class missions typically require the complexity of a Venus swing-by as part of either the outbound or return transfer segment. As conjunction-class trajectories offer increased benefit (an order of magnitude greater Mars stay time), at lower cost (significantly lower propulsive and radiation shielding requirements) and lower risk (due to the decreased time in zero gravity), they serve as the focus for the trajectories examined in this investigation.

The conjunction trajectories explored in this paper include both Earth-Mars and Mars-Earth legs in which the trip time is constrained to specific durations between 120 and 270 days, in 10 day increments. This provides an overview of savings in trip time which can be gained through increases in propulsive delta-v, and can be useful for planning both crew and cargo transfers. An example of such a trajectory is provided in Figure 1. In addition, for the crew transfer from Earth to Mars, the option to provide a free return abort capability may offer benefits. In such a transfer, the spacecraft is placed on a trajectory from the Earth to Mars which also returns to Earth at a specified point in the future, without the need for any major additional action by the spacecraft (in practical usage, mid-course corrections would likely be necessary). Typically such trajectories employ a resonance in the period between the transfer trajectory and the orbit of the Earth about the Sun (Patel et al. 1998; Landau and Longuski 2006). In this paper, free return trajectories with a 2 year period or 2:1 Earth to spacecraft revolution about the Sun resonance, and with a 1.5 year period, or 3:2 Earth to spacecraft revolution about the Sun resonance, are investigated. The 2:1 resonant trajectories are called 2-year free return trajectories, as the period of time from Earth departure to Earth return will be 2 years if the abort option is taken. Similarly, the 3:2 resonant trajectories are called 3-year free return trajectories as the time from Earth departure to return is 3 years in the case of an abort. This paper also explores an option for a propulsive abort trajectory, in which if the crew opts to forgo capture at Mars, an abort burn is performed which places the spacecraft on an interplanetary trajectory that will return the crew to Earth. For the purposes of this paper, such trajectories have an Earth departure to Earth return time (in the case of an abort) of slightly over 2 years. Both the free return and the propulsive abort trajectories offer the prospect for the crew to return home without capture at Mars in the case of failure of some system already at Mars or otherwise required for the safe return of the crew to Earth. Both abort trajectory options clearly place a requirement on the outbound spacecraft of being able to support the crew during the lengthy return to Earth, and in the case of the propulsive abort trajectories add an additional critical event in the case of an abort. In addition, the consequences of two years or greater exposure to the space environment (radiation and zero gravity effects) must be accommodated, possibly with significant detrimental impacts to the overall feasibility of the overall mission. Whether such abort trajectories are employed is a decision to be analyzed by mission planners and eventually determined by program decision makers. The information in this paper is intended to serve as a reference in assisting with such decisions.

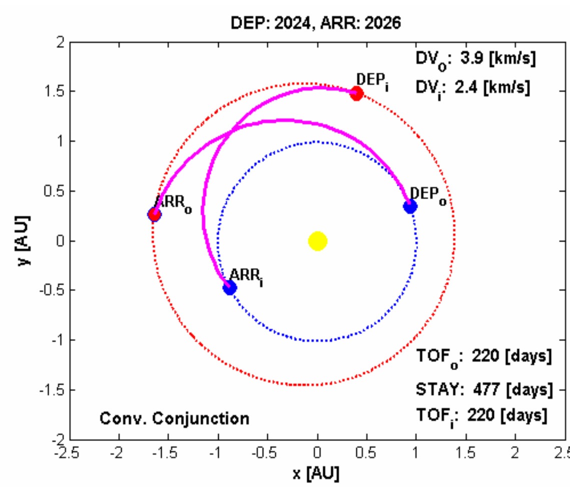

Figure 1. Example outbound (Earth-Mars) and inbound (Mars-Earth) conjunction trajectories relative to the orbits of the Earth and Mars for an Earth departure in 2024 and an Earth return in 2026. Both trajectories are conjunction transfers with transit durations of 220 days. The *x* and *y* axes are in Astronomical Units (AU).

The interplanetary trajectories examined in this paper make use of the J2000 ephemerides for determining planetary positions (Vallado 1997). The delta-v's are assumed to be impulsive in nature, and are computed based on departure from a 500 km altitude circular orbit (about either the Earth or Mars). The entry velocities are calculated based upon an entry interface at 125 km altitude. A patched conic method with the assumption that the sphere of influence radii of planets are negligibly small relative to the interplanetary path lengths traveled is used for the trajectory calculation (Battin 1999). All trajectories (conventional inbound / outbound, free return, and propulsive abort) were optimized for minimum departure energy (and thus minimum departure delta-V), since this parameter has a direct impact on mission mass requirements. In more detailed trajectory design, attempts to also optimize other parameters such as entry velocity would likely be worthwhile. Sequential quadratic programming (SQP) is used for the optimization. All interplanetary trajectory code was written and executed in MATLAB.

Aeroassist modeling

Aeroassist maneuvers will likely be required for human Mars exploration to decrease vehicle mass. Aeroassist maneuvers performed at both Mars arrival and Earth return virtually eliminate the propellant required to slow mission assets down in an all-propulsive architecture. Previous analyses estimate that the use of aeroassist maneuvers may offer significant reductions in the initial mass of a crewed Mars vehicle, potentially by greater than 50% (Draper Laboratory 2005, Braun et al. 1990). This reduction in vehicle mass implies a large derived reduction in launch cost and architecture complexity which may be mission enabling.

Aeroassist trajectories use a planetary atmosphere to alter the vehicle velocity vector non-propulsively. Aeroassist trajectories dissipate vehicle energy through heat transfer, creating an extreme thermal environment that a successful entry vehicle must survive. The peak heat rate, integrated heat load, and deceleration environment experienced by an entry vehicle are functions of vehicle mass, aerodynamics, geometry, and the vehicle state at atmospheric interface. High-speed steep entries incur the highest heat rates and largest peak decelerations, while high-speed shallow entries incur the largest integrated heat loads. In all cases, the aerothermal environment becomes more extreme with increasing entry velocity magnitude. However, an entry vehicle may utilize aerodynamic lift to minimize the severity of the aerothermal and deceleration environments. Two types of aeroassist maneuvers are relevant to human Mars missions: entry and aerocapture.

Entry maneuvers use a planetary atmosphere to decelerate from high velocity and descend to the planet's surface (Figure 2). Entry may occur from inbound hyperbolic trajectories (direct entry) or from orbit. Entry from orbit requires a small propulsive maneuver to lower the periapsis of the vehicle orbit into the atmosphere. Aerocapture uses a planetary atmosphere to decelerate from a hyperbolic trajectory to a bound orbit during a single atmospheric pass. Aerodynamic drag incurred during the atmospheric pass reduces the vehicle's energy to a desired value. After the vehicle exits the atmosphere with the correct energy, a propulsive maneuver is required to raise the periapsis of the vehicle's orbit out of the atmosphere. In this investigation, aerobraking (as performed by the series of recent robotic Mars orbiters including the Mars Reconnaissance Orbiter) is not considered a viable architectural option due to its required 3-6 month duration.

Entry trajectory analysis of aeroassist maneuvers at Earth and Mars was performed with the three-degree-of-freedom version of the Program to Optimize Simulated Trajectories (POST) (Brauer et al. 1977). The linear feedback control option was used to fly deceleration-constrained entry trajectories by utilizing lift control through bank angle modulation. Historic aerodynamic data was used when available for this analysis. For non-heritage vehicles, aerodynamic analysis was performed using the tangent cone method option in the Aerodynamic Preliminary Analysis System (APAS) (Sova and Divan 1991). The fidelity of the aerodynamic data in the hypersonic regime generated by APAS is on the order of the fidelity of the entry trajectory analyses performed. Aeroheating calculations were performed with two approximate stagnation point heating methods. Stagnation point convective heating was calculated using Chapman's equation (Chapman 1958) and stagnation

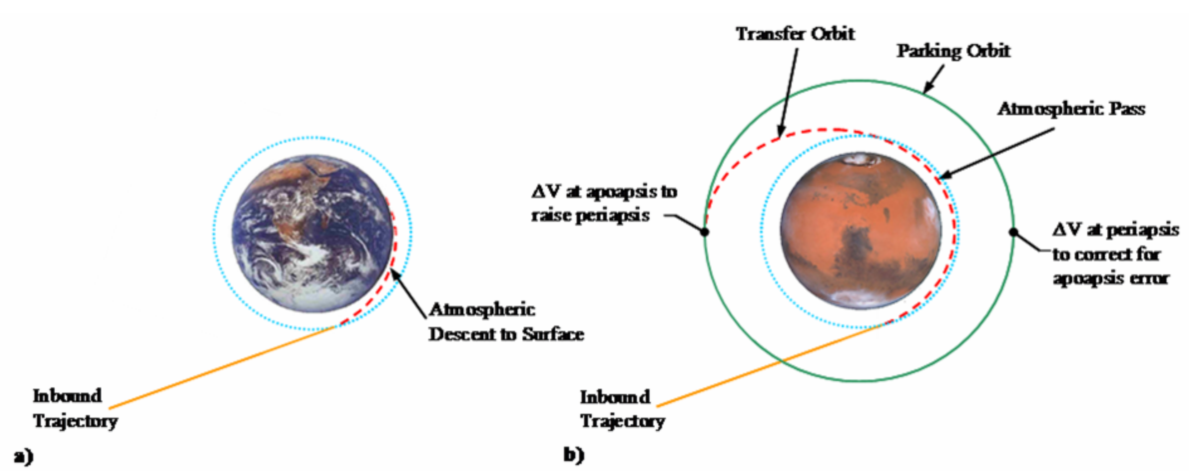

Figure 2. Human Mars mission aeroassist trajectories: (a) direct entry and (b) aerocapture.

point radiative heating was estimated using the Tauber-Sutton approximation (Tauber and Sutton 1991). The results for both calculations were summed to find the total stagnation point heat rate and integrated heat load as a function of time for each trajectory.

Conjunction trajectory options for Earth-Mars and Mars-Earth transfers

In this section, we explore conjunction trajectory options for Earth-Mars and Mars-Earth transfers with transfer durations set in 10 day increments from 120 to 270 days. It is worth noting that a number of previous studies have suggested 180 days as a conjunction transfer time (Zubrin et al. 1991; Hoffman and Kaplan 1997; Draper Laboratory 2005), which is within this range. 270 days represents the maximum Type I (less than 180 degree transfer about the Sun) transfer duration. The values of departure delta-v and arrival entry velocity are presented for each of the trajectories. In addition, the Mars stay times associated with the outbound (Earth-Mars) transits are presented relative to a fixed duration inbound (Mars-Earth) return transfer, followed by the sensitivity of Mars stay time to the inbound transfer duration.

Earth-Mars outbound trajectories

Figure 3 shows the Earth departure delta-v for conjunction transfers with durations of 120 to 270 days. These trajectories are suitable for cargo flights as well as crew flights (if an abort option is not selected) to Mars. In observing the delta-v graph, one can see that in any particular departure opportunity a trade exists between delta-v and flight time, although the minimum delta-v only rarely corresponds with the maximum transfer time. It is useful to consider the delta-v required to be able to make an Earth-Mars transfer in every opportunity. This delta-v could be used to size a system to operate across all opportunities, supporting a continuous sequence of Mars missions. In this case the delta-v required if transit time is not an issue is just under 4 km/s (3.98 km/s in our analysis). This delta-v is sufficient to enable a transfer of 200 days or less in every opportunity. Decreasing the transit time to 180 days or less in every opportunity would require only a minor increase in delta-v (to 4.08 km/s in our analysis); whereas, a decrease to a transit time of 160 days or less in every opportunity requires an Earth departure delta-v of 4.27 km/s.

Figure 4 shows the Mars arrival entry velocity for the same set of trajectories. In contrast to the departure delta-v's, the arrival entry velocities show a stronger correlation between lower entry velocities and longer transit times. The minimum required entry velocity capability to support missions across all opportunities over the range of transfer times investigated is approximately 6.09 km/s, driven by the 270 day transfer in the 2031 mission opportunity. A system designed around a 6.09 km/s entry velocity would support significantly faster transfers in certain opportunities (for example a 160 day transfer in 2035). To enable minimized Earth departure energy transfers of 180 days or less in all opportunities, a Mars entry velocity capability of 8.57 km/s is required (significantly higher than that experienced to date in the robotic Mars exploration program).

In viewing the Mars entry velocity and Earth departure delta-v information together, it appears that Mars entry velocity may be a stronger constraint on transit times than Earth departure delta-v. Given that cargo missions are likely to be insensitive to transit times, it appears that developing cargo transportation systems such that they can provide an Earth departure delta-v of approximately 4 km/s and support Mars entry velocities up to approximately 6.1 km/s (with appropriate design margin required in each case) would be a reasonable architectural approach. For crew missions utilizing 180-day or less minimized departure energy conjunction trajectories, development of a technology plan to increase the Mars entry velocity capability (to 8.5-9.0 km/s) is required to enable faster Earth-Mars transit times in the more challenging opportunities.

Mars-Earth inbound trajectories

Figure 5 shows the Mars departure (Trans-Earth Injection) delta-v for each Earth return opportunity from 2020 to 2037 for transfers with return transit durations of 120 to 270 days, based upon return from a low Mars orbit. It should be noted that due to the nature of the arrival and departure dates, a crew transiting to Mars in one Earth departure opportunity would first be able to return to Earth in the next Earth return opportunity. For example, a crew transiting from Earth to Mars in the 2024 Earth departure opportunity would be able to return in the 2026 Earth return opportunity (or any subsequent one, in the case of longer duration stays). In analyzing the data, again a trade is found to exist in any given opportunity between delta-v and transit duration, with the minimum delta-v coming about through increased trip times in some but not all opportunities. If any transit duration (within the range analyzed) is allowed, an Earth return delta-v of 2.56 km/s is required, which is driven by the 2037 opportunity for a 270 day transit. Increasing the delta-v slightly to 2.75 km/s reduces the maximum transit time across all opportunities to 200 days, a significant reduction, and would enable faster transfers in a number of opportunities. Further increasing the delta-v to 3.03 km/s allows for return transit times of 180 days or less in all opportunities.

The Earth entry velocities for the same set of Mars-Earth transfers are shown in Figure 6. Here again a stronger correlation is found between decreased transit time and increased Earth entry velocity, particularly in the opportunities featuring higher entry velocities. While entry velocities are only plotted up to 15 km/s, the peaks exceed 18 km/s in some opportunities (for 120 day transfers in 2026 and 2028). An Earth entry velocity capability of 12.41 km/s is sufficient to enable Earth return in any opportunity if transit duration is not a constraint (limited by the 260 day transfer in 2030). This is in family with the 12.8 km/s Earth entry successfully completed by the Stardust robotic sample return capsule in January 2006. Outside of the 2028 and 2030 opportunities, an entry velocity capability of 11.8 km/s would be sufficient to enable Earth return in any opportunity

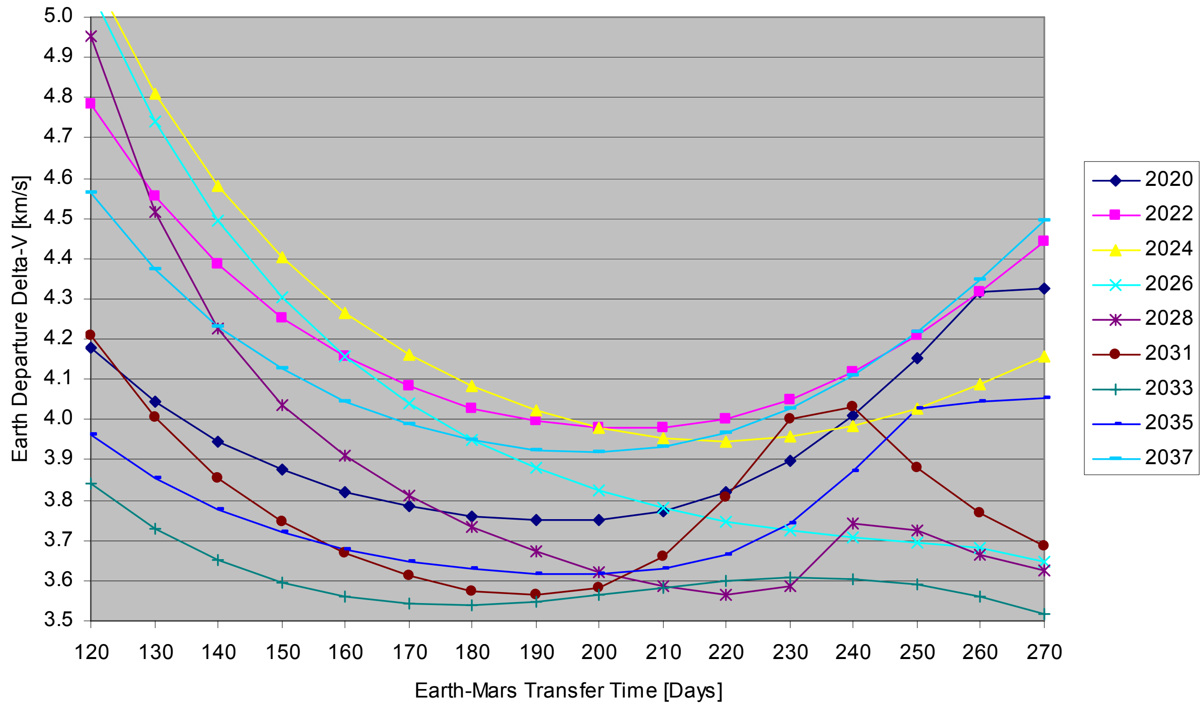

Figure 3. Low Earth orbit departure Delta-V for Earth-Mars conjunction trajectories with transit times from 120 to 270 days, over Earth departure opportunities from 2020 to 2037.

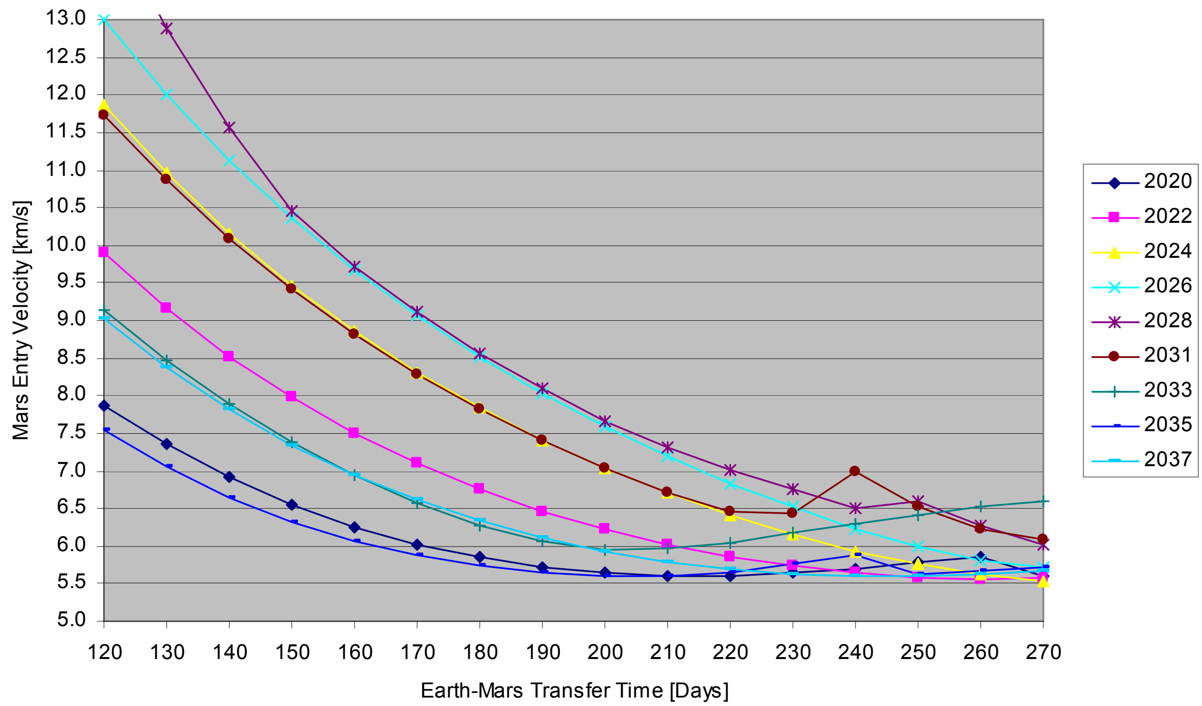

Figure 4. Mars arrival entry velocities for Earth-Mars conjunction trajectories with transit times from 120 to 270 days presented for Earth departure opportunities from 2020 to 2037.

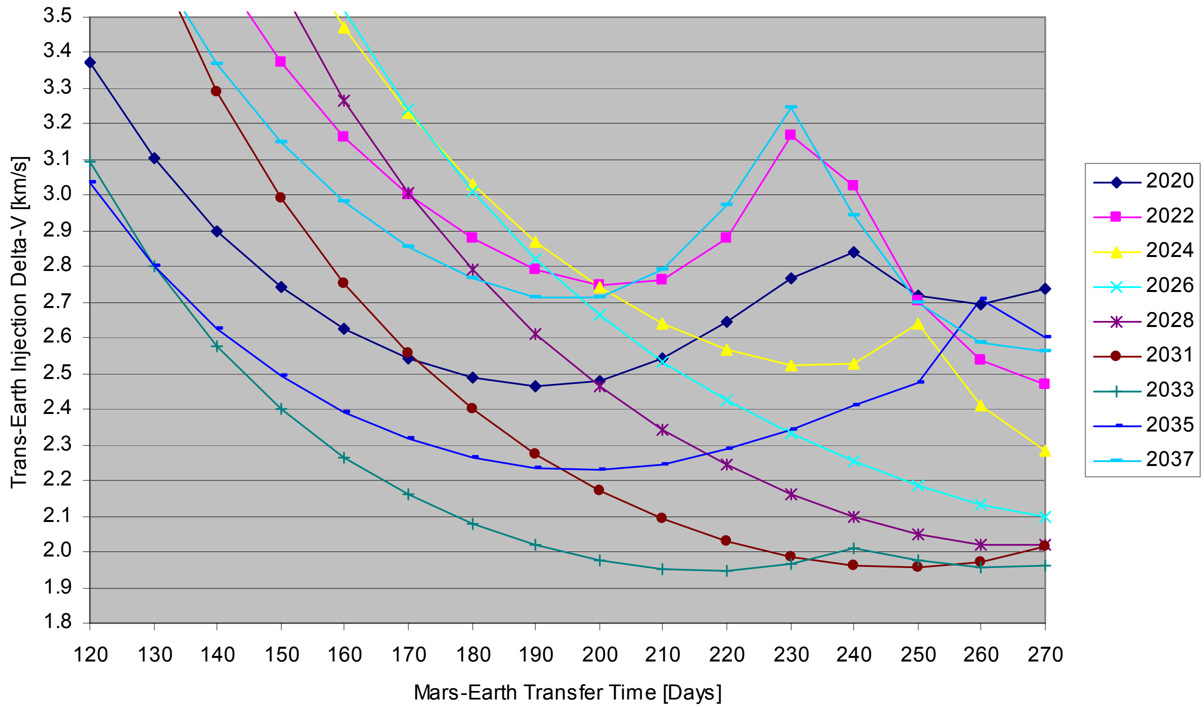

Figure 5. Low Mars orbit departure Delta-V for Mars-Earth conjunction trajectories with transit times from 120 to 270 days presented for Earth return (Mars departure) opportunities from 2020 to 2037.

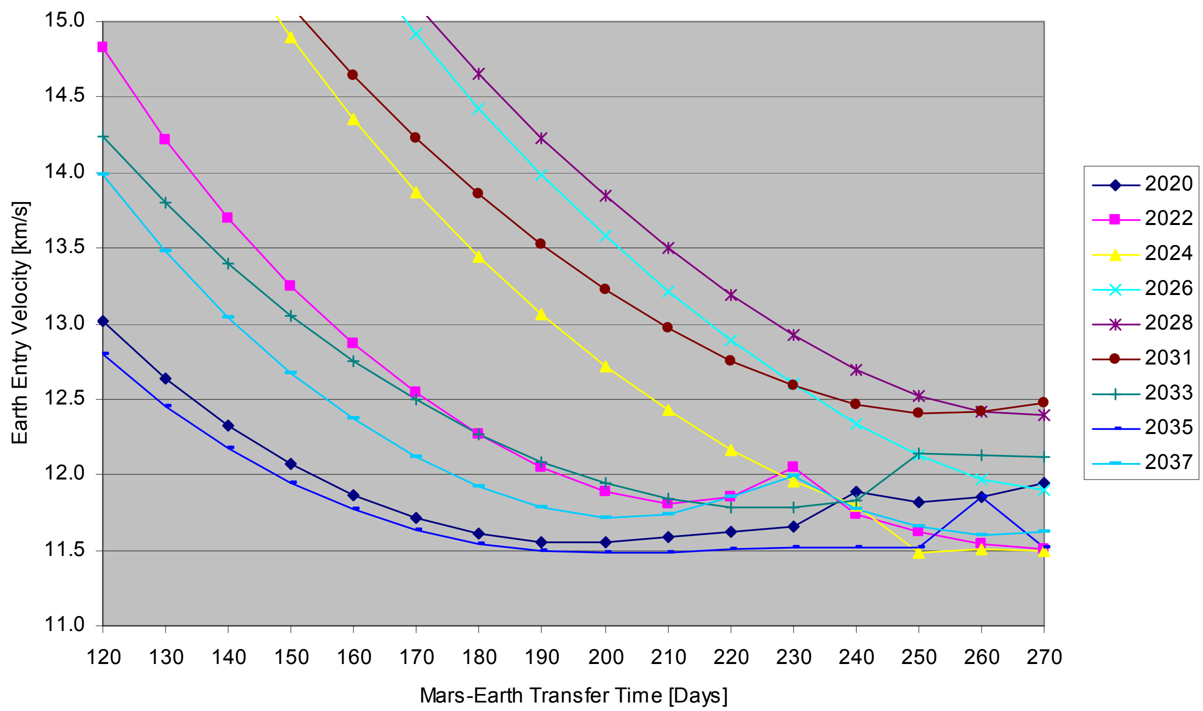

Figure 6. Earth arrival entry velocities for Mars-Earth conjunction trajectories with transit times from 120 to 270 days, over Earth return (Mars departure) opportunities from 2020 to 2037.

if transit duration constraint is not a constraint. For crew transfer, decreasing Mars-Earth transit time is highly desirable. In order to return in 180 days across all opportunities, an Earth entry velocity capability of 14.65 km/s is required (driven by the 2028 opportunity). If an Earth entry limit of 14 km/s were imposed, the 2028 opportunity would be constrained to a 200 day return transfer and 2026 would be constrained to a 190 day return, with the remainder of opportunities able to support 180 days or less. Further reducing the Earth entry velocity capability to 13 km/s would require a 230 day return in 2028, a 220 day return in 2026, a 210 day return in 2030, a 190 day return in 2024, and enable returns of 180 days or less in the remaining opportunities. Clearly a strong trade exists between the entry velocity capability of the Earth entry vehicle (along with the technology development associated to achieve this) and the requirements associated with supporting the crew for somewhat longer durations.

Mars stay time associated with conjunction transfers

In addition to impacting the in-space duration, propulsive, and entry velocity requirements, the choice of Mars mission trajectories will also determine the stay time available at Mars. The stay time available will impact both the quantity of exploration that can be performed and the systems required to support the crew while at Mars. Figure 7 provides information on the Mars stay time resulting from the choice of outbound Earth-Mars trajectories. The values are provided relative to an Earth return transfer of 270 days in the immediately following return opportunity (i.e., the earliest return opportunity available). Figure 8 shows the change in stay time of various return transfers relative to the 270 day transfer presented in Figure 7. There can be significant variation in stay time both from opportunity to opportunity and within an opportunity based upon outbound transit time, with stay times ranging from just over 400 days to just over 600 days. The tendency is for faster Earth-Mars transit times to result in longer Mars stay times, although this trend is stronger in some opportunities than in others. On the other hand, decreasing Mars-Earth transit time can either increase or decrease Mars stay time, although the sensitivity of stay time to Mars-Earth transit time appears to be lower than to Earth-Mars transit time. While we believe it is unlikely that surface duration considerations will be a major driver in trajectory option selection (within the family of conjunction-class trajectories), including considerations of the increased benefit of longer surface durations in trajectory trades is worthwhile, in particular in terms of the change in stay time with varying Earth-Mars transit times. Since exposure to radiation and zero-g is significantly reduced when on the Mars relative to in-transit, this may be an important consideration.

Earth-Mars transfer trajectory abort options

The trajectories options discussed in the previous section could be used for either cargo or crew transfers to and from Mars. In the specific case of crew transfer to Mars, it may be desirable to provide an option for an abort without capture into Mars orbit. Whether and how the need would arise to conduct such an abort would depend greatly upon the design of the human Mars exploration architecture, including the degree to which other contingency options exist. Examples that could potentially lead to such an abort include the failure of prepositioned systems required to either support the crew at Mars or return them to Earth, or failure of systems on-board the outbound craft that are required in order to successfully capture into Mars orbit and/or possibly descend to the surface. It should be noted that the abort trajectory options presented here by no means provide a rapid return to Earth option – the total time spent in the aborting spacecraft can approach or even exceed the total duration of a nominal mission, and would feature a significant increase in exposure to radiation and zero gravity as compared to the nominal mission, possibly limiting the feasibility of such aborts. As such and again depending upon the Mars architecture employed, including the option for one of these aborts will have a significant impact upon the design of the outbound spacecraft. In addition, the abort option does not provide a mitigation for failure of crew support systems onboard the outbound spacecraft; in such a case it may be most advantageous to arrive at Mars as planned so that the crew can make use of other assets and resources present there (Zubrin et al. 1991; Hoffman and Kaplan 1997; Draper Laboratory 2005).

The abort trajectory options investigated in this paper include free return trajectories with orbital periods of 2 years and 1.5 years, and propulsive abort trajectories covering a range of impulsive abort delta-v's during a swing-by at Mars. Other abort options, including free return trajectories with other periods (such as 1 or 3 years) or combined swing-bys of Mars and Venus have been proposed (Okutsu and Longuski 2002); however, these are not included as they tend to have some combination of higher propulsive and entry velocity requirements, longer in-space transit times, infrequent mission opportunities, and higher radiation shielding requirements (in the case of abort trajectories involving Venus swing-bys).

The free return trajectories employ the periodicity inherent in the Earth's orbit around the Sun to return to the point in space where they departed from Earth at the same time that the Earth returns to that point in space. In the case of the 2-year free return investigated in this paper, the outbound spacecraft is on a 2 year period orbit, such that the spacecraft completes one orbit about the Sun (if arrival at Mars is aborted), while the Earth completes two revolutions about the Sun (see Figure 9). This free return represents the shortest Earth return time of any of the practical Earth-spacecraft resonant period free return trajectory options. Although a one year period orbit is theoretically possible, this requires the perihelion of the transfer orbit to be significantly inside the orbit of the Earth, resulting in a very large Earth departure delta-v requirement. For the three year free return option presented, the spacecraft is placed on a 1.5-year period heliocentric orbit for transfer to Mars, and in the

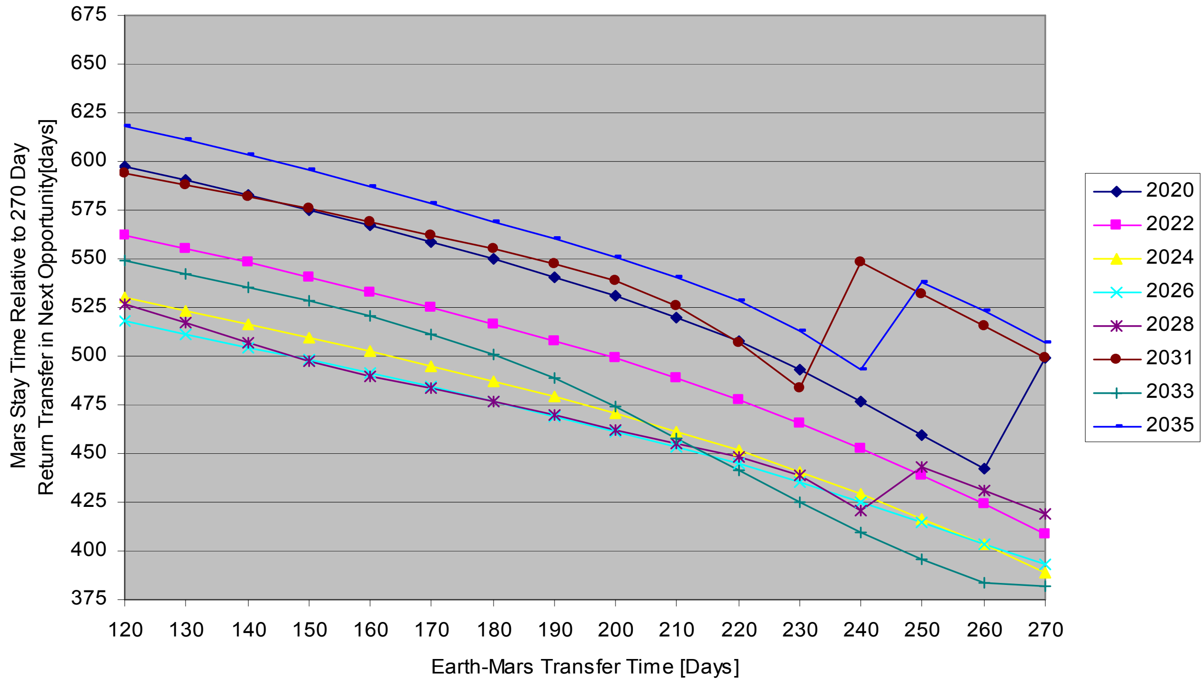

Figure 7. Mars stay times for Earth-Mars conjunction trajectories with transit times from 120 to 270 days presented for Earth departure opportunities from 2020 to 2035. The stay times are computed assuming a Mars-Earth return transfer of 270 days in the immediately following Earth return opportunity (e.g., the 2022 Earth return opportunity in the case of a 2020 Earth departure).

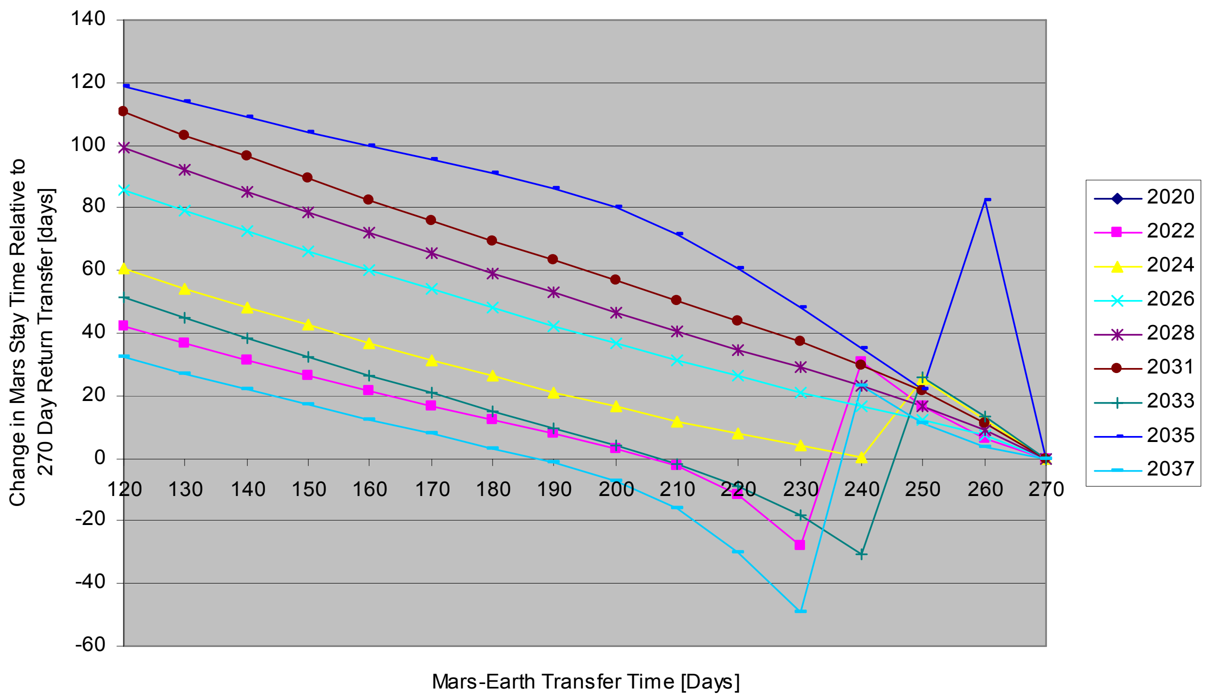

Figure 8. Change in Mars stay time for Mars-Earth conjunction trajectories with transit times from 120 to 260 days relative to a 270 day Mars-Earth transfer, presented for Earth return (Mars departure) opportunities from 2022 to 2037. Negative numbers represent a decrease in Mars stay time, positive numbers an increase.

case of the abort makes two complete orbits about the Sun while the Earth makes three revolutions. The lower energy of the transfer orbit makes this the lowest Earth departure delta-v option in a number of opportunities, although at the expense of increasing the transfer time to Mars and the time required to be spent in space in the case of an abort. While free return trajectory options with longer orbit periods exist (3 year period, 4 year period, etc.), they exhibit high Earth departure delta-v's due to their higher energy transfer orbits without offering any benefits in terms of time in-space during an abort.

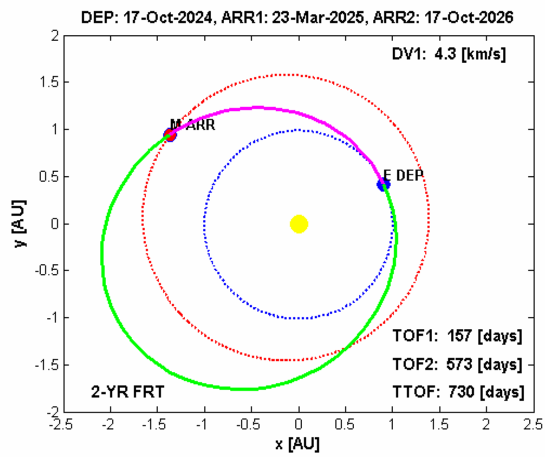

Figure 9. Example 2-year free return trajectory. The outbound spacecraft starts on an Earth-Mars transfer with a total orbital period of 2 years (magenta), and continues along that same trajectory after by-passing Mars in the case of an abort (green).

Although the 2-year free return trajectories are relatively more attractive in terms of in space duration, they suffer from high Mars entry velocities in a number of opportunities. As such, the potential to reduce these Mars entry velocities through decreasing the initial transfer orbit energy and then employing a propulsive maneuver at Mars (in case of an abort) to transition to a trajectory that will return the crew to Earth was investigated, with a total in-space time similar to that of a 2-year free return trajectory (see Figure 10). In addition to decreasing the Mars entry velocity, such propulsive abort maneuvers were found to offer the additional benefit of decreasing the Earth departure delta-v. In order to decrease the impact upon the outbound spacecraft of including this option, we suggest that propulsion systems already onboard the spacecraft for nominal mission maneuvers (such as Mars landing or Earth return propulsion), be employed to perform the abort. As such, we investigated a range of abort delta-v's from 600 m/s, representative of the low end of delta-v required of Mars landing systems (Draper Laboratory 2005), up to 2,700 m/s which appears to be towards the upper end of anticipated trans-Earth injection delta-v's assuming a circular departure orbit as discussed in the previous section (see Figure 5).

Figure 11 presents the delta-v required for departure from Earth for the free return and propulsive abort trajectory options investigated. The 2-year free return trajectory consistently has an Earth departure delta-v of approximately 4.3 km/s across all opportunities (ranging from 4.27 to 4.35 km/s). While there is greater variability from opportunity to opportunity, the propulsive abort options consistently provide a monotonic decrease in Earth departure delta-v with increasing abort delta-v, approaching the 2-year free return trajectory in the limit as the abort delta-v approaches zero. The 3-year free return (1.5-year period heliocentric orbit) trajectory exhibits the unique pattern of featuring a relatively low delta-v in many opportunities, and then a significantly higher delta-v in some opportunities (most notably 2022 and 2037, and to a lesser extent 2035). This behavior occurs because in some opportunities a 1.5-year period orbit must have its perihelion inside the orbit of the Earth in order to travel far enough out to reach Mars, resulting in increased angles between the desired heliocentric velocity of the orbit and the Earth's heliocentric velocity, and consequently increased Earth departure delta-v. In short, the 1.5-year period orbit is distant from the minimum energy Mars transfer in those opportunities. It is interesting to note that, with the exception of the 3-year free return at its peaks, all of these delta-v's are quite similar to the conjunction transfer trajectory options presented previously (less than a 10% increase in the case of 2-year free returns, see Figure 3). As such, Earth departure propulsive considerations are unlikely to preclude the inclusion of an abort option in the set of feasible human Mars exploration trajectories.

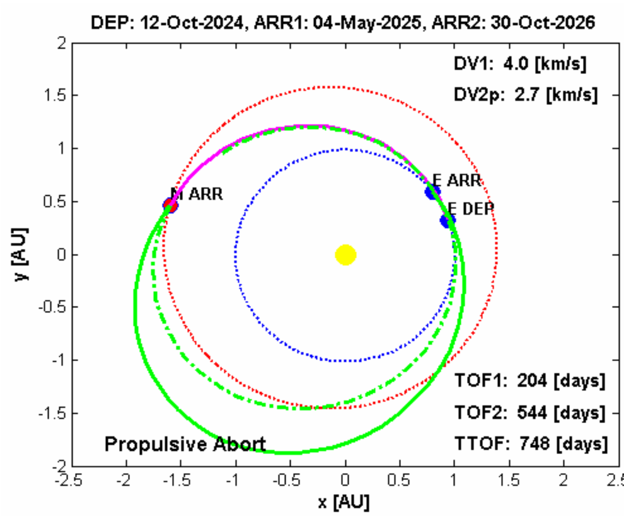

Figure 10. Example propulsive abort trajectory (with 2.7 km/s abort delta-v). The outbound spacecraft starts on an Earth-Mars transfer (magenta), which if no action were taken would not return to Earth (dashed green). In the case of an abort a propulsive Mars swing-by maneuver shifts the trajectory to one that does arrive at Earth (solid green).

The nominal Mars entry velocity (without abort) for trajectories that allow for the possibility of a Mars abort are presented in Figure 12. Here there is a pronounced variation in entry velocity from opportunity to opportunity, and the entry velocities themselves can become quite high. The 2-year free return has the highest entry velocity, peaking over 12 km/s in 2031 and 2033, and staying above 9 km/s for all

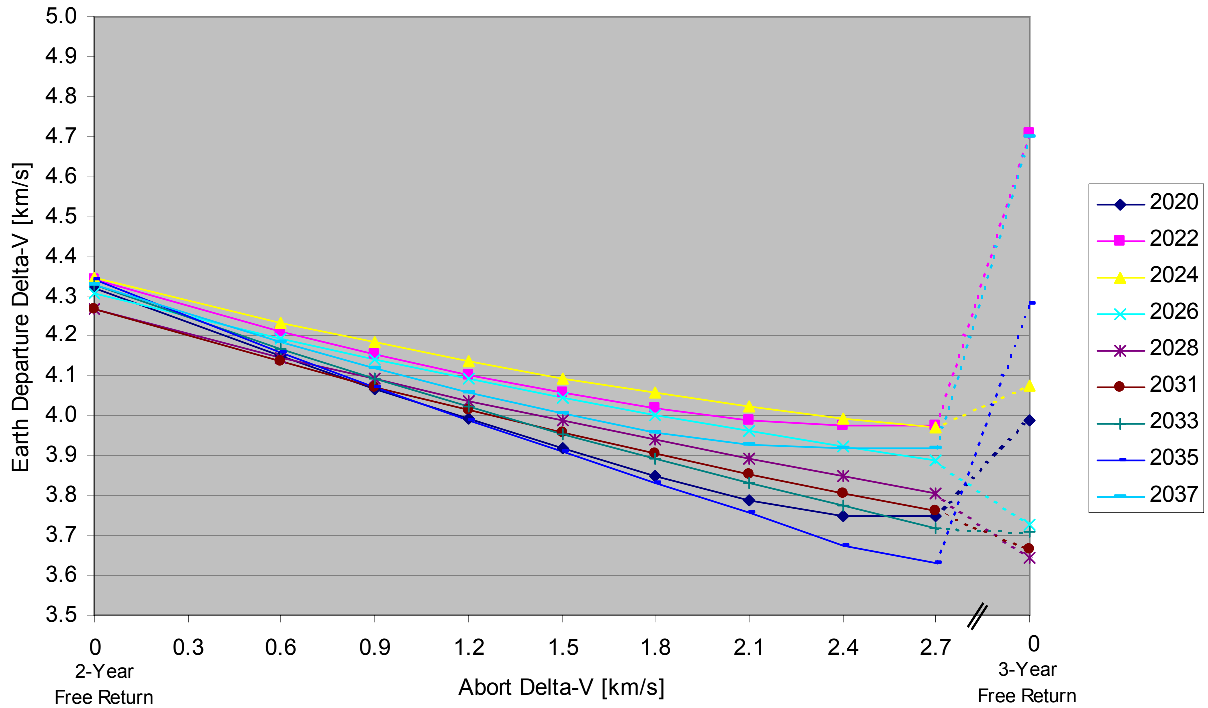

Figure 11. Low Earth orbit departure Delta-V for Earth-Mars free return and propulsive abort trajectories presented for Earth departure opportunities from 2020 to 2037.

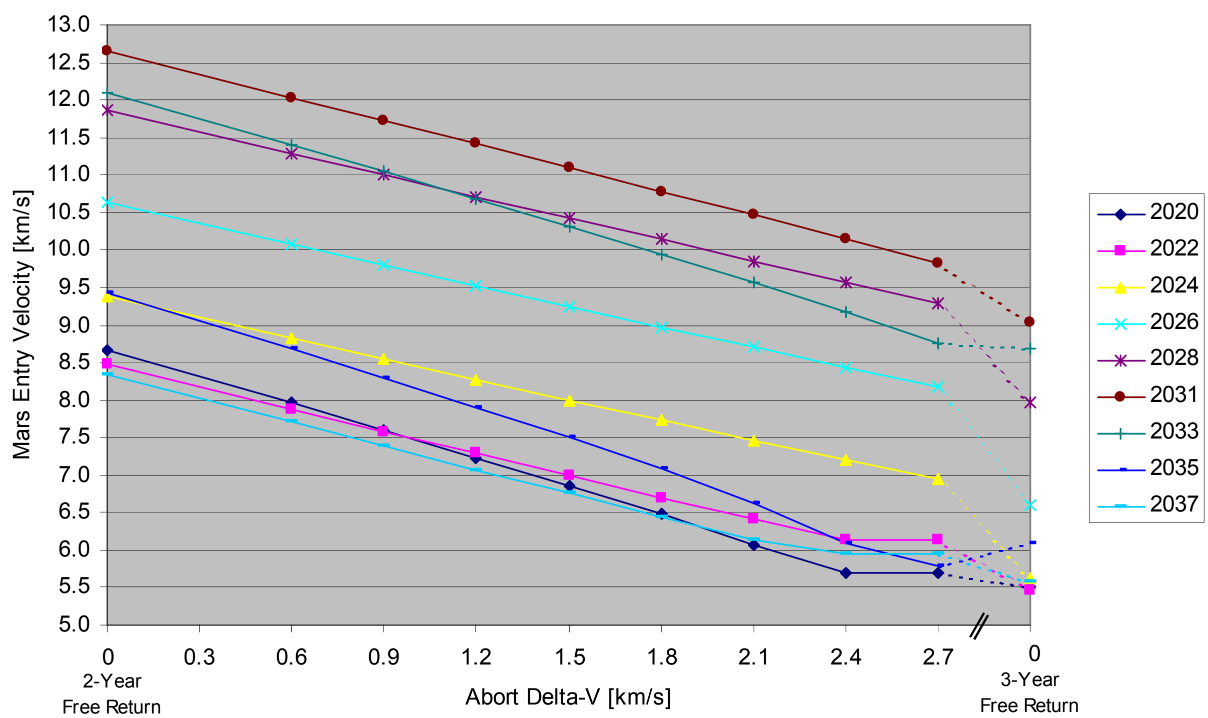

Figure 12. Nominal Mars arrival entry velocities (without abort) for Earth-Mars trajectories allowing for free return and propulsive aborts presented for Earth departure opportunities from 2020 to 2037.

of the opportunities from 2024 to 2035, inclusive. As will be discussed in the following section, such entry velocities may present a challenge for Mars aeroassist systems, and may preclude the use of the 2-year free return in a number of mission opportunities. Including a propulsive abort capability can lower the Mars entry velocities. By including an abort delta-v of 2.7 km/s (consistent with the nominal trans-Earth injection delta-v capability of a vehicle designed to bring the crew back from low Mars orbit at the end of a mission), the Mars entry velocity can be kept below 10 km/s in all mission opportunities. Utilizing instead an abort delta-v of 1.8 km/s (consistent with an Earth return vehicle departing a highly elliptic Mars orbit), the entry velocity will peak at 10.78 km/s in 2033, and be below 9 km/s other than in the 2028, 2030, and 2033 opportunities. From a Mars entry velocity perspective the 3-year free return option provides a distinct advantage in that it remains below 9 km/s in all opportunities.

Unlike the fixed transfer durations of the trajectories in the previous section, the Earth-Mars transit times for the abort trajectories vary based upon the opportunity as presented in Figure 13. For the most part, these trajectories offer comparable or faster transit times to the trajectories analyzed in the previous sections, with the exception of the 3-year free return. The 3-year free return exhibits close to a factor of two variation in transit time, from a minimum of 137 days in 2033 to a maximum of 262 days in 2035. While such transfer durations are not infeasible, having lower transfer durations than those exhibited by these lengthy 3-year free return transfers would be desirable.

Beyond the considerations of transit to Mars, abort trajectories by their very nature must also consider their Earth return leg in case an abort is opted for. The Earth entry velocities, presented in Figure 14, are relatively benign, when considering that the peak entry velocities of approximately 12.5 km/s are in line with the minimum entry velocity capability required for ensuring Mars return in all opportunities (even without constraints on Mars-Earth transit time). The Earth entry velocities in the abort cases (Figure 14) are benign relative to nominal Mars-Earth return trajectories (Figure 6) because in the abort case, at Earth arrival the heliocentric orbits of the Earth and the spacecraft are nearly tangential, minimizing the magnitude of the Earth relative velocity of the spacecraft ($V_{infinity}$). The abort orbits are nearly tangential as a result of minimizing the Earth departure energy of the original trajectory and (in the case of propulsive abort trajectories) the abort delta-v required at Mars. In contrast, nominal Mars-Earth trajectories tend to be nearly tangential to the heliocentric orbit of Mars at departure (to minimize Mars departure energy) but not at arrival at Earth (due to the constraints on the Mars-Earth transit durations).

For an abort to be feasible the crew must be supported during the duration of their roundtrip transit to Earth. Figure 15 presents the total in-space time in the case of an abort (this is the total time from Earth departure to Earth return, as this value will size the crew support requirements of the outbound spacecraft in order to support aborts). What can be seen is that the propulsive aborts all have very similar in-space durations to that of the 2-year free return, increasing marginally with increasing abort delta-v. The 3-year free return of course represents a 50% increase in abort time over the 2-year free return. In terms of these durations, two years is somewhat less than the duration of a complete round-trip Mars mission. In the two year case then, it may be possible to make use of consumables (and equipment) that was intended for the nominal mission in the case of the abort, although this would tend to preclude the use of prepositioned or in-situ consumables during the nominal mission. Additionally, options to include emergency rations (which can significantly reduce the mass of food), reduce consumption of water (such as through decreased washing allocations), and the conversion of propellants to consumables (oxygen and water, both of which benefit from propellants that include hydrogen and oxygen in their composition), should be investigated in considering how an architecture could enable such an abort mode. Supporting the crew on this abort return may indeed be one of the most challenging aspects of employing these abort trajectories, particularly if the 3-year free return is selected (based upon its lower Mars entry velocity requirement.)

As in the case of the conjunction trajectories discussed earlier, the choice of outbound Earth-Mars trajectory will impact the stay time available at Mars. Figure 16 presents the stay times for the abort trajectories investigated (in the case where the abort is not performed and the crew stays at Mars), relative to the subsequent 270-day return transfer as in the previous section (see Figure 8 for changes resulting from other return durations). Again, this is not likely to be a major driver in trajectory selection, but provides insight into changing exploration capability resulting from various trajectory selections. In general, the abort trajectories have higher surface stay times than the 180-day conjunction transfers, due to their even faster outbound transit times, with the exception of the 3-year free return, which in some opportunities can have significantly shorter mission durations. For propulsive abort trajectories, as the abort delta-v decreases (with the 2-year free return in the limit) the Mars stay time consistently increases, although the sensitivity varies with opportunity. These increases in stay time correspond with decreases in outbound transit duration.

Aeroassist maneuver considerations for human Mars mission trajectories

Human Mars missions utilizing aeroassist maneuver trajectories require a vehicle able to withstand the intense deceleration and aerothermal environment encountered during atmospheric entry. Uncrewed mission elements may perform aerocapture or direct entry at Mars arrival. Crewed mission elements will likely perform aerocapture at Mars arrival to provide operational flexibility in mitigating the large uncertainty inherent in the Mars atmosphere (Draper Laboratory). In addition, aerocapture will allow time for final system check–out in orbit before descent to the surface as well as changes to landing site if required. At the end of the

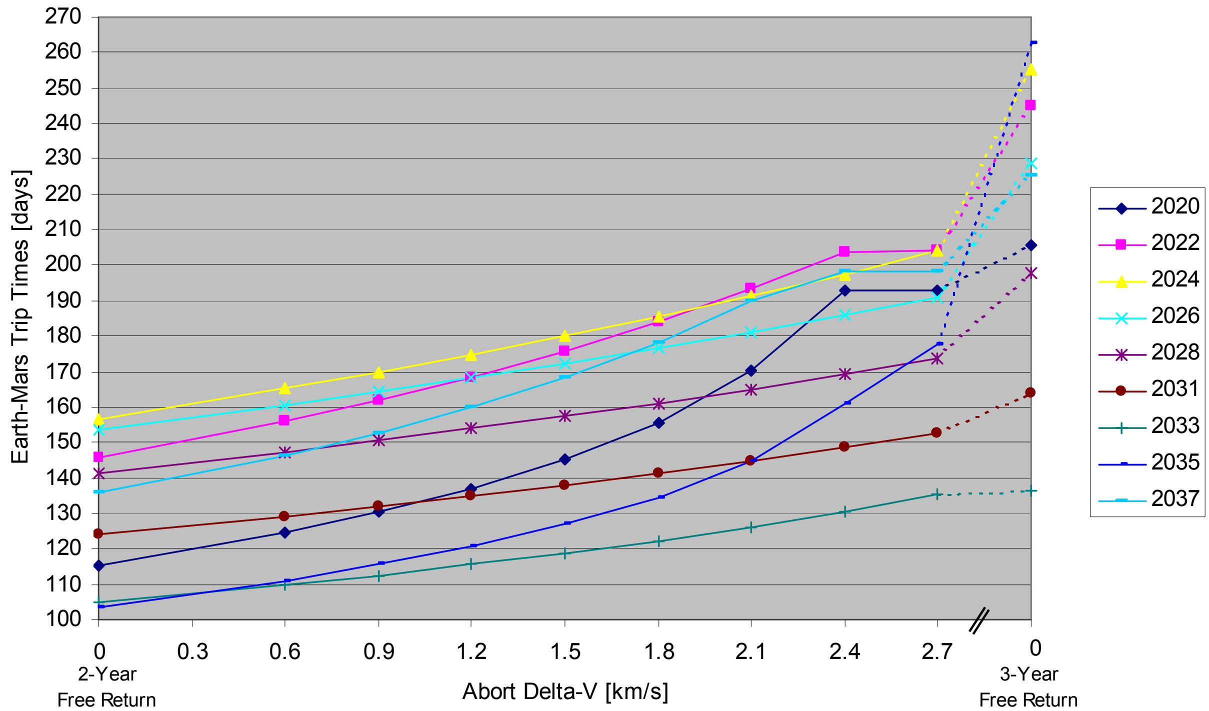

Figure 13. Earth-Mars transit times for Earth-Mars free return and propulsive abort trajectories presented for Earth departure opportunities from 2020 to 2037.

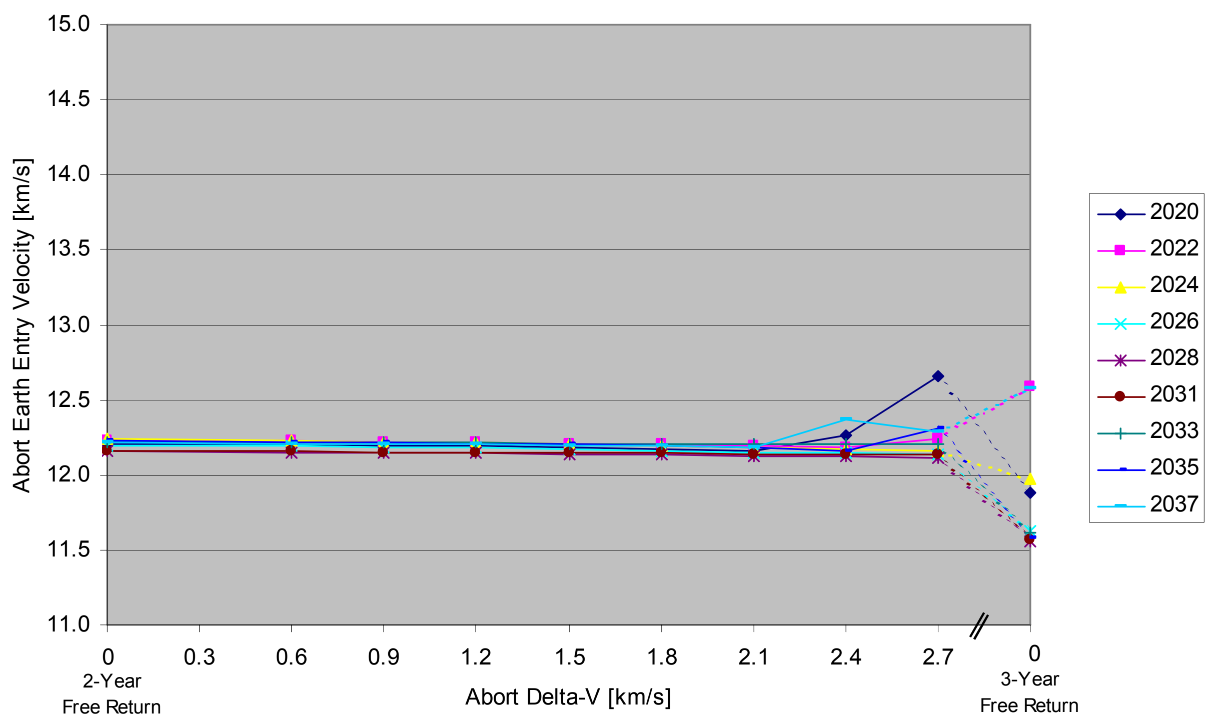

Figure 14. Earth arrival entry velocities (in the case of an abort) for Earth-Mars free return and propulsive abort trajectories presented for Earth departure opportunities from 2020 to 2037.

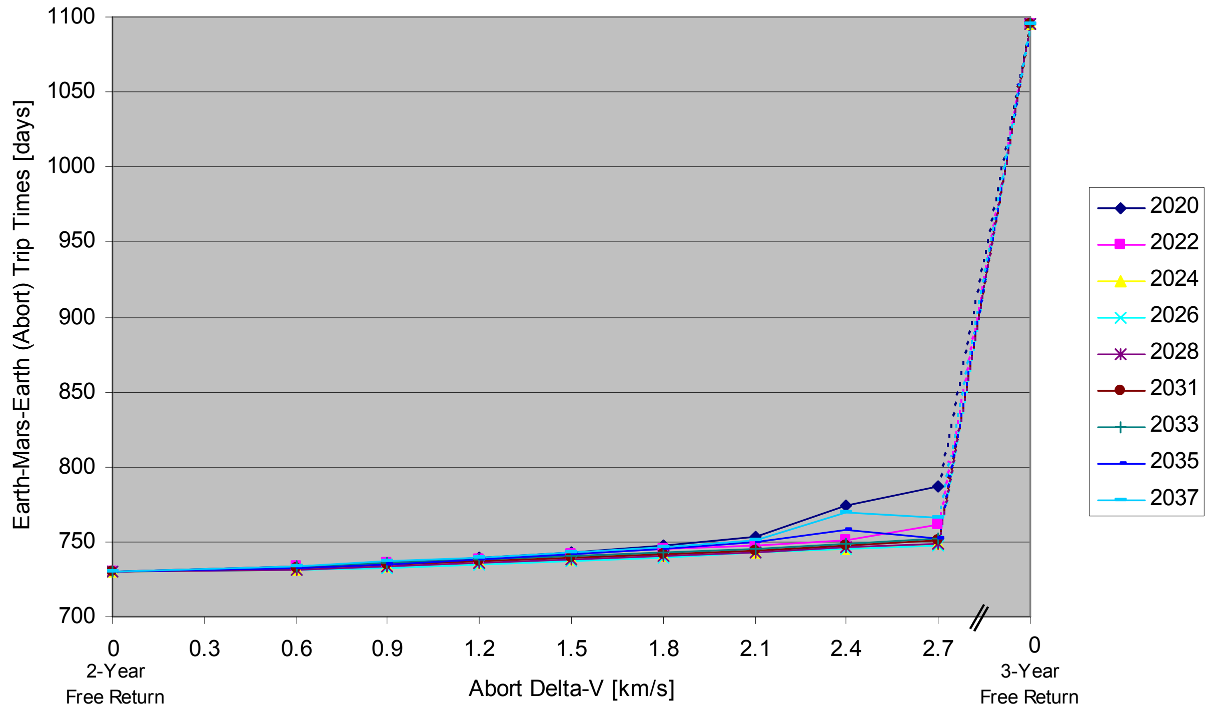

Figure 15. Total in-space times in case of an abort for Earth-Mars free return and propulsive abort trajectories presented for Earth departure opportunities from 2020 to 2037.

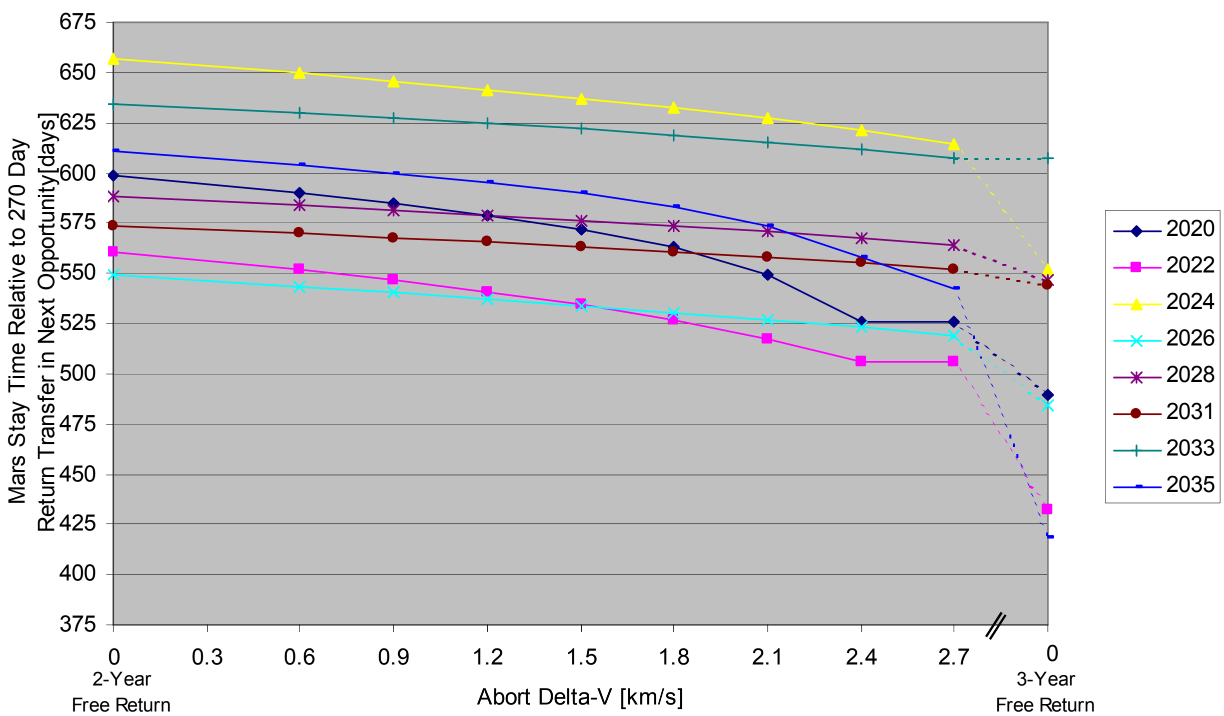

Figure 16. Nominal Mars stay times for Earth-Mars trajectories with free return and propulsive abort options presented for Earth departure opportunities from 2020 to 2035. The stay times are computed assuming a Mars-Earth return transfer of 270 days in the immediately following Earth return opportunity (e.g., the 2022 Earth return opportunity in the case of a 2020 Earth departure).

crew's Mars stay, they will depart for Earth. As the return vehicle nears Earth, the crew will likely transfer to a dedicated entry vehicle which will then perform direct entry at Earth. The development of practical, reliable, high performance aeroassist subsystems will demand significant technology investment. The magnitude of the required technology investment is related to the severity of the aerothermal and deceleration environment the aeroassist vehicle must pass through, which is directly related to the velocity (mission design), mass, and geometry of the vehicle at Mars or Earth arrival.

Aeroassist at Mars arrival

Human Mars missions require delivery of large mass payloads to orbit or the Mars surface, with corresponding entry masses ranging from several tens of metric tonnes to perhaps over 100 metric tonnes. Large vehicles will be required to transport large volume, large mass payloads to parking orbits and the surface, such as human habitats, ascent vehicles and return vehicles. The payload mass will drive the size of aeroshells, with 100 metric tonne entry masses perhaps requiring heatshields with forebody diameters of 15 m or greater (Braun et al. 1990; Cruz et al. 2005; Draper Laboratory 2005; Wells et al. 2006; Braun and Manning 2006). Significant technology investment is required to prepare for Mars aeroassist of this class of system (Braun and Manning 2006). The manufacture and qualification of large diameter heatshields presents a major engineering challenge. The large heatshields required for aeroassist maneuvers at Mars will drive the launch vehicle shroud requirements, assuming these systems are launched fully assembled. Heat shield assembly in orbit (or use of a large inflatable entry system) introduces a number of challenges, such as qualification of a sectional heat shield assembled in low Earth orbit. In addition, vehicles performing entry at Mars directly from an interplanetary trajectory will experience a heating environment which becomes more severe with increasing entry velocity.

An L/D of 0.3 represents an achievable value for a blunt body entry vehicle flying at a realistic angle of attack. Several blunt body shapes exist that can achieve this performance (Braun and Powell 1991). While entry vehicles with L/Ds greater than 0.3 are conceivable for large vehicles, they often achieve a higher L/D by decreasing drag rather than by increasing lift. This can pose a severe problem at Mars, where atmospheric density is low and it is difficult to decelerate the vehicle to the proper velocity before it impacts the surface or exits the atmosphere.

Figure 17 shows the range of entry flight-path angles and velocities that provide acceptable aerocapture trajectories for a 100 mt vehicle with a 15 m aeroshell generating an L/D of 0.3, including a 5 g deceleration limit for crew factors reasons. In order to ensure successful aerocapture, some margin must be left in entry flight-path angle to account for navigation uncertainty at Mars arrival. Current robotic Mars missions have demonstrated navigation accuracies of approximately +/- 0.5 degrees or better, resulting in a robust capability to achieve an entry corridor width of 1 degree (Braun and Manning 2006; Braun and Powell 1991). Based on this limit, to maintain adequate corridor margin for aerocapture within the 5 g constraint, it is recommended that entry velocities be limited to 9 km/s or less. Direct entry at Mars is subject to the same navigation uncertainty as aerocapture. To maintain a corridor of 1 deg, the entry vehicle previously described (L/D = 0.3) must have an entry velocity of 9 km/s or less. For Mars entry velocities above 9 km/s, assuming a 5 g acceleration limit and a 1 deg entry corridor constraint, a higher L/D vehicle is required, implying the need for a different aerodynamic configuration. Direct entry at high velocity also produces an extreme heating environment with stagnation point heat rates approaching 500 W/cm^2. In comparison, current Mars entry experience is limited to heat rates of approximately 100 W/cm^2 (Milos et al. 1999). The required heating magnitude coupled with the size of the heat shield will present a significant thermal protection system (TPS) development challenge. To minimize the TPS technology investment required for direct entry, the lowest possible entry velocity and environments that maintain a low radiative heating contribution are preferred. For these reasons we suggest a Mars entry velocity limit of 9 km/s for the purposes of conceptual-level Mars mission design.

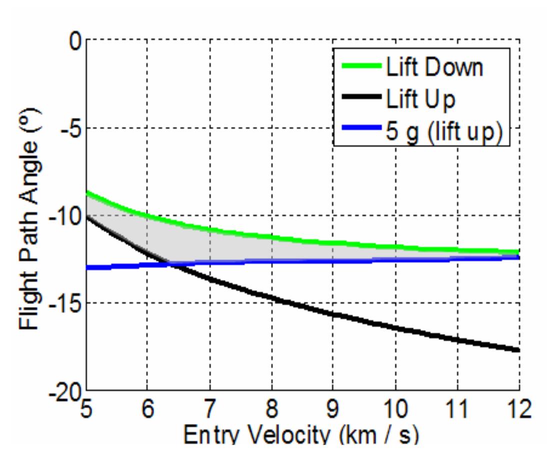

Figure 17. Aerocapture corridor width as a function of Mars entry velocity for a 100 mt vehicle with a 15 m aeroshell generating an L/D of 0.3 (Braun and Manning 2006).

Aeroassist at Earth return

Returning a human crew to Earth from Mars requires investment in a different set of aeroassist technologies than those required for Mars arrival. Earth entry will occur at much higher velocity (12+ km/s) in a dense atmosphere. While the dense atmosphere eases L/D requirements, the crew deceleration environment requires a greater L/D, and the higher entry velocity and density increase both the peak heat rate and integrated heat load significantly, placing additional stress on the TPS. The crewed earth entry vehicle function will likely be performed by an upgraded version of the Orion Crew Exploration Vehicle (CEV) currently under development by NASA, which has a 5 m diameter.

A blunt body entry vehicle returning from Mars at 12.5 km/s may experience laminar stagnation point heat rates in excess of 400 W/cm^2. Direct entry at Earth at 14 km/s produces stagnation point heat rates well above 1000 W/cm^2 (Putnam et al. 2005). These estimates of the aerothermal environment may be doubled at positions approaching the vehicle shoulder due to turbulent heating augmentation. There are currently no human-rated heatshield materials available that can withstand the extreme heating environment encountered when returning from Mars. However, several promising new forebody TPS materials have recently been successfully demonstrated by the robotic exploration program and several other materials are being considered for the CEV and other planetary exploration programs (Laub 2003).

The development of the CEV for both LEO missions and lunar return missions implies the development and qualification of at least one new heatshield material. This material will be able to withstand direct entry for human lunar return, implying tolerance of entry velocities up to 11 km/s. Any forebody TPS material developed for human lunar return will likely be applicable for entry velocities up to 12.5 km/s, and should be developed with this requirement in mind. This velocity range includes the lowest Mars return velocities, and is therefore the minimum technology investment required for human return from Mars.

To facilitate fast transfer return trajectories from Mars (180 days) in all opportunities, the Earth return vehicle must be able to withstand entry at velocities up to 14.7 km/s. Entry at this velocity may produce laminar stagnation point heat rates in excess of 2000 W/cm^2 and integrated laminar stagnation point heat loads in excess of 100 kJ/cm^2 (Putnam et al. 2005). At these velocities, radiative heating accounts for 50% or more of the total heat pulse, creating a heating environment that is difficult to model and test in ground-based facilities. Developing and qualifying a heatshield that can withstand entry at 14 km/s represents a significant engineering challenge. At Earth return velocities higher than 14 km/s, radiative heating becomes the dominant mechanism of heat transfer and TPS mass fraction begins to dominate entry vehicle mass. Increased entry vehicle mass is a significant penalty to incur for additional interplanetary trajectory flexibility as the entry vehicle must be transported to Mars and back. In addition, limitations in our ground-based test facilities makes the challenge of qualifying new TPS materials becomes more difficult as the Earth return entry velocity increases.

While TPS development and qualification for Mars return will require significant investment, NASA's current CEV moldline provides adequate aerodynamic deceleration performance. Figure 18 shows the hypersonic L/D required for a given peak deceleration at Earth. The current CEV design calls for an Apollo-style blunt body with an L/D of approximately 0.3. Figure 18 shows that this is adequate to limit peak deceleration to 5 g's even at entry velocities of 14 km/s.

The minimum technology required for human Mars return at Earth will be an extension of the systems under development for the lunar exploration program. Systems currently under development can likely accommodate Earth return speeds as high as 12.5 km/s, providing return trajectories in most opportunities for transfers of less than 210 days. To increase mission flexibility and allow fast returns for human Mars missions at all opportunities, it may be necessary to invest significant resources to develop and qualify a TPS that can withstand Earth entry at 14 km/s. This development and qualification represents a major scientific and engineering challenge that will have significant cost and schedule requirements. Above 14 km/s, TPS development will become even more difficult and a new vehicle moldline may be required to maintain a 5-g acceleration limit.

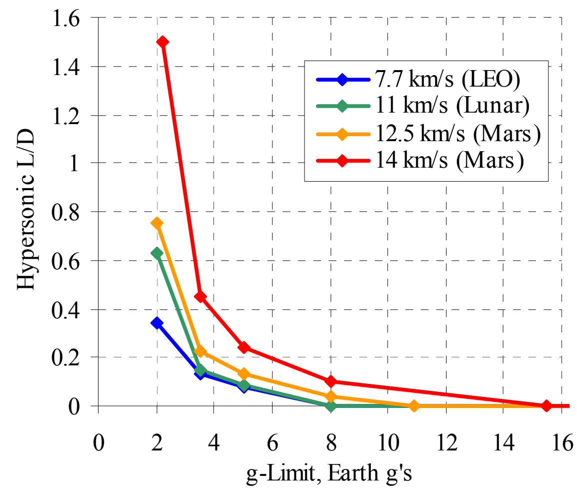

Figure 18. Required lift-to-drag ratio for direct entry with a given g-limit (Putnam et al. 2005).

Human Mars mission trajectory summary

A summary of the resulting trajectory options incorporating both propulsive requirements and Earth or Mars entry system requirements follows. Based upon the Mars entry corridor width lower limit of 1 degree and the resulting 9 km/s upper limit on Mars entry velocity, Earth-Mars trajectories presented in Table 1 are feasible in the given opportunities. Using a conjunction trajectory, Earth-Mars transit times of 180 days or less are feasible in all opportunities. The 2-year free return is only feasible in a subset of opportunities based upon the entry corridor limit, while the three year free return is feasible in all opportunities. The propulsive abort can be achieved for relatively low delta-v in approximately half of the opportunities, with substantially higher abort delta-v's required in the remaining opportunities.

For Earth entry, a 12.5 km/s capability appears readily achievable, while an Earth return entry velocity capability up to 14 km/s will require significant technology investment. Table 2 shows the change in Mars-Earth transfer time that can be achieved through increasing the Earth entry velocity limit. Increasing the limit to 14 km/s can keep transit times under 200 days in all opportunities, as opposed to requiring transfers greater than 200 days in the 2024-2030 opportunities with a 12.5 km/s entry velocity limit.

Table 1. Minimum Earth-Mars transit time, free return feasibility, and required propulsive abort delta-v by opportunity for 9 km/s Mars entry velocity limit.

Mission opportunity	Minimum conjunction transfer time [days]	2-year free return feasible?	3-year free return feasible?	Propulsive abort delta-v [km/s]
2020	≤ 120	Yes	Yes	≤ 0.6
2022	≤ 140	Yes	Yes	≤ 0.6
2024	≤ 160	No	Yes	≤ 0.6
2026	≤ 180	No	Yes	≤ 1.8
2028	≤ 180	No	Yes	> 2.7
2031	≤ 160	No	Yes	> 2.7
2033	≤ 130	No	Yes	≤ 2.7
2035	≤ 120	No	Yes	≤ 0.6
2037	≤ 130	Yes	Yes	≤ 0.6

Table 2. Minimum Mars-Earth transit time by opportunity for 12.5 km/s and 14 km/s Earth entry velocity limits.

Mission opportunity	Minimum conjunction transfer time [days], 12.5 km/s Earth entry velocity limit	Minimum conjunction transfer time [days], 14 km/s Earth entry velocity limit
2020	≤ 140	≤ 120
2022	≤ 180	≤ 140
2024	≤ 210	≤ 170
2026	≤ 230	≤ 190
2028	≤ 260	≤ 200
2031	≤ 240	≤ 180
2033	≤ 170	≤ 130
2035	≤ 130	≤ 120
2037	≤ 160	≤ 120

Table 3 presents the required capabilities in terms of departure delta-v and entry velocity in order to support Earth-Mars or Mars-Earth transfers in every opportunity with a transfer time within the constraint. For example, in order to support Earth-Mars transfers of 180 days or less in every opportunity, an Earth departure delta-v capability of 4.08 km/s is required and a Mars entry velocity capability of 8.57 km/s is required. Similarly, to support Mars-Earth transfers of 180 days or less in every opportunity requires a Mars departure delta-v capability of 3.03 km/s and an Earth entry velocity 14.65 km/s. These capabilities are required in order to support transfers in all opportunities – in many opportunities, a system with such a capability may be able to execute significantly faster transfers.

Conclusions

A variety of trajectory options for human Mars missions have been analyzed and presented. In addition to conjunction class trajectories, 2-year free return, 3-year free return, and propulsive abort trajectory options may be feasible. In many cases, a strong trade exists between the interplanetary trajectory chosen and its Earth or Mars entry system requirements. An Earth departure delta-v of 4.08 km/s enables a 180-day or faster Earth-Mars transfer in every opportunity; with less than a 10% increase required to allow potential abort opportunities. Mars entry velocity is a significant constraint on Earth-Mars transfer options, particularly in the case where abort opportunities are also desired. Transfers less than 180 days are possible across all opportunities for a Mars entry velocity constraint of 9 km/s. The Earth entry velocity is tightly coupled to the Mars-Earth return transfer duration, with an entry velocity as high as 14 km/s required to enable transfers below 200 days in every opportunity. A 12.5 km/s Earth entry velocity constraint increases the Mars-Earth transfer duration by 10 to 60 days depending on the mission opportunity, relative to the comparable 14 km/s Earth entry velocity transfer. Overall it appears that conjunction-class trajectories are feasible, including options for abort paths for the crew in most opportunities (depending on the abort duration allowable). Attention must be paid to entry velocity constraints driven by aeroassist systems in the selection and detailed design of trajectories for human Mars missions.

Table 3. Required departure delta-v and entry velocity capabilities to support Earth-Mars or Mars-Earth transfers in every opportunity within maximum transfer time constraint.

Maximum transfer time constraint [days]	Required Earth departure delta-v [km/s]	Required Mars entry velocity [km/s]	Required Mars departure delta-v [km/s]	Required Earth entry velocity [km/s]
≤ 120	5.11	14.45	5.42	18.24
≤ 130	4.81	12.89	4.79	17.47
≤ 140	4.58	11.57	4.28	16.80
≤ 150	4.40	10.45	3.86	16.18
≤ 160	4.27	9.72	3.52	15.63
≤ 170	4.16	9.11	3.24	15.12
≤ 180	4.08	8.57	3.03	14.65
≤ 190	4.02	8.09	2.87	14.23
≤ 200	3.98	7.67	2.75	13.84
≤ 210	3.98	7.30	2.75	13.50
≤ 220	3.98	7.00	2.75	13.19
≤ 230	3.98	6.76	2.75	12.92
≤ 240	3.98	6.50	2.75	12.70
≤ 250	3.98	6.50	2.72	12.52
≤ 260	3.98	6.27	2.69	12.42
≤ 270	3.98	6.02	2.69	12.41

Directory of supporting data

root directory
- wooster_mars_2007_0002.pdf this file
- earth-mars-conjunction.txt trajectory data for conjunction transfers from Earth to Mars
- mars-earth-conjunction.txt trajectory data for conjunction transfer from Mars to Earth
- earth-mars-abort.txt trajectory data for transfer from Earth to Mars with option for abort to Earth

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