Delta-Vs and Design Reference Mission Scenarios for Mars Missions

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ABSTRACT

Before setting out on any long journey, it is important to first have an idea about how to get there, how long it might take, and how much it will cost. Primarily, the answers depend upon where you are starting and where you wish to go. In the formulative stages of any mission to Mars, having quick estimates of answers to these basic questions will aid in the efficient exploration of the trade space. In this paper, we present a "mileage chart" of sorts illustrating the range of ΔV's and times-of-flight (TOF) between various starting and stopping points between Earth and Mars. This paper expands upon a chart from a previous work by the authors. We discuss the methodologies used to calculate or estimate expected values and reasonable ranges, including some more detailed specific examples.

INTRODUCTION

The calculations for ΔV and TOF greatly depend on the propulsion technology used – whether it be traditional chemical propellants, leading to near-impulsive maneuvers and ballistic trajectories, or solar-electric propulsion (SEP), leading to low-thrust optimized transfers.

The primary product of this paper is a set of tables [see appendix for full page tables], herein referred to as the Table. Separate "mileage charts" and mission scenarios are given for each type of propulsion – chemical and SEP. It is an expansion of tables 1 and 2 from a previous paper by the authors [1]. The ΔV's and ranges generated were also used as inputs for a cost model that was developed at JPL that acknowledged and attempted to quantify the significant contribution of propulsion on total cost for small Mars missions [2].

The starting/stopping points, or "bus stops", analyzed are: Low-Earth Orbit (LEO), Geosynchronous Transfer Orbit (GTO), Lunar Transfer Orbit (LTO), Earth Escape, Mars Transfer (C3 ≈ 15 km^2/s^2), Mars Capture ($V_\infty = 0$ km/s), 2-sol orbit, areostationary orbit, Phobos, and Low-Mars Orbit (LMO). Note that the bus stops do not specifically include Mars entry. Entry trajectories typically do not need large deceleration maneuvers but rather enter the atmosphere hyperbolically, relying on aerodynamic braking and/or retropropulsion to land safely. Therefore, landed missions can use the ΔV requirements to get to Mars Transfer, then coast ballistically. Small maneuvers such as biasing or correction maneuvers are subsumed in the values given.

Each pair of starting and stopping points has a range of possibilities that affect both the ΔV and TOF. These include the specific launch opportunity, specific parameters of the orbits (inclination, node, etc.), mission requirements, etc. This paper discusses the key considerations for each type of transfer and how estimates were made for a "reasonable" range of high and low values, as well as a "most likely". The Table is a detailed lookup reference with all of the bus stops for both ΔV and TOF ranges for both propulsion types. This yields 90 possible transfer combinations and their associated ranges, which are discussed in the sections that follow.

The longest transfers – from near-Earth to near-Mars (which are found in the lower left region of the Table), require significant amounts of ΔV from any propulsion system. For chemical propulsion, > 4 km/s of ΔV leads to a 3:1 ratio of propellant to dry mass and usually benefit from staging. Much beyond that and some systems will not converge. For SEP, > 10 km/s of ΔV only needs a ~1:1 propellant/dry ratio, however, the real challenge is to find thrusters with sufficient lifetimes to process that much propellant.

"BUS STOPS" FROM EARTH TO MARS

The points included as starting, stopping, and intermediate "bus stops" in the Table are not exhaustive but were chosen as representative and suitable to characterize a wide range of potential missions.

Low-Earth Orbit (LEO)

LEO can be defined as any nearly-circular orbit with an attitude from 185 km to as high as 2000 km and any inclination, but for calculations here it is assumed to be 200 km circular and 28.5° inclined.

Geosynchronous Transfer Orbit (GTO)

200 km x 35,786 km. This is the common delivery orbit for satellites destined for geostationary orbit (GEO), which complete the transfer using their own propulsion. The orbital period is 10.5 hours. It also typically has an inclination of ~27°. The orientation of the line of apsides is variable and depends on the launch conditions and the desired GEO longitude.

Lunar Transfer Orbit (LTO)

There are many potential cislunar transfers that may require longer duration for reduced propulsion requirements or meet other objectives, but the most basic is an elliptical orbit with an apogee at the lunar distance – 200 km x 384,000 km. The period is about 10 days and the C3 is -2 km^2/s^2.

Earth Escape

This is the transition point between bound and unbound orbits at Earth's sphere-of-influence (SOI). V_∞ = C3 = 0. A rideshare to escape may be possible as part of a launch vehicle upper stage disposal burn that chooses to burn to escape rather than controlled reentry. Earth/Sun Lagrange point missions are also near escape trajectories.

Mars Transfer (Trans-Mars Injection)

A trans-Mars injection (TMI) burn provides the velocity and asymptote needed to ballistically coast to Mars intercept with only minor course corrections. Every 26 months, low-energy type I/II trajectories are available with a minimum C3 of 8 – 16 km^2/s^2.

SEP missions will optimize their Mars transfer energy based on launch vehicle, mass, thrust, and other mission objectives. Optimal C3 values are often 2 - 10 km^2/s^2, but for consistency, the Mars Transfer calculations were done using C3 = 15 km^2/s^2 for both chemical and SEP.

Mars Arrival (V_∞ = 0)

Represents the arrival, or rendezvous, with the Mars gravitational sphere of influence. Note that this is NOT EQUIVALENT TO MARS ENTRY. Landed missions typically arrive at Mars hyperbolically (V_∞ ~ 3 km/s) and use atmospheric braking.

Mars arrival is not a typical stopping point for chemical missions as the V_∞ = 0 threshold is crossed during the MOI capture burn. For SEP, however, the low-thrust trajectory does rendezvous with Mars and transitions from heliocentric cruise to Mars-centric spirals.

2-sol Mars Orbit

A highly-elliptic 2-sol orbit is a representative capture orbit to begin an aerobraking campaign down to LMO, although capture orbits of 1 - 4 sols are not uncommon. A 2-sol orbit is 300 km x 57,826 km with a period of 49.3 hours. As a starting point, a typical aerobraking campaign takes 6-12 months and would require ~200-300 m/s for targeting, corridor control, and periapsis pop up.

Areostationary Orbit

Areostationary is a circular, equatorial orbit at 17,031 km analogous to a geostationary orbit at Earth. It has a 24.6 hr. period (1 sol) and remains fixed over one point. Small inclinations are also possible, creating a figure eight ground track. This orbit can also be used as a surrogate orbit for Deimos, which is at 20,062 km x 1.8°. It is about a 100 m/s transfer between them.

Phobos Orbit

Phobos has a mean orbital altitude of 5980 km and inclination of 1°. It has period of 7.66 hours.

Low Mars Orbit (LMO)

LMO is generally any near-circular orbit < 1000 km. They are often used for scientific observations, with the most common being a sun-synchronous orbit (SSO) with a near polar inclination of ~93°. 300 km circular is used as a reference LMO.

CHEMICAL PROPULSION TRANSFERS

Transfers using chemical propulsion (typically Lox-H2, Lox-RP1, MMH-NTO ("biprop"), hydrazine ("monoprop"), or solid propellants) usually have a high thrust-to-weight ratio, and the resulting maneuver can be treated as largely impulsive. Maneuvers are most efficient when the spacecraft is moving the fastest, lower in the gravity well (Oberth Effect [3]). In some instances, it is possible to break a maneuver into multiples in order to keep burn locations closer to the optimal point and reduce gravity losses. The ΔV values given in the Table are rounded up to account for ~2-3% in gravity losses. Additional ΔV should be carried for additional gravity losses, trajectory correction maneuvers (TCMs), orbital maintenance, and other margins. Around 100 - 200 m/s is a good starting point.

Transfers from LEO

LEO as a starting point for Mars missions is not recommended both due to the excessive amounts of ΔV needed from so low in the gravitational well, as well as the radiation environment.

LEO to GTO. A standard Hohmann-type transfer with a large kick and a ~5-hour coast to the GEO belt, most often performed by the launch vehicle upper stage, with separation occurring after completion. In some cases, excess capability in the launch vehicle is used to boost beyond GEO (super-synchronous) in order to reduce the propulsion needs of the satellite. ΔV variation is due to potential inclination changes and transfers beyond GEO (e.g. super-synchronous).

LEO to LTO. In its simplest form this is similar to the GTO transfer but with a greater apogee (out to lunar orbit). A purely Hohmann transfer takes 3.2 km/s and 5 days, but it is not uncommon to go slower (or faster), requiring less (or more) ΔV. Low-energy lunar transfers go well beyond lunar orbit, taking advantage of solar perturbations, and may take weeks [4][5]. ΔV variation is due to the potential need for inclination changes and phasing with the moon.

LEO to Escape. Pure launches or burns to escape are not common and mostly serve as a reference point. Near-escape trajectories may occur for Lagrange point missions, some interplanetary SEP missions, or launch vehicle upper stage disposal. ΔV variation is due to transfers slightly short of or beyond escape (e.g. Lagrange points).

LEO to Mars Transfer. TMI burns are commonly performed in LEO by the launch vehicle upper stage within a few hours of launch. In the event of a rideshare to LEO, the spacecraft or a kick stage would have to perform TMI. ΔV variation is due to the range of possible C3's, which are opportunity and transfer specific. Minimum C3 values across a full Mars Cycle vary from 8 – 16 km^2/s^2 [6]. ΔV [in km/s] can be found by:

$$\Delta V = \sqrt{\frac{2\mu_e}{6578} + C3} - 7.78 \qquad (1)$$

where $\mu_e = 3.986e5\ km^3/s^2$.

LEO to Mars Arrival. Similar to Earth escape, this is more of reference point than a destination. It is equivalent to the loosest possible capture orbit at Mars, or reducing V_∞ to 0. For a ballistic transfer from LEO, it would entail a TMI burn, a 7-13 month coast, followed by an MOI near Mars for a loose capture. ΔV is about 200 m/s less than capture to a 2-sol orbit.

LEO to 2-sol Orbit. After TMI and cruise, a MOI is performed at 300 km altitude to capture to an elliptical 2-sol orbit with an inclination greater than the absolute value of the incoming declination. The total ΔV from LEO is the sum of LEO to Mars Transfer (C3 dependent) and Mars Transfer to 2-sol orbit (V_∞ dependent).

LEO to Areostationary Orbit. This is a combination of LEO to Mars Transfer and Mars Transfer to Areostationary, which includes a 3-burn capture.

LEO to Phobos. This is a combination of LEO to Mars Transfer and Mars Transfer to Phobos.

LEO to LMO. The total ΔV for this transfer is > 6.5 km/s and is on the very limit of feasibility for a single spacecraft. It would need to be over 90% propellant by mass.

Transfers from GTO

GTO is significantly higher in Earth's gravity well than LEO which makes it a much better starting point for missions to Mars. However, the orientation of the ellipse is effectively random and usually dictated by the primary mission going to GEO. The launch can also occur at any time and not necessarily during optimal Earth-Mars transfer seasons.

For these reasons the secondary spacecraft must employ a series of maneuvers, phasing orbits, loitering, and potential lunar flybys to align for and perform a TMI during the optimal window [7][8].

GTO to LTO. If the line of apsides were aligned then this would be a simple, single-burn boost at perigee. However, the direction of the GTO apogee and the location of the Moon vary and need to be phased, either through waiting (up to 28 days) or intermediate phasing maneuvers.

GTO to Escape. The ΔV to reach escape velocity from GTO perigee is just 770 m/s, where direction does not matter. That is only ~90 m/s above the ΔV need to get to 384,000 km (lunar distance).

GTO to Mars Transfer. TMI from GTO would likely be the final burn in a series of maneuvers to alter the orbit and phase for the proper departure asymptote and timing. The most efficient location would be during the final perigee passage. ΔV's in the Table are the sum total of those burns, and are primarily affected by the variation in the required C3, with some contribution to the phasing strategy employed [9]. In general, the ΔV can be found by using Eq. (1), but replacing the 7.78 with 10.24.

GTO to Mars Arrival. Mars Arrival is more of reference point than a destination, equivalent to $V_\infty = 0$. ΔV is about 200 m/s less than capture to a 2-sol orbit.

GTO to 2-sol Orbit. The total ΔV from GTO is the sum of GTO to Mars Transfer (C3 dependent) and Mars Transfer to 2-sol orbit (V_∞ dependent).

GTO to Areostationary Orbit. This is a combination of GTO to Mars Transfer and Mars Transfer to Areostationary, which includes a 3-burn capture.

GTO to Phobos. This is a combination of GTO to Mars Transfer and Mars Transfer to Phobos.

GTO to LMO. This is a combination of GTO to Mars Transfer and Mars Transfer to LMO. The nominal total ΔV of ~4 km/s is about the same as the ΔV from LEO to Mars Transfer.

Transfers from LTO

Lunar transfers are a very energetic orbit and are just a few m/s below Earth escape. In practice, missions to the moon may employ a wide range of trajectories to meet their objectives, which means that any secondaries must carefully plan the optimal separation point and Earth departure strategy. As with GTO, the primary challenge is to loiter, phase, and maneuver such that a TMI burn can be performed

LTO to Escape. At perigee, only an additional 90 m/s would be required to escape. Since "escape" is also used to represent near-escape and Lagrange points, up to a few hundred m/s additional could be required.

LTO to Mars Transfer. As with TMI from GTO, phasing and loitering may be required to align for a proper departure burn. The total ΔV to achieve TMI is primarily dependent on C3, and can roughly be calculated using Eq. (1) and replacing the 7.78 with 10.92. Multiple lunar flybys or low-energy transfers may reduce the ΔV needed at the cost of additional time.

LTO to Mars Arrival. Similar to GTO to Mars Arrival.

LTO to 2-sol Orbit. Similar to GTO to 2-sol Orbit.

LTO to Areostationary Orbit. Similar to GTO to Areostationary Orbit.

LTO to Phobos. Similar to GTO to Phobos.

LTO to LMO. This is a combination of LTO to Mars Transfer and Mars Transfer to LMO.

Transfers from Escape

Escape as a starting point for a chemical rideshare is much more challenging than GTO or LTO in that the natural orbit will not return fly by Earth for the secondary to perform its departure burn. Performing TMI beyond Earth's SOI would require many additional km/s versus a burn close to Earth.

Escape to Mars Transfer. There are two ways to address the problem of TMI from an escape trajectory. The first is for the secondary to perform a small maneuver after separation and return for another Earth flyby to perform TMI there. The second is to separate quickly after the escape maneuver and perform TMI as quickly as possible so as to maintain some efficiency for a burn close to Earth. Either method relies on the direction and date of the escape being favorable for Mars transfer.

Escape to Mars Arrival. This is essentially $V_\infty = 0$ at Earth to $V_\infty = 0$ at Mars, or SOI to SOI. Without the ability to take advantage of the Oberth effect by performing departure and capture near the planets, the pure Hohmann transfer ΔV would be 5.6 km/s (see Figure 1). Utilizing the Oberth effect on either end could reduce this to < 2 km/s.

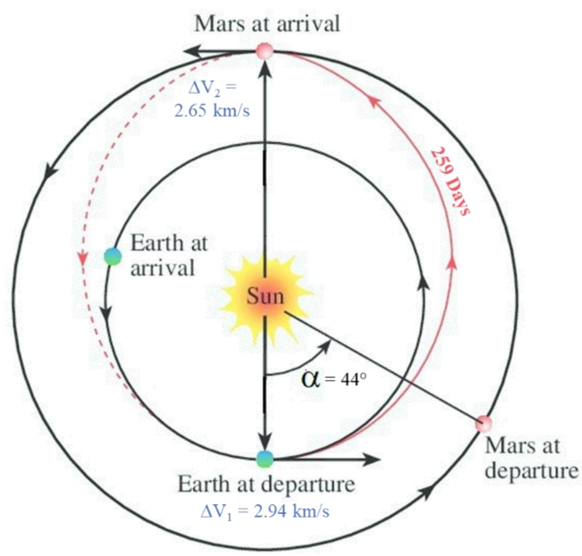

Figure 1 - Pure Hohmann Transfer from Earth to Mars. This is impractical as it negates the use of the Oberth effect at either planet and more than doubles the ΔV needed.

Escape to 2-sol Orbit. Similar to GTO to 2-sol Orbit.

Escape to Areostationary Orbit. Similar to GTO to Areostationary Orbit.

Escape to Phobos. Similar to GTO to Phobos.

Escape to LMO. This is a combination of Escape to Mars Transfer and Mars Transfer to LMO.

Transfers from Mars Transfer (TMI)

Trans-Mars injection is the most common starting point for a Mars mission with a dedicated launch vehicle. Once the proper departure velocity is achieved, only minor TCMs are needed to reach Mars after a 7-13 month cruise. The spacecraft would then be responsible for entry targeting (landers) or MOI (orbiters).

Mars Transfer to Mars Arrival. As noted, Mars Arrival (V_∞ = 0) is not a common destination for ballistic transfers. ΔV would be about 200 m/s less than 2-sol for the

Mars Transfer to 2-sol Orbit. The magnitude of the MOI needed to achieve the highly elliptical (e = 0.87) 2-sol orbit is dependent on the arrival V_∞ at Mars, which varies with the heliocentric trajectory. The minimum V_∞ varies from 2.5 km/s to > 4.2 km/s over a Mars Cycle, which accounts for the range of ΔV's given. ΔV [in km/s] can be found by:

$$\Delta V = \sqrt{\frac{2\mu_m}{3696} + V_\infty^2} - 4.675 \quad (2)$$

where μ_e = 42828 km^3/s^2. MOI ΔV to a 1-sol orbit (r_a = 33,800 km) is about 100 m/s more, whereas a 4-sol orbit (r_a = 96000 km) is about 50 m/s less.

Mars Transfer to Areostationary Orbit. It is not possible to achieve an equatorial orbit directly from an incoming trajectory with a nonzero declination. It is also more efficient to perform MOI capture maneuvers close to the planet. For these reasons, transfers to areostationary typically consist of three or more maneuvers. First, capturing to an inclined (equal to incoming declination) elliptical orbit with apoapsis beyond areostationary. Second, a combined periapsis raising maneuver and inclination reduction. Third, a circularization burn with necessary phasing to target the desired longitude. Other strategies may be employed, but the ΔV variation is primarily driven by the incoming declination and velocity. An extra month is added to the interplanetary cruise for the high end TOF to account for the multiple maneuvers.

Mars Transfer to Phobos. As with areostationary, Phobos also lies near the equatorial plane. A multi-burn strategy is needed to reduce the incoming inclination and circularize at Phobos. There is also an added need to match phase with the moon itself and perform rendezvous. The gravity is so low that an orbital insertion is not needed in the traditional sense. ΔV variation is driven by plane change and incoming velocity.

Mars Transfer to LMO. MOI directly to LMO can be performed by one large maneuver, or broken into multiple in order to avoid excessive gravity losses. It is generally advisable to keep burn times under 30 minutes. It is possible to achieve direct MOI to inclinations greater than the incoming declination. For near-equatorial orbits, a multi-burn plane change is generally needed. ΔV variation is primarily driven by incoming V_∞. ΔV [in km/s] can be found by:

$$\Delta V = \sqrt{\frac{2\mu_m}{3696} + V_\infty^2} - 3.4 \quad (3)$$

where μ_e = 42828 km^3/s^2.

Transfers from Mars Arrival (V_∞ = 0)

Mars Arrival as a starting point would not be very common as it is an intermediate point for most missions. It is also not well defined. The most likely scenario would be that of a SEP-powered host approaching Mars from a heliocentric transfer. Separating on arrival could allow a chemical-powered secondary to target a different orbit and inclination. For calculations, we will assume a drop-off point of around 1 million km, with a varying flight path angle (FPA).

Mars Arrival to 2-sol Orbit. A true V_∞ = 0 asymptote aimed at Mars (FPA ≈ 90°) would only need ~140 m/s to capture to a 2-sol orbit with an MOI at 300 km. A more circular approach (FPA ≈ 0) would be like a 2-burn orbital transfer with a total ΔV over 300 m/s.

Mars Arrival to Areostationary Orbit. Starting from the SOI, a secondary spacecraft can freely choose the inclination which is helpful for equatorial orbits. It is also more efficient to perform MOI directly to areostationary orbit rather than down at 300 km and subsequently raise periapsis.

Mars Arrival to Phobos. As with areostationary, direct access to an equatorial orbit is possible from V_∞ = 0. Direct MOI at Phobos's orbit is the most efficient method. Additional ΔV could be needed for maneuvers near the SOI and for phasing to rendezvous with Phobos.

Mars Arrival to LMO. The ΔV needed for MOI at LMO from V_∞ = 0 represents the minimum possible ΔV to achieve LMO without the use of aerobraking. ΔV variation is due to the range of possible drop-off states around the SOI.

Transfers from 2-sol Orbit

Areocentric orbit transfers from the highly elliptic 2-sol orbit usually consist of two combination maneuvers, one at each apse, to match the desired parameters and efficiently change inclination if required. Sometimes intermediate orbits may be employed.

2-sol Orbit to Areostationary Orbit. The first maneuver is at apoapsis to raise the periapsis to 17,031 km and reduce most (~95%) of the inclination to 0°. The second maneuver at the new periapsis circularizes and finishes the inclination change. Extra ΔV may be needed for nodal rotation or phasing in areostationary.

2-sol Orbit to Phobos. Similar 2-burn transfer strategy as with areostationary. Variation in ΔV due to amount of inclination change needed and phasing.

2-sol Orbit to LMO. Can be performed with one maneuver at 300 km if no inclination change is needed. Otherwise, and inclination change is first performed at apoapsis followed by a circularization burn at periapsis. Inclination accounts for most of the ΔV variation.

Transfers from Areostationary Orbit

Since areostationary is in or near the equatorial plane, it is well suited for transfers to the Martian moons. Deimos is just 3000 km above areostationary and requires just 100 m/s for the transfer.

Areostationary Orbit to Phobos. A near Hohmann transfer is all that is required to transfer to Phobos orbit. Some ΔV is needed for phasing and Phobos rendezvous.

Areostationary Orbit to LMO. A purely equatorial transfer requires the minimum ΔV, whereas the inclination change drives the requirement higher. The nominal value is given for a 30° change.

Transfers from Phobos

Phobos to LMO. Being even closer to Mars, inclination changes are even more costly. An equatorial transfer is around one third of the ΔV needed to go to a polar LMO. The nominal value is given for a 30° change.

SOLAR ELECTRIC PROPULSION TRANSFERS

Transfers using SEP are significantly different from their ballistic counterparts. Electric propulsion can provide 5 to 10 times greater specific impulse (Isp) than chemical, but produces up to 3 *orders of magnitude* less thrust. In order to gain the large ΔV's necessary for interplanetary transfers, SEP thrusters must operate nearly continuously for many weeks to months. This is fundamentally different than conventional trajectories that essentially have one large maneuver to escape Earth and another large braking maneuver to capture into orbit upon arrival. There are numerous methods to calculate and optimize the durations, locations, and directions of low-thrust maneuvering to find optimal transfers using electric propulsion [10][11][12]. For additional discussion on the application of SEP to Mars missions see previous papers by the author [13][14].

The fundamentally different way in which low-thrust missions work leads to vastly different trajectories, ΔV's, and TOFs. There is also a much wider array of possibilities in estimating ΔV and TOF ranges. Key parameters affecting the performance of SEP missions include: the thrust and Isp of the engine (along with their variation with respect to power), the power provided by the solar arrays, total propellant load, and the dry mass of the vehicle.

For the sake of ΔV and TOF estimation for this table, best practices and some rules-of-thumb were employed in order to provide reasonable values. For TOFs, an acceleration level of 0.1 - 0.2 mm/s^2 at 1 AU was assumed. (It has been found that SEP system designs that provide accelerations in this range are usually near mass optimal). ΔV's are usually inversely proportional with TOF, so one must pick a reasonable value or look for a "knee-in-the-curve". Minimum ΔV typically occurs at an infinite time which is not practical and are not used. On the other end, some types of transfers have a maximum ΔV corresponding to a minimum possible TOF. These values are given when available.

Heliocentric low-thrust trajectories were simulated and optimized using MALTO, a rapid, medium-fidelity low-thrust optimizer [15]. This tool can quickly calculate trajectories from Earth to Mars under a variety of conditions and constraints. MALTO also has the capability to add a circular capture spiral down to a desired orbit using the methods of Melbourne and Sauer [16]. This method analytically approximates the propellant mass and time necessary to complete the transfer, and is part of the optimization process. Where appropriate, MALTO data from thousands of simulations was used to provide ΔV and TOF ranges for the Table over a range of conditions.

Transfers from LEO using SEP

LEO as a starting point for a SEP-powered transfer to Mars is rarely a good idea. The primary challenge is the significant duration of the spirals just to leave Earth, which can be on the order of years. The other significant challenge is the duration spent passing through the Van Allen radiation belts and the accumulation of radiation dose. Significant shielding and preventive measures would be needed to assure success.

LEO to GTO. This is a spiral trajectory that begins circular and ends up highly elliptical, which is difficult to optimize without specific mission objectives. It is also not a common thing to do since a more likely destination would be GEO itself, which would require a ΔV around 5 km/s to spiral from LEO. GTO is common for chemical transfers, and is just given here for reference. This transfer would also have very significant radiation dosage.

LEO to LTO. As with GTO, spiraling from a circular orbit to a highly elliptical one rarely serves a purpose and is difficult to optimize. ΔV varies significantly with the

TOF allowed. If the moon were the desired destination, it would make a lot more sense for a circular spiral to lunar rendezvous, requiring around 7 km/s of ΔV.

LEO to Escape. "Escape" occurs when the continuously-thrusting SEP mission crosses the point of negative energy with respect to Earth to positive energy. An optimal Earth departure spiral from LEO can range from purely circular to partially elliptical depending on performance and objectives. The key is the amount to time it takes to raise the spacecraft velocity to escape velocity. The total ΔV needed can be approximated by the circular velocity of the departure orbit (about 7.7 km/s in LEO).

LEO to Mars Transfer. Mars Transfer is less defined for a SEP mission since trajectory does not necessarily reach a point where a ballistic coast to Mars is possible. Nearly continuous thrusting is needed for targeting and rendezvous. The values given in the table are approximately those needed to achieve a hyperbolic excess velocity (V_∞) of about 3.5 km/s.

LEO to Mars Arrival. Mars Arrival for a SEP trajectory is the point at which V_∞ crosses 0 and the spacecraft begins its spiral downwards. It is effectively at rest with respect to Mars and co-traveling around the sun. A transfer to here from LEO requires first the long spiral to escape Earth, followed by a heliocentric transfer accelerating away from Earth's orbit and matching velocity with Mars. The thrust duty cycle will typically be > 80-90%.

LEO to 2-sol Orbit. This is a combination of LEO to Escape and Escape to 2-sol orbit.

LEO to Areostationary Orbit. This is a combination of LEO to Escape and Escape to Mars Arrival.

LEO to Phobos. This is a combination of LEO to Escape and Escape to Areostationary Orbit.

LEO to LMO. This is a combination of LEO to Escape and Escape to LMO. It is the longest possible transfer included here and the highest ΔV more than 16 km/s. Even with SEP, this represents a significant amount of propellant, in addition to multiple years of transfer time.

Transfers from GTO using SEP

GTO may be a much better starting point for SEP and reasonably common as a rideshare. It is significantly higher in the gravitational well and affords easier departure from Earth.

GTO to LTO. This is a transfer between two highly elliptic orbits and is not likely to be a common need. Optimization of such a spiral trades the efficiency of thrust arcs near Earth with the long durations of coasting around apogee.

The ΔV can be approximated through the following method. First, we introduce a normalized parameter

$$u = (T/m_0) \cdot 0.9t \tag{4}$$

where T is the thrust in Newtons, m_0 is the initial spacecraft mass in *grams*, and t is the transfer time in seconds. The *0.9* accounts for the eclipse fraction and can be varied. Using the normalized parameter, we approximate the ΔV required from the engine to escape using a rational polynomial,

$$\Delta V(u) = \frac{p_1 u^2 + p_2 u + p_3}{u^3 + q_1 u^2 + q_2 u} \tag{5}$$

which expects u and ΔV to have units of km/s, and where $p_1 = 1.425$, $p_2 = -3.838$, $p_3 = -0.6359$, $q_1 = -4.263$, and $q_2 = 4.002$. Note that we require $u \geq u_{\min}$, where $u_{\min} = 3.2$ km/s corresponds to a minimum-time transfer from GTO to LTO.

GTO to Escape. The spiral to escape is highly elliptical and has a strong dependence on the amount to time allowed for the transfer. Optimization is balance of thrust arcs and coast arcs. As with GTO to LTO, the ΔV can be approximated through a similar method using a rational polynomial, as detailed in [13]. The same normalized parameter, u, is calculated using Eq. (4). The rational polynomial of Eq. (5) has an additional term in the denominator

$$\Delta V(u) = \frac{p_1 u^2 + p_2 u + p_3}{u^3 + q_1 u^2 + q_2 u + q_3} \tag{6}$$

where u and ΔV have units of km/s, and $p_1 = 25.34$, $p_2 = -208.8$, $p_3 = 500.4$, $q_1 = -3.909$, $q_2 = -15.91$, and $q_3 = 83.15$. We require $u \geq u_{\min}$, where $u_{\min} = 3.7$ km/s corresponds to a minimum-time transfer from GTO to Escape.

Figure 2 shows an example of Eqs. (5) and (6) for a starting acceleration level of 0.2 mm/s^2.

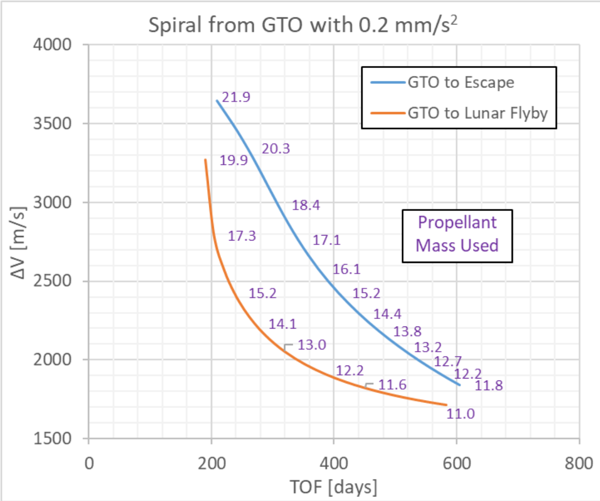

Figure 2 - ΔV vs. TOF for spirals from GTO. This example uses an acceleration level of 0.2 mm/s^2. Propellant is calculated using a 100 kg wet mass and 1500 sec. Isp.

GTO to Mars Transfer. This is a combination of GTO to Escape and Escape to Mars Arrival.

GTO to Mars Arrival. This is a combination of GTO to Escape and Escape to Mars Arrival.

GTO to 2-sol Orbit. This is a combination of GTO to Escape and Escape to 2-sol orbit.

GTO to Areostationary Orbit. This is a combination of GTO to Escape and Escape to Areostationary Orbit. A more detailed treatment of this transfer is the subject of [13].

GTO to Phobos. This is a combination of GTO to Escape and Escape to Phobos.

GTO to LMO. This is a combination of GTO to Escape and Escape to LMO.

Transfers from LTO using SEP

Starting at Lunar transfer orbit is similar to GTO with the added benefit of being even higher in the gravity well, making it that much easier to escape. Lunar flybys are also more easily targeted, which may reduce ΔV even further.

LTO to Escape. The spiral to escape is highly elliptical and has a strong dependence on the amount to time allowed for the transfer, similar to GTO to Escape. Optimized escape spirals would create a pareto front of ΔV vs. TOF similar to those in Figure 2.

LTO to Mars Transfer. This is a combination of LTO to Escape and Escape to Mars Arrival.

LTO to Mars Arrival. This is a combination of LTO to Escape and Escape to Mars Arrival.

LTO to 2-sol Orbit. This is a combination of LTO to Escape and Escape to 2-sol orbit.

LTO to Areostationary Orbit. This is a combination of LTO to Escape and Escape to Areostationary Orbit.

LTO to Phobos. This is a combination of LTO to Escape and Escape to Phobos.

LTO to LMO. This is a combination of LTO to Escape and Escape to LMO.

Transfers from Escape using SEP

Escape, and just beyond (C3 = 0-5 km^2/s^2), is a good starting point for SEP missions as it avoids lengthy spirals to leave Earth, utilizes the performance of the launch vehicle, and still allows for the best use of the highly-efficient SEP system.

Escape to Mars Transfer. Mars Transfer is not a destination for a SEP transfer, but more of an intermediate point. It is a continuation of the spiral to escape and represents the first part of the interplanetary portion where the heliocentric velocity is increased. The values in the Table represent a point where C3 ≈ 15 km^2/s^2 is crossed to be comparable to the ballistic transfers.

Escape to Mars Arrival. The ΔV for a low-thrust transfer from $V_\infty = 0$ at Earth to $V_\infty = 0$ at Mars is very close to the Hohmann transfer value (see Figure 1) and only varies slightly with different performance values and from opportunity to opportunity. The ΔV can be reduced almost linearly for the V_∞ at either end is positive. For example, $V_\infty = 0$ at Earth and $V_\infty = 2$ km/s at Mars (suitable for a direct entry) would reduce the ΔV values in the Table by ~ 2 km/s. A deeper discussion of analytic solutions to low-thrust ΔV estimation can be found in [17].

Escape to 2-sol Orbit. The values in the Table are a combination of Escape to Mars Arrival and Mars Arrival to 2-sol orbit. Due to the high eccentricity of the capture orbit, a more dedicated optimization algorithm may offer more refined results.

Escape to Areostationary Orbit. The transition through the Mars SOI using low-thrust is complex and not always intuitive. Thrusting to match Mars's velocity (i.e. bringing V_∞ to zero) is combined with targeting to begin the spiral in to Mars. This kind of combined maneuvering at the transition between gravitational spheres-of-influence leads to a total ΔV that is slightly lower than the sum total of the Mars rendezvous and spiraling down from escape.

An analytic approximation of the benefit of capturing from a heliocentric orbit down to a circular orbit is described by Melbourne and Sauer [16]. MALTO, utilizing this approximation, was used to simulate transfers from Escape to Areostationary and the other circular orbits.

Escape to Phobos. See Escape to Areostationary Orbit.

Escape to LMO. See Escape to Areostationary Orbit. Note that for spirals that go all the way to LMO, shadowing becomes a significant factor during the final orbits (weeks to months). The duty cycle can be reduced by as much as 40% to account for eclipses. This is primarily manifest in increased TOF.

Transfers from Mars Transfer (TMI) using SEP

Low-thrust optimization software such as MALTO have the ability to select the launch C3 for a given launch vehicle that will result in the maximum delivered mass for given mission parameters. The optimum is generally lower than the ballistic C3, usually around $0 – 10\ km^2/s^2$. However, in cases where there is an excess of launch vehicle capability, or a shorter TOF is desired, it is possible to have a much higher C3. As C3 increases, the ΔV required by the SEP system decreases at first, but then begins to increase as more and more ΔV will be required to slow down at Mars arrival.

Mars Transfer as a starting point, and for the values in the Table, is taken to be a typical ballistic TMI – around $15\ km^2/s^2$, which is a bit higher than mass-optimal, but might be a feasible starting point as a rideshare with other Mars-bound missions.

Mars Transfer to Mars Arrival. The additional boost of a positive C3 at Earth allows for less ΔV from the SEP system to complete the transfer. The lowest possible ΔV comes from the equivalent to a ballistic transfer from Earth followed by the ΔV needed to rendezvous with Mars (i.e. the second half of a Hohmann transfer). Varying conditions and dates cause the ranges given in the Table.

Mars Transfer to 2-sol Orbit. The ΔV and TOF values given in the Table were simulated over a range of parameters using MALTO. The loosely-captured nature of a 2-sol orbit leads to ΔV's and transfer times that are only slightly higher than a transfer to Mars Arrival, taking advantage of combined maneuvering and efficiencies of a close approach. A longer spiral with shorted thrust arcs leads to the minimum ΔV but the longest total TOF. The orientation of the elliptical orbit may lead to increased shadowing near periapsis and lead to the higher values of ΔV and TOF.

Mars Transfer to Areostationary Orbit. MALTO simulations over a range of parameters was well suited to predict the ΔV and TOF necessary for this transfer. SEP is well-suited to target areostationary orbit as equatorial orbits do not cost any additional ΔV and circular spirals are naturally efficient.

Mars Transfer to Phobos. Similar to Mars Transfer to Areostationary. Ranges were estimated using MALTO.

Mars Transfer to LMO. Similar to Mars Transfer to Areostationary. Ranges were estimated using MALTO.

Transfers from Mars Arrival ($V_\infty = 0$) using SEP

Mars Arrival represents the start of the spiral-down phase of a SEP mission. In the weeks or months leading to the rendezvous with Mars, the desired plane (both inclination and node) is targeted such that the spiral results in the desired final orbit.

Mars Arrival to 2-sol Orbit. A very rough way to approximate this transfer is to calculate the ΔV for a spiral from 100,000 km circular to a 2-sol orbit. Even this method leaves a trade off between TOF and ΔV. In some cases, a plane change may also be needed, increasing the TOF.

Mars Arrival to Areostationary Orbit. The ΔV and TOF values for this transfer were estimated by extracting the spiral portions of MALTO trajectories, noting that MALTO uses the Melbourne and Sauer method to approximate a spiral down to a circular orbit starting from a heliocentric transfer.

Mars Arrival to Phobos. The same method was used here as with Mars Arrival to Areostationary – extracting the spiral data from MALTO trajectories.

Mars Arrival to LMO. The same method was used here as with Mars Arrival to Areostationary – extracting the spiral data from MALTO trajectories.

Transfers from 2-sol Orbit using SEP

Spirals from a 2-sol elliptical orbit are not likely unless MOI was performed using chemical propulsion. In which case, the spirals to the following orbits start out highly elliptical and transition to circular as inclination is also matched. An optimized thrusting profile will balance the minimization of transfer time with the maximization of thrusting efficiency.

2-sol Orbit to Areostationary Orbit. The variation in the ΔV for this transfer primarily depends on the inclination of the initial elliptical 2-sol orbit.

2-sol Orbit to Phobos. Similar to the transfer to areostationary, with most of the inclination change initially, followed by more circular, equatorial spiraling near the end.

2-sol Orbit to LMO. Both of the orbits have the potential for an arbitrary inclination and node. During the spiral down, the non-spherical gravitational perturbations will also affect the node and must be accounted for in the optimized thrust profile.

Transfers from Areostationary Orbit using SEP

Transfers between two circular orbits, even with different inclinations, can be estimated quite accurately using the Edelbaum approximation [18]. The resulting transfer is circular with nearly continuous thrust and an optimized plane change profile.

Areostationary Orbit to Phobos. Since both orbits are coplanar, the Edelbaum approximation for the ΔV is reduced to the difference between the circular velocities at either orbit.

Areostationary Orbit to LMO. The variation in the ΔV for this transfer primarily depends on the inclination of the desired LMO. An Edelbaum approximation was used to simulate a nominal 30° change, whereas the high and low values for the table are for 90° and 0°, respectively.

Transfers from Phobos using SEP

Phobos to LMO. Similar to Areostationary to LMO.

CONCLUSION

The primary intent of this paper is to serve as a quick reference for any transfer between Earth and Mars. The Table on the last page can be printed or kept readily on hand to look up representative values and potential ranges of ΔV's and TOF's for any transfer combination. Having two sections, one for chemical and one for SEP, allows the user to quickly see the difference that propulsion type has on a potential mission, both in ΔV and duration. Basic rocket equation calculations with estimated Isp's and dry masses will give propellant estimates and wet mass. This information can be used for feasibility and to assess technology needs.

Many of the starting/stopping combinations do not represent rational missions, but are included for completeness. The respective sections throughout this paper describe the nature of each transfer combination, assumptions, methods, and causes for the values and variations provided in the Table, as well as other pertinent considerations.

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ΔVs and Times-of-Flight for Earth to Mars Transfers

Chemical Propulsion	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High
To $_v$/From$^{->}$ (*TOF - From $_v$ /To$^{->}$*)	**LEO**			**GTO**			**LTO**			**Escape**			**Mars Transfer**			***Mars Arrival***			**2-sol orbit**			**Areostationary**			**Phobos**			**LMO**		
Low-Earth Orbit (LEO)				*0 hrs*	*5.3 hrs*	*7 hrs*	*3 days*	*5 days*	*15 days*	*5 days*	*10 days*	*20 days*	*5 days*	*6 days*	*10 days*	*6 mos.*	*10 mos.*	*13 mos.*	*6 mos.*	*10 mos.*	*13 mos.*	*6 mos.*	*10 mos.*	*14 mos.*	*6 mos.*	*10 mos.*	*14 mos.*	*6 mos.*	*10 mos.*	*14 mos.*
Geosynch Tranfer Orbit (GTO)	2.4	2.5	3																											
Lunar Transfer Orbit (LTO)	3.1	3.2	3.6	0.7	0.7	1.5																								
Escape	3.2	3.3	3.5	0.7	0.8	0.9	0.1	0.2	0.3				n/a	n/a	n/a															
Mars Transfer (C3 ~ 15)	3.9	4	4.2	1.4	1.5	1.9	0.7	0.9	1.3	0.7	0.9	3																		
Mars Arrival (V_∞ = 0)	*4.5*	*4.8*	*5.7*	*2*	*2.3*	*3.4*	*1.3*	*1.7*	*2.8*	*1.3*	*1.7*	*4.5*	*0.6*	*0.8*	*1.5*				*25 hr*	*50 hr*	*100 hr*									
2-sol orbit	4.7	5	5.9	2.2	2.5	3.6	1.5	1.9	3	1.5	1.9	4.7	0.8	1	1.7	0.2	0.2	0.3				*25 hr*	*35 hr*	*70 hr*	*20 hr*	*28 hr*	*56 hr*	*0 hr*	*25 hr*	*30 hr*
Areostationary Orbit	5.6	6	6.8	3.1	3.5	4.5	2.4	2.9	3.9	2.4	2.9	5.6	1.7	2	2.6	0.8	0.9	1.1	0.6	0.7	0.8				*8 hr*	*8 hr*	*16 hr*	*6 hr*	*6 hr*	*12 hr*
Phobos	6	6.3	7.5	3.5	3.8	5.2	2.8	3.2	4.6	2.8	3.2	6.3	2.1	2.3	3.3	1	1.1	1.3	0.8	0.9	1	0.7	0.7	0.8				2 hr	2 hr	4 hr
Low Mars Orbit (LMO)	6	6.5	7.5	3.5	4	5.2	2.8	3.4	4.6	2.8	3.4	6.3	2.1	2.5	3.3	1.4	1.5	1.8	1.3	1.3	1.5	1.7	1.9	2.7	1.2	1.7	3.3			
ΔV [km/s]																														

TOF

Solar Electric Propulsion	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High	Low	Most Likely	High
To $_v$/From$^{->}$ (*TOF - From $_v$ /To$^{->}$*)	**LEO**			**GTO**			**LTO**			**Escape**			***Mars Transfer***			**Mars Arrival**			**2-sol orbit**			**Areostationary**			**Phobos**			**LMO**		
Low-Earth Orbit (LEO)				*9*	*18*	*36*	*12*	*24*	*40*	*12*	*24*	*45*	*19*	*33*	*57*	*23*	*39*	*69*	*24*	*40*	*74*	*26*	*42*	*73*	*25*	*42*	*73*	*26*	*44*	*75*
Geosynch Tranfer Orbit (GTO)	2.8	3.8	5.5				*6*	*12*	*24*	*7*	*15*	*24*	*14*	*24*	*36*	*18*	*30*	*48*	*19*	*31*	*53*	*21*	*33*	*52*	*20*	*33*	*52*	*21*	*35*	*54*
Lunar Transfer Orbit (LTO)	3.5	6.4	8	1.6	2.2	3.2				*2*	*4*	*6*	*9*	*13*	*18*	*13*	*19*	*30*	*14*	*20*	*35*	*14*	*20*	*32*	*15*	*22*	*34*	*16*	*24*	*36*
Escape	6.8	7.5	8	2	2.6	3.7	0.8	1.7	3				*7*	*9*	*12*	*11*	*15*	*24*	*12*	*16*	*29*	*12*	*16*	*26*	*13*	*18*	*28*	*14*	*20*	*30*
Mars Transfer (C3 ~ 15)	10.3	11.3	12	5.5	6.4	7.7	4.3	5.5	7	3.5	3.8	4				11	13	16	*11*	*15*	*21*	*11*	*15*	*21*	*12*	*16*	*23*	*15*	*18*	*25*
Mars Arrival (V_∞ = 0)	12.4	13.2	13.8	7.6	8.3	9.5	6.4	7.4	8.8	5.6	5.7	5.8	2.6	3.2	4.4				*0.5*	*1*	*5*	*0.7*	*1*	*2*	*1.5*	*3*	*4*	*3*	*5*	*8*
2-sol orbit	12.9	13.9	15	8.1	9	10.7	6.9	8.1	10	6.1	6.4	7	3	3.6	4.8	0.5	0.7	1.2				*1*	*2*	*5*	*1.5*	*2.5*	*6*	*3*	*5*	*9*
Areostationary Orbit	13.2	14.2	15.5	8.4	9.3	11.2	7.2	8.4	10.5	6.4	6.7	7.5	3.6	4.3	5	0.8	0.9	1	0.9	1.3	2				*1*	*2*	*3*	*4*	*8*	*16*
Phobos	13.9	14.9	16.2	9.1	10	11.9	7.9	9.1	11.2	7.1	7.4	8.2	4.3	4.9	6	1.5	1.6	1.9	1.4	2	3	0.7	0.7	0.8				*2.5*	*9*	*18*
LMO	15	16.1	17.5	10.2	11.2	13.2	9	10.3	12.5	8.2	8.6	9.5	5.6	6.3	7	2.7	2.9	3.2	2.2	2.8	4	2	2.6	4.6	1.3	3.3	5.2			
ΔV [km/s]																														

TOF [months]

Notes and assumptions:

Ranges given are for "reasonable" mission designs. (e.g. non-infinite trip times, Isp/thrust of typical thrusters, nominal plane changes). ~ 1-sigma range.

ΔVs on Left/Lower portion, TOFs on Right/Upper portion. Transfers are intended to be in Earth -> Mars direction, but can be assumed to be reciprocal.

Mars V_∞ =0 isn't a typical intermediate point for ballistic transfers. The ΔVs shown are for an MOI to become loosely captured.

The point "C3=15" is not well-defined for low-thrust, except where it is the point where a LV has injectected the S/C.

SEP TOFs based on accelerations of 0.1 - 0.2 mm/s^2 at 1 AU. Thrust profiles optimized. Some shadowing where appropriate.

Ballistic TOFs for Earth escapes are approx. to the SOI crossing. TOFs for in-system ballistic transfers are the durations from peri- to apoapsis, where defined

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LEO - 200 x 200 km	**Arrival** - V_∞ = 0 km/s
GTO - 200 x 35786 km	**2-sol** - 300 x 57826 km
LTO - 200 x 384000 km	**Areo** - 17031 km x 0 deg
Escape - Vinf = 0 km/s	**Phobos** - 5980 km x 1 deg
TMI - C3 ≈ 15 km^2/s^2	**LMO** - 300 x 300 km