Transformational Propulsion for In-Space Fast Transits

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NASA has ambitious mission objectives requiring much faster transits to and from Mars, interstellar probes and a gravity lens observatory. However, there has been very little investment in transformational propulsion system development activities since the 1970s. There have been dozens of concepts proposed for high acceleration propulsion solutions, but nearly all have been limited to paper studies. A quick assessment was completed in 2023 to determine the potential of transformational propulsion concepts to enable faster transits, with a focus on crewed missions to and from Mars. Far-term conventional nuclear thermal propulsion is limited to very modest improvement in transfer times. Advanced nuclear propulsion options have significant performance potential, but limited interim evolutionary payoff for NASA. Advanced electric propulsion, both solar and nuclear, offers transformational performance potential with high payoff during interim progress for power production, conversion, and heat rejection technologies. Study results and recommendations for near-term investments are presented herein.

I. Nomenclature

α	=	mass-to-power ratio, kg/kW
ΔV	=	change in velocity, km/s
η	=	efficiency
C_3	=	characteristic energy, km^2/s^2
F	=	thrust
g	=	acceleration of gravity, 9.81 m/s^2
I_{SP}	=	specific impulse
P	=	power

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II. Introduction

The RL10 engine represents the State-of-the-Art (SOA) in high performance exploration class propulsion. The newly operational Space Launch System (SLS) uses an RL10 engine for the Interim Cryogenic Propulsion Stage (ICPS) for Artemis 1, 2 and 3. The SLS Block 1B evolution will employ four RL10 engines on the Exploration Upper Stage (EUS). This is 60 years after the RL10 was first flown in 1963; first tested in 1959. There are significant advancements in materials, tweaks to propellant chemistry and major shifts in manufacturing. The EUS will leverage the RL10C-X that includes 3-D printed elements to reduce production times and cost. However, chemical propulsion is largely still deflagration of a combustion product passed through a choked nozzle and expanded for thrust. Electric propulsion is similar with new materials, new propellants, and slightly larger scales, but is fundamentally the same technology for Hall Effect thrusters and gridded ion engines since first flown in 1972 and 1964 respectively. Ambitious propulsion technology programs for transformational space transportation existed in the past, such as Project Orion ending in the early 1960s and the Nuclear Engine for Rocket Vehicle Application (NERVA) project that ended in the early 1970s. For the past 50 years, transformational propulsion system development has largely stagnated. There continues to be a large number of concepts with viable technology options, many coming from the NASA Innovative Advanced Concepts (NIAC) program, Department of Energy (DOE) studies, derived systems from Advanced research Projects Agency- Energy (ARPA-E) investments, etc., but most are limited to paper studies with no integrated strategic system development planned. The NASA Space Technology Mission Directorate (STMD) requested an assessment of technology options to enable transformational space transportation beyond where evolutionary system performance plateaus and recommend near-term or sustained investments to validate the efficacy of these new concepts and/or make meaningful progress towards the maturation of a far-term fast transit capability.

III. Fast Transits Envisioned Future State

NASA and the aerospace community has a range of missions that benefit and are potentially enabled through transformational propulsion. For decades, the driving exploration transportation challenge is for rapid crewed transits to Mars. Transit times to Mars drives the total energy required, with decreasing times exponentially increasing the Earth departure mass. Mission duration also impacts crew health, overall system reliability performance and logistical and commodity burdens. Traditional reference approaches are multi-year missions that never deviate far from Hohmann near-impulsive trajectory solutions as shown in figure 1.[1]

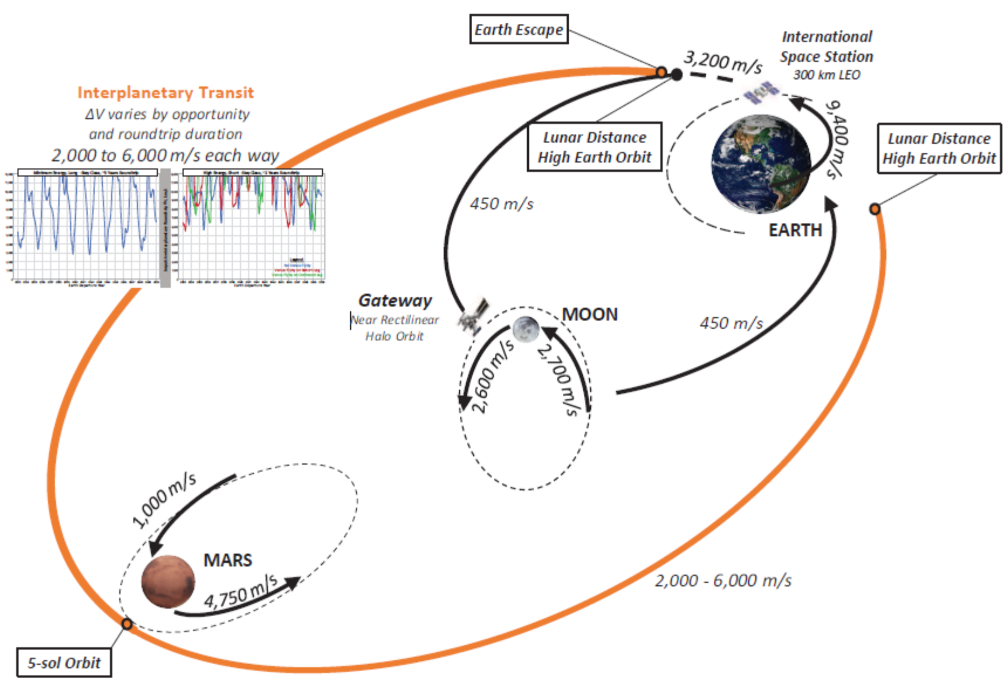


Figure 1: Notional Evolutionary Earth-Moon and Earth-Mars energy map. [1]

The NASA Exploration Systems Development Mission Directorate (ESDMD) Mars Architecture Team (MAT) has been conducting trade studies for the range of Mars transportation propulsion options with sensitivity analyses to the roundtrip mission duration and Mars vicinity stay time.[2] However, the total trip interplanetary durations considered are all approximately 2–3-year missions. The ESDMD studies are limited to these roundtrip duration ranges due to the rapid increase in mission ΔV, and resulting Earth Departure mass requirements, as the transit durations are pushed far from the optimized impulsive solution. Propulsion options primarily considered include chemical propulsion, nuclear thermal propulsion (NTP), and hybrid approaches combining chemical propulsion with either Solar Electric Propulsion (SEP) or Nuclear Electric Propulsion (NEP). The electric propulsion assessments are highly sensitive to the allowable transfer times and the propulsion power level. Propulsion solutions all have a clear knee in the curve where the propulsion system becomes impractical for the increasing ΔV requirements. The 2039 Mission opportunity sensitivity assessment by technology is shown in figure 3. The results are normalized to an 850-day roundtrip mission with a minimum Mars vicinity stay time of 50 sols.

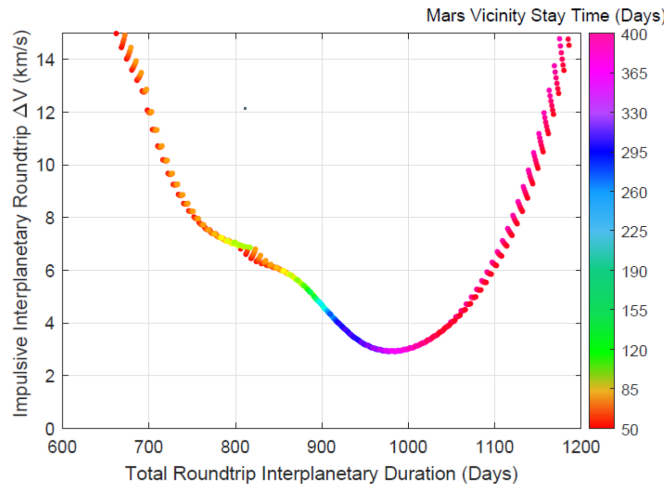


Figure 2: Interplanetary ΔV sensitivity to mission duration. [2]

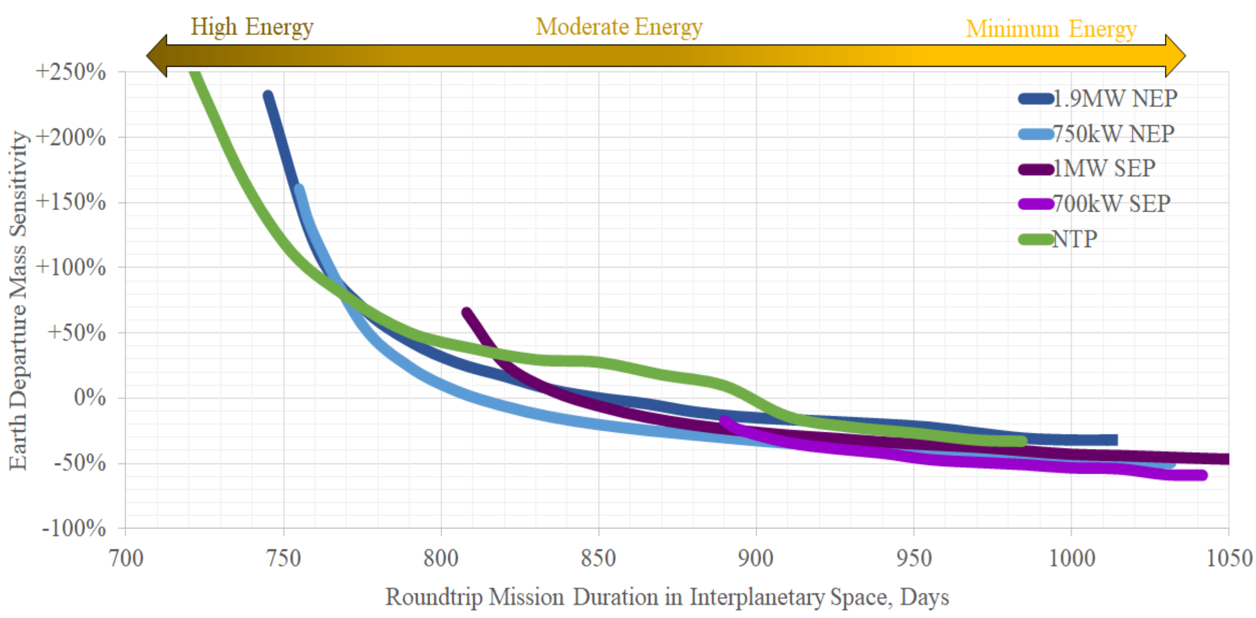


Figure 3: Notional Evolutionary Earth-Moon and Earth-Mars energy map. [2]

None of the propulsion solutions considered for the transformational. Even the use of Nuclear Thermal Propulsion is only a factor of two increase in specific impulse. While significant, it is not a viable technology to obtain transfer times much less than two years. The ongoing studies highlight that the envisioned future state for the next twenty years at NASA, at best, will be limited to approximately two-year mission durations. The motivation for this study is to take a longer-term vision to determine what approaches may eventually enable truly transformational propulsion solutions are crewed roundtrip missions less than 1 year, ideally much less than 1 year; acknowledging the challenges are exponentially increasing.

IV. Performance and Technology Trades

The scope of the performance trades initially included investigating chemical propulsion limitations, advanced nuclear thermal propulsion, electric propulsion, both solar and nuclear, and pulsed nuclear technologies. To simplify the analysis, the trades all assumed a departure energy or C_3 of 0 km^2/s^2. However, the trades included a range of Mars arrival and departure conditions that ranges from unconstrained flyby solutions, reference MAT 5-sol orbits, and down to 500km circular orbits. Also, trades included Earth return arrival conditions of unconstrained returns, direct entry limited by evolutionary heat loads on the thermal protection system and returning to the Lunar Distance High Elliptical Orbit (LDHEO). The analyses ranged from one-way transits to Mars from 200 days down to as low as 25 days and round-trip mission durations between 400 days and 50 days. High thrust, solutions included specific impulses ranging from current LOx/LCH4 (360s) capabilities through advanced (1800s) Nuclear Thermal Propulsion. Low-thrust finite burns ranged in specific impulse from 2,000s up to 20,000s, and included fixed and variable specific impulse. After preliminary results, the study focused on aggressive approaches for low mass-to-power propulsion solutions, consistent with earlier approaches for rapid transits to Mars.[3, 4] All the detailed ground rules and assumptions, and complete set of results, are provided in the Appendix.

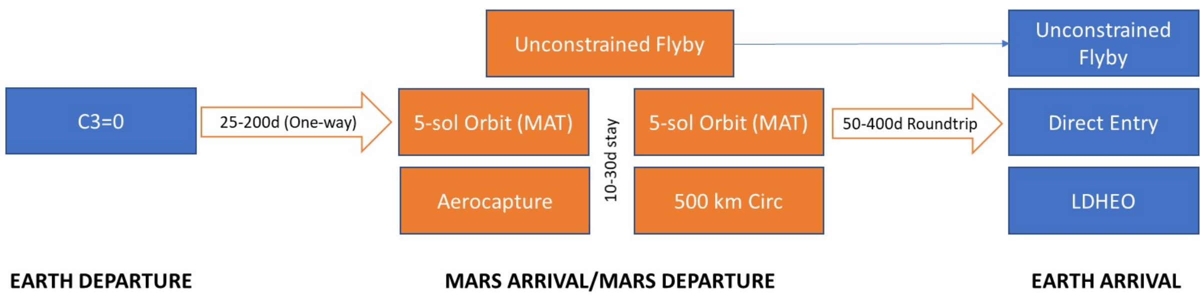


Figure 4: High level trade space of departure and arrival conditions.

The first portion of the study focused on determination of the near-impulsive high thrust ΔV requirements for the easier one-way transfers to Mars and then added the capture and return mission requirements. The ΔV for the one-way transits to Mars with an unconstrained flyby asymptote to as low as 3km/s, remained relatively flat down to ~4 months transits, but then rises rapidly by an order of magnitude approaching 25-day transits. The roundtrip missions are daunting, from approximately 6.6km/s for the 850-day missions in figures 1 and 2 to again, an order of magnitude increase to obtain 6-month round-trip missions.

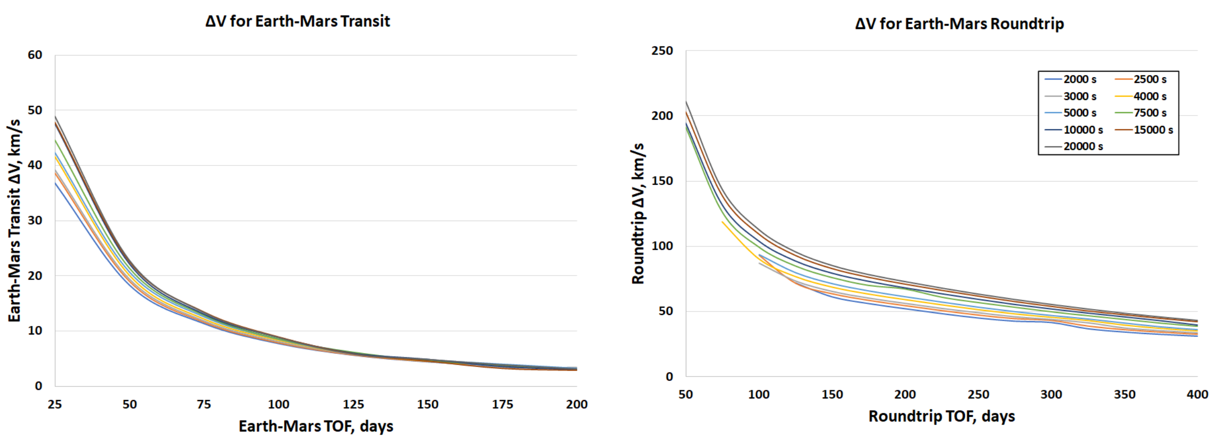


Figure 5: Mission ΔV requirements as a function of one-way (left) and roundtrip (right) transfer times.

The remarkably high ΔV requirements for very fast transits quickly highlighted that traditional chemical propulsion and even far-term advanced nuclear thermal propulsion solutions are completely non-viable for 6-month missions and likely true for a 1-year round-trip mission. A quick calculation using the rocket question provides insight into practical limitations.

$$MR = \frac{m_f}{m_0} = e^{\frac{\Delta v}{g I_{SP}}} \tag{1}$$

For a launch vehicle, propellant mass fractions are viable and can exceed 90%. Mars transportation solutions can leverage staging of drop-tanks throughout the mission. However, longer duration systems and in-space vehicle mass fractions are typically much lower. An advanced mars transportation is likely to be closer to launch vehicle with respect to propulsion system performance and propellant mass fractions. The Mars Transportation Architecture Study (MTAS) designs had a calculated 90% propellant mass fraction for the Nuclear Thermal Propulsion approach and 75% propellant mass fraction for the NEP/Chem hybrid solution.[5] If we bound the required payload mass fraction to then be at least 10% and ideally greater than 25% of the departure mass, it becomes clear that only very high specific impulse solutions are viable. Far-term advanced nuclear thermal propulsion may one-day be viable for 1-year roundtrip missions but are not plausible for 6-month roundtrip missions.

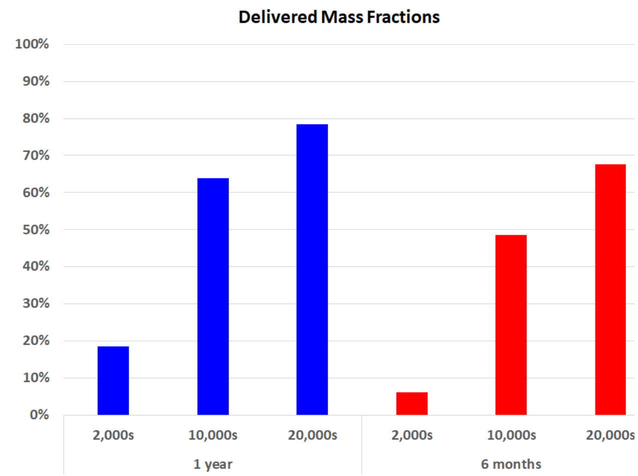


Figure 6: Delivered mass fractions limited by specific

As we evolve propulsion solutions up the specific impulse curve, with temperature becoming the driving limitation for nuclear thermal propulsion solutions, we can only plausibly achieve ~3-month transits to Mars. For high thrust propulsion solutions, the departure maneuvers to Mars are typically chosen to reduce the trip-time until the ΔV starts to increase relatively rapidly. If we hold the departure mass constant, we can reduce our transit time by ~30% as we change technologies from traditional chemical propulsion to near-term (900s) nuclear thermal propulsion, with a doubling of the specific impulse. However, doubling the specific impulse again (1800s), with exponentially more challenging technology requirements, we can only gain another ~15% trip time reduction. We are at diminishing returns for trip time reductions as we approach the most advanced concepts for plausible nuclear thermal solutions as illustrated in figure 7. Holding departure mass constant, the propulsion system specific impulse must increase at the same rate as the mission ΔV for viability.

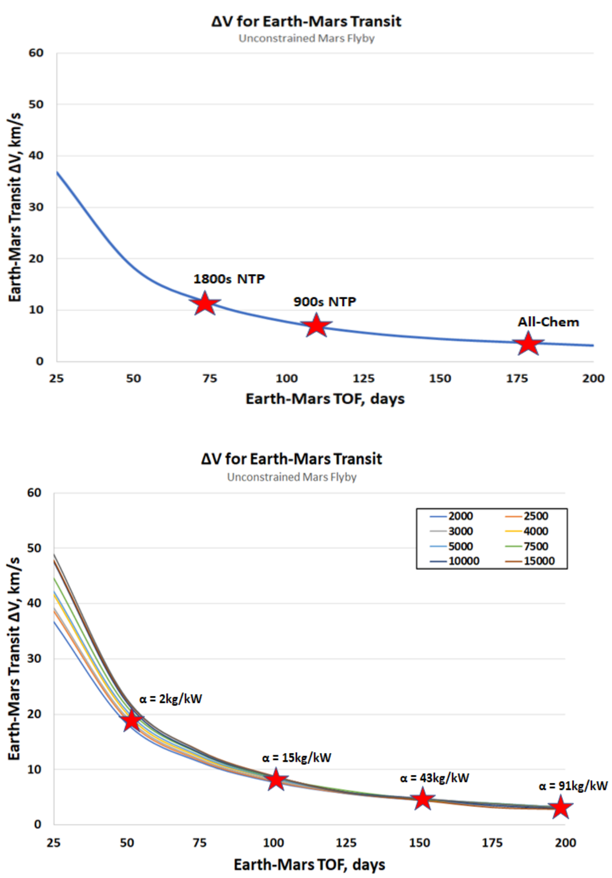


Figure 7: Constant departure mass trip times by key driving parameters for NTP (top) and EP (bottom).

With chemical propulsion and advanced nuclear thermal propulsion approached essentially eliminated from the viable trade space, the study focused on electric propulsion and advanced nuclear alternatives. To communicate the driving key performance parameters, results focused on required effective alpha. The effective alpha required to achieve rapid transits to Mars is also shown in figure 7; relatively insensitive to specific impulse for long transfer times, but highly sensitive at the most aggressive mission performance targets.

NASA is making investments in both Nuclear Electric Propulsion and Nuclear Thermal Propulsion through the STMD Space Nuclear Propulsion Program. For NTP, the largest investment is for the DARPA DRACO partnership for a rapid NTP demonstration in 2027.[6] In addition to DRACO, the Space Nuclear Propulsion (SNP) program is investing in fuel development and qualification to achieve higher temperatures necessary to obtain the 900s specific impulse target for the program beyond DRACO.

A. Nuclear Electric Propulsion Focus

Electric propulsion has known approaches to achieve specific impulses greater than 2,000s with relatively mature concepts. For Nuclear Electric Propulsion system development, there are multiple Critical Technology Elements as illustrated in Figure 8.[7] The driving elements are the reactor and coolant system (CTE 1), the power conversion subsystem (CTE 2), power management and distribution (CTE 3), the electric propulsion subsystem (CTE 4) and the primary heat rejection system (CTE 5). Solar electric propulsion (SEP) alternatives have reduced complexity with CTE 4 (i.e. the thrusters) essentially the same regardless of the power source. Solar arrays would replace both the reactor and cooling subsystem and the power conversion elements, CTEs 1 and 2, and eliminate the need for a separate power conversion heat rejection system.

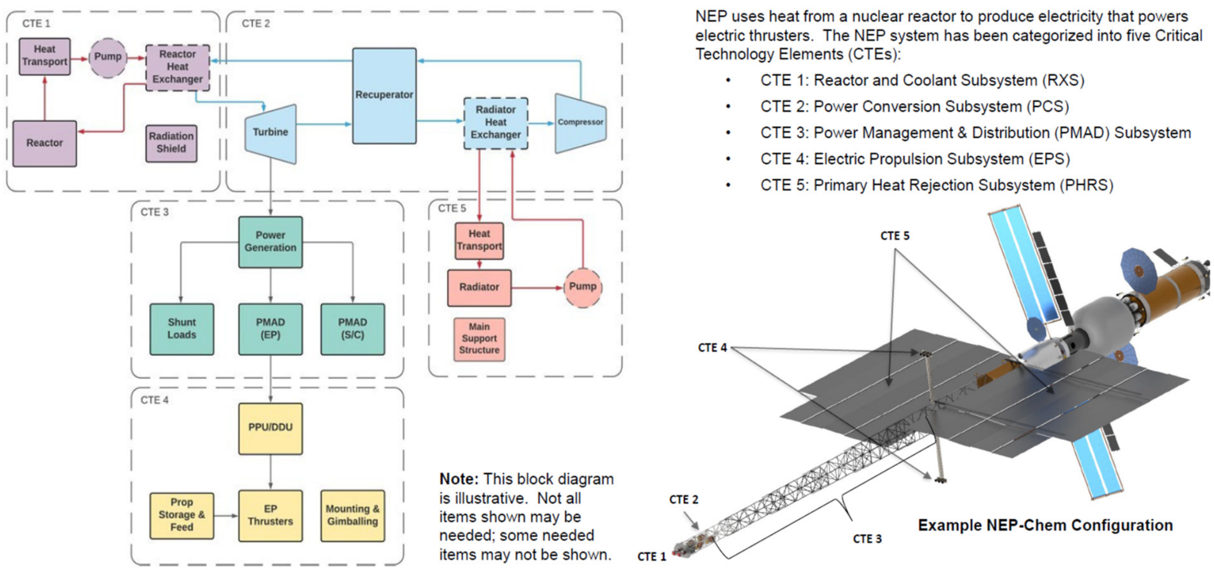


Figure 8: Critical Technology Elements for Nuclear Electric Propulsion.[7]

NASA established a technology maturation plan (TMP) with an approach to mature all the NEP critical technology elements.[7] The CTEs are then subdivided into major assemblies such as Brayton engines, radiators, thrusters, power processing units, etc. Many of the CTEs are interdependent and optimization of an NEP system requires interface control. The SNP office performed a series of community wide Technical Interchange Meetings (TIMs) regarding each of the CTEs. Based on the community input, the program established threshold and goal metrics for Key Performance Parameters (KPPs) of each of the elements, with the subsystem alpha a driving measure of performance. Table 1 includes the current state-of-the-art (SOA) alpha of each of the CTEs in addition to the threshold and goal metrics of performance. State of the art is defined as requiring only engineering development opposed to fundamentally new technologies to achieve the desired performance. This study team evaluated the limits for far-term evolutionary technology maturation and far-term technology potential with a shift in the underlying technology. Since the SNP office has a defined path to achieve the goal performance metrics, the threshold goals are assumed to be near-term achievable with the mid-term representing the planned goal performance. From figure 7, electric propulsion approaches are already competitive with conventional propulsion options. The goal performance metrics are not planned to achieve transformational performance, i.e total system alpha < 5 kg/kW and target < 1kg/kW for 6-month roundtrip missions.

Table 1: MW Class NEP Subsystem SOA and target alpha metrics.

Subsystem	SOA	Near Term	Mid-Term
CTE-1: Power Generation	16.5	6	4.5
CTE-2: Power Conversion	10	3	2
CTE-3: PMAD	6	4	2
CTE-4: Propulsion	5.5	5	2
CTE-5: Heat Rejection (w/o deployment mechanisms)	22 (13)	(6)	(2.5)
Total System, kg/kWe	51	24	13

1) Heat Rejection (CTE-5)

The heat rejection subsystem is the biggest opportunity for system alpha improvement. The SOA for large scale space heat rejection is the International Space Station (ISS), shown in figure 9. The biggest return on investment is simply to increase the heat rejection temperature from the SOA 300k to 600K. This increase in temperature is an evolutionary approach largely retaining conventional systems and materials. We could continue to gain performance for the heat rejection system by transitioning to new materials and transport fluids but increasing the heat rejection temperature will be penalize the power conversion efficiency. Without any fundamental shift in the underlying technology, there is low risk with a direct path to achieve a significant reduction in alpha. However, the heat rejection system alone will prohibit ever achieving transformation electric propulsion system performance without also a transformational approach in heat rejection.

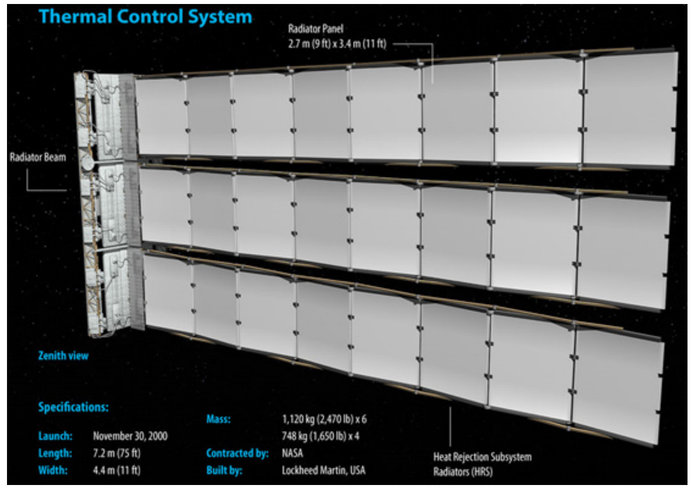


Figure 9: ISS Heat Rejection System.

Multiple concepts have been proposed for a transformational heat rejection system. First, one could consider a droplet radiator. While the technology has been in the literature for over a decade, there has been no technology development to raise the concept maturity. Droplet radiator technologies have the potential to provide step change in overall nuclear power system performance through dramatically increasing the temperature at which a radiator system can operate, thereby shrinking the mass by opening the working fluid to free space. While these technologies have been examined in their simplest form, an open jet of liquid droplets in free space vacuum is challenged by its working fluid thermophysical properties.

A slight variation of the droplet radiator may include shaped jet radiators that could create planar sheets of working fluid that are transported through an optically transparent cavity. In order to provide efficient radiation, the working fluid could radiate through the optical cavity at 1000ºC -1500ºC. The thin shell optical cavity will allow for transmission of the radiated energy through a sapphire cover. The internal atmosphere may be held above the ambient vacuum with an inert atmosphere to reduce or vapor-pressure related losses from selected working fluids. Examples include lead-antimony, sodium-potassium alloys or mixtures. In the example for a lead-antimony jet-sheet radiator verses a NaK-78 traditional radiator; the reduction in radiator area is highly significant. Here, the planar-sheet radiator area operating at 1300ºC could be as little as 7-8% of the area required for a traditional NaK78 based radiator operating at 550ºC. In other words, a 92 % to 93% reduction in radiator area.

Oscillating heat pipes have also received significant attention with clear performance advantages. A notional concept to achieve transformational heat rejection could be a dual-purpose structural oscillating heat pipe. All of the NEP concept designs, one example in figure 8, include large truss structures for standoff distance between the reactor and primary vehicle. Conventional radiators are often deployed form the truss structure. If the truss structure itself could be comprised of higher temperature oscillating heat pipes, to meet surface area and view factor constraints, one could theoretically achieve a primary heat rejection performance of ~0 kg/kW at large scale.

2) Propulsion Subsystems (CTE-4)

NASA and the Air Force has intermittently invested in high power electric propulsion concepts. Traditionally the propulsion systems have outpaced the power system development. There are also a range of technology options

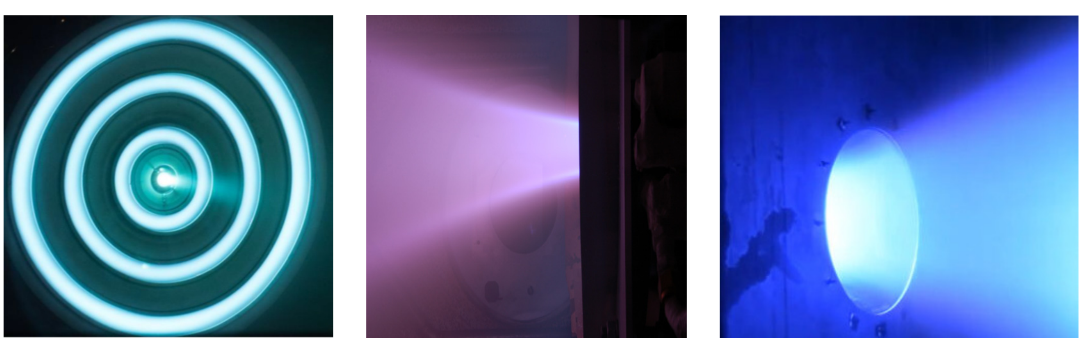


Figure 10: X-3 Hall Effect Thruster (left), MPD (center) and VASIMR (right) thruster options.

including an array of mature concepts such as Hall Effect Thrusters (HET) and Gridded Ion Engines, or fewer higher power devices such as a MagnetoPlasmaDynamic (MPD) thrusters, a Pulsed Inductive Thruster (PIT) or the Variable Specific Impulse Magnetoplasma Rocket (VASIMR). The reference case in the MTAS report leveraged an array of 100kWe HETs based on the NASA-457M demonstrated for performance and 100 hours of operation. under the NASA NextSTEP High Power Electric Propulsion investment. The MTAS reference design had a calculated subsystem alpha of ~2.5kg/kWe. The X-3 Hall Effect thruster, also designated XR-100, with concentric nested channels is designed with a wide power range up to 250kWe and is anticipated to yield an alpha below 2 kg/kWe for the subsystem, approaching ~0.5kg/kWe for the thruster itself and the Power Processing Unit (PPU) and flow control system adding another ~1kg/kWe. The X-3 project demonstrated a thruster specific power of 1.25kg/kWe with a path for mass reduction. HET technology is anticipated to be a highly competitive option at megawatt class system performance. As the power level increases, the scalability of the MPD offers packaging advantages. E.g. an MPD architecture may leverage 3 x 500kWe MPDs over 15 x 100kWe HETs. Princeton University and the Jet Propulsion Laboratory (JPL) have been making small investments in lithium based MPDs. If systems begin to exceed 10s of megawatts, the complexity of HET architectures and eventually even the MPDs may be become impractical for vast arrays of thrusters. In these cases, a device, such as VASIMR, may offer clear performance advantages at these very high-power levels. Figure 11 provides notional mass-to-power performance for total system options based on the various thrusters. At 100MWe class systems, an alpha much less than 1kg/kWe appears plausible.

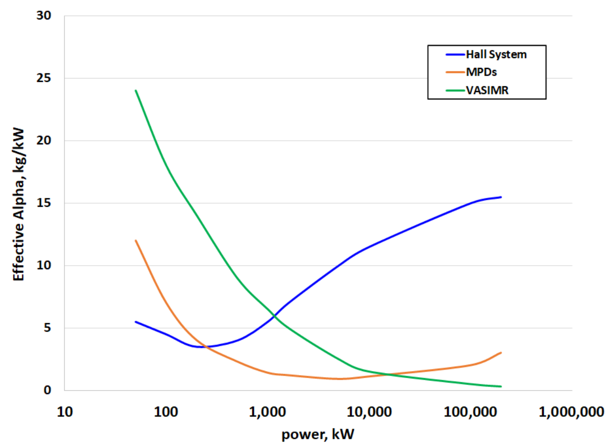


Figure 11: Notional propulsion system performance.

3) Power Management and Distribution (CTE-3)

A high-power NEP system will require a Power Management and Distribution (PMAD) system for interface the power from CTE-2 to the electric propulsion subsystem (CTE-4). The PMAD subsystem must be able to shunt power, as requirement for the thermal management system and the overall spacecraft. The major assemblies include generators and the CTE-2/CTE-3 interface, the transmission lines, switching network and transformers to condition the power. Based on the SNP TIM, a mid-term alpha of 2kg/kWe is plausible in support of the HET architecture and interface requirements.[8] The target of 2kg/kWe PMAD alpha for MWe class NEP is consistent with previous studies on PMAD architecture.[9] There are no currently proposed far-term PMAD approaches to exceed 2kg/kWe for NEP.

4) Power Conversion and Generation (CTE-2 and CTE-1)

The power generation and conversion system are tightly coupled for an optimized system. The enabling technology, more than all others, of transformational propulsion, lies in the ability to generate vast amounts of energy and effectively convert that energy into thrust with extremely low system mass. This is simply Newton's second law combined with electric propulsion performance with thrust proportional to power enabling very high accelerations given a high power-to-mass ratio or low alpha.

$$F = ma \qquad \& \qquad P = \frac{g(FI_{SP})}{2\eta} \tag{2}$$

There is very little experience with operational space nuclear power systems. The flight experience is limited to the SNAP-10A from 1965 that demonstrated only 500We. The most recent earnest investment in the technology development of a space nuclear reactor was during the NASA Prometheus project 20 years ago. At the time, it was predicted a 200kW system could achieve a power system alpha between 30-40 kg/kW. The nuclear fission core mass was 1,569kg for a core alpha of 7.8kg/kW before including the shielding, control system and the power conversion system for an integrated power generation system alpha of 16.5kg/kW.[10] There are practical and theoretical limitations if we stay with a Brayton engine based energy conversion that is not anticipated to yield a total system alpha less than 10kg/kW even with higher temperature and alternative fluid options. The near-term development approach is to leverage higher maturity concepts for a closed-cycle Brayton He-Xe working fluid.

While Brayton based system are never anticipated to achieve transformational alphas, <<10 kg/kWe, there have been a wide range of novel concepts that have been studied for transformational performance. Alterative lower-TRL options exist including a Magnetohydrodynamic (MHD) system with a projected alpha approaching 1 kg/kW.[11] Even lower system alpha systems have been proposed using Fissioning Plasma Core Reactor (FPCR) with MHD power generation.[12] The high operating temperatures of an MHD generator (efficiencies near 30%) can be combined with heat recovery steam cycles for overall system efficiencies approaching 60%.[13] These were earlier studies, however, there are now a large number of concepts that could theoretically achieve a power generation alpha as low as 0.05 kg/kW and energy conversion approaches down to 0.07 kg/kWin or 0.4 kg/kWe for the combined energy generation and conversion.[14] These are all limited to paper studies without any meaningful technology maturation to date.

Table 2: NEP Subsystem Far-Term Concept Potential

Subsystem	SOA	Far-Term Evolutionary	Transformative
CTE-1: Power Generation	16.5	1	0.05
CTE-2: Power Conversion	10	0.9	0.35
CTE-3: PMAD	6	2	0.07
CTE-4: Propulsion	5.5	2	0.5
CTE-5: Heat Rejection (w/o deployment mechanisms)	22 (13)	1	0
Total System, kg/kWe	51	7	0.97

5) Integrated System

The integrated system performance potential assessed by the technology team as a part of the study determined the limits of performance for evolutionary subsystem technologies may be able to achieve an alpha approaching 7. Transformational shifts in the critical technology elements are necessary to achieve an alpha less than 1, but a path is notionally identified to achieve an alpha ~1kg/kWe.

B. Solar Electric Propulsion Variant Deep Dive

For solar electric propulsion, the CTEs 1 and 2, power generation and conversion respectively, are primarily captured under the solar array system. The propulsion system may need heat rejection, but due to the relatively high efficiencies, compared to the power conversion system, more than 80% of the heat rejection burden is eliminated without the nuclear thermal heat load.

1) Power Generation (CTE-1)

Advanced, very large-scale solar arrays are potentially viable non-nuclear electric power generators for crewed missions to Mars. These arrays, coupled with batteries, fuel cells, or other energy storage technologies, could provide power for the mission entirety: spacecraft operations, environmental control and life support, solar electric propulsion, and other spacecraft systems. Herein a look is taken at these arrays with an emphasis on specific mass; also known as 'alpha' and defined as the array mass per unit power generated. Alpha has the unit kg/kW and is the inverse of the specific power.

Figure 12 shows array alpha versus total array power output for several different space solar power generators. The values of alpha are for beginning of life (BOL) performance at 25°C, with degradation due space environments effects nor performance changes due operational temperature not included. The chart is grouped into three broad categories: (i) State-of-the-art (SOA) technology, defined as ≥TRL6 technologies that have either already been flown or are nearly ready to fly; (ii) Evolutionary (highly probable) technology, defined as TRL2-6 technologies that would be ready for flight within *ca.* 10 years; (iii) Revolutionary (speculative) technology, defined as ≤TRL2 technologies that may be ready for flight in 50+ years. These three categories give an assessment of power limits and mass allocations for solar powered generators based on what is possible today, what is likely to be possible in the mid-term, and what might be possible in the long term.

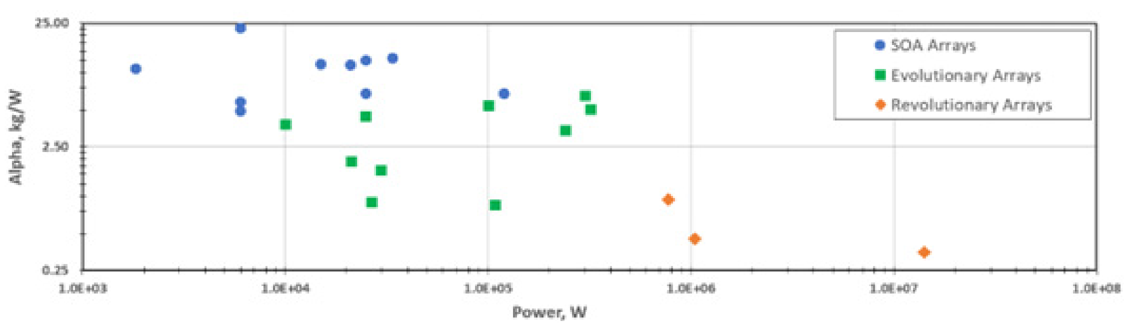


Figure 12: Alpha (kg/kW) of SOA, Evolutionary and Revolutionary Solar Arrays.

A general trend can be seen in Figure 12. SOA arrays are generally in the small to mid-sized power generation range (i.e. <34kW) and are representative of relatively high alphas (i.e. >5kg/W). For these arrays, the alpha is comparatively independent of the power level. SOA arrays are of two basic constructions: rigid or semi-flexible, where the difference is whether the collection of rigid solar cells is affixed onto a rigid or flexible substrate. The more traditional fully rigid arrays, those with rigid cells on rigid substrate, represent the higher SOA alphas. The semi-flexible arrays, those with rigid cells on flexible substrates, represent the lower values of alpha. The evolutionary, mid-term arrays are in the mid to high power generation range (i.e. 25kW to 350kW). These arrays split down two paths, those lowering the alpha of mid-sized power arrays (i.e. <50kW @ <2.5kg/W) and those growing array size, but with little or no change to alpha (i.e. >100kW @ >5kg/W). Evolutionary arrays are comprised of advanced semi-flexible designs, fully flexible designs, defined as arrays employing flexible solar cells and flexible substrates, as well as concentrator designs, defined as arrays employing a concentration of sunlight onto the solar cells. For these mid-term arrays, the alpha is often unchanged or even increasing with increasing power level due to the need for additional structure to provide stiffness for the mission application of these larger arrays. The revolutionary, long-term arrays, for the scope of this study, push towards those with very high powers (i.e. >750kW) and very low alphas (i.e. <1kg/W) – those arrays which could support solar electric propulsion crewed missions to Mars. These arrays are fully flexible and based on theoretical, but physically possible material sets. The following sections detail the literature, data, and rationale of each category of solar array and the scatter points figure 12 is based.

SOA Technology: ≥TRL6, flown or nearly flight ready

Current state of the art solar arrays for space, Table 3, are all based on cells formed from the group III-V atoms of the periodic table. These elements are compound semiconductors with excellent optoelectronic performance, high photon absorption, and high electron mobility.[15, 16] High optical absorption allows for reasonably thin solar cells and these compounds are often grown by depositing a reactive vapor onto a substrate, generating thin layers of single-crystal material. These compounds are typically intolerant to crystal defects and as such must be synthesized carefully and with a minimum of faults, dislocations, or inclusions. They are grown on lattice-matched substrates to reduce these defects. Today's triple junction solar cells are comprised of Ge (germanium), GaAs (gallium arsenide), and InGaP2 (indium gallium phosphide). The elemental semiconductor germanium is very closely lattice matched to GaAs, which is similar in lattice size to InGaP2, and these semiconductors' bandgaps are well-tuned to absorb a large portion of the incident solar radiation. These materials are deposited on top of a commercially available germanium substrate by chemical vapor deposition in high quality and can regularly yield >30% conversion efficiencies with areal mass densities in the range of 840g/m^2.

Those cells are somewhat fragile due to the inherent brittleness of the single-crystal material and the desire to make the wafer substrate as thin as possible for mass and material (cost) savings. A coverglass is adhered to the cell both to increase the physical robustness as well as to shield the cell from the deleterious effects of ionizing radiation in space. A typical SOA coverglass has a density of 2.60 g/cm^3 or 198.1g/m^2 for a representative 3mil thick glass. To create an array, the cells are bonded physically to a substrate and metal tabs are spot welded to make series connections up to the desired bus voltage. The substrate is typically a polyimide sheet that is then further bonded to the structure of the array that is designed to mission vibration and shock loads. Those secondary structures are generally one of two styles: fixed honeycomb structure for rigid arrays or fiberglass mesh for semi-flexible arrays.

Table 3: Summary of Representative Solar Arrays

Solar Array	Power, W	Alpha, kg/kW
Rigid Panel Small (Hypothetical)	6,000	22.73
Rigid Panel Mid	25,000	12.50
Flexible fold-out	120,000	6.67
Flexible Roll Out Small (Hypothetical)	6,000	4.85
Flexible Roll-out Mid	25,000	6.67
ISS ROSA	34,000	13
Ultraflex Array (Hypothetical)	6,000	5.71
Lucy (Ultra Flex)	21,000	11.43
Insight (Ultra Flex)	1,840	10.65
Psyche	17,000	11.53

Rigid arrays, those with the rigid II-V cells on a rigid substrate, are ubiquitous throughout space applications. As mentioned above and seen in **Error! Reference source not found.**, their alpha values are relatively high at 12.5 to nearly 23 kg/kW. More advanced is the semi-flexible designs, which is at the cutting edge of SOA and truly represents the best, with respect to alpha, of what is possible today. These are those arrays, in the SOA version, with rigid III-V solar cells on a flexible substrate. They represent the lower alphas (<12kg/kW but still >4.85kg/kW), however; are more limited in the ability to withstand acceleration loads (e.g., from propulsive engine firing), and typically have lower stiffness and lower natural frequencies. In one semi-flexible array design currently used in space, flexible panels are shaped into a 'pie-slice' shape, or 'gore' and bonded together near the vertices to create a fan-folded structure that deploys into a circular shape. This structure has the commercial name "UltraFlex" and is advantageous because the stowable volume is modest and array structure has potential to host many cells, ultimately leading to high power capability. Lucy, Insight, and Cygnus (ISS commercial resupply program) utilize the UltraFlex design concept. Alpha on these as flown designs is modest at ~11.5kg/kW but the hypothetical SOA has an alpha half that at 5.71kg/kW and, as will be shown in section 0, there is scaling potential to much higher power generation as well as a still lower alpha.

A second SOA semi-flexible design is comprised of a fiberglass mesh to which polyimide sheets bearing the solar cells are laminated. This type of array structure can be either rolled around a mandrel that is sized such that the brittle cells do not exceed their radius of curvature, or even Z-folded into a flat package for launch, in which hinges to fold the arrays are located between cells, and the cells themselves remain flat and avoid the physical stress from being flexed. The structure is deployed in space by releasing twin carbon fiber booms that are stored under tension and when released will unfurl the solar array blanket. This design is referred to as the "Roll Out Solar Array" or ROSA and will be the updated ISS solar array wing technology. Alpha of the flown module, 13kg/kW is in family with the UltraFlex, and there is also significant scaling potential.

When considering SOA solar arrays for fast crewed transit to Mars, small size is the primary constraint. The largest SOA arrays reach power levels of just a couple hundred kW, whereas a crewed electric propulsion vehicle requires a power generation of at least ~600kW, with some estimating as much as ~15MW. Alpha also remains high at >~5kg/kW, with the solar cell assemblies (cell + covergalss + interconnect) accounting for more the 50% of this alpha (>2.5kg/kW). This then drives heavier support substrates and, more importantly, more massive structural and deployment systems.

Evolutionary Advancements: TRL2-6, highly probable, flight in *ca.* 10 years

Evolutionary versions of current array technologies are shown in Table 4. These arrays are largely comprised of evolved semi-flexible designs, fully flexible designs, and concentrators. Early generation 'PowerSails', an advanced full flexible array, are also in this category. These evolutionary arrays are trending in either of two directions: (i) toward larger arrays with higher power generation while maintaining SOA alpha (i.e. >100kW @ >4kg/W), or (ii) toward lower mass arrays with improved alpha while maintaining SOA power (i.e. <100kW @ <2.5kg/W). The stiffness requirement on the array is an important parameter and defines why evolutionary arrays are split into these two categories. As the an array size increases, assuming strength and stiffness requirements stay the same and there are no major changes to the core array materials used, alpha grows *via* increased structure. Banik *et. al.* generalized this for all flexible array technologies, stating that metrics such as w/kg imply linear scaling which may not be appropriate. Instead, flexible array technology should put more emphasis on structural scaling parameters. This is an important point to keep in mind when drawing scaling conclusions from alpha tables herein, especially for those describing flexible technologies. It is also important to note that the higher alpha arrays listed in 4 are at a higher TRL than the lower alpha arrays. That is, the lower alpha PowerSails are significantly less mature than, for example, the evolved UltraFelx (called MegaFlex) and the evolved ROSA.

Table 4: Summary of high-probability evolutionary arrays.

Solar Array	Power, W	Alpha, kg/kW
MegaFlex SEP Tug	300,000	5.20 - 6.45
MegaFlex Aggressive	10,000	3.80
ROSA/FACT	25,000	3.50 - 4.40
ROSA Evolved	242,000	3.40
Compact Telescoping Array	100,000	5.40
O'Neil Fresnel Array (small)	21,1000	1.52 - 1.90
O'Neil Fresnel Array (large)	321,400	4.1 - 5.00
Scout PowerSail Conservative	29,559	1.61
Scout PowerSail Moderate	26,645	0.89
JAXA OKEANOS Power Sail	108,254	0.84 - 1.30

MegaFlex is an evolved version of the UltraFlex array discussed in the SOA section, which adds an additional fold to increase the diameter to sizes up to 30 meters in diameter. The evolutionary MegaFlex is targeting 300kW or higher, and structural test items have been successfully deployed under space-like conditions in a vacuum chamber. Analytical methods predict the extensibility of the array into a few scenarios, most relevant herein is the prediction of an SEP Space Tug.

Using SOA 28.9% efficiency solar cells with an array strength requirement of withstanding acceleration of up to 0.10g with a minimum frequency of 0.1 Hz, the 300kW solution showed an alpha of 6.45kg/kW. An aggressive extrapolation of the possible mass of a Megaflex array was also calculated, assuming the best-case conditions and highest specific power from reference and increasing the cell efficiency to an assumed next-generation efficiency of 36% at BOL. For a wing diameter of 5 meters, a minimum possible alpha of 3.82 was calculated.

An engineering design study of a 500-kW SEP Mars mission looked at use of a next-generation ROSA solar array, using two wings of 242 kW, where each wing consisted of ten individual ROSA panels mounted on a rotatable boom for solar tracking. Each wing had a mass of 825 kg, for an alpha value (listed in the table as “ROSA/evolved”) of 3.4 kg/kW.

Another lightweight technology under development in the near term is the Compact Telescoping Array or CTA, developed under a NASA SBIR, which uses a Z-folded flexible photovoltaic blanket supported by a telescoping boom. Array designs have been analyzed at power levels ranging from 7 kW to 613 kW. The expected value of alpha ranges from 5.4 kg/kW for an array of 100kW, rising to 6.5 kg/kW at the higher power of 613 kW, where again the higher value of alpha at the larger size is due to stiffness requirements. A design study for a Solar Electric Propulsion Mars mission using the CTA for primary power, assuming 36% efficient next-generation IMM solar cells, calculated 7.8 kg/kW for the 500 kW solar array wing. This alpha rises to 9.3 kg/kW for the engineering design of the fully integrated system (including, along with the photovoltaic array, all structure, mechanisms, wiring, and harnessing, and incorporating a 30% engineering growth factor) at a 1-MW BOL power level.

Another important class of array that is under development, especially for deep space robotic missions, are concentrators. Concentrator arrays use mirrors, lenses, etc. to focus [concentrate] more sunlight onto a solar cell. These concepts have been around for quite some time. As a reference point, the 1996 SCARLET array showed a path to 2kW of power generation with an alpha of 12.8kg/W. Recent and coming advancements make this a worthwhile technology to capture. The ROSA/FACT array is a design incorporating lightweight IMM solar cells with a low concentration “trough” concentrator, assembled on the ROSA array structure discussed above. Although primarily designed for outer planet missions, the detailed design study showed a 25.62 kW array with a mass of 113.7 kg, for alpha of 4.43 kg/kW. At a higher concentration ratio, using today’s efficiencies, O’Neill *et al.* estimate that 1 kg/kW is possible at the cell blanket (not including array deployment, support structure, etc.) with pop up 25X ultralight Fresnel lenses. This pop-up concentrator is compatible with Compact Telescoping Array as part of the EESP project. Further, a >4X low concentration system is designed to be compatible with ROSA arrays and can also net a total of 0.9 kg/kW at the blanket level. This translates to specific masses of ~1.9 kg/kW for a 22.1kW array, sliding down to ~5.0 kg/kW for a 321.4kW array.

In addition to growing the structure to increase cell packing density or using concentrators to add more light to cells, the cells themselves are undergoing incremental advances currently and are expected to evolve towards greater conversion efficiency within the 10-year time frame. These improvements generally take on two forms: efficiency enhancement through process optimization and creating more semiconductor junctions on a single cell so that high voltage can be attained, and more spectral photons can be absorbed. Theoretical efficiency limits for solar cells, known as the detailed balance limits, are well established for both single and multiple junction solar cells. For an ideal, single junction cell, which is limited only by radiative recombination, the maximum efficiency achievable (at room temperature and without solar concentration) has been established to be 30%. Multijunction cells can reach much high theoretical efficiencies, detailed balance limits under unconcentrated and concentrated Air Mass 1.5 (AM1.5) sunlight are shown in Table 5.

Table 5: Detailed balance limit of the efficiency of tandem solar cells.

Solar Cell Junctions	Theoretical Efficiency (1 sun)	Theoretical Efficiency (45.9k:1)	Laboratory Demonstrated Peak (>1cm^2)	Utilization Factor	Solar Cell Junctions
1J	30%	40%	24.4	81%	1J
2J	42%	55%	30.3	72%	2J
3J	49%	63%	37.9	77%	3J
4J	53%	68%	38.8	73%	4J
Infinite	68%	86%	NA	NA	Infinite

Estimations of future efficiency enhancement are predicated on estimating the “learning curve” for a technology, essentially the track record of advancement over time and simple extrapolation. The National Renewable Energy Laboratory (NREL) is a nationally recognized testing laboratory equipped to accurately determine solar cell performance and subsequently publish an annual review of record solar cell efficiency advancements, although it should be noted that the efficiencies quoted are under terrestrial sunlight conditions, AM1.5, which results in slightly higher efficiency than that measured under space illumination conditions. This publication was used to estimate learning curves for various technologies relevant to space solar array advancement for this paper. The best data exists for triple junctions of monolithic III-V semiconductors, which are the dominant state of the art space solar cell, and mature enough to suggest that incremental improvement over time is all that is expected. Crudely, we expect approximately 0.5%/year improvement in technology. In this case the percent is absolute percentage points on efficiency, e.g after 10 years, one expects 5% added to current efficiencies, yielding approximately 35-37% large areas cells at scale production. However, new technologies that include junctions greater than three hold promise of even higher efficiencies. Already 4J and 5J are becoming commercially available and assuming a similar learning curve to the 3J, may reach approximately 40% conversion efficiency with large area cells.

The PowerSail is another category of array, which will ultimately form the basis for the speculative arrays that could be used for SEP Crewed Transit to Mars, described in the next section. The PowerSail is a photovoltaic array concept based on thin-film, fully flexible blankets. In short, it is the marriage of the 'Solar Sail' with thin-film solar cells and flexible protective coatings. This concept has been pervasive in the community for decades, with works as early as 1990 predicting alphas approaching 1.15kg/kW when utilizing thin-film technology for Mars expeditions. Early protypes by AFRRL in 2003 showed potential for 50k to 100kW at 4kg/kW. More recent work from JAXA has begun to make PowerSails a reality, with an early tech demo flight of 'IKAROS' in 2010. Those this flight only included 300W of a-Si solar power generation, it gives near term credibility to the concept. With continued advancements in Solar Sail, thin-film solar cells, and flexible coatings, is feasible, and highly likely, this type of technology will be ready for mission within a decade.

Table 6 shows three evolutionary PowerSail arrays; two calculated specifically for this study and one from literature published by JAXA. The first of the calculated arrays is dubbed 'Scout PowerSail Conservative'. It is based on the NEAScout Solar Sail and includes technologies that are in development today, which are expected to be scaled and flight worthy within 10 years. As of 2017, NASA and Jet Propulsion Laboratory-Caltech predict for mid- to far-term (5-10 years) solar cell efficiencies will reach >37%, consistent with the estimations made for this paper. This, along with recent advancements in inverted metamorphic multijunction (IMM) solar cells, led to the use of a thin-film, 37% IMM (or similar epitaxial lift-off, whereby the more massive substrate the cell is deposited upon is removed from the functional cell) at 250g/m^2 for this case. This areal density is achievable today and, thereby, likely a bit high for ~10 years out. Nonetheless, it represents a good conservative estimate with the assumption that development would focus more on cell efficiency and the fabrication and qualification of such a large thin-film array. Coating materials are based on SOA polyimides, such as Optinox®SR, with the assumption that improvements will be made to increase longevity in the space environment (e.g. UV reflectors) from single years to 10+ in the deep space environment. The array substrate is based on a SOA 3µm thick colorless polyimide 1 (CP1) solar sail that is deployed utilizing the NEAScout, boom-based deployment system. A 25% mass penalty was applied to the deployment system to account for the added weight of the solar cells and power routing (copper traces). The second of the calculated arrays is dubbed 'Scout PowerSail Moderate'. It builds off the conservation array to include a perovskite solar cell with 30% efficiency and an areal density of 50g/m^2. This is based on an extension of the efficiency learning curve for a monolithic, all-perovskite tandem, or 2-junction, solar cell. Improvements were also assumed for the substrate material by reducing the sail thickness by 50%. Finally, improvements to the deployment system were incorporated by reducing the mass penalty from 25% to 15%. These two PowerSails give bounds for what is likely possible within 10 years. Table 6 summarizes the assumptions made. In both cases, a 300V bus voltage was assumed to support direct drive options for electric propulsion according to the current crewed Mars SEP design reference mission. Currents were calculated accordingly. Copper traces, which would be embedded in the PowerSail substrate or in a separate, thin-film wiring harness, were sized per the current and added to the overall mass. 20% mass growth allowance with 15% mass margin (35% total) was added to each array.

Table 6: Evolutionary PowerSail calculation assumptions and inputs.

Array	Cell Type; Efficiency; Area Density	Coating Type; Area Density	Substrate; Area Density	Deployment System
Scout PowerSail Conservative	Epi Lift; 37%; 250g/m^2	Polyimide; 36g/m^2	3µm CP1; 4.6g/m^2	3.7kg
Scout PowerSail Moderate	Perovskite; 30%; 50g/m^2	Polyimide; 36g/m^2	1.5µm CP1; 2.3g/m^2	3.4kg

The bounded alpha for this small 'Scout PowerSail', 0.89 to 1.61kg/kW is in good agreement with JAXA's most recent work on the PowerSail mission 'OKEANOS', an explorer to the Jupiter Trojan asteroid. The JAXA team built a prototype showing a path to a 108.2kW array at ~1.30kg/kW, with potential to reach 0.84kg/kW. Though not directly stated, these arrays appear to be based off SOA CIGS technology (~10-12% efficiency AM0; ~110g/m^2).

Transformational Advancements: ≤TRL2, speculative, flight in >50 years

To speculate on what very large scale, revolutionary designs could look like in 50+ years, scaled PowerSail arrays were sized (Table 7). These arrays are based on theoretical and, in some cases, 'not-yet-invented' components; but components that are bounded by physics and feasible given current design paths. The following details the assumptions that were made with **Error! Reference source not found.** summarizing the input parameters.

For the solar cells, It is instructive to consider the approach of evolutionary development in solar cell improvement above but for a longer time frame. The detailed balance limitation on solar cell performance has a ceiling of approximately 65% conversion under AM0 conditions for an infinite number of junctions, which is of course not feasible. The current best utilization factor, the ratio of record performance to the theoretical limit is about 81% for silicon solar cells, which have been under development for close to 50 years, and therefore can stand as a model system for where current technologies may exist in 50 years. Therefore, we may expect that a "many-junction" solar cell (~6J or more) may reach an 80% utilization in that time frame and reach at most about 50% conversion efficiency. To estimate the mass of this hypothetical solar cell we need to estimate the minimum thickness that it would require to absorb photons.

Table 7: Summary of transformational arrays.

Solar Array	Power, W	Alpha, kg/kW
Cruiser PowerSail Moderate	774,933	0.94
Cruiser PowerSail Aggressive	1,041,788	0.45
Early Mars Crewed PowerSail	14,131,000	0.35

To estimate the minimum thickness of an excellent, yet realistic solar cell material we first examine the absorption coefficient of broadband light incident on the semiconductor. Methylammonium lead triioidide ($NH_3CH_3PbI_3$) exists in the perovskite structure and has the fastest growing learning curve of any solar cell material reported. It also has an extremely high absorption coefficient for above band gap energy light, at approximately 10^5 cm^{-1}, which is similar to other high-performance semiconductors for optoelectronics, such as GaAs and InP. Because of the blackbody nature of the incident solar radiation, there exists an optimal band edge absorption energy of approximately 1.4eV for a single junction solar cell to produce the greatest power output. Assuming a very steep absorption profile for above band gap energies, a material as thin as 100nm will absorb nearly 100% of available photons. GaAs has a mass density of 5.3 g/cc and $NH_3CH_3PbI_3$ has a mass density of about 4.1 g/cc and therefore the mass of a 25% single junction solar cell could be fundamentally as low as 0.6 g/m^2. That value essentially scales linearly with the number of junctions sought. Because the efficiency gains scales sub-linearly with the number of junctions, as described above, the alpha value is optimized at the cell level for 1J or 2J device that has high utilization factor and is also ultrathin, such as $NH_3CH_3PbI_3$, or its relatives. This must be balanced against deployment area and, thereby, the mass of the booms and deployment system.

Table 8: Transformational PowerSail calculation assumptions and inputs.

Array	Cell Type; Efficiency; Area Density	Coating Type; Area Density	Substrate; Area Density	Deployment System
Cruiser PowerSail Moderate	5J Thin-film; 45%; 7.2g/m^2	SiO; 26.3g/m^2	Graphene; 0.1g/m^2	39.0kg
Cruiser PowerSail Aggressive	6J Thin-film; 50%; 3.6g/m^2	SiO; 2.1g/m^2	Graphene; 0.05g/m^2	40.7kg

For these cells, the active material must be synthesized *via* means designed to deposit thin layers on a substrate, which would be used as the PowerSail substrate as well. In essence, a monolithically integrated PowerSail. A substrate of interest is based on graphene or a graphene-like material – exceedingly thin and low mass, but with enormous tensile strength. The theoretical areal density for a single layer of graphene has been reported to be $7.4x10^{-4}$g/m^2,which has a predicted tensile strength ~200x that of steel. Speculative works have begun to look at theoretical graphene based solar sails. Breakthrough Starshot, an interstellar travel challenge championed by Breakthrough Initiatives, is also pushing for revolutionary sail material improvements. Starshot has set a target for the Starchip light-sails at 10m^2 with a mass <1 gram or less than 0.1g/m^2, something that might be met with graphene. A single layer of graphene may be infeasible as a substrate material, and several additional layers may be required to effectively produce it and enable its use also a current conduction layer. A 2-dimensional structure has unusual conduction habits, but many layers of graphene are being used in place of transparent conducting oxides in solar cells in laboratory research. Reports that graphene can reach a "bulk" resistivity of $5x10^{-4}$ Ohm-cm have been made for sheets as low as 4 layers, essentially adding 0.003 g/m^2 to the mass of the array. Even if the required amount of graphene were numerous layers at 0.1g/m^2, this mass represents <<0.1% of the overall PowerSail mass, essentially negligible to the alpha. The solar cell and largely the copper harnessing masses dominate.

As with other technologies that require cover-materials to protect from radiation, these cells would also require radiation protection to be useful for a round-trip through the van Allen Belts to Mars. In this scenario an additional monolithic radiation protection layer could be added, albeit it would add significant mass to the structure. Optimally, heavier atoms would be desired for stopping power, but recent work by Kirmani et al. showed that a 1-micron layer of vapor deposited silicon monoxide (SiO) added significant protection to a perovskite solar cell irradiated with 0.05 MeV protons and 1MeV electrons. At a density of 2.1 g/cc, this leads to an added mass of 2.1g per m^2 of solar array.

Higher array voltages of 600V for the 'moderate' Cruiser PowerSail and 1,200V for the 'aggressive' were assumed. Studies on solar electric propulsion crewed transit to Mars have revealed spacecraft power generation levels from ~600kW to ~15MW. ~775kW and ~1MW were assumed for 'moderate and 'aggressive' accordingly. Currents were calculated thereby. Higher voltages were used to keep the mass of copper traces on the PowerSail manageable. Further improvement of the array alpha may be attainable by the utilization of the graphene substrate itself or *via* a standalone high temperature superconducting harness as electrical conductors. At ~1MW, even at 1.2kV, the copper to combine the solar cell strings and route power back to the spacecraft bus alone accounts for >60% of the PowerSail mass. Nonetheless, even with copper, sub-1kg/kW with potential to reach ~0.35kg/kW is theoretically possible.

The solar cruiser deployment system was utilized for the calculations with a 15-20% mass penalty for stronger booms and motors. There is little doubt that this SOA deployer could be improved and mass driven downward over the next 50 years, however; this scope was not explored herein. 35% mass growth allowance with 15% mass margin (50% total) was added to each array. Using Solar Electric Propulsion, the overall system performance potential is shown in table 9.

Table 9: SEP Subsystem Far-Term Concept Potential

Subsystem	SOA	Far-Term Evolutionary	Transformative
CTE-1: Power Generation	6.67	0.84	0.35
CTE-2: Power Conversion	-	-	-
CTE-3: PMAD	2	2	0.07
CTE-4: Propulsion	5.5	2	0.5
CTE-5: Heat Rejection	4.4	0.2	0
Total System, kg/kWe	19	5	0.92

C. Nontraditional Propulsion Discussion

The study team surveyed a wide range of alternative propulsion systems. The survey included open literature concepts as well as the inventory of all past NASA Innovative Advanced Concepts (NIAC) applicable investments. The range of options generally fell into two categories, extremely low alpha propulsion systems (e.g. the Propagating Magnetic Wave Plasma Accelerator (PMWAC) with a thruster $\alpha = 6E\text{-}5$ kg/kW.) or pulsed nuclear approaches. Pulsed nuclear solutions do offer a compelling advantage over very low alpha propulsion systems regarding scalability. At the very low alpha capability limits, the payload masses must fit within the margin of capability. i.e. if the system can only deliver very small masses per kilowatt of power, then incredibly high power levels are necessary for delivering large payloads. Consider a 50-day transit to Mars viable at 1kg/kW and the propulsion system has an integrated performance of 0.9 kg/kW for a net payload capability of 0.1kg/kW. For a 100,000kg payload, the power level required would equal a gigawatt of power. The notional SEP systems may lose credibility much beyond 2 MWe of power, even in the far-term. NEP systems may lose credibility much beyond 200MWe class systems. Pulsed systems can scale their effective power level simply by increasing the pulse rate (i.e. carrying more of the "fuel" to be detonated). Pulsed detonation systems can therefore scale to multiple gigawatts of effective power for large mass transits while also maintaining high accelerations or achieve even faster transit times for small payloads if the minimum system mass is acceptable. Another finding from the nontraditional propulsion concept survey is the lack of clear interim steppingstones of utilities prior to fielding a full-scale operational system; inhibiting modest yet sustained investments.

V. Investment Recommendations

Regarding investment recommendations, it was clear that there are a wide range of technology concepts that have merit towards enabling transformational propulsion. Most concepts are limited to only cursory paper studies and would benefit from at least in-depth modeling and analyses; ideally bench-top testing of efficacy where appropriate. It is therefore recommended to make and protect low-level sustained investments. Investments on the order of 1% of the Space Nuclear Propulsion program applied towards transformational propulsion could provide evidence of viability for technology concepts under each of the critical technology elements. Promising concepts would then become credible for existing early stage and technology maturating funding opportunities (e.g. SBIRs, STTRs, GCD, etc.) Also, even incremental improvements in any of the CTEs would feed directly into the SNP evolutionary architectures and may offer advantages for a range of other applications (e.g. heat rejection, solar power, etc.) well beyond propulsion.

VI. Summary Results and Conclusions

This rapid study identified a technology path to achieve transformational propulsion. The current technology investments, through evolutionary investments are not likely to achieve transformational performance for nuclear propulsion. Evolutionary solar propulsion investments may result in near-transformational propulsion capabilities. Near-term and evolutionary solar electric propulsion systems offer performance advantages over near-term and evolutionary nuclear electric propulsion system out to Mars distance. All near-term evolutionary approaches require trip-times longer than 1-year for viability. Far-term evolutionary NEP and SEP enables roundtrip transit times of 250 days and 220 days respectively. Transformative performance for either NEP or SEP enables one-way and roundtrip transits of 50 days and 100 days respectively when returning to a Lunar Distance Highly Elliptical Orbit, with viable round-trip mission less than 3 months if direct Earth entry velocities can be tolerated. The full results of the various trades are provided in the appendix.

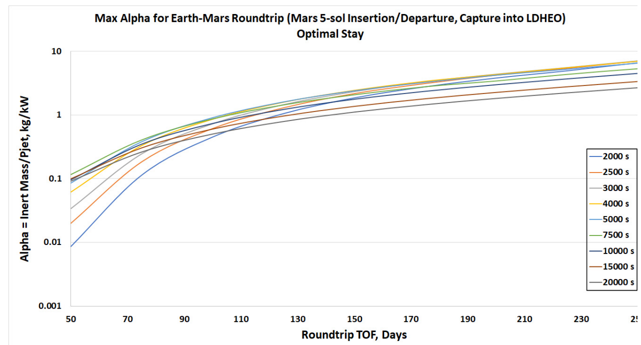


Figure 13: Highest delivered alpha capability for Mars 5-sol Insertion/Departure and return to LDHEO.

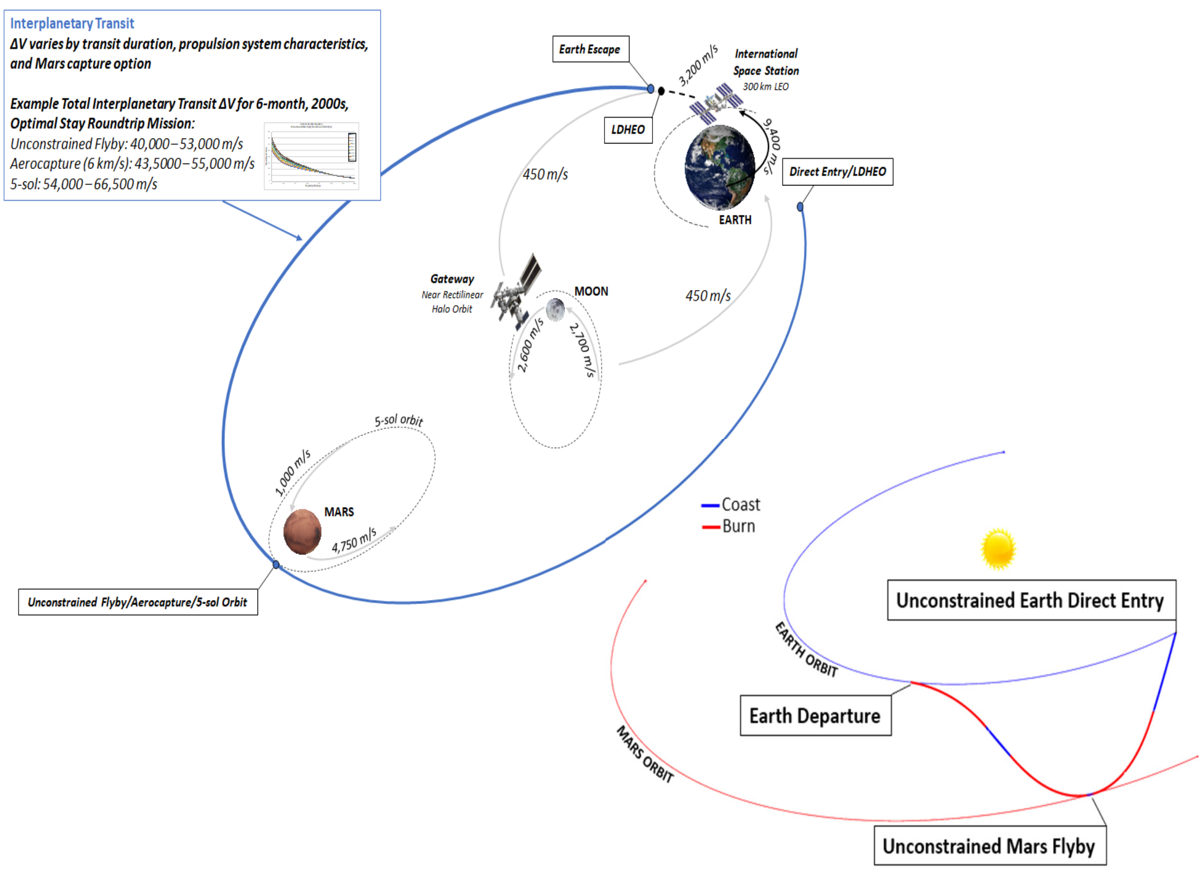


Figure 14: Transformational Earth-Moon and Earth-Mars energy map (top left) and a 50-day round-trip flyby trajectory (bottom right).

Appendix – Full Results

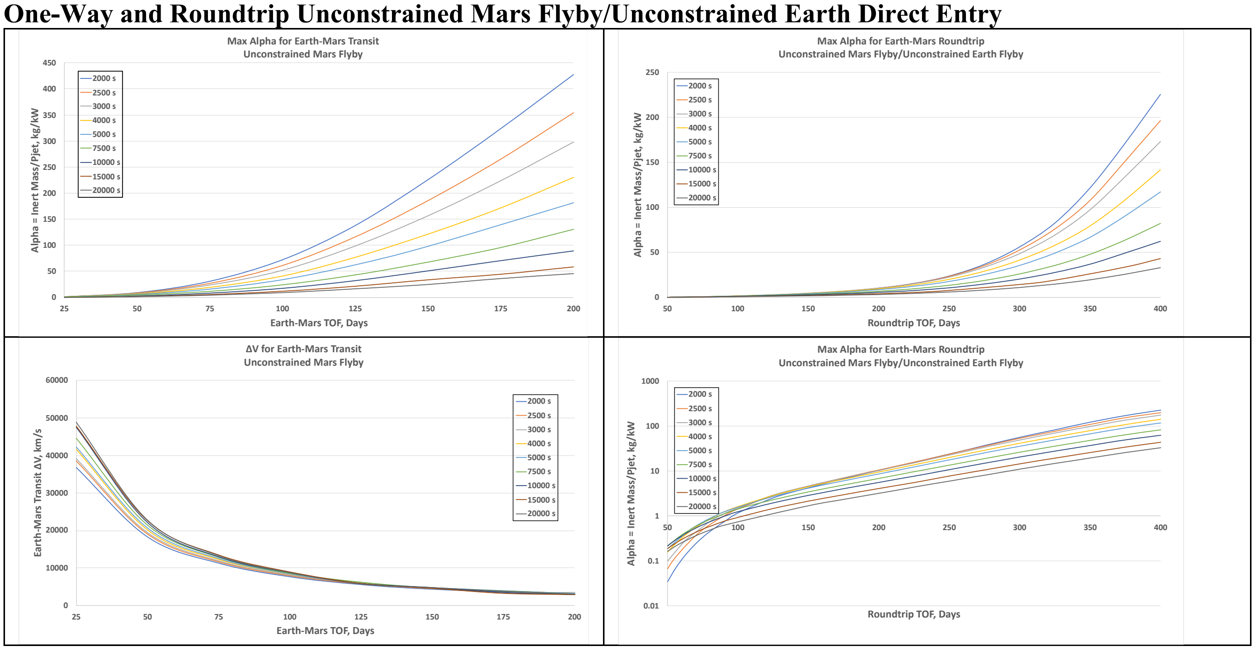
Trajectory models were created using Copernicus for all mission permutations depicted in Figure 4. Copernicus is a trajectory design and optimization tool in which a series of segments is used to model a 3-Deegree of Freedom trajectory. Copernicus was conceived at the University of Texas at Austin by Dr. Cesar Ocampo and is now primarily developed at NASA.[17] The ground rules, assumptions, and constraints for the various capture and departure methods as well as how these were implemented in the Copernicus trajectory models are provided in A-1. The 2033 mission opportunity was use as the reference date for these models.

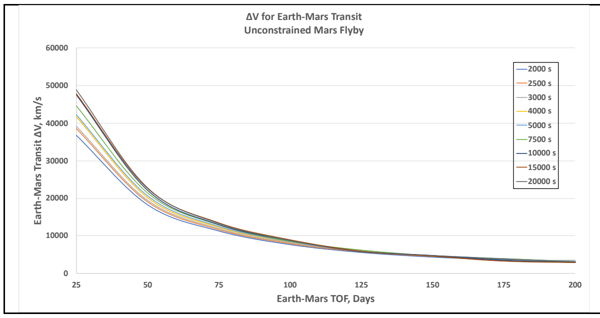
Table A-1 Implementation of GR&A in Trajectory Models

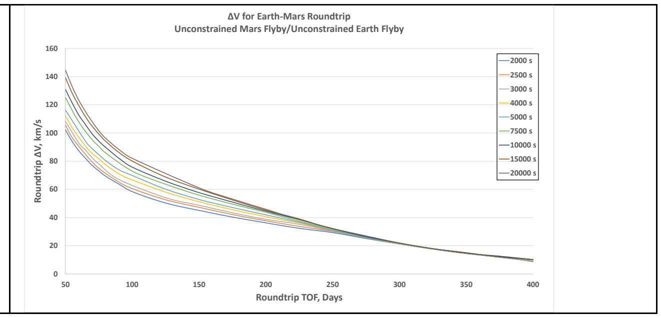
Constraint/GR&A		Trajectory Model Implementation
	Low Thrust Engine	Internal Power Source. Isp will be traded parametrically. Power set at 2MW, will remained fixed
Earth Departure Condition	C3 = 0km^2/s^2	C3 = 0 km^2/s^2, Rp = 6778.12 km, Tau = 0 s, |inc|,=28.5, raan and aop unconstrained (EARTH J2000). T0 is optimized within +/- 365 day bounds
Mars Arrival/Departure Conditions	Unconstrained Flyby	Flyby altitude held constant at 50 km, no upper limit on arrival hyperbolic excess velocity
	Aerocapture	Altitude = 50 km, max velocity varies based on specific case (<6, 10, 20 km/s), FPAC=0º. Raan, dec, azmc are all unconstrained - assuming inclination can be altered aerodynamically (MARS J2000). Dt = -0.5 days for coast prior to aerocapture for reorientation and checkout
	5-sol MAT Orbit	250 km x 443876.22 km (5-sol) over 35º landing site (inc = 35º, aop = 90º). Burn initiated at perigee (TA = 0º). Raan is optimized
Earth Arrival Conditions	Unconstrained Flyby	Flyby altitude held constant at 50 km, no upper limit on arrival hyperbolic excess velocity
	Direct Entry	Velocity <= 14 km/s at 121 km, FPAC = -5.86º (EARTH J2000). Dt = -0.5 days for coast prior to direct entry for reorientation and checkout
	LDHEO	400 km x 393622 km. Inc, Raan, AOP are optimized. A -0.5d burn from TA = 0º added for reorientation and checkout

The following plot show the maximum alpha and total transit ΔV plots for all mission permutations for both the one-way and roundtrip missions.

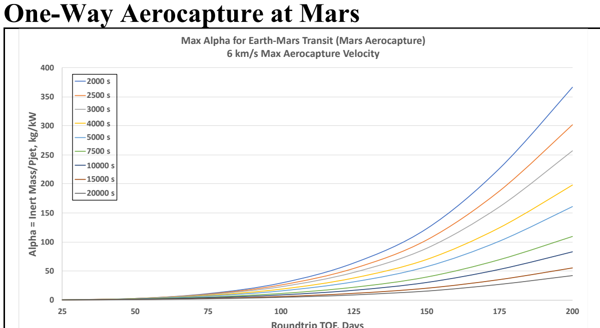
One-Way and Roundtrip Unconstrained Mars Flyby/Unconstrained Earth Direct Entry

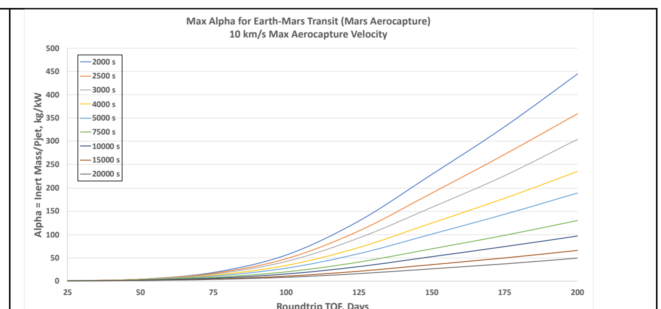


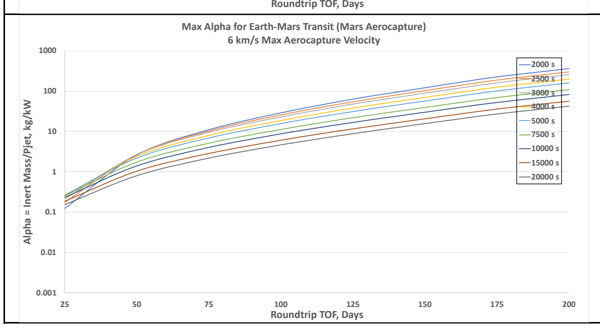


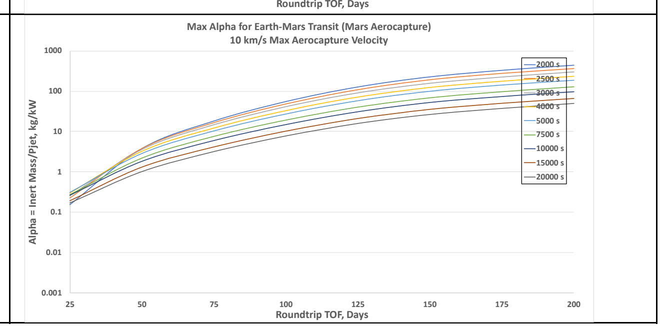


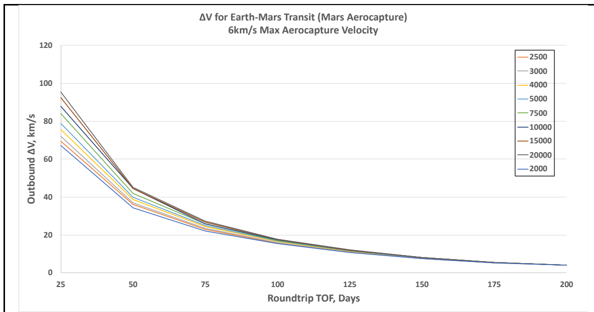
One-Way Aerocapture at Mars

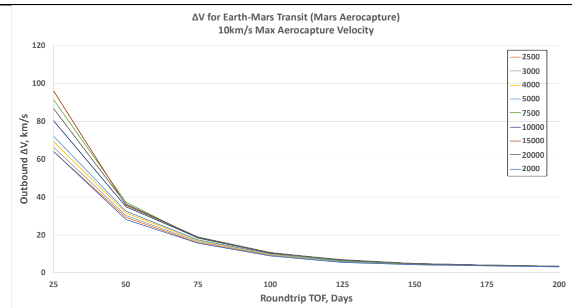


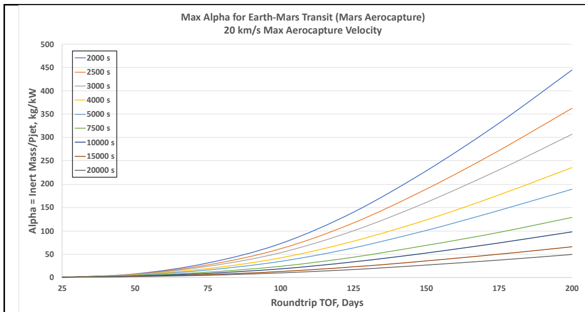


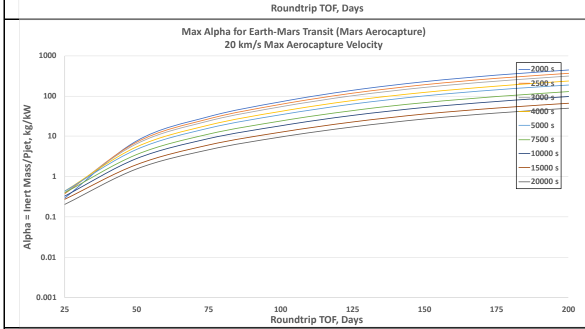


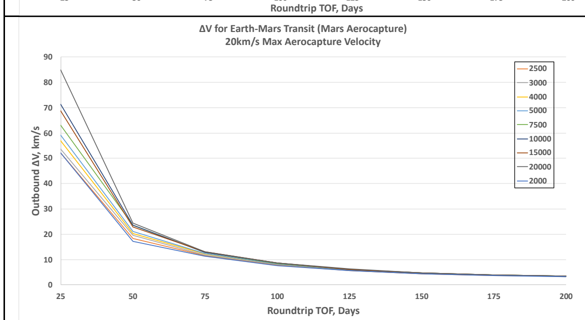




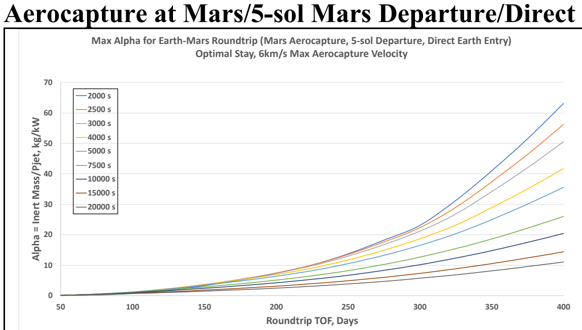


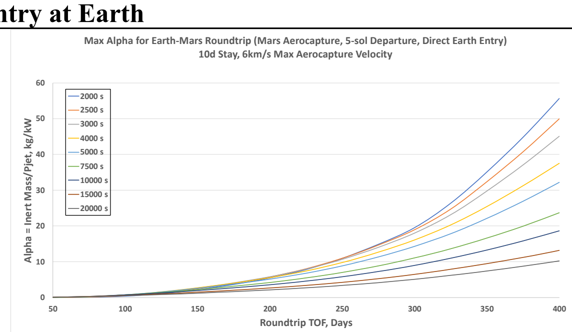


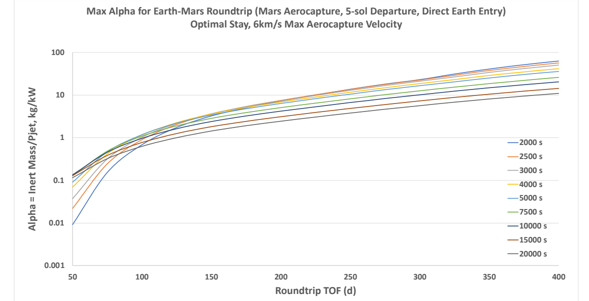


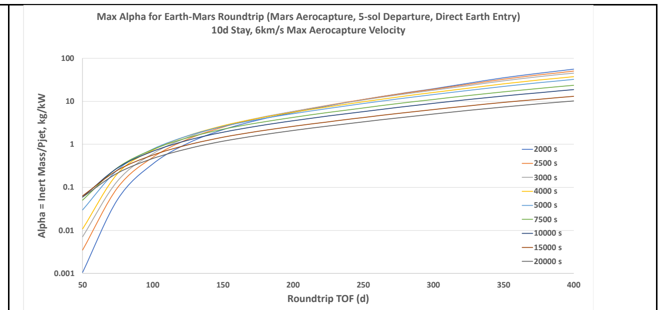


Aerocapture at Mars/5-sol Mars Departure/Direct Entry at Earth






Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
Optimal Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF (d)
2000 s
2500 s
3000 s
4000 s
5000 s
7500 s
10000 s
15000 s
20000 s


Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF (d)
2000 s
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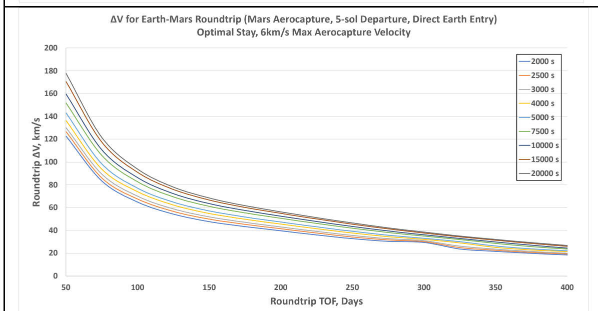
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
Optimal Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
2000 s
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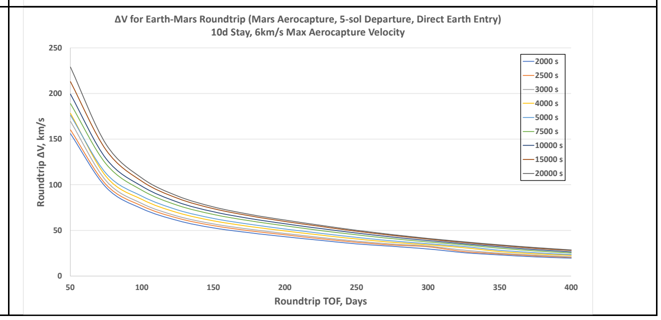
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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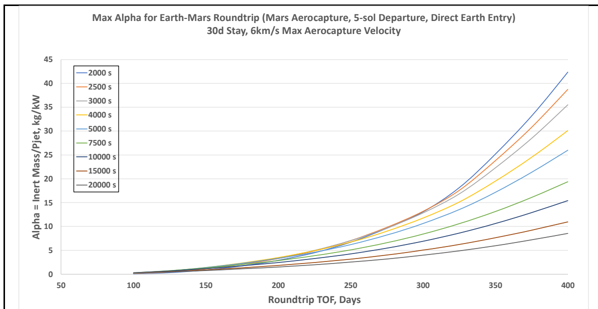
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
30d Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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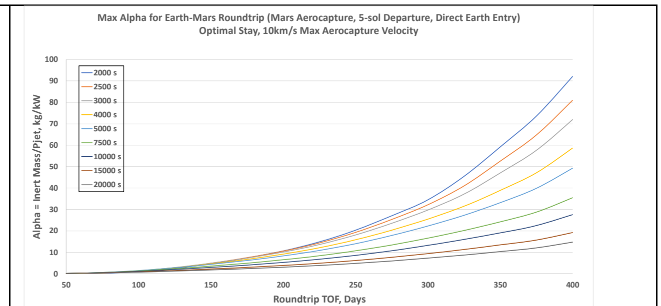
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
Optimal Stay, 10km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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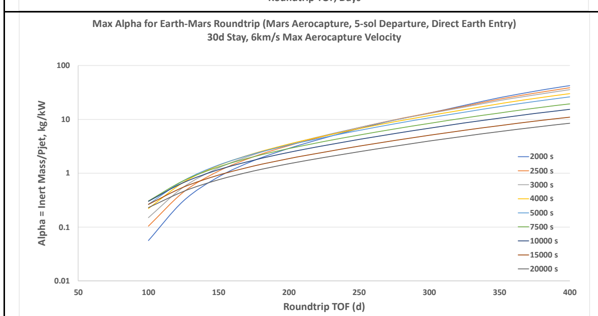
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
30d Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF (d)
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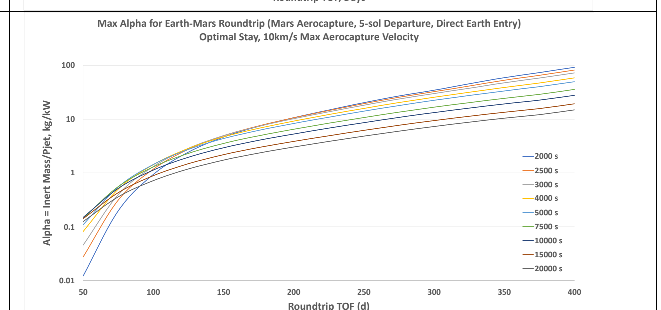
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
Optimal Stay, 10km/s Max Aerocapture Velocity
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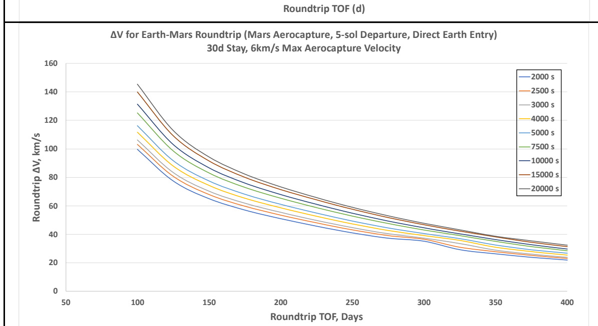
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
30d Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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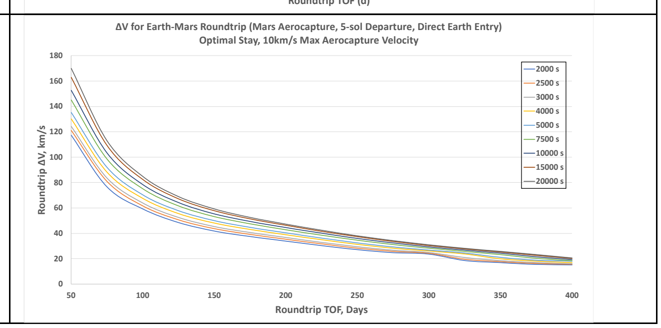
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
Optimal Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
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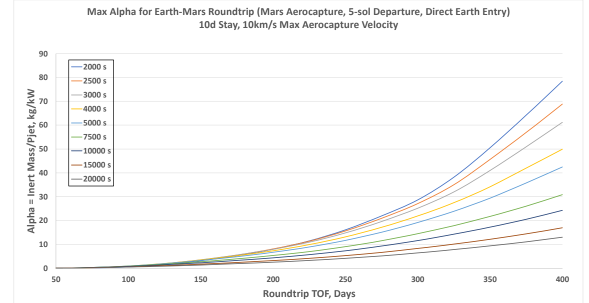
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
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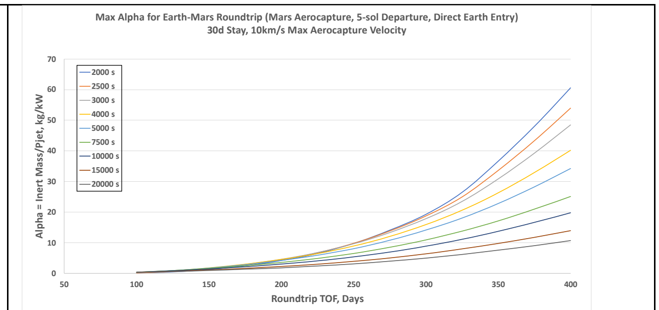
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30d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
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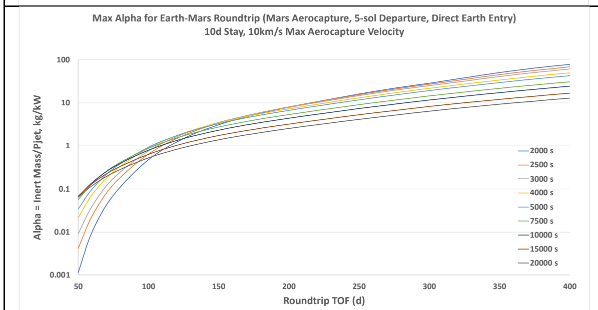
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF (d)
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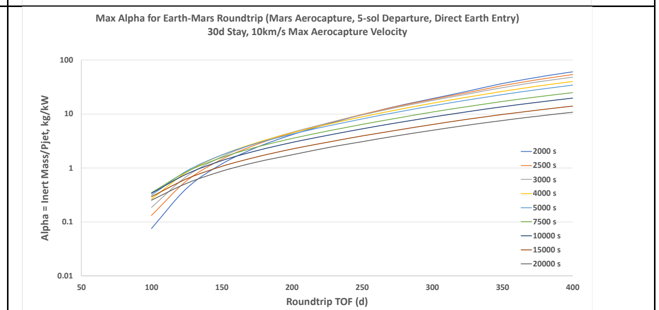
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
30d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF (d)
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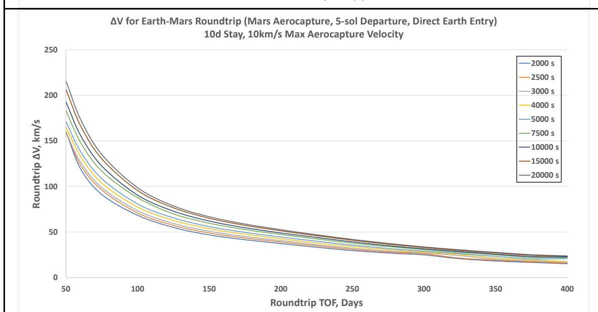
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
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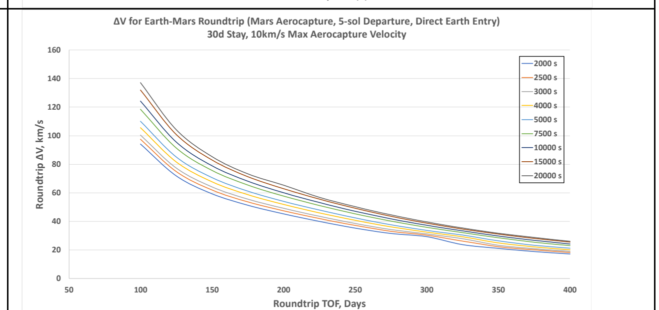
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
30d Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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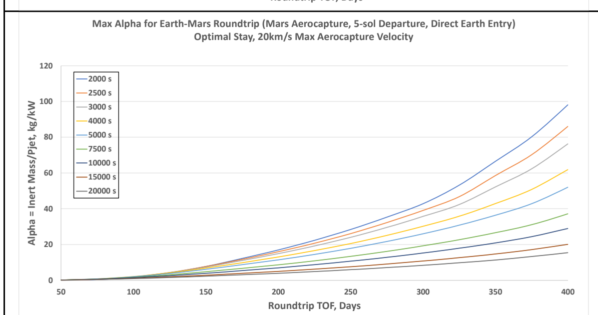
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Optimal Stay, 20km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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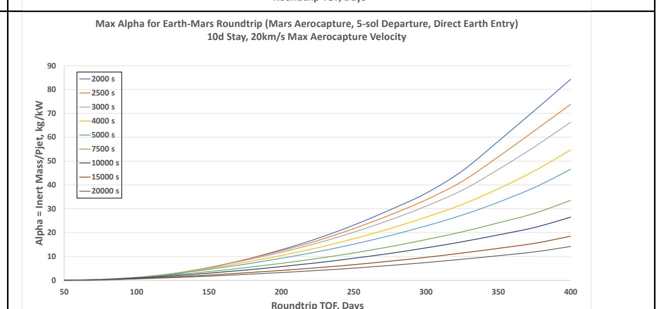
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 20km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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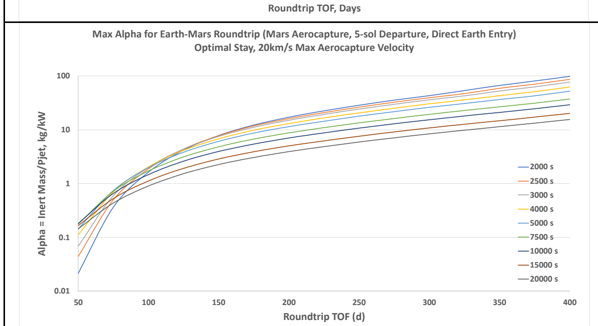
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
Optimal Stay, 20km/s Max Aerocapture Velocity
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Roundtrip TOF (d)
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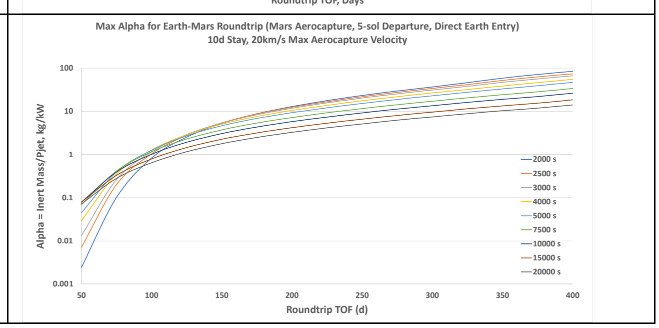
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Direct Earth Entry)
10d Stay, 20km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF (d)
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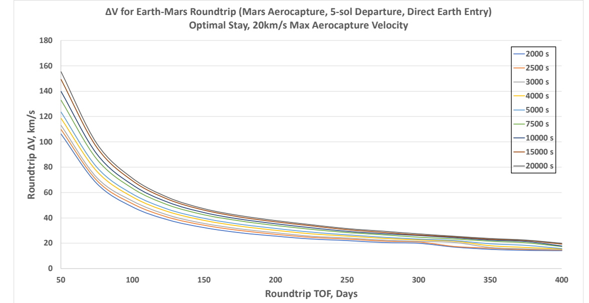
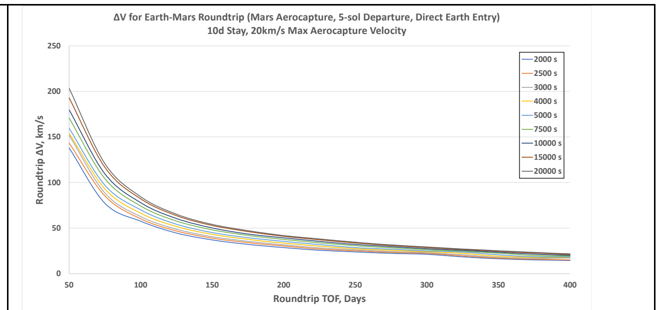
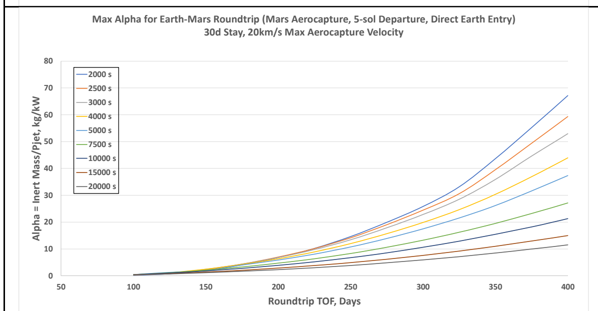
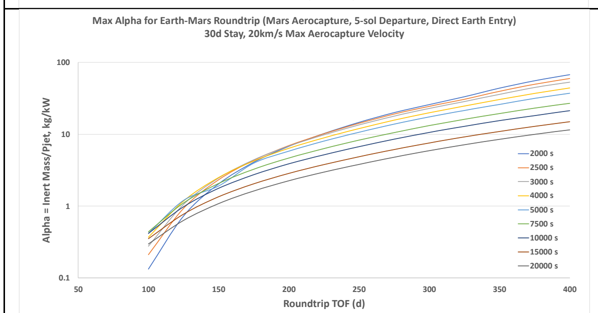
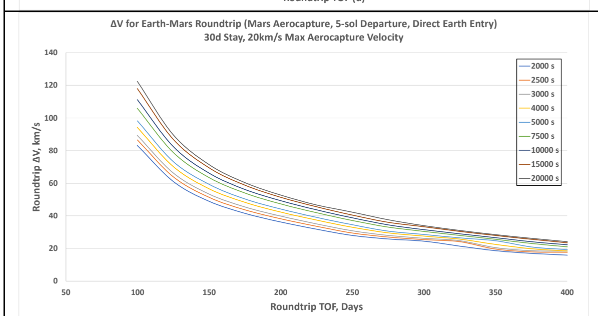


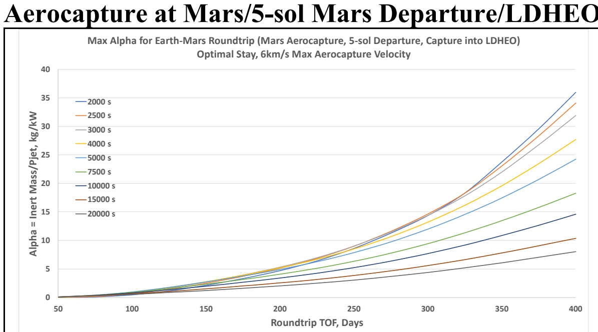


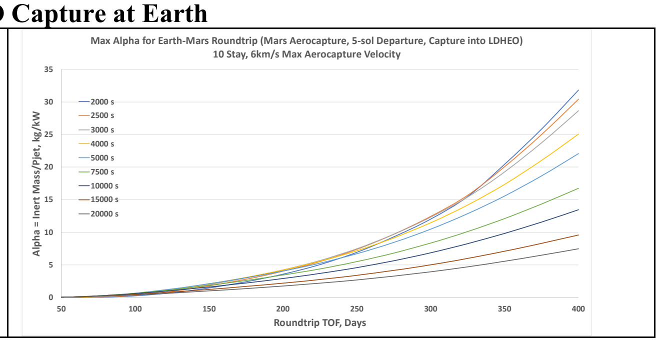


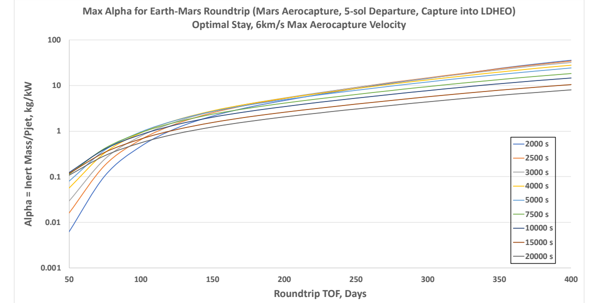




Aerocapture at Mars/5-sol Mars Departure/LDHEO Capture at Earth






Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
Optimal Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
2000 s
2500 s
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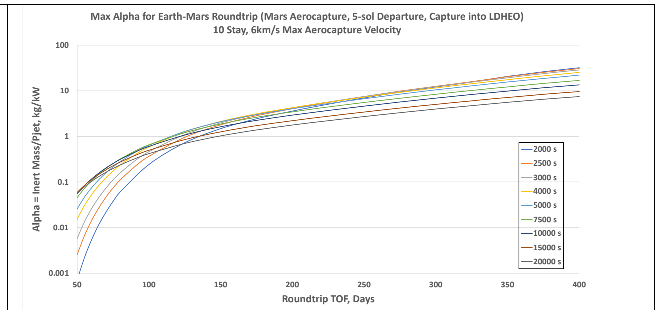
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
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Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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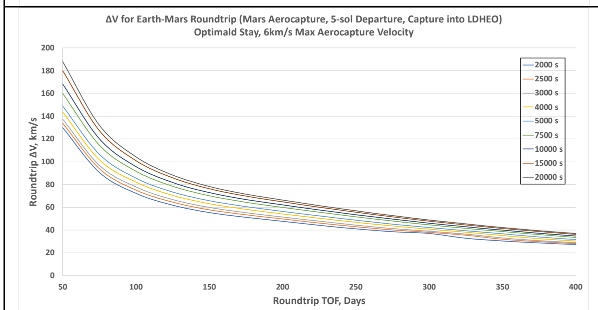
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
Optimald Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
2000 s
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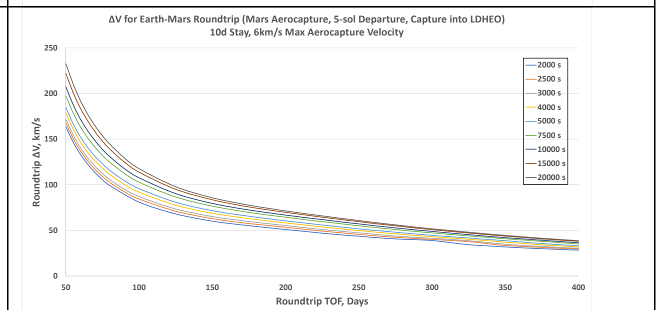
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
10d Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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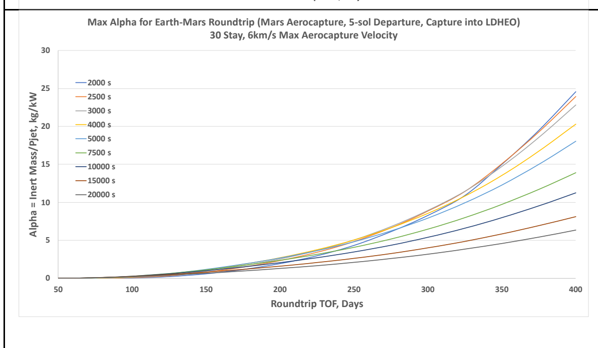
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
30 Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
2000 s
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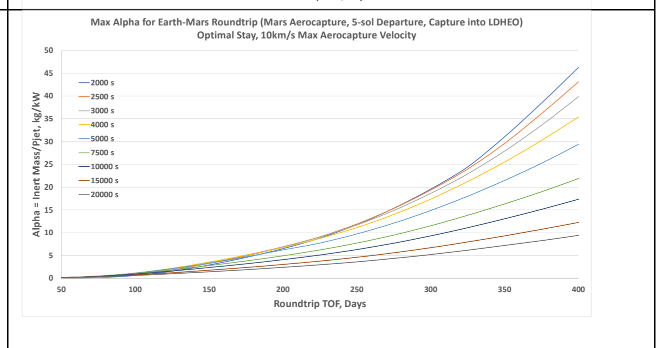
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
Optimal Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
2000 s
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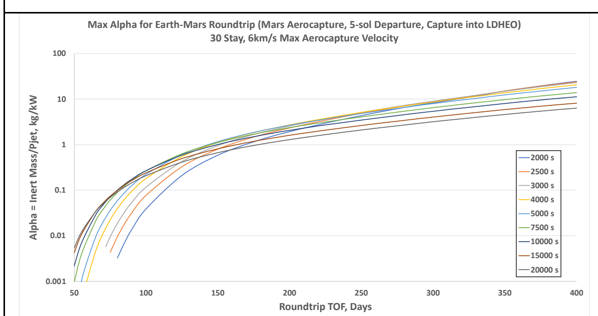
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
30 Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
2000 s
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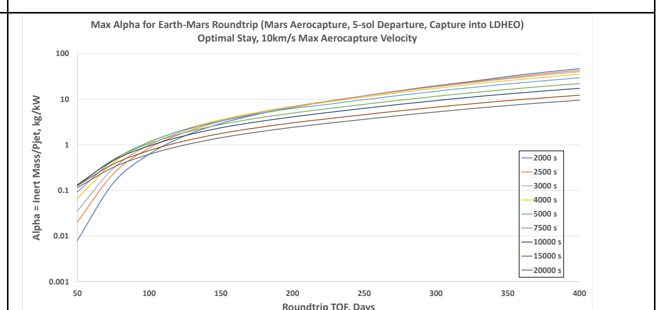
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Optimal Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
2000 s
2500 s
3000 s
4000 s
5000 s
7500 s
10000 s
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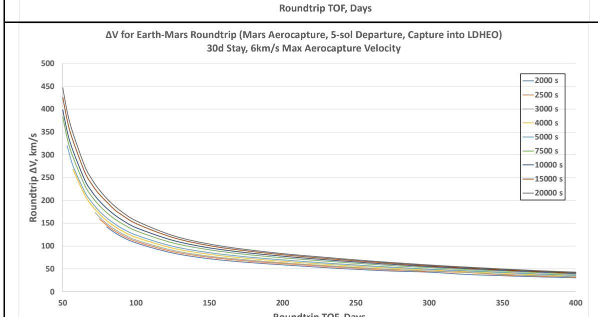
20000 s


ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
30d Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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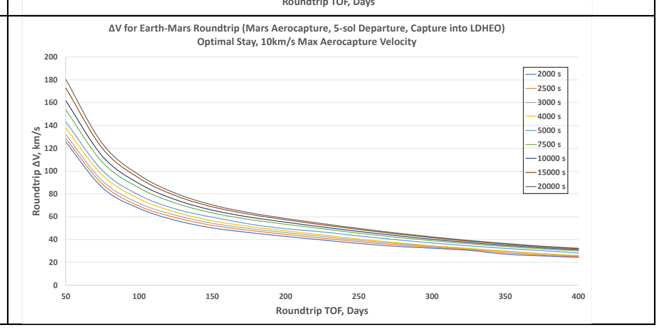
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
Optimal Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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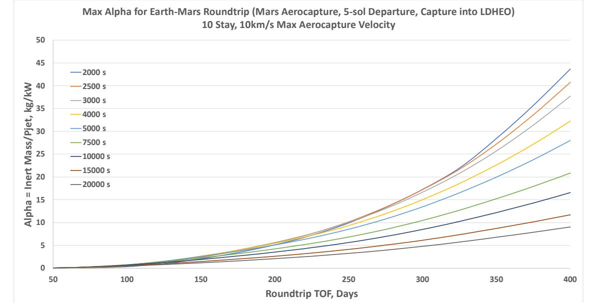
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
10 Stay, 10km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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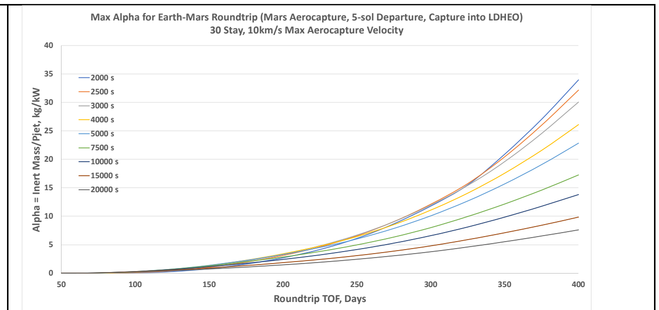
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
30 Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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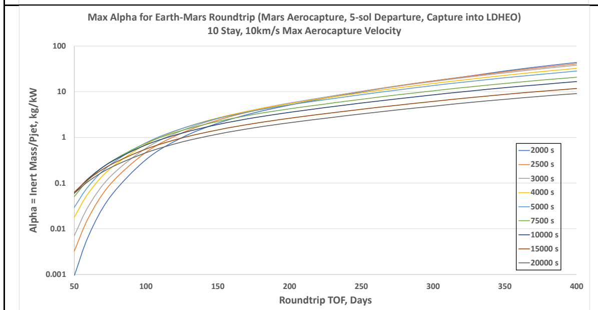
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
10 Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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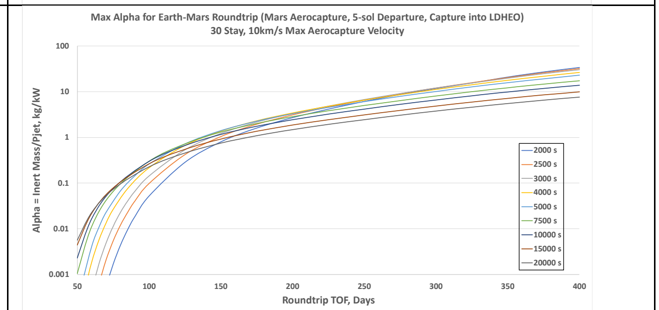
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
30 Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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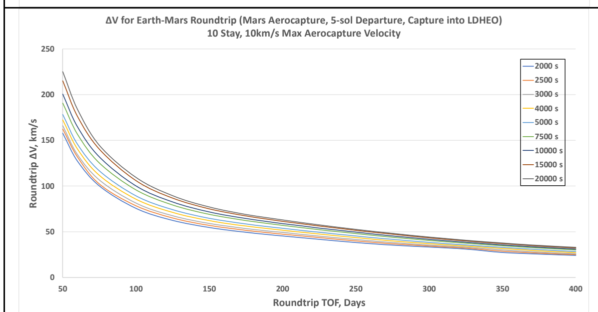
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
10 Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
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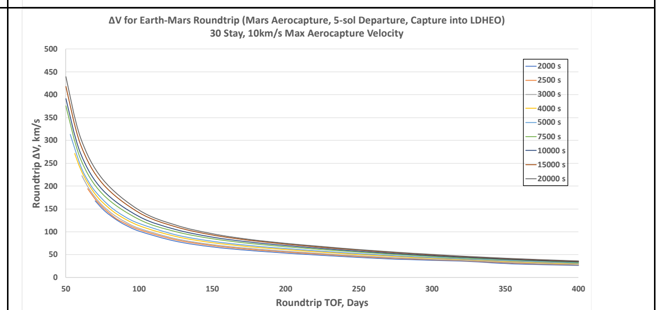
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
30 Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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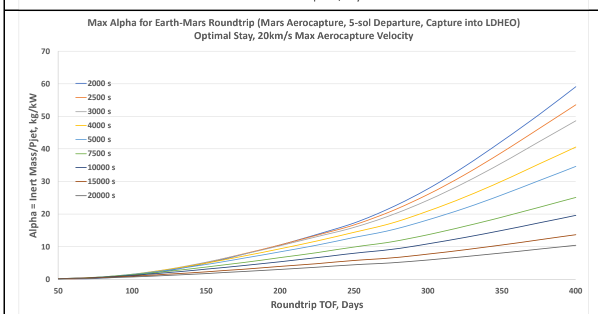
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
Optimal Stay, 20km/s Max Aerocapture Velocity
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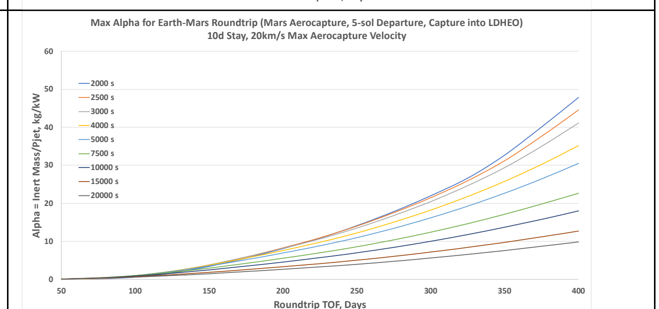
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
10d Stay, 20km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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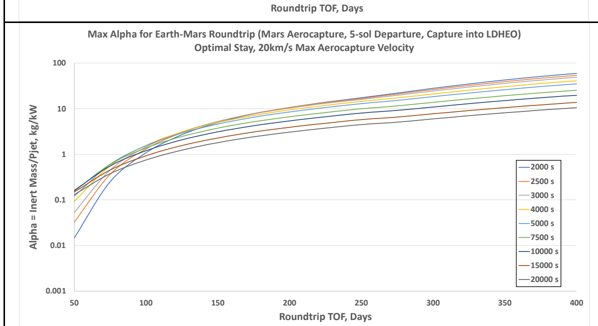
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 5-sol Departure, Capture into LDHEO)
Optimal Stay, 20km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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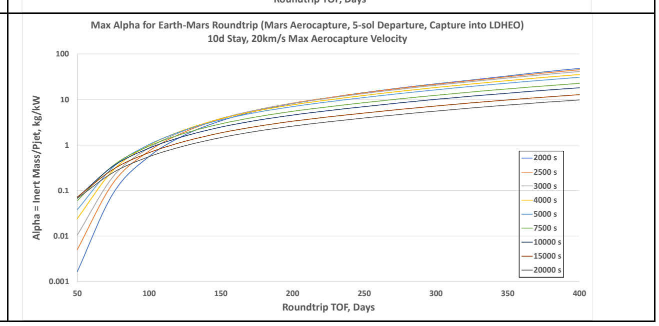
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Alpha = Inert Mass/Pjet, kg/kW
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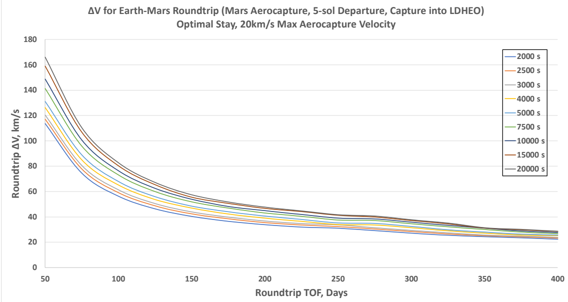
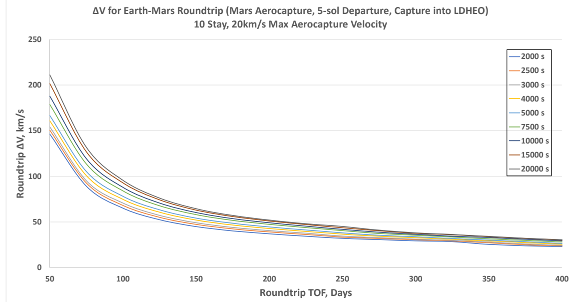
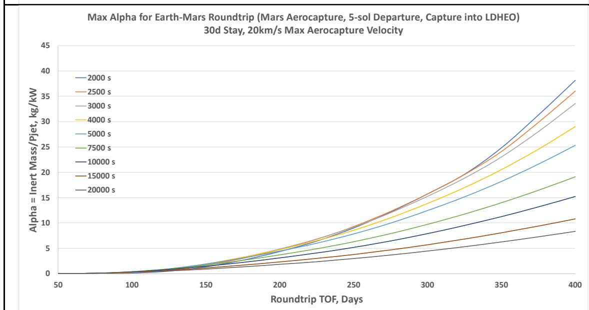
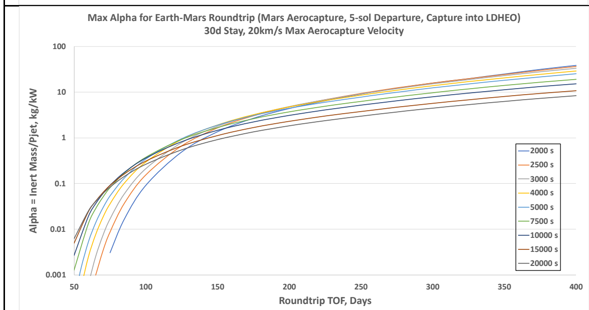
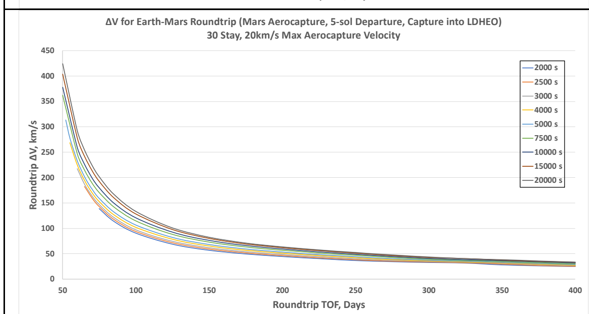


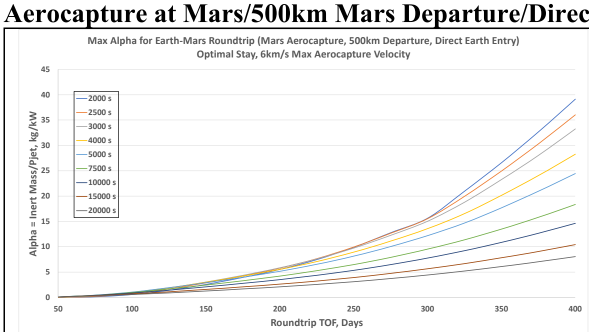


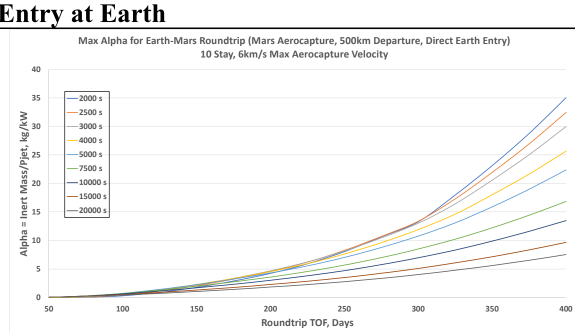


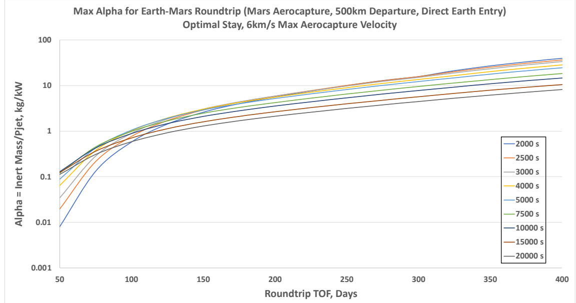




Aerocapture at Mars/500km Mars Departure/Direct Entry at Earth






Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
Optimal Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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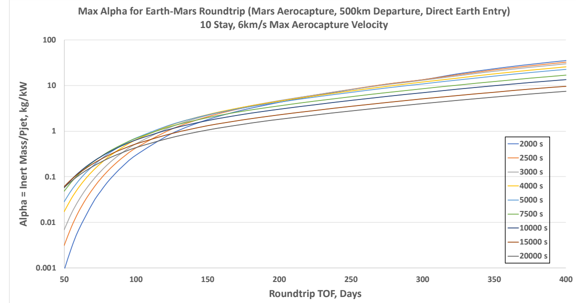
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
10 Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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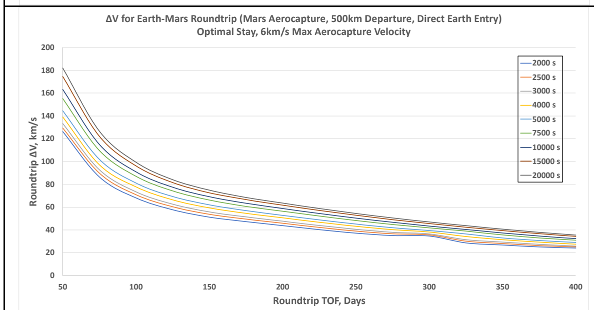
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
Optimal Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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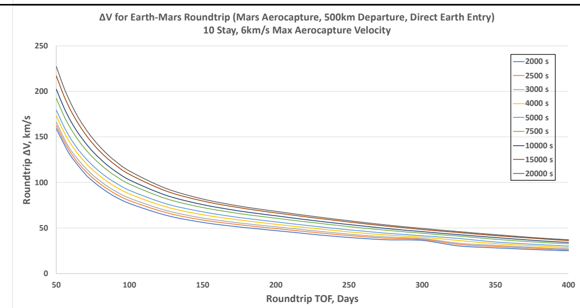
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
10 Stay, 6km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
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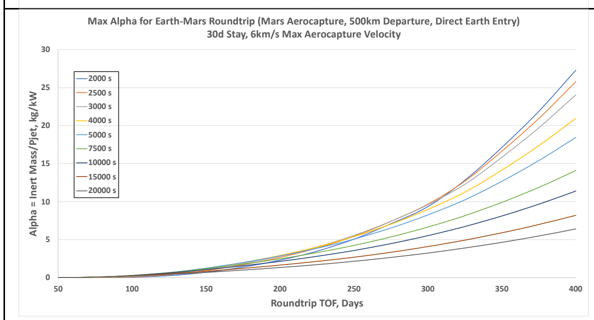
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
30d Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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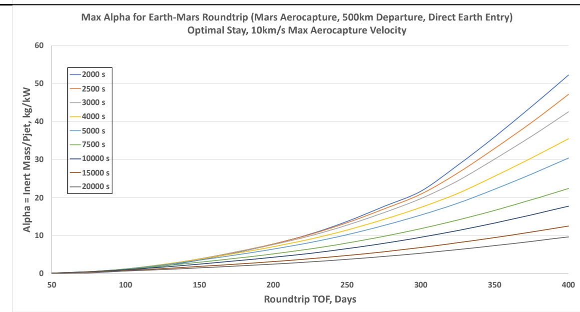
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
Optimal Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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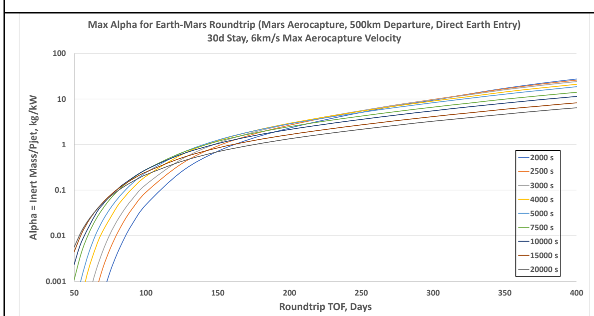
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
30d Stay, 6km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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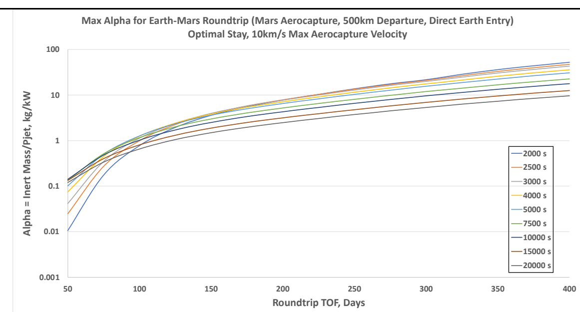
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
Optimal Stay, 10km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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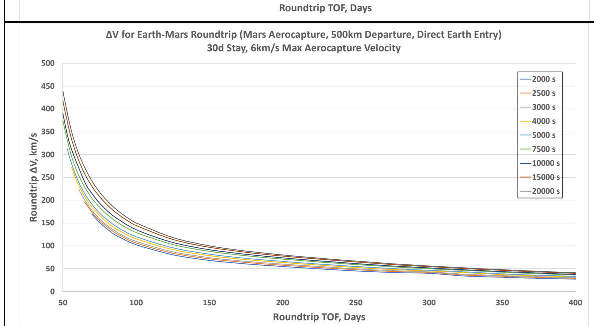
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
30d Stay, 6km/s Max Aerocapture Velocity
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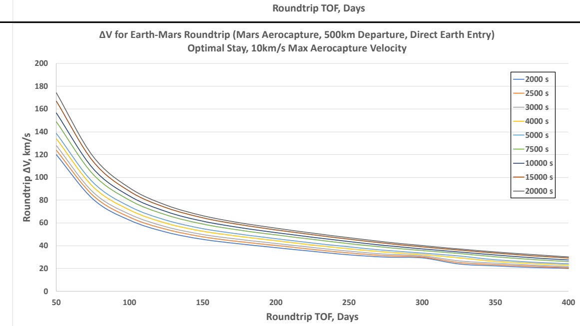
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
Optimal Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
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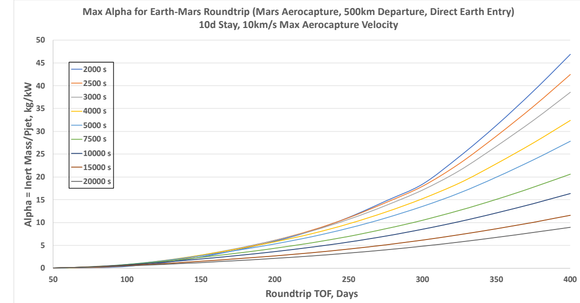
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
10d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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50 100 150 200 250 300 350 400
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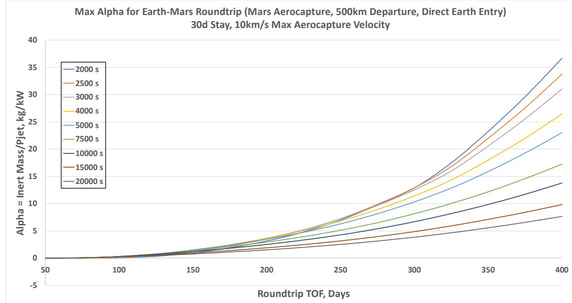
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
30d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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50 100 150 200 250 300 350 400
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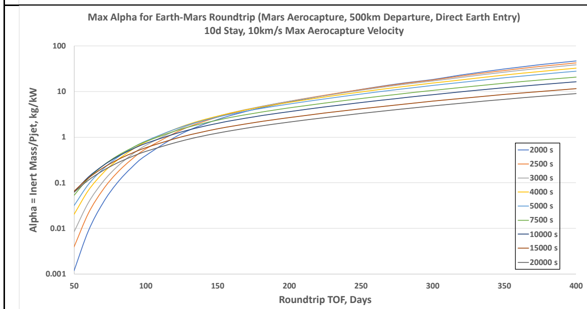
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
10d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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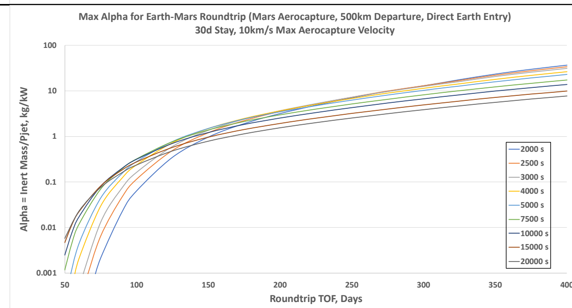
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30d Stay, 10km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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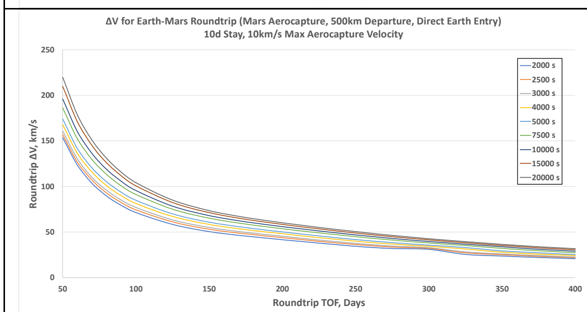
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10d Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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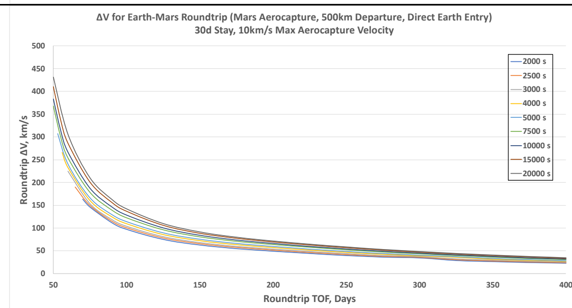
ΔV for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
30d Stay, 10km/s Max Aerocapture Velocity
Roundtrip ΔV, km/s
Roundtrip TOF, Days
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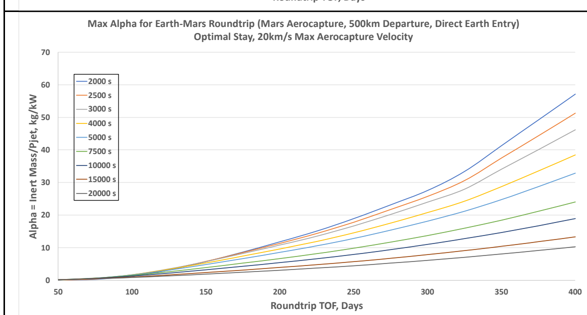
Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Direct Earth Entry)
Optimal Stay, 20km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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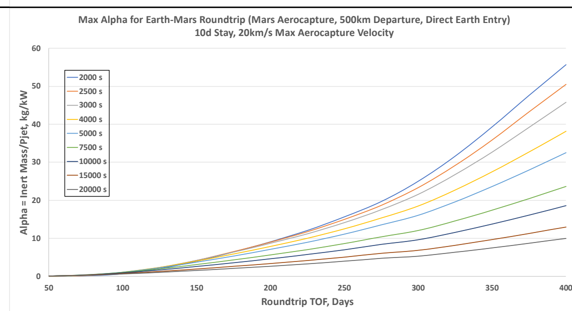
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10d Stay, 20km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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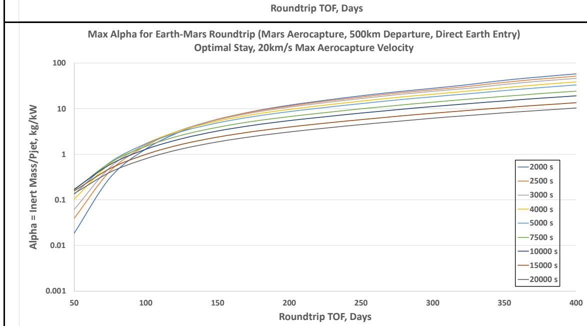
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Optimal Stay, 20km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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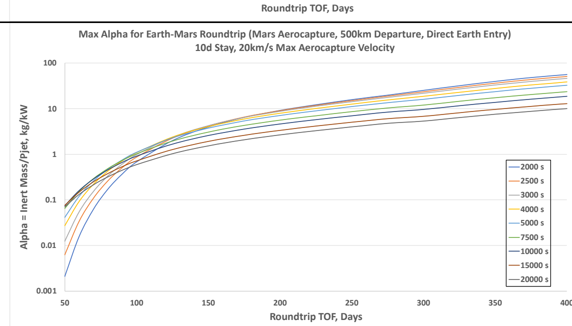
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10d Stay, 20km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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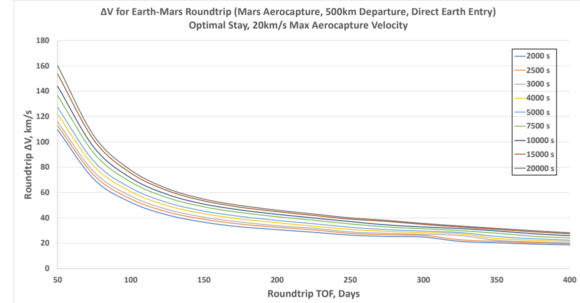
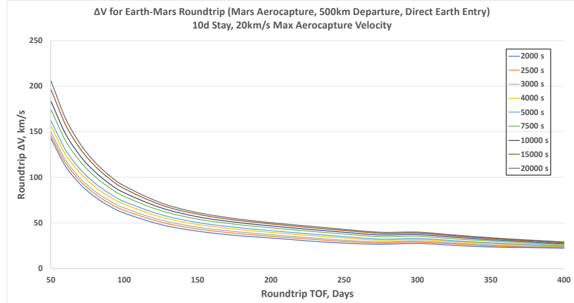
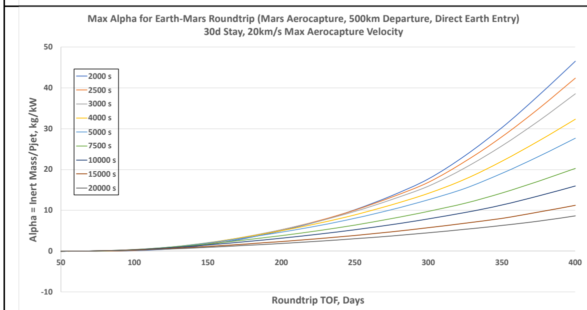
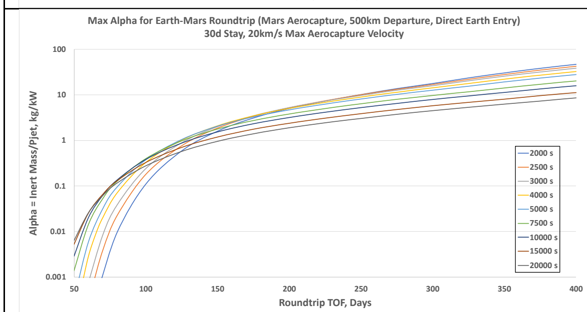
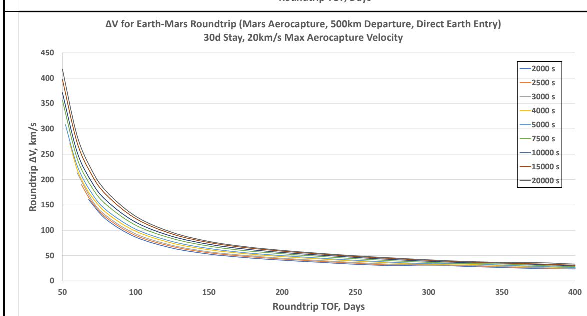


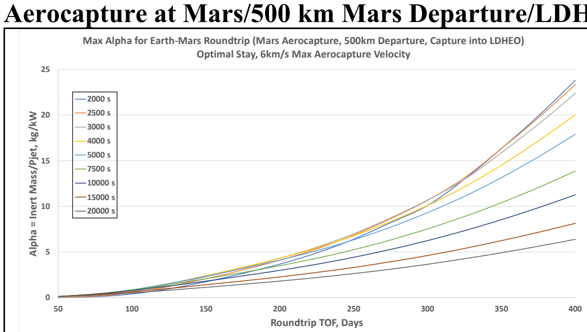


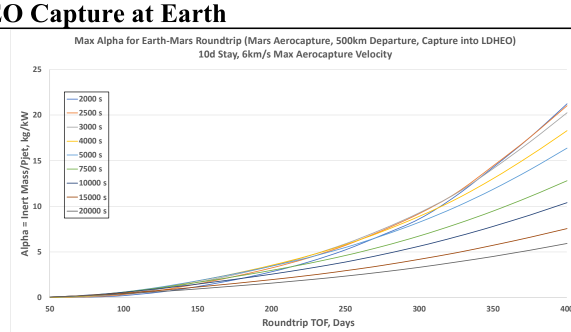


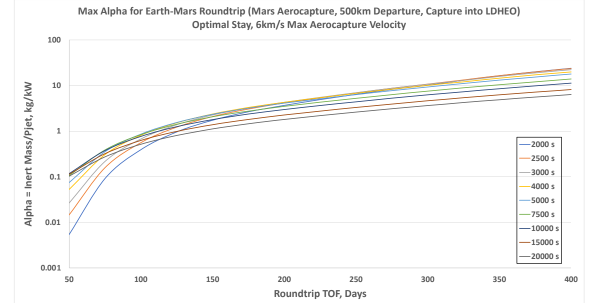




Aerocapture at Mars/500 km Mars Departure/LDHEO Capture at Earth






Max Alpha for Earth-Mars Roundtrip (Mars Aerocapture, 500km Departure, Capture into LDHEO)
Optimal Stay, 6km/s Max Aerocapture Velocity
Alpha = Inert Mass/Pjet, kg/kW
Roundtrip TOF, Days
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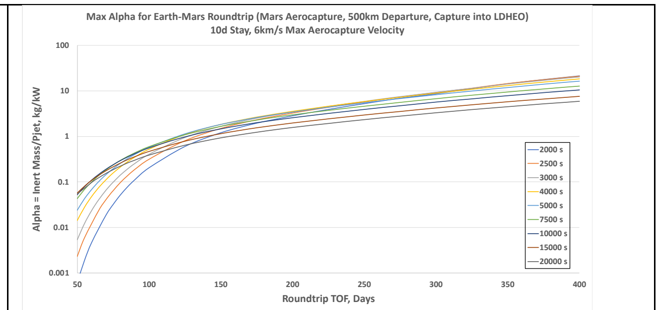
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10d Stay, 6km/s Max Aerocapture Velocity
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Roundtrip TOF, Days
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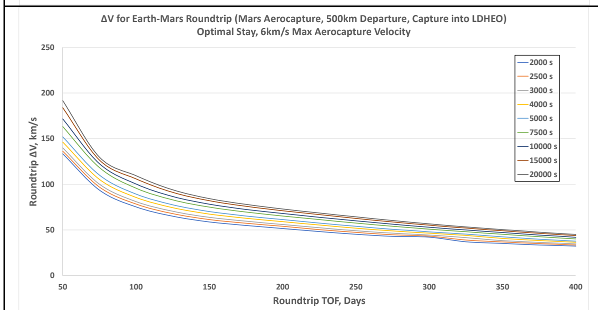
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Optimal Stay, 6km/s Max Aerocapture Velocity
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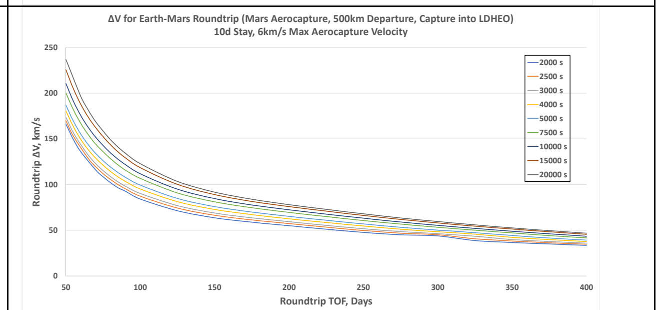
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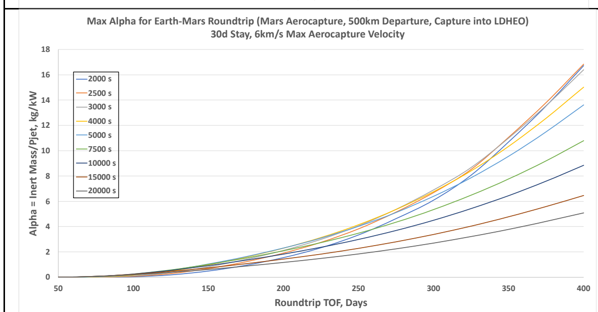
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30d Stay, 6km/s Max Aerocapture Velocity
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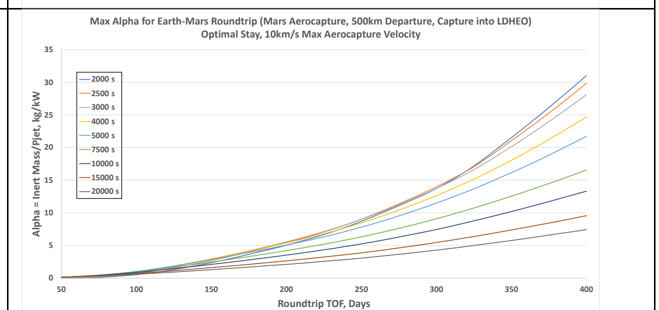
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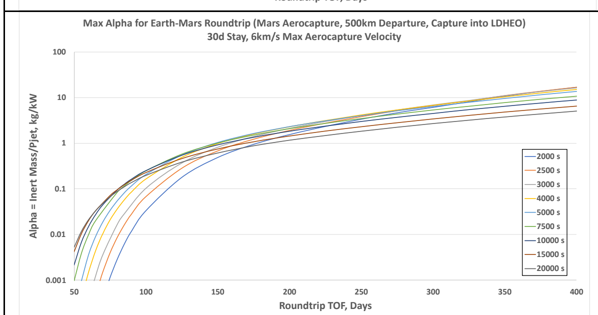
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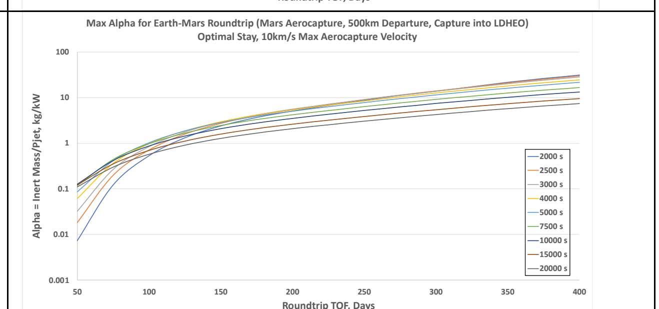
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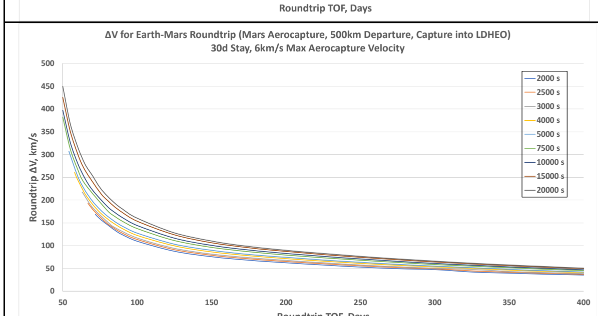
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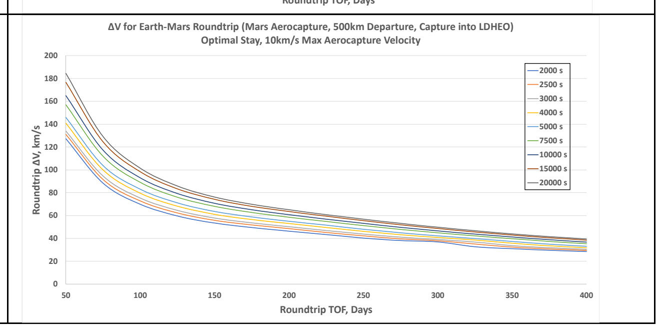
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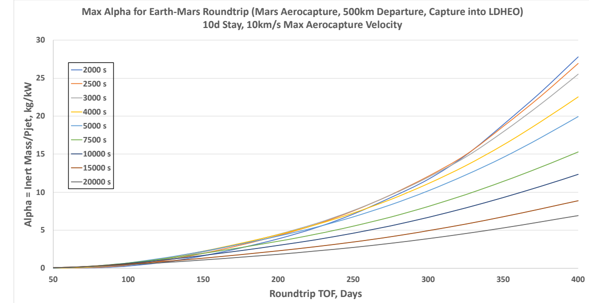
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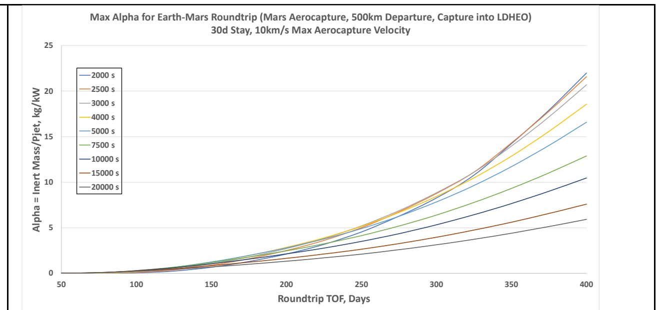
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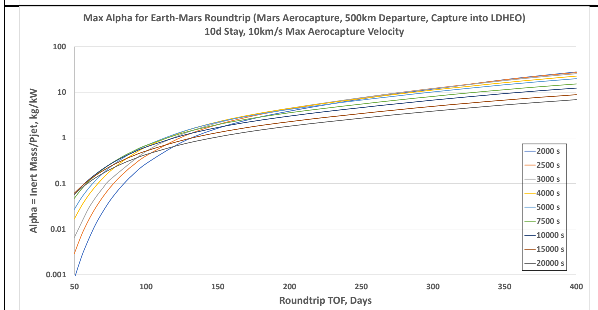
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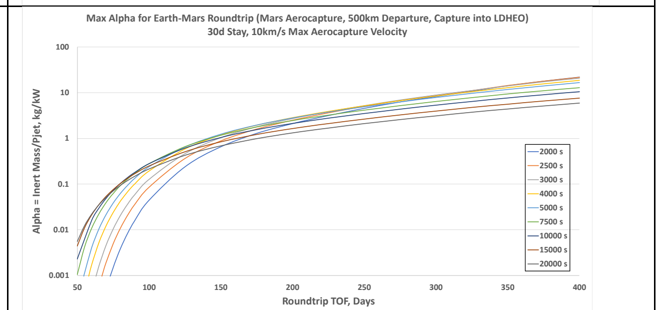
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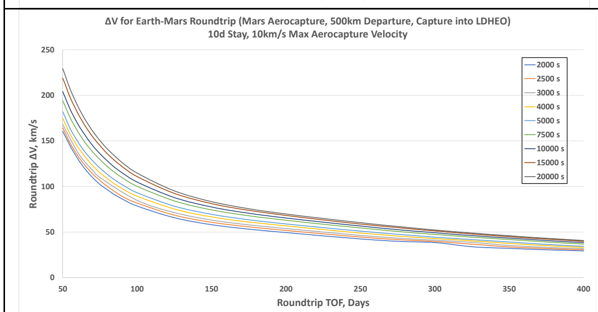
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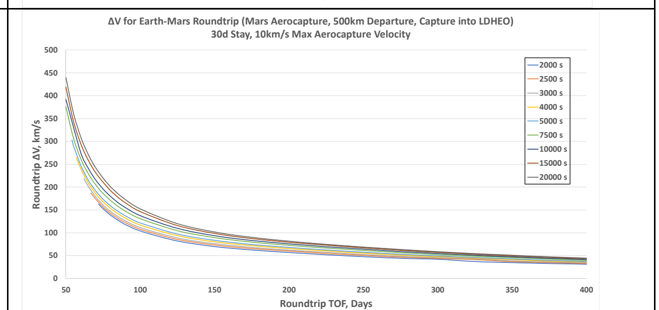
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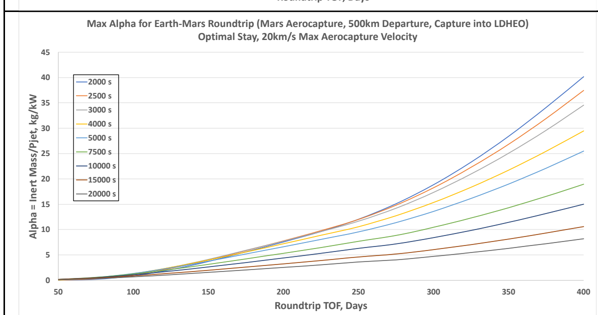
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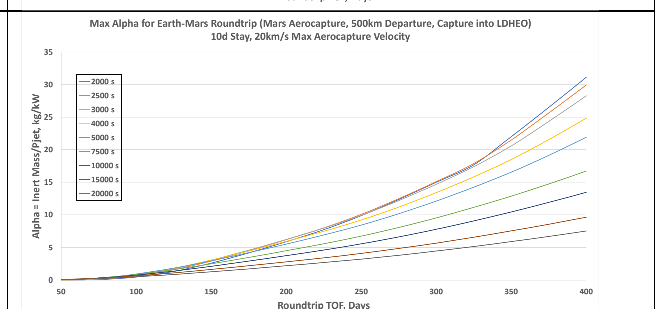
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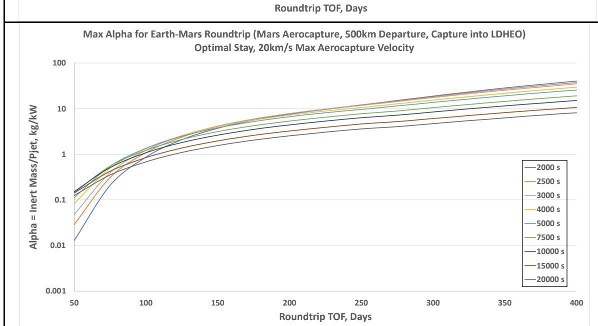
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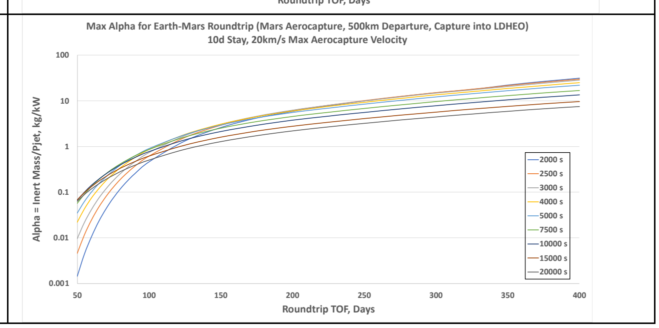
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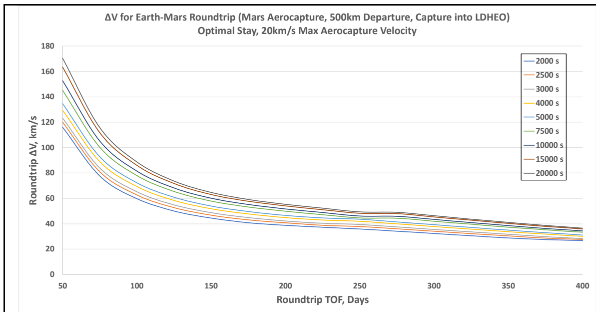
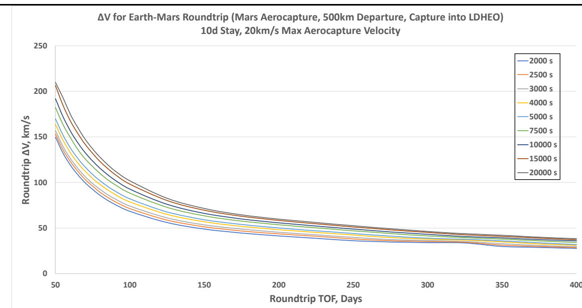
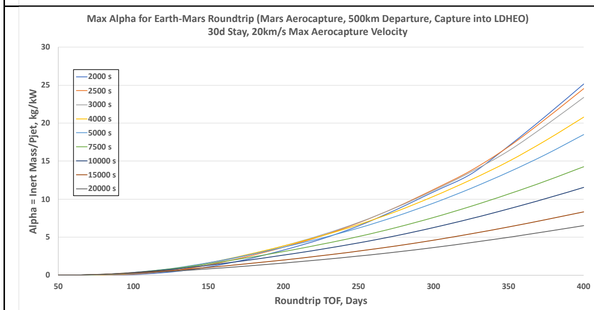
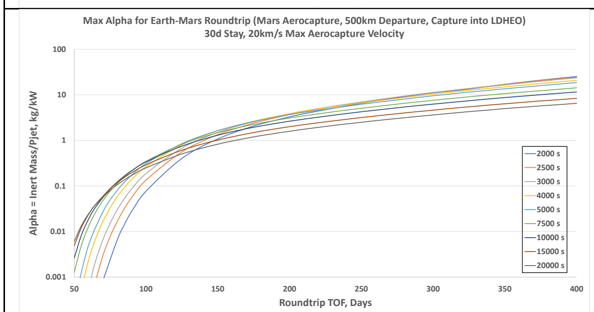
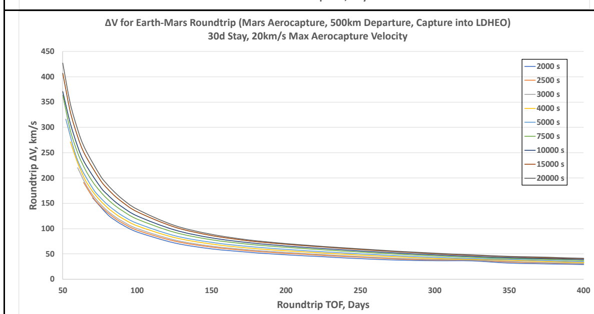


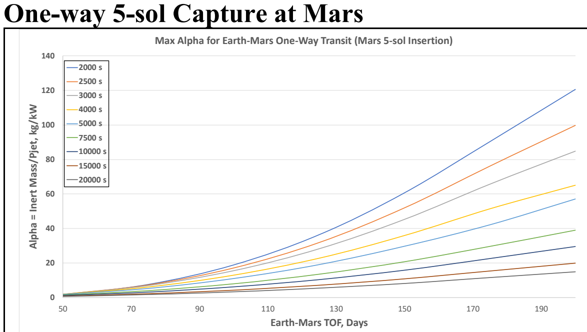


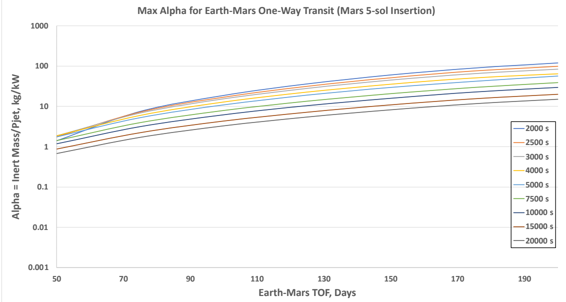


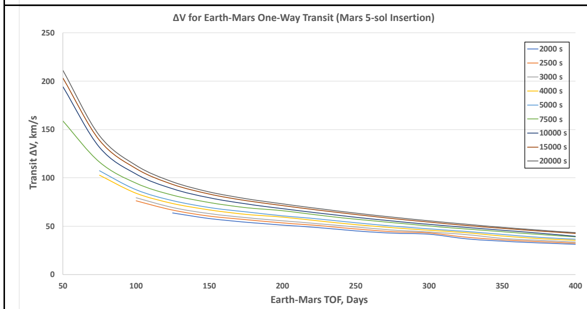




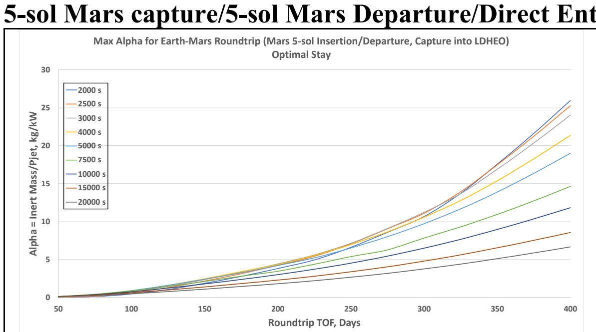
One-way 5-sol Capture at Mars

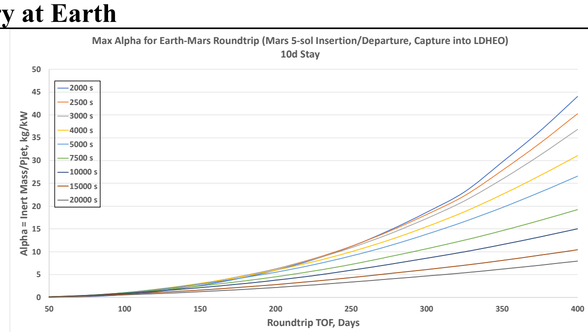


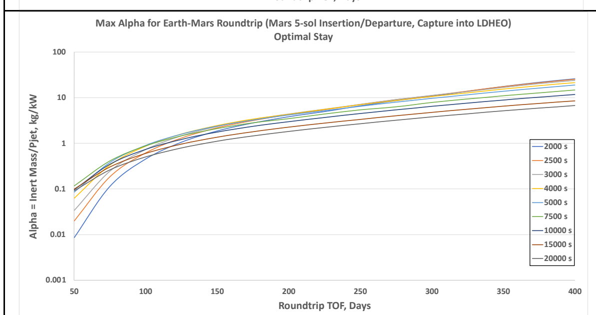


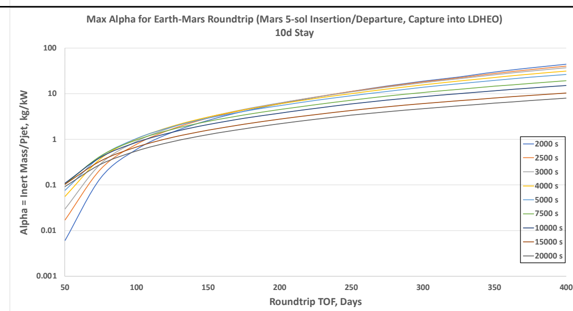


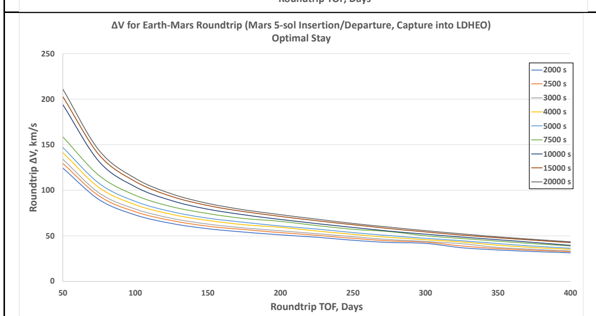
5-sol Mars capture/5-sol Mars Departure/Direct Entry at Earth

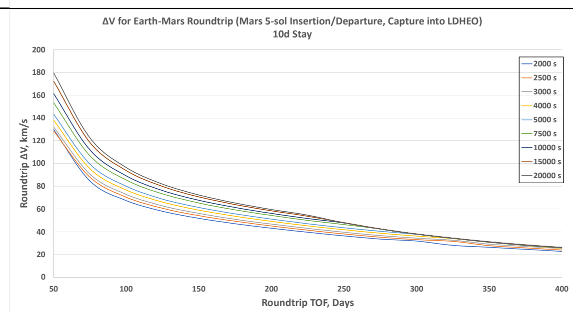


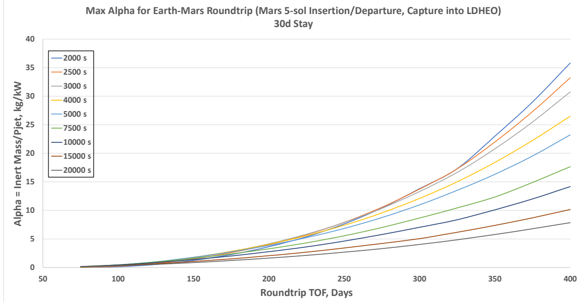


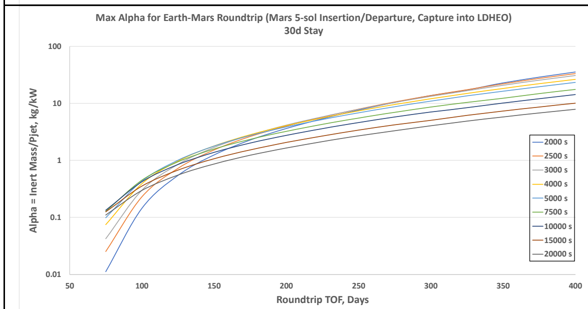


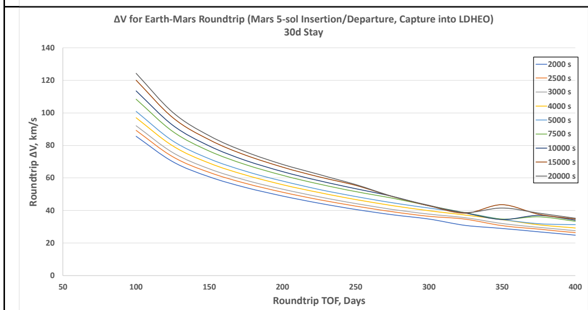




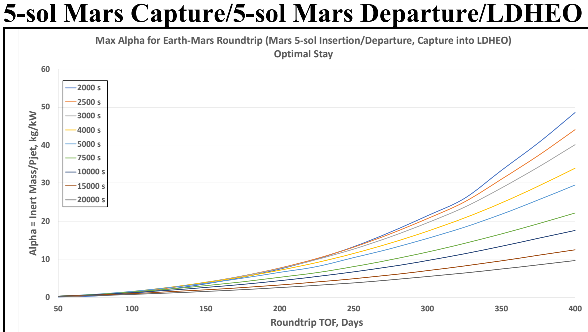


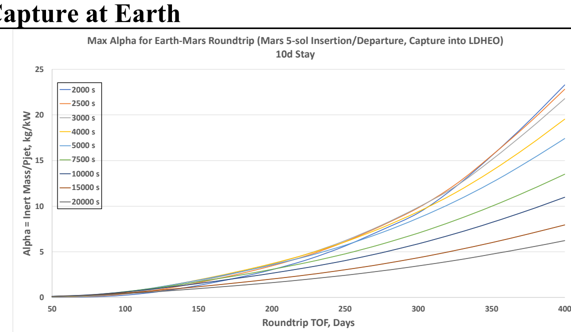


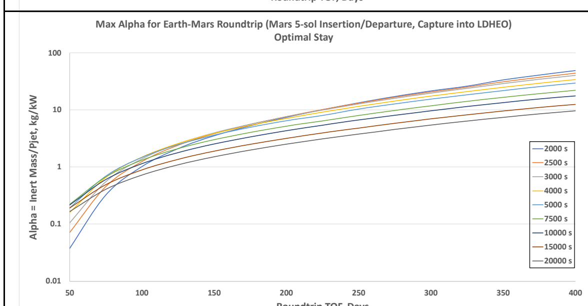


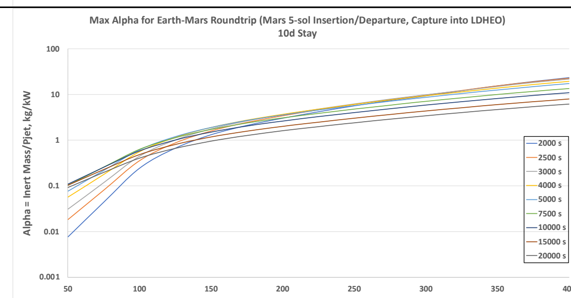


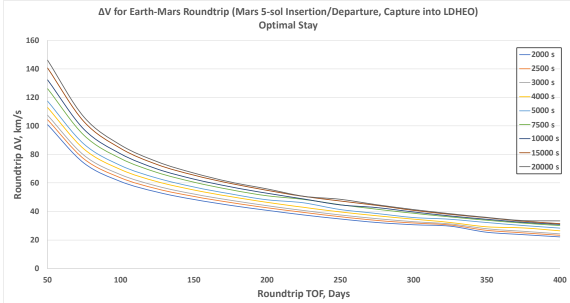
5-sol Mars Capture/5-sol Mars Departure/LDHEO Capture at Earth

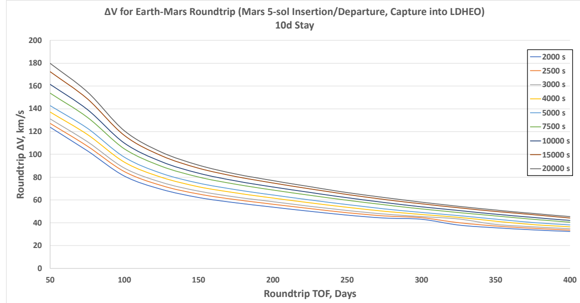


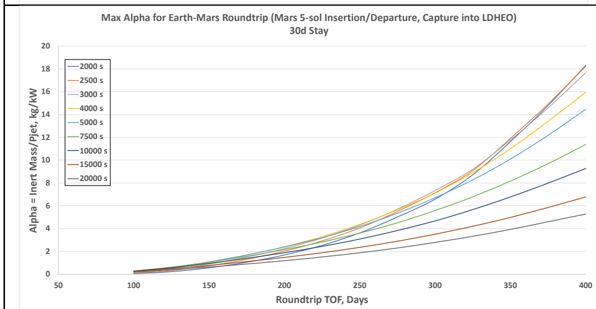


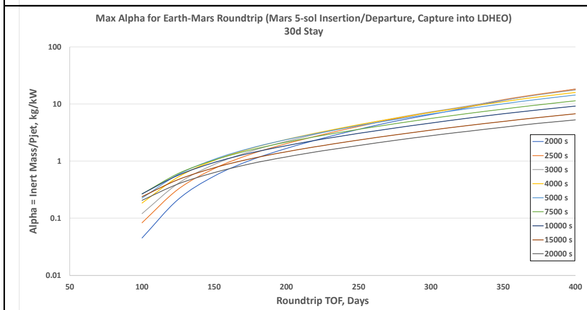


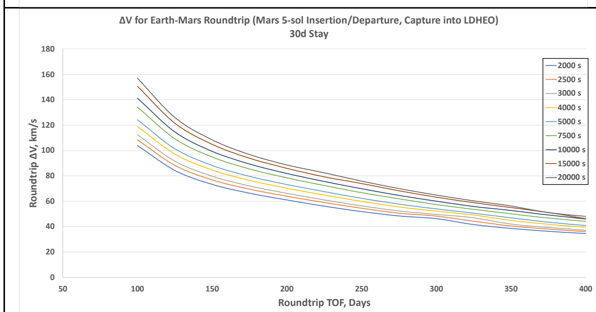












Acknowledgments

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