AAS 15-xxx

# SEP MISSION DESIGN SPACE FOR MARS ORBITERS

**Ryan C. Woolley[*] and Austin K. Nicholas[†]**

The advancement of solar-electric propulsion (SEP) technologies and larger, light-weight solar arrays offer a tremendous advantage to Mars orbiters in terms of both mass and timeline flexibility. These advantages are multiplied for round-trip orbiters (e.g. potential Mars sample return) where a large total ΔV would be required. In this paper we investigate the mission design characteristics of mission concepts utilizing various combinations and types of SEP thrusters, solar arrays, launch vehicles, launch dates, arrival dates, etc. SEP allows for > 50% more mass delivered and launch windows of months to years. We also present the SEP analog to the ballistic Porkchop plot – the "Bacon" plot.

## INTRODUCTION

The allure of solar electric propulsion (SEP) has been obvious for decades – an order-of-magnitude increase in specific impulse (Isp) compared to conventional chemical propulsion. For interplanetary missions which require significant ΔV this translates to a marked decrease in required propellant mass and hence, total launch mass.[1] This allows for more payload mass on smaller launch vehicles which can reduce cost, and in some cases enables the mission altogether. One of the principle drawbacks of SEP is that it provides 2-3 orders of magnitude less thrust when compared to chemical propulsion. Low-thrust trajectories are inherently less efficient and in many cases can take substantially more time versus their impulsive counterparts. The analysis of low-thrust trajectories requires the use of optimization techniques whose results are highly dependent on mission parameters such as thruster type, power level, sun distance, dry mass, etc.

One application where SEP really shines is for a potential Mars Sample Return (MSR). The most common architecture calls for a multi-mission campaign which includes a round-trip orbiter that would retrieve the samples from low-Mars orbit and would return them to Earth.[2] Typical ballistic trajectories require 3-4 km/s of ΔV, which equates to a fuel mass fraction of more than 75% for chemical propellants. The use of high Isp SEP thrusters can reduce the fuel mass significantly and many studies have shown that much smaller, cheaper architectures are possible – including the possibility of using just one medium-size launch vehicle for both the orbiter and lander.[3,4,5,6,7,8]

There are various reasons that the incorporation of SEP has not yet advanced past the conceptual level. Some reasons include: perceived immaturity of thruster technology, lack of availability of low-mass, high-power solar arrays, and perceived technology development cost risk. In addition to the technology risks, many early analyses with undersized thrusters and lower powers show very

[*] Mission Design Engineer, Inner-Planets Mission Analysis Group, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109.
[†] Systems Engineer, Assigned Pre-Projects Systems Engineering Group, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Drive, Pasadena, CA 91109.

long flight times that increase operational costs, lifetime concerns, and radiation risks. When these underpowered systems are forced to have flight times comparable to ballistic trajectories most of the advantages of SEP are marginalized and the perceived risk increase is warranted.

In this paper we explore the use of mature, high-power Hall-effect and ion thrusters over a wide array of powers and other parameters in order to characterize where SEP works best and quantify its true benefit. By exploring a wide range of parametric combinations we are able to create a better map of the tradespace we seek to explore. This allows us to see the "lay of the land" in order to evaluate where our desired mission might fit best, and to know whether any "peaks, valleys, plateaus, or cliffs" may lay in the vicinity. By plotting performance parameters versus launch and arrival dates we can create the SEP analog to a ballistic Porkchop plot – the "Bacon Plot."

## ANALYSIS METHODS

In order to gain the large ΔV's necessary for interplanetary transfers SEP thrusters must operate nearly continuously for many weeks to months. This is fundamentally different than conventional trajectories which essentially have one large maneuver to escape Earth and another large breaking maneuver to capture into orbit upon arrival. There are numerous methods to calculate and optimize the durations, locations, and directions of low-thrust maneuvering to find optimal transfers using electric propulsion.[9,10,11,12] The fact that solar panels are being used as an energy source introduces the added complication of diminishing power, and hence engine performance, as the spacecraft travels further from the sun.

In order to find a suitable, optimized SEP trajectory a number of parameters and constraints must be specified. These include engine performance, solar arrays, dry masses, dates, and time-of-flight (TOF) constraints. Other details may also be included such as forced coast periods, mass and power margins, launch vehicle capabilities, etc. If constraints are set too tightly or initial conditions are not specified properly the optimizer may fail to converge on a feasible solution.

Within the range of possible trajectories we must specify the parameter(s) to optimize. This is usually a maximum arrival mass, minimum departure mass, minimum transfer time, or some combination of these. The nature of SEP trajectories is such that there is always a Pareto frontier amongst optimal trajectories representing the trade between performance and time. Figure 1 shows a typical performance vs. time Pareto frontier for a SEP trajectory. On this curve there is a minimum time solution where thrust is continuous throughout the trajectory. This results in a large propellant mass penalty due to the inefficiencies of thrusting at many points along the trajectory. At the opposite end of the curve increased flight time allows the optimizer to only thrust at the most efficient locations – approaching the efficiency of an impulsive transfer at the extreme. Often there is a "knee in the curve" which represents a good trade between propellant mass and time-of-flight.

A new Pareto frontier is created for any change in mission parameters such as engine type or number, power level, payload mass, etc. In order to find the true optimum it necessary to iterate between specific spacecraft systems design and trajectory optimization – since mass, power, propellant, and time are all interrelated. In order to facilitate this rapid iteration between system and trajectory design it was necessary to generate a large SEP trajectory database under a wide range of parameters.

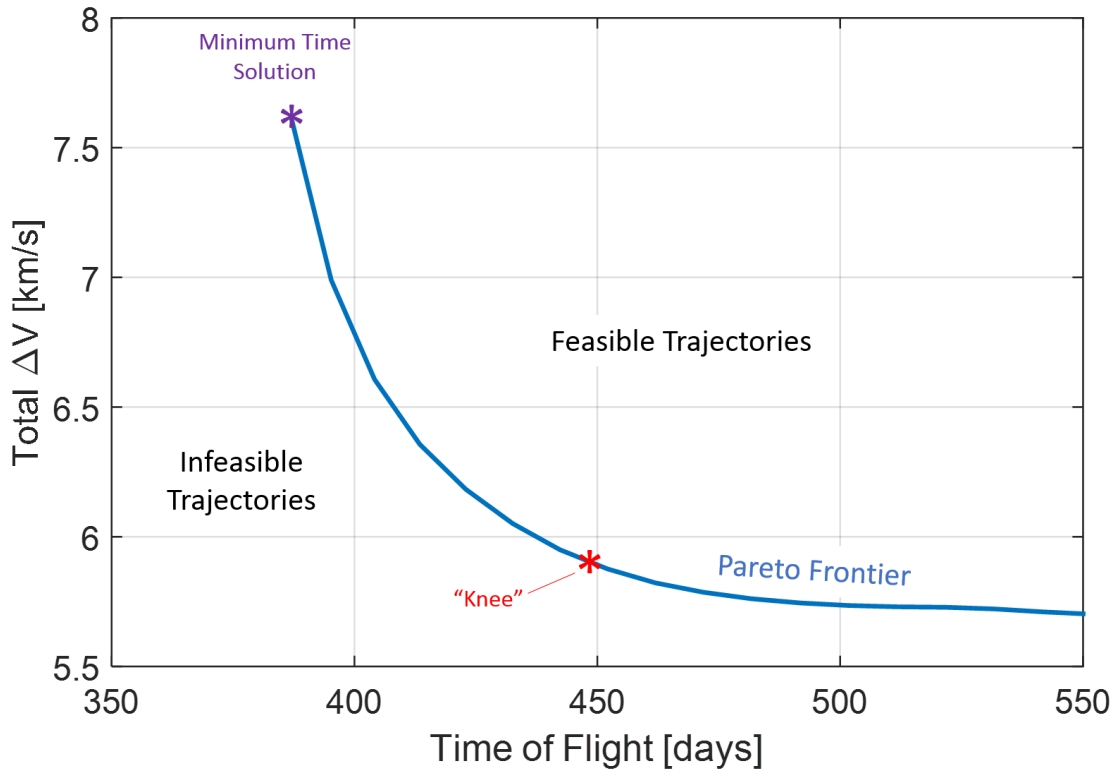


**Figure 1. ΔV vs. Time curve for an Earth-to-Mars transfer using SEP, including spiral to final orbit. This Pareto curve is a feature of all low-thrust transfers - trading performance for time. Mission designers must choose how to define the "knee in the curve" to provide optimality within mission constraints.**

## Data Generation

The mission design analysis was carried out using MALTO, a fast, medium-fidelity low thrust optimizer developed at JPL.[13] This tool generally exhibits robust convergence and can be run in parametric mode with fast, accurate results. We used MATLO to generate hundreds of thousands of trajectories by sweeping through parameters such as engine type and quantity, power level, time-of-flight, launch year, launch vehicle, etc. All of the variations are centered on a Sample Return Orbiter that would be launched in the 2020's. Common assumptions include:

- A positive C3 launch (i.e. no Earth spirals)
- Sample rendezvous in low-Mars orbit (320 km circular)
- 700 W allocation for non-thruster power needs
- 300 kg payload allocation

The parameter levels for the runs are shown in Table 1. Not all possible combination have been run. Each configuration takes a custom input file and some human-in-the-loop interaction to get going, in addition to the computation time itself. A data run of 5,000 trajectories typically requires 4-10 hours on a laptop PC. Configurations were run as needed based around current orbiter studies. For most cases, power levels were swept at 1 kW increments, time-of-flight at 10 days, and entry

mass at 250 kg. These resolutions gave sufficient data for interpolation for mission optimization (discussed in the following section).

**Table 1 -Mission parameters for MALTO trajectory simulations.**

| Discrete Parameters | Value(s) |
|---|---|
| Launch Vehicle | F9, Atlas V 421, 431, 551, FH |
| Launch Years | 2022, 2024, 2026, 2028 |
| Return Years | 2028, 2030, 2033 |
| Engine | BPT-4000, NEXT, XIPS, ARM |
| Engine Mode | Hi Thrust, Hi Isp |
| Number of Engines | 1, 2, 3, 4 |
| Earth Arrival V-infinity | 1.5 km/s, 4.5 km/s |

| Swept Parameters | Value(s) |
|---|---|
| Power Levels | 4 – 38 kW |
| Time of Flight (Heliocentric) | 200 – 600 days |
| Entry Mass at Earth (Inbound only) | 250 – 2500 kg |

**SEP Thrusters**

For interplanetary missions there are many options for high power ( > 4 kW) SEP thrusters that have been flown or are at a high maturity level and ready to be incorporated into near-term missions. For our studies we considered two commercial thrusters – Aerojet's XR-5 (was BPT-4000) and L-3's XIPS; and two NASA-developed thrusters – NEXT (NASA's Evolutionary Xenon Thruster) and ARM (candidate thruster for the Asteroid Redirect Mission). Pertinent characteristics of each engine are shown in Table 2. Many of these thrusters have been proposed for NASA missions. The implications for their use in future mission including heritage, risk, credibility, and modifications is further discussed by Oh, et. al.[14]

**Table 2 - Properties of SEP engines considered.**

| Thruster | Type | Max Isp | Max Thrust | Max Power | Total Mass | Peak Efficiency |
|---|---|---|---|---|---|---|
| ARM | Hall | 3200 sec | 490 mN | 14 kW | 66 kg | 55% |
| NEXT | Ion | 4000 sec | 230 mN | 6.9 kW | 53 kg | 64% |
| XR-5 | Hall | 2000 sec | 250 mN | 4.8 kW | 32 kg | 51% |
| XIPS | Ion | 3500 sec | 175 mN | 5 kW | 37 kg | 60% |

Each engine has a performance curve that relates power to Isp and thrust levels. The Isp versus power curve is shown in Figure 2. Isp and thrust are inversely proportional for a given efficiency and power level. Some engines have the ability to be flown in either a high thrust or high Isp mode. Increased thrust tends to reduce flight times and total $\Delta V$, whereas higher Isp tends to decrease the total fuel mass and increase payload. In general, we found that higher Isp led to the most mass delivered, provided that the mission is not power limited.

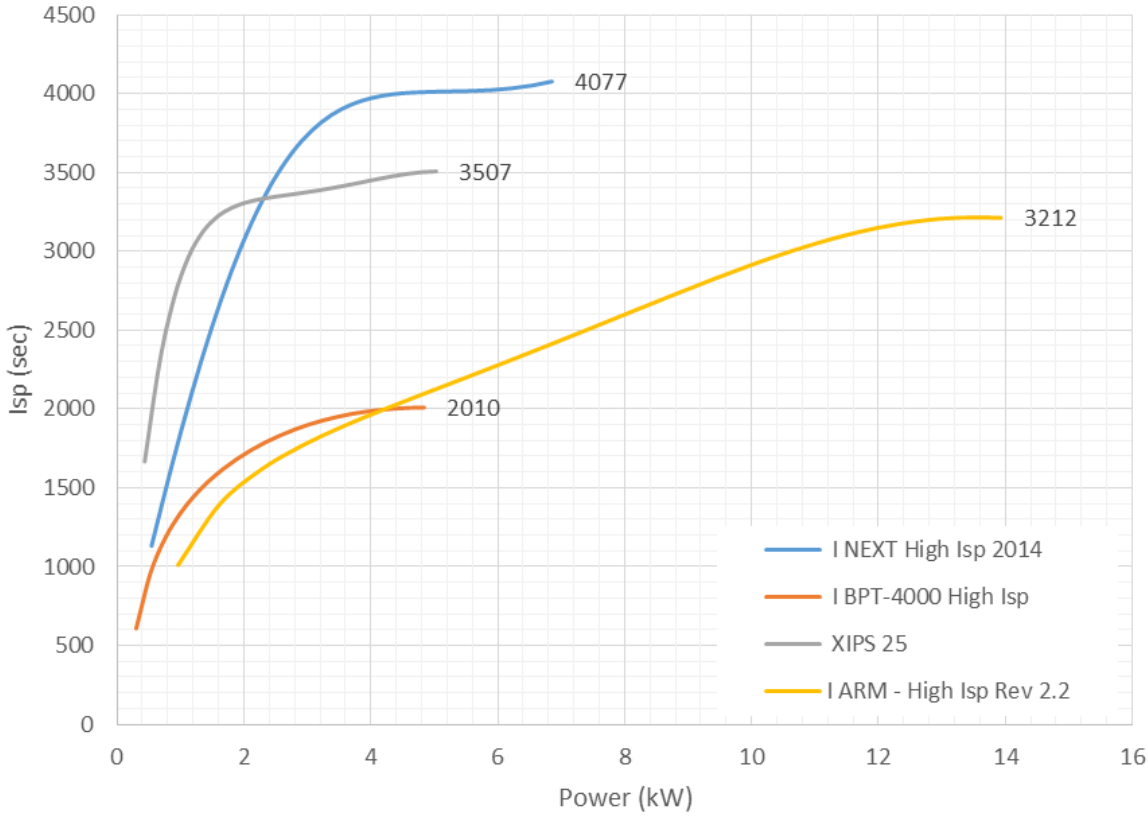


**Figure 2. Isp vs. Power for the four thrusters considered. Note that Isp increases with power up to the thruster's max power, or saturation.**

## OPTIMIZED PARAMETER SWEEPS

Creating a preliminary design for a sample-return orbiter that is fully converged requires an iteration between spacecraft subsystem masses (tanks, power, etc.) and round-trip trajectories that are consistent with those mass and power levels. Full optimization would require iterating over:

- Segment flight durations – heliocentric outbound/inbound, spiral in, spiral out
- Thruster selection – type, number, operating mode (high Isp vs. high thrust)
- Power levels – solar array type, size, and efficiency
- Masses – to accommodate thruster selection, power systems, propellant loads

The construction of end-to-end trajectories of all the permutations required to fully explore the space is simply not feasible. Even if the continuous parameters such as power level, masses, stay times, and flight durations of each segment were discretized, the number of trajectories needed would be the full factorial combination of the variable parameters. In practice, a much simpler approach is typically used – a thruster set, power level, and notional mass are selected to create a single trajectory. Then the spacecraft subsystems are designed to meet that allocation, resulting a in a point design that may be far from optimal. Neither of these extremes gives a desirable result of a nearly optimal spacecraft design produced in reasonable time.

In order to make the optimization problem more tractable, the outbound (Earth-to-Mars) and inbound (Mars-to-Earth) portions of the trajectory were analyzed independently. MALTO was used to create a large database using the parameters shown in Table 1. The segments are "linked" in the spacecraft design tool by matching mass and power levels. Exact matching requires multi-dimensional interpolation of the trajectory databases. This process is described by Bailey, et. al.[15]

**Earth-to-Mars (Outbound) Trajectories**

Outbound trajectories begin on a launch vehicle (LV) at Earth with a launch energy (C3) greater than zero, subject to the capabilities of the LV itself. After arriving at Mars the spacecraft begins a circular spiral to lower its orbit down to a 320 km science orbit. It should be noted that MALTO does not calculate the actual spiral, but rather approximates the duration and fuel mass required based on an analytic algorithm developed by Melbourne and Sauer.[16]

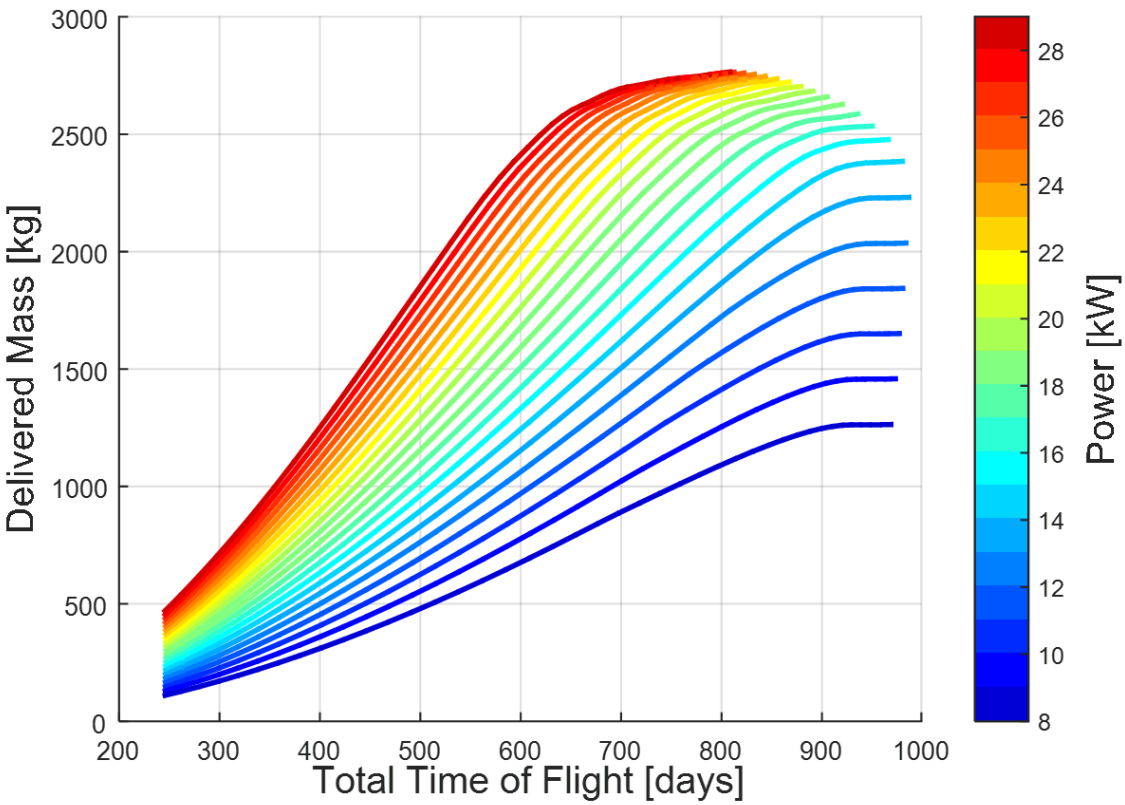


**Figure 3. Maximum delivered mass to Low Mars Orbit on a Falcon 9 and using an ARM thruster operating in high Isp mode. The data are from the 2024 launch opportunity. The time of flight includes both the heliocentric transfer and the spiral to the final orbit.**

MALTO was run in parametric mode, sweeping through a range of power levels and times-of-flight to create the full set of trajectories. This is repeated for each thruster set (type, number, mode), launch year, and launch vehicle. For each trajectory with its given power and TOF constraints, MALTO seeks to find the launch and arrival dates that maximizes the total delivered mass

to Low Mars Orbit (LMO). All of the relevant details of the optimized trajectory are then recorded. Early work on this process was documented by Kowalkowski.[17]

The delivered masses for optimized trajectories are shown in Figure 3. This particular case is for a 2024 launch on a Falcon 9 launch vehicle with one ARM engine operating in high Isp mode. However, the same plot for other launch years, engines, and launch vehicles all look very similar – with delivered mass increasing with power and TOF, quickly at first and then tapering off. Note that for every data point (power and TOF pair) the launch date and C3 are free parameters to be optimized. For the shorter flight times the C3 values are typically high (25 $km^2/s^s$ +) whereas for the longest flight times C3 will be less than 2 $km^2/s^s$. When more time is available the optimizer trades the performance of the LV upper stage (300-400 sec. Isp) for time on the much more efficient SEP thruster.

For the case considered in Figure 3, 14 kW is required to fully power the ARM engine and achieve maximum Isp and thrust. The power system size on the colorbar scale is the power to the thruster at 1 AU. As the trajectory progresses towards Mars that power diminishes. In order to keep the thruster fully "saturated" it is necessary to select an array size that provides 14 kW at the maximum heliocentric distance. For these trajectories, that is greater than 30 kW. Adding power above the saturation level does not improve performance. In fact, there is a point of diminishing returns as added power does not result in significant performance enhancement. This can be seen in the plot as the orange and red lines become very close together, especially at higher flight times. The mission optimization tool must be able to trade the mass penalty of added power versus the increased performance to find the ideal level. (Also bear in mind that this must also be simultaneously balanced with the optimal power level for the return leg for round trip orbiters, which is discussed in the next section.)

The key feature of low-thrust trajectories – a performance vs. time Pareto front (see Figure 1) – is also exhibited here. Along with that front there is also a "knee in the curve" where the performance lines begin to flatten out at high TOFs. This knee occurs around 700 days for the highest powers and above 900 days for powers below 15 kW. The "sweet spot" for SEP trajectories is to select a design near that point. If we force the trajectory to be much shorter then performance will take a major hit and the benefits of SEP won't be realized. If a faster trajectory must be used, it is far better to add thrust by adding more engines or power rather than forcing an undersized system. On the contrary, if mission requirements are such that a faster trajectory may be initially selected, then there is a wealth of margin available to accommodate mass growth by simply allowing the flight time to increase.

**Mars-to-Earth (Inbound) Trajectories**

For inbound trajectories we do not have the initial mass constraint of a launch vehicle performance curve. It is conceivable that a trajectory begins with any mass value in Mars orbit and begins its spiral and return to Earth. This adds mass as a parameter to sweep in MALTO along with TOF and power. For the purposes of regularization, we use MALTO in reverse mode by specifying the final mass to be delivered to Earth and seek to minimize the propellant mass, and hence, initial mass. An example of a portion of an inbound parameter sweep is shown in Figure 4. This is the analog to the outbound scenario in Figure 3 – utilizing one ARM thruster in high Isp mode. The plot shows the required propellant mass to deliver 500 kg to Earth with a maximum $V_\infty$ of 1.5 km/s.[*] Similar plots can be made for each mass level considered.

[*] Trajectories were constrained to $V_\infty$ = 1.5 km/s in order to allow a late decision between direct entry or divert to a stable lunar distant retrograde orbit (DRO) for later retrieval.

As expected, propellant mass decreases with increasing time and power. The diminishing returns on power are evident as the lines are spaced more closely as power increases. The knee is also very pronounced for each curve (which are marked by red stars). For flight times shorter than the knee points the propellant mass increases almost vertically and trajectories quickly become infeasible. For longer flight times the mass decrease is only modest. In order to reduce the degrees of freedom in mission optimization, we often simply select the knee point for a given power level.

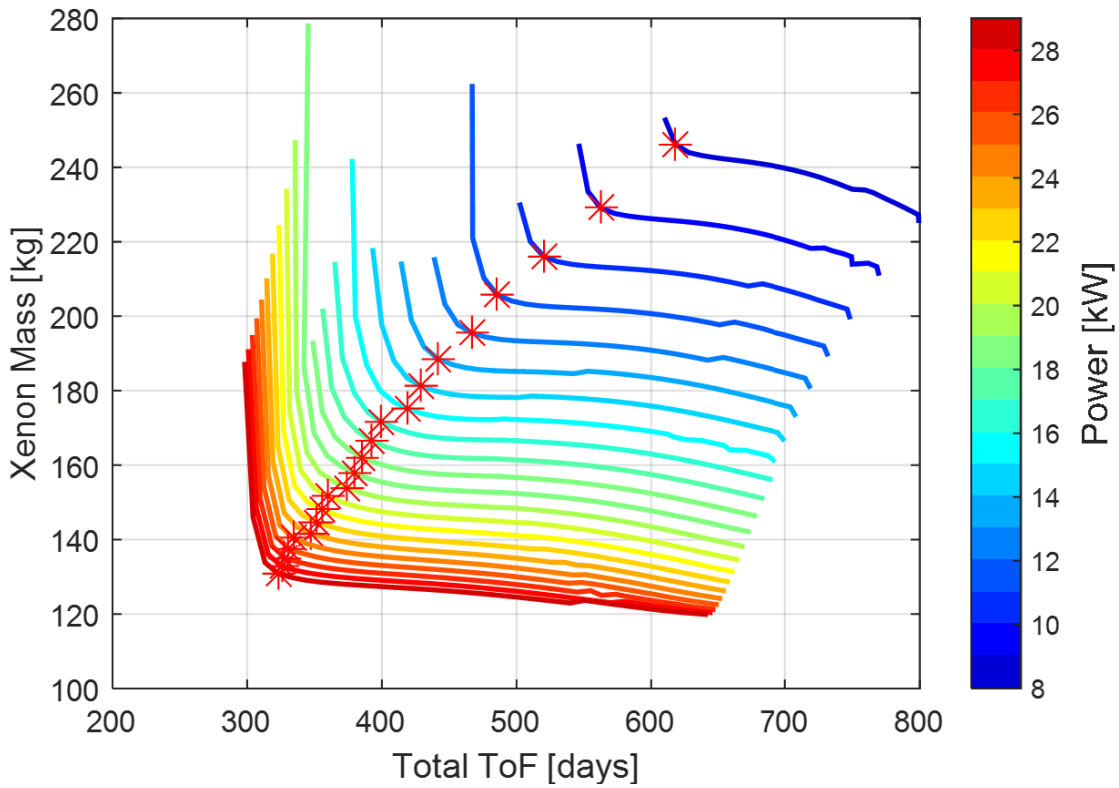


**Figure 4. Xenon Propellant mass required to return 500 kg of dry mass to Earth from low Mars orbit using one ARM thruster. The data is for the 2033 return opportunity. The red stars show the locations of algorithmically selected "knees" for each power level. Maximum $V_\infty$ at Earth arrival is constrained to 1.5 km/s.**

## Trajectory Synthesis with Spacecraft Design

One of the main challenges of designing an optimal SEP mission is that the spacecraft design and the trajectory design are strongly coupled. For instance, adding power increases the system mass but may also improve trajectory performance. Adding mass or changing the thruster can change $\Delta V$ and time of flight substantially. The trade of high thrust vs. high Isp modes within a given thruster is often unintuitive. For this reason, it is crucial to jointly optimize the trajectory with the spacecraft.

Although not the topic of the paper, a brief description of our spacecraft design tool is given for context. At the core is a set of parametric relationships which estimate subsystem masses. The process of creating a mission begins with selecting some basic parameters such as thruster set, initial power level, launch year, desired LV, etc. (See Figure 5). From there a first estimate of

subsystem masses is made using parametric relationships. The trajectory module then builds up the spacecraft backwards in time. The spacecraft dry mass must be delivered to Earth at the end of a sample return mission. This mass is passed to the trajectory database where a return trajectory is interpolated based on power and mass. The propellant required for this leg sets the mass in Mars orbit prior to departure for Earth. Mass for Mars maneuvers and activities is then added to find the total mass that needs to be delivered to orbit from an outbound Earth-to-Mars trajectory. This trajectory is then interpolated based on power level and mass. (Our current implementation uses excess LV capability to reduce outbound trip duration). After a loop of the trajectory module is completed desired margins are added and the required propellant mass is sent to the spacecraft module as an input for another iteration. This process continues until a fully-converged, fully-optimized mission is found. Additional optimizations over power, thruster modes, payloads, etc, are performed automatically by iterating on the entire model.

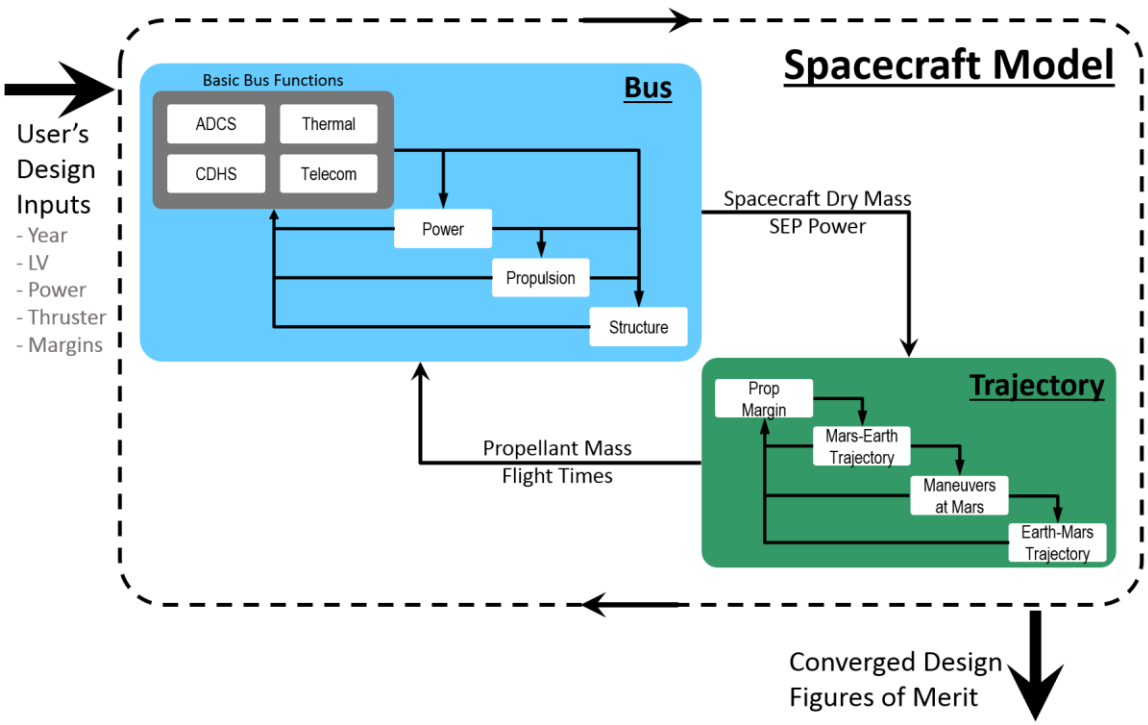


**Figure 5. Process flow for spacecraft mission design tool. Not only are the spacecraft subsystems interrelated, but the SEP trajectories themselves are part of the iteration loops.**

## LAUNCH/ARRIVAL DATE SPACE – "BACON PLOTS"

This process of mission design using a large, separable database of parametrically optimal trajectories works very well for creating and sizing a viable, self-consistent mission that can be compared to other point designs and further refined. However, it is not without its limitations. As mentioned, the actual dates of the trajectory are not selected, nor can they be specified. Each trajectory in the database represents an optimization of a given time-of-flight originating in a specified opportunity. In fact, most of the inbound trajectories are not even considered since we only use an

algorithmically selected trajectory from the "knee" in the curve, which means that the departure and arrival dates of the return trajectory are fixed for a specified mass and power level.

If we plot the actual dates of departure and arrival for the trajectories contained in Figure 3 we get the points shown in Figure 6. Each power level follows a nearly monotonically decreasing ridge of arrival dates for increasing departure dates. If we wished to constrain arrival, for example, to no later than July of 2025, there would be no way to do that. For those kinds of constraints we need to fill in the whole launch date/arrival date space for every power level.

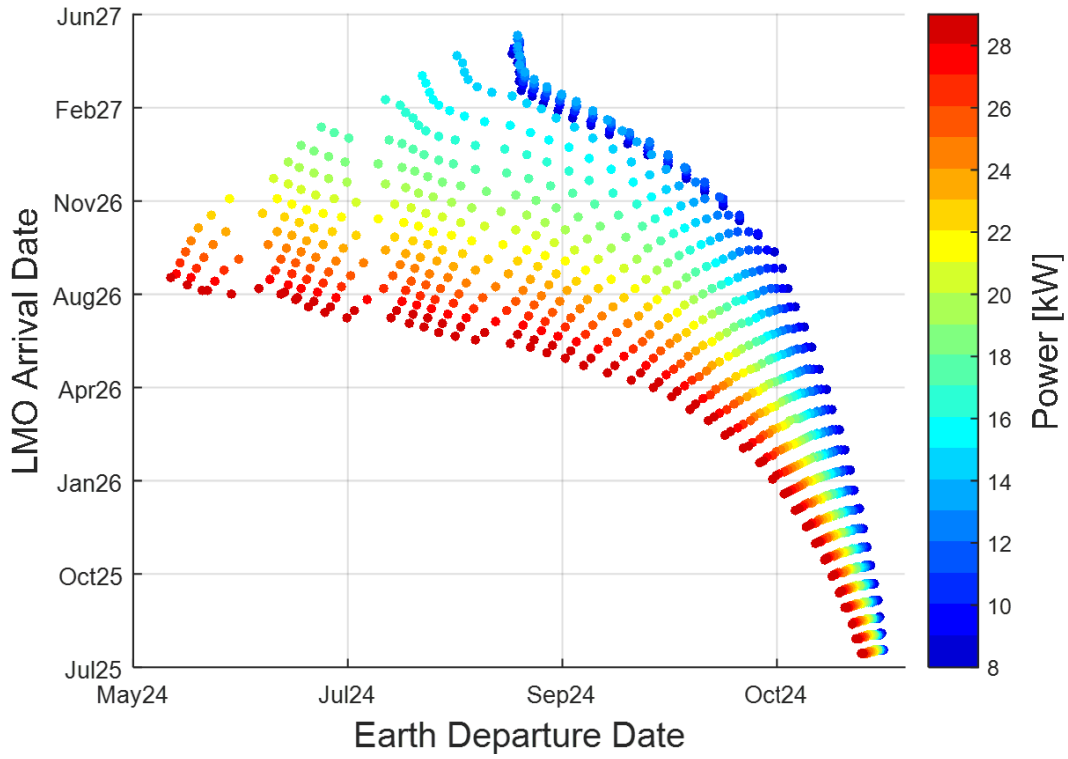


**Figure 6. Launch and arrival dates associated with trajectories in optimized database (see Figure 3). Launch in 2024 on Falcon 9 with one ARM engine in high Isp mode.**

## Porkchop Plots

For two-impulse ballistic transfers to Mars, Lambert's problem assures us that a unique solution exists for every combination of launch date and arrival date. The solution parameters can be plotted as contours versus launch date/arrival date in what is commonly known as a Porkchop plot (due to its shape), illustrated in Figure 7. For low-thrust trajectories the solutions are no longer unique. The continuous maneuvers create an infinite number of solutions. In order to find a unique, optimized solution, a set of assumptions are required. These include the thruster type and number, the power level, and the launch vehicle or inert mass. On a more fundamental level these assumptions are surrogates for specific thrust and Isp as a function of heliocentric distance.

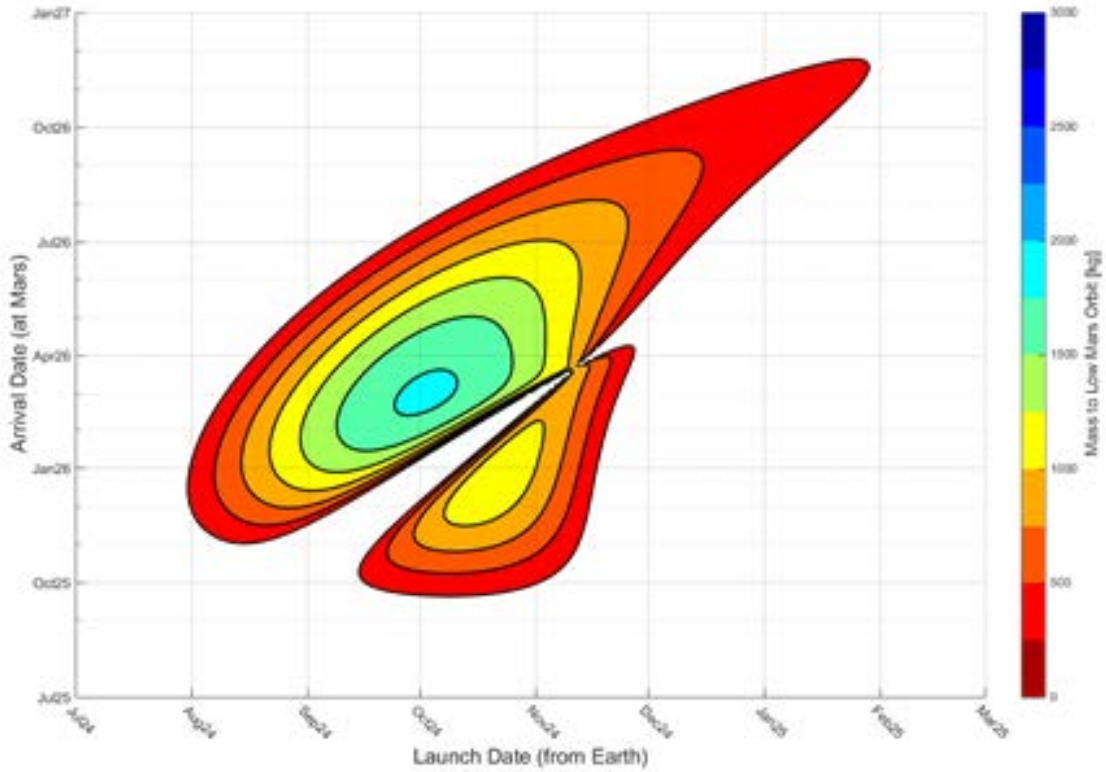


**Figure 7. 2024 Earth-to-Mars "Porkchop plot" showing the maximum mass delivered to LMO by a Falcon 9. Assumes Isp = 325s, 1-sol capture orbit, and 6 months for aerobraking.**

Once a set of assumptions are defined, it is possible to draw contours of the parameters of interest such as mass delivered or fuel used in the launch/arrival date space. Rather than a bi-lobed structure, these contours create continuous bands with better performance as flight time increases. These plots look more like Bacon rather than porkchops – hence our moniker "Bacon Plots". With these plots it is possible to readily answer some of the typical questions that arise in early mission formulation:

- What is the latest/earliest we can depart/arrive?
- How long is my launch period?
- What does it cost to constrain some date or duration?
- What happens if dates slip?

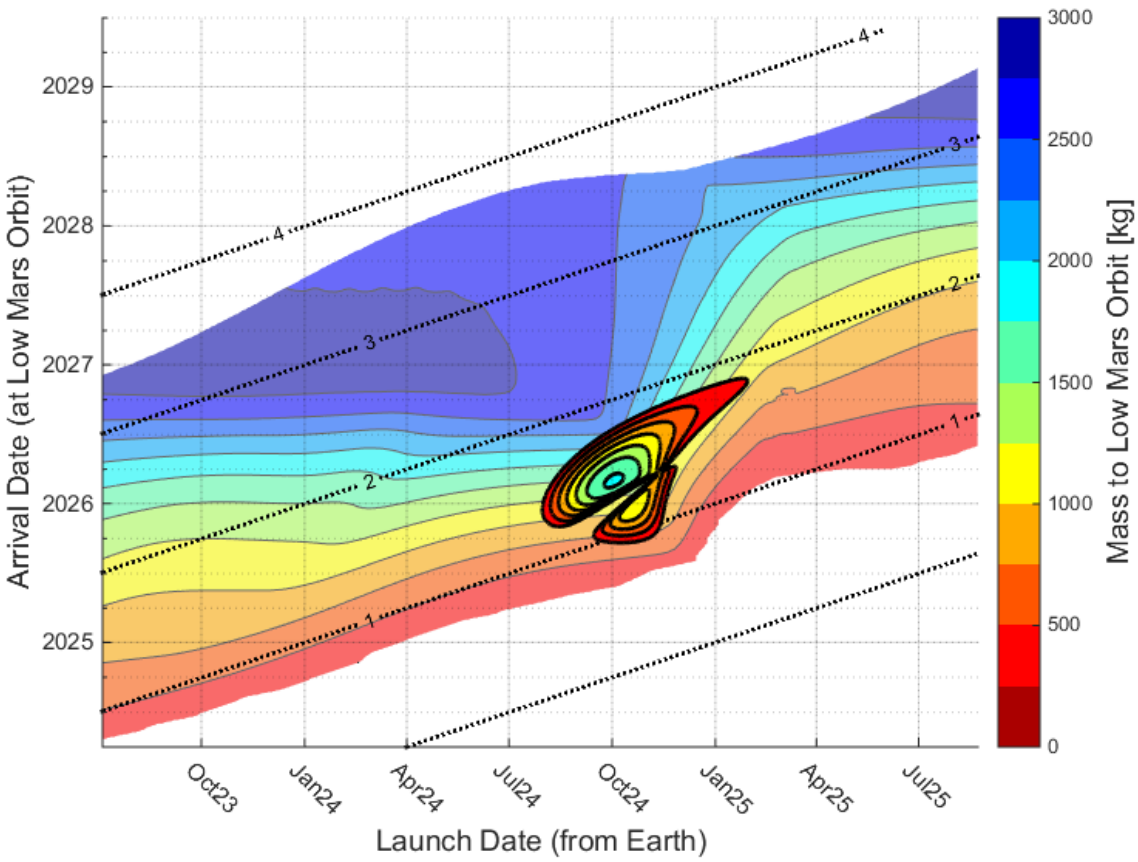


**Figure 8. 2024 Earth-to-Mars "Bacon Plot" with ballistic "Porkchop plot" overlay. SEP parameters: 1 ARM engine operated on high Isp mode with 23 kW at 1 AU. Contours show maximum delivery mass to LMO using a Falcon 9. SEP provides better performance at nearly every LD/AD point and up to 75% more mass for 1 extra year of flight time. Diagonal lines indicated total time-of-flight (years).**

A representative Bacon plot from Earth to Mars is shown in Figure 8. It is specified by one ARM engine in high Isp mode and 23 kW of power at 1 AU. The plot shows contours for the total mass delivered to low Mars orbit using a Falcon 9 launch vehicle. The launch C3 is optimized by MALTO and is typically high for shorter flight times and lower for longer flight times. Here the maximum delivered mass is around 3000 kg in 2.5 years (which includes both the heliocentric portion and the spiral at Mars). This is compared to 1750 kg using chemical propulsion and aerobraking which takes about 1.5 years. Maximum mass rises steeply versus time-of-flight at first then levels off above 2 years. Also note the horizontal nature of the mass bands between 1500-2500 kg to the left of the ballistic data. Arrival dates and delivered masses are constant through most of 2023 and 2024 – creating a launch window over a year in duration.

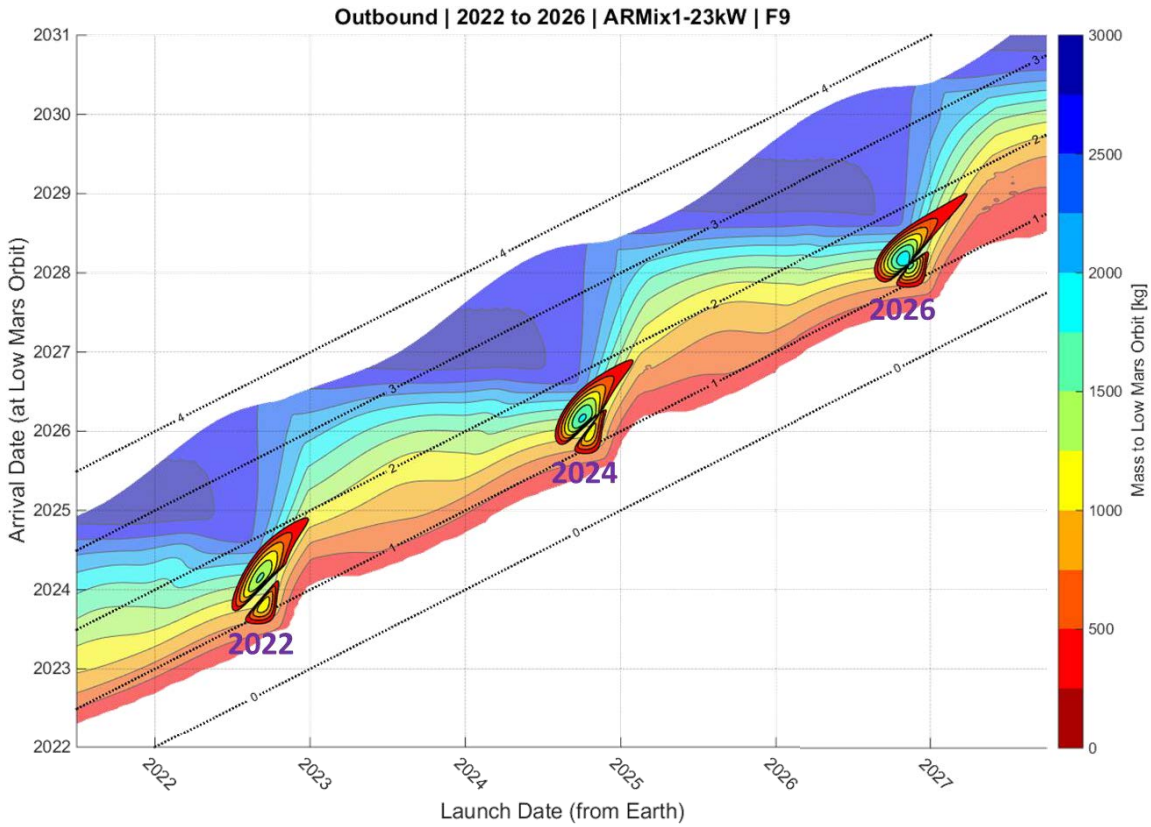


**Figure 9. Expanded Bacon plot covering three ballistic launch opportunities (2022-2026). Other assumptions are the same as Figure 8.**

Figure 9 shows a Bacon plot covering three Earth-Mars launch opportunities spanning six years. The ballistic porkchop plots are overlaid. As expected, the Bacon plot exhibits periodic behavior with the Mars-Earth synodic period (~26 months). It is interesting that the ballistic porkchops lie on top of the SEP Bacon plot. This is due to the assumed use of areobraking for the ballistic system which serves to save mass and increase flight time. However, for this particular case, SEP still outperforms chemical, even when areobraking is used.

It is notable that the SEP trajectories provide *continuous* launch opportunities over this entire six-year window. Figure 10 shows the optimized delivery mass as a function of launch date (from Earth). From this plot, it can be seen that even at two-year total flight times, SEP out-performs chemical substantially around the optimal launch periods each opportunity. Moving to longer flight times around 3 years allows for the launch period to be significantly longer, on the order of years to continuous, depending on mass. For instance, 2000 kg can be delivered to low Mars orbit for a flight time under 3 years with a launch *on any day*! This could enable launch operations similar to Earth missions, with less pressure to launch in short periods of time or enable Mars missions to be flown in between ballistic opportunities. In contrast, ballistic opportunities allow at most 1750kg to be delivered, and only in a very short launch window of approximately 1 month duration.

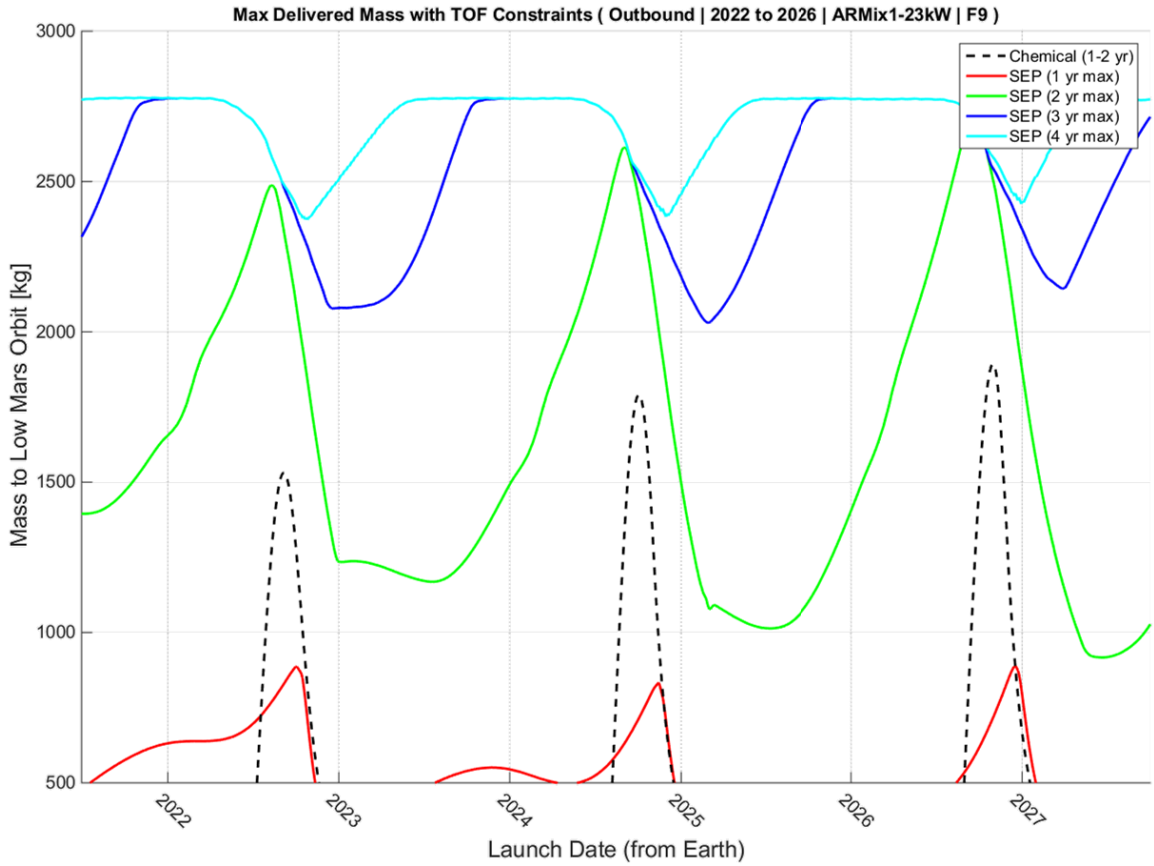


**Figure 10. Subset of data from Bacon plot in Figure 9 optimizing delivered mass as a function of launch date with time of flight constraints. Comparison to ballistic shown in the dashed black line.**

Figure 11 shows the Bacon plot for a Mars-Earth trajectory. The spacecraft assumptions are the same as before. Instead of using a Falcon 9 launch vehicle curve as an initial condition, Earth-Mars trajectories assume departure from a low mars orbit and a terminal constraint of 1250kg at Earth arrival and a maximum hyperbolic excess velocity of 1.5 km/s. The figure of merit is propellant required to meet these constraints.

It should be noted that the ballistic contours are labeled with fuel mass values between 1400 and 2000 kg, which is well above the color scale used for the SEP contours. Unlike the Earth-Mars plots, the ballistic porkchops do not overlap the SEP contours in either mass or time of flight. This is because areobraking cannot be used on the Mars-Earth trajectory. This dramatically increases propellant use and moderately decreases flight time for chemical options.

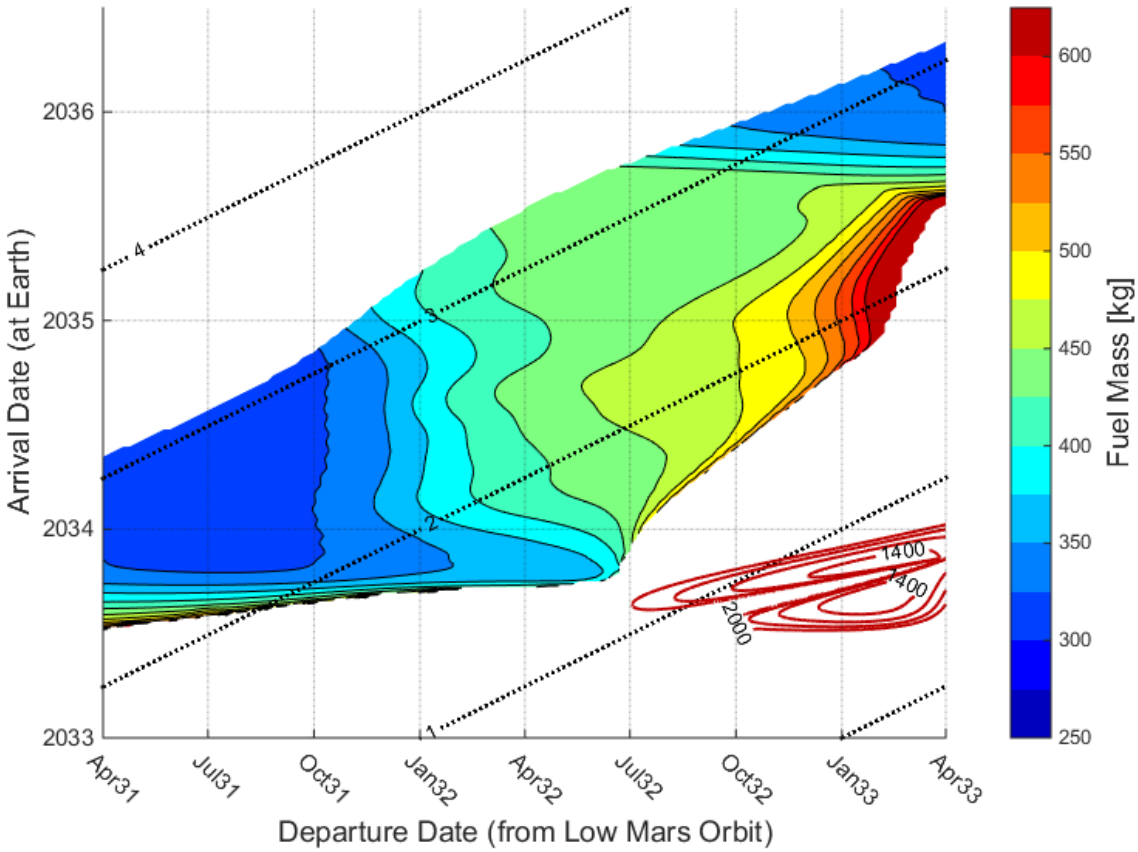


**Figure 11. 2033 Mars-to-Earth "Bacon Plot" with Ballistic Contours. SEP parameters: 1 ARM engine (14 kW class) operated on high Isp mode with 23 kW $P_0$ at 1 AU. Contours show the fuel mass needed to deliver 1250 kg from Low Mars Orbit to Earth with a maximum $V_\infty$ of 1.5 km/s. The ballistic contours (red) assume Isp=325s. They require > 1400 kg of fuel and far exceed the colorbar limits. While SEP always requires more time, it also requires 5-10 times less fuel. Diagonal lines indicated total time-of-flight (years).**

Figure 12 shows the sensitivity of return trajectory propellant use as a function of arrival date (at Earth), with contours at various total time of flight constraints. The most apparent result from this plot is that the optimal chemical options require approximately 4 times more propellant than chemical options do. Also consider that this propellant needs to be carried to Mars as "payload" of the Earth-Mars trajectory. This further highlights why SEP is so beneficial on round-trip missions.

Because the Mars-Earth SEP trajectories do not have the launch vehicle degree of freedom available, they have a higher minimum time of flight limited by the mass flow rate of the engine. In this case, the minimum flight time is approximately 1.5 years. As allowable flight time increases to 2 years, the propellant mass decreases significantly and the possible arrival dates increase to cover about one year around the optimal. With more allowable flight time, any arrival date becomes accessible.

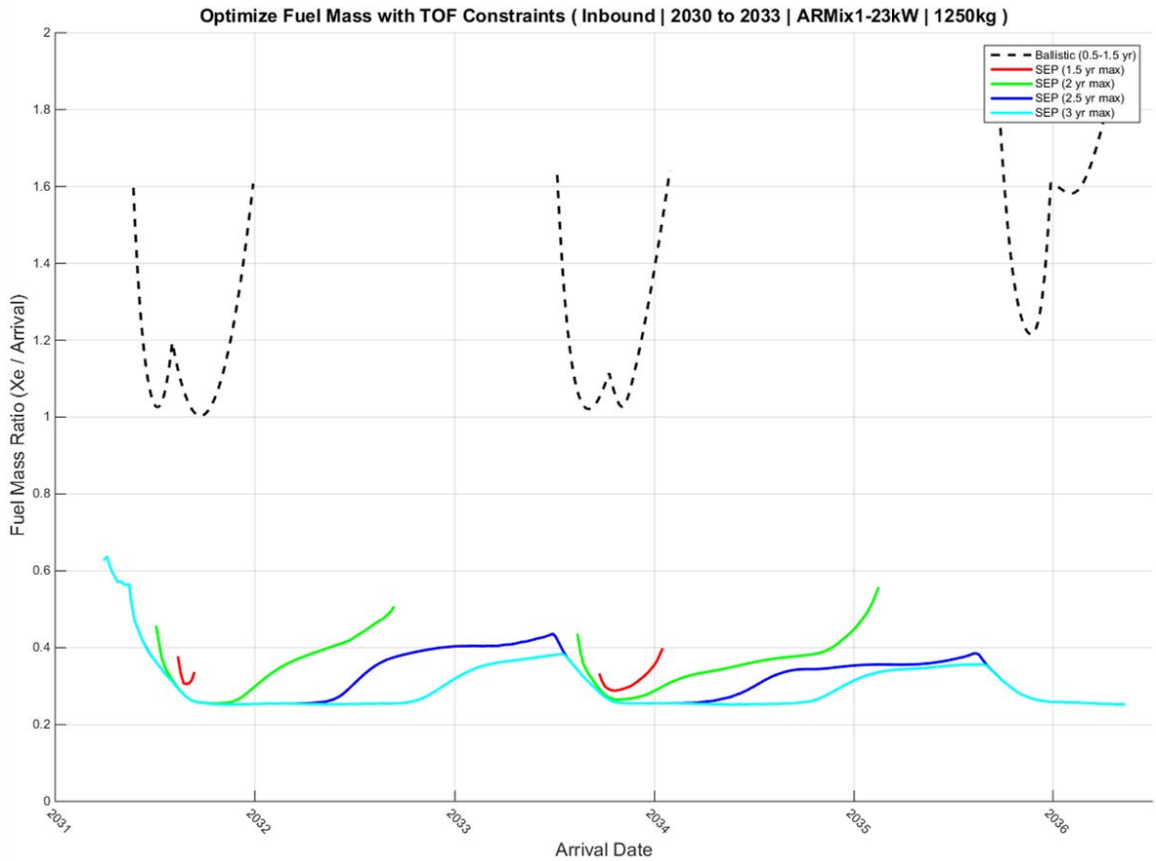


**Figure 12. Subset of Figure 11, optimizing fuel mass ratio as a function of Earth arrival date, with contours of time-of-flight constraints. Ballistic options are shown in dashed black lines. Arrival mass is 1250 kg.**

## CONCLUSIONS

It is difficult to plan SEP missions using blanket heuristics such as: "More power is always better", "Shorter trip times always require more fuel", or "Higher Isp is preferable to higher thrust". These statements simply are not always true. It is important to properly match the SEP system parameters to mission requirements in order to let the system work most effectively. One of the greatest features of SEP missions is their flexibility. Mass, power, and flight-time are all tradable. There are not as many rigid constraints (i.e. "SEP is squishy"). We find that when sized and used properly, SEP offers a tremendous advantage over traditional propulsion for Mars missions, and especially for Sample Return.

## ACKNOWLEDGMENTS

This research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration. The authors would like to thank Rob Lock and the members of the Sample Return Orbiter team at JPL for their helpful insights and comments. Early work on these concepts was also performed by Theresa Kowalkowski, Zach Bailey, and Erick Sturm.

## APPENDIX

### Bacon Plot

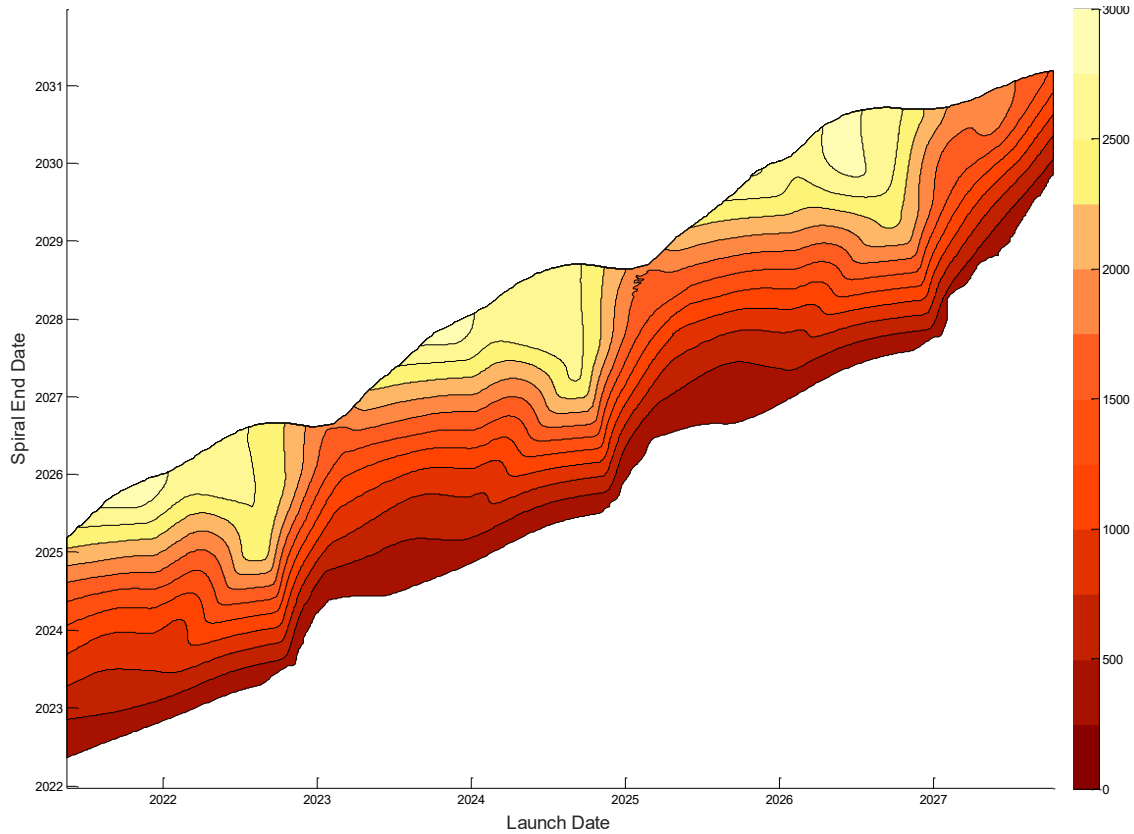


**Figure 13 - Outbound Bacon plot similar to Figure 9 with "bacon" color scheme.**

## REFERENCES


[1] P. Patel and D.J. Scheeres, "Maximizing Payload Mass Fractions of Spacecraft for Interplanetary Electric Propulsion Missions," *Journal of Spacecraft and Rockets* Vol. 43, No. 4, 2006.

[2] R. Mattingly, "The Many Faces of the Mars Sample Return Mission Architecture," AAS-05-066, 27th Annual AAS Guidance and Control Conference, February 2005.

[3] S.N. Williams and V. Coverstone-Carroll, "Mars Mission Using Solar Electric Propulsion," *Journal of Spacecraft and Rockets*, Vol. 37, No. 1, Jan. – Feb. 2000.

[4] J. R. Brophy and D. H. Rodgers, "Ion Propulsion for a Mars Sample Return Mission," 36th AIAA Joint Propulsion Conference and Exhibit, AIAA Paper 2000-3412, Huntsville, AL, July 2000.

[5] B. B. Donahue, S. E. Green, V. L. Coverstone, and B. Woo, "Chemical and Solar-Electric-Propulsion Systems Analyses for Mars Sample Return Missions," *Journal of Spacecraft and Rockets* Vol. 43, No. 1, 2006.

[6] D.Y. Oh, R.R. Hofer, I. Katz, J.A. Sims, N.Z. Warner, T.M. Randolph, R.T. Reeve, and R.C. Moeller, "Benefits of Using Hall Thrusters for a Mars Sample Return Mission," *31st International Electric Propulsion Conference*, IEPC-2009-217, Ann Arbor, MI, Sep. 2009.

[7] R. Bertrand, J. Bernussou, S. Geffroy, and R. Epenoy, "Electric Transfer Optimization for Mars Sample Return Mission," *Acta Astronautica*, Vol. 48, No. 5-12, pp. 651-600, 2001.

[8] J.W. Dankanich, L.M. Burke, and J.A. Hemminger, "Mars Sample Return Orbiter/Earth Return Vehicle Technology Needs and Mission Risk Assessment," IEEE Aerospace Conference, Big Sky, MT, March 2010.

[9] R.S. Nah, S.R. Vadali, and E. Braden, "Fuel-Optimal, Low-Thrust, Three-Dimensional Earth–Mars Trajectories," *Journal of Guidance, Control, and Dynamics,* Vol. 24, No. 6, November–December 2001.

[10] C.A. Kluever, "Efficient Computation of Optimal Interplanetary Trajectories Using Solar Electric Propulsion," *Journal of Guidance, Control, and Dynamics,* Vol. 38, No. 5, May 2015.

[11] Y. Gao, "Low-Thrust Interplanetary Transfers, Including Escape and Capture Trajectories." *Journal of Guidance, Control, and Dynamics,* Vol. 30, No. 6, pp. 1814-1818, 2007

[12] A. A. Quarta, G. Mengali, G. Aliasi "Multi-revolution transfer for heliocentric missions with solar electric propulsion." *Advances in Space Research* 55, 647-659, 2015.

[13] J. A. Sims, P. A. Finlayson, E. A. Rinderle, M.A. Vavrina, and T. D. Kowalkowski, "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design," AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, Colorado, Aug. 2006.

[14] D.Y. Oh, J. S. Snyder, D. M. Goebel, R. R. Hofer, and T.M. Randolph. "Solar Electric Propulsion for Discovery-Class Missions", Journal of Spacecraft and Rockets, Vol. 51, No. 6 pp. 1822-1835. 2014.

[15] Z.J. Bailey, T.D. Kowalkowski, R.E. Lock, E.J Sturm, and R.C. Woolley, "Round-Trip Solar Electric Propulsion Missions for Mars Sample Return," 24th AAS/AIAA Space Flight Mechanics Meeting, AAS Paper 14-365, Santa Fe, NM, Feb. 2014.

[16] Melbourne, W. G. and Sauer, C. G., "Performance Computations with Pieced Solutions of Planetocentric and Heliocentric Trajectories for Low-Thrust Missions," Space Programs Summary 37-36, vol. IV, Jet Propulsion Laboratory, Pasadena, California, pp. 14–19, December 31, 1965.

[17] T.D. Kowalkowski, Z.J. Bailey, R.E. Lock, E.J Sturm, and R.C. Woolley, "Robotic Mars Exploration Trajectories Using Hall Thrusters," 24th AAS/AIAA Space Flight Mechanics Meeting, AAS Paper 14-364, Santa Fe, NM, Feb. 2014.