Optimized Low-Thrust Missions from GTO to Mars

Ryan Woolley
Jet Propulsion Laboratory, California Institute of Technology
4800 Oak Grove Drive, Pasadena, CA 91109
818-393-8279
ryan.c.woolley@jpl.nasa.gov

Zubin Olikara
Jet Propulsion Laboratory, California Institute of Technology
4800 Oak Grove Drive, Pasadena, CA 91109
818-354-2114
zubin.p.olikara@jpl.nasa.gov

***Abstract*—A small spacecraft has three primary methods to get to Mars: by a dedicated launch vehicle, by sharing a ride with another Mars-bound mission, or via rideshare to Earth orbit and making its own way to the Red Planet. The third option may be the most attractive in terms of cost effectiveness and frequency of access, which are the main drawbacks of the first two, respectively. In the past 5 years, over 50 satellites have been launched from U.S. soil en route to geosynchronous orbit and beyond. Many of these launches do not use the full capability of the launch vehicle, leaving an opportunity for a small secondary spacecraft to get a ride to geosynchronous transfer orbit (GTO). Starting from an ESPA-ring rideshare to GTO, a solar-electric propulsion (SEP) powered spacecraft can make its own way to Mars orbit to perform a useful mission. In this paper, we explore the mission design space for small, ESPA-class (200-450 kg) spacecraft transferring from GTO to Mars orbit. This is accomplished by creating tools that jointly optimize low-thrust trajectories and spacecraft subsystems to create feasible mission concepts. An example is given of an Areostationary telecom orbiter that reaches Mars in 2 years with a dry mass of 200 kg.**

TABLE OF CONTENTS

1. INTRODUCTION

Utilizing recent advances in long-lived, sub-kilowatt class solar-electric propulsion (SEP) engines [1][2], a small spacecraft could potentially make its own way from GTO to Mars orbit to perform high-quality science and telecommunications for a very low cost [3]. A ride to GTO could come through various rideshare opportunities with government or commercial payloads bound for geostationary orbit. From 2002 through June of 2018, there were 168 launches of large U.S. launch vehicles – Atlas 5, Delta IV, and Falcon 9. The number of launches annually has increased from < 5/yr before 2008, to > 15/yr since 2014, and continues to rise. Figure 1 shows the distribution of orbital targets for these launches. In order to get to geostationary orbit, various intermediate transfer orbits may be used: a basic Hohmann-like transfer (GTO), a high-apogee beyond geostationary (GTO+, or super-synchronous), a high perigee (GTOp), or direct transfer to circular geostationary (GEO). The sum total of these orbits accounts for nearly half of all evolved expendable launch vehicle (EELV) launches to date. These large orbits are common and represent a good opportunity for a rideshare to commence an interplanetary mission.

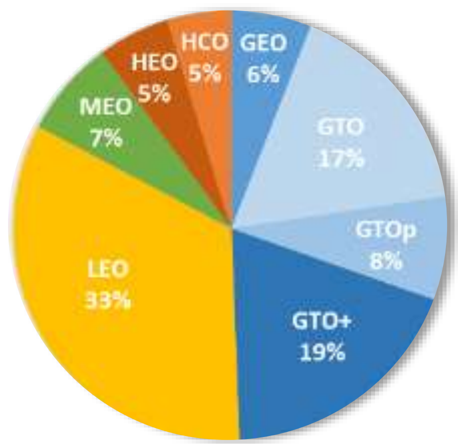

Figure 1 - Destinations for U.S. EELV's since 2002. Shades of blue (right hand side) represent orbits to GTO or beyond. They account for ~50% of all launches, with launch rates now averaging as high as one per month.

There are multiple ways for secondary payloads to be accommodated along with the primary. One of the most common for secondaries more than 100 kg is the EELV Secondary Payload Adapter (ESPA) ring, created by Moog. It provides a common interface connection between the LV and primary spacecraft for up to 4-6 secondaries. The standard ESPA is rated for 220 kg per spacecraft, whereas the ESPA Heavy and ESPA Grande can carry 322 kg and > 465 kg, respectively [4].

A self-propelled spacecraft going from GTO to Mars must: 1) escape from Earth orbit, 2) traverse interplanetary space from Earth to Mars, and 3) capture into the desired orbit at Mars. The three trajectory phases are shown in Figure 2. For low-thrust trajectories, the total ΔV can be as high as 8-10 km/s! Luckily, SEP thrusters are very efficient, achieving specific impulses (I_{sp}) of greater than 1500 seconds. This can lead to a spacecraft with a dry mass fraction of 60% or more. This is important when a mission is constrained to a maximum wet mass of a few hundred kilograms. A large dry mass fraction means more useful spacecraft in orbit at Mars.

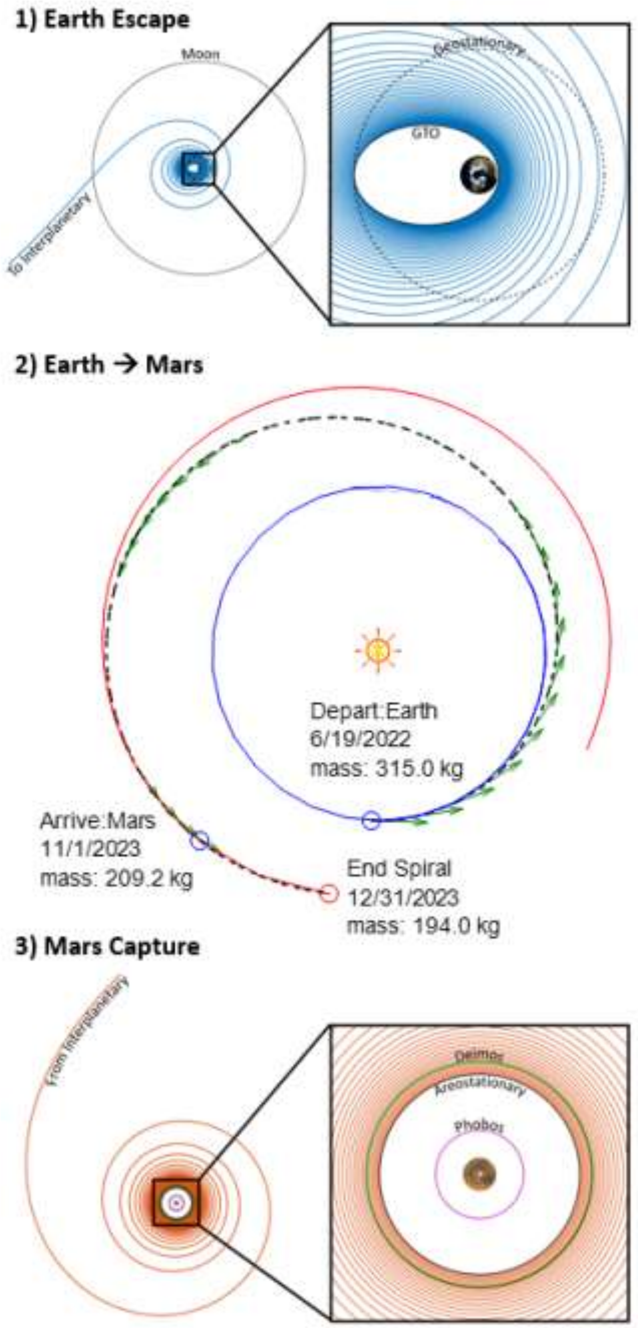

Figure 2 - Three trajectory phases from GTO to Mars Orbit. 1) Low-Thrust spiral out from GTO to Earth escape. 2) Heliocentric transfer from Earth to Mars. 3) Spiral down to high Mars orbit.

In order to construct an optimal mission architecture using SEP, it is essential to employ methods that simultaneously optimize both the flight system and trajectory design [5][6]. Low-thrust trajectories are extremely flexible, but they are also very sensitive to driving parameters such as power level, thrust, mass, and time-of-flight (TOF). In fact, specific characteristics of a chosen thruster, such as throttle profiles and rated lifetime, can greatly affect the design of a mission.

The mission design process consists of an iteration loop between a spacecraft module and a trajectory module. These modules can be as simple or complex as desired. In this paper, we describe the tool creation process for an efficient method to design a small satellite mission from GTO to Mars orbit. The trajectory module consists of a combination of parametric curve fits and a large database of optimized low-thrust trajectories, described in Section 2. The spacecraft module, covered in Section 3, has basic parametric models of key spacecraft subsystems that vary with wet mass. Section 4 gives an example mission designed with this method – an orbiter to Areostationary orbit starting from an ESPA rideshare to GTO.

2. Mission Design Database

As mentioned, there are three phases to the transfer from GTO to Mars orbit: Earth-centric spirals to escape, heliocentric transfer to Mars rendezvous, and Mars-centric spiral down to the desired orbit. Alternate options exist beyond basic low-thrust transfers to complete each mission phase. These include high-thrust chemical maneuvers, rideshare on other propulsive elements, and gravitational assists at the Moon or other bodies, but these are beyond the scope of the work presented here [7][8].

Due to the nature of (nearly) continuous low-thrust maneuvers, there are many options in the methods of optimization. During early concept and feasibility studies, rapid, low-to-medium fidelity methods are sufficient to hone in on optimal architectures. What is important is to capture the key trends and relationships when trading masses, time-of-flight (TOF), propellant, etc. Rapid exploration of the trade space over a range of key parameters will allow mission designers to determine what is possible and set proper constraints and figures-of-merit (FOMs) for the optimization of their mission as a whole.

SEP Thrusters and Power

It is difficult to discuss the optimization of low-thrust trajectories without knowing the detailed properties of the engines being used. To first order, it may be sufficient to assume a constant I_{sp} and efficiency[1]. This is particularly true when the power level remains nearly constant over the trajectory, such as during spirals at Earth or Mars (the Martian eccentricity, however, does cause a fair bit of variation). In this case, the thrust is linearly proportional to input power. However, in reality, most thrusters vary in Isp and efficiency as power varies. In fact, the power-processing unit (PPU) can often vary both current and voltage to create

[1] Thruster efficiency is defined as the ratio of the kinetic energy of the exhaust particles (or jet energy), to the electrical input power to the propulsion system. Typical efficiency values range from 40 – 65%. Peak efficiency is usually attained near the maximum input power of the thruster, and decreases as power drops.

multiple operating throttle points. Trajectory optimization routines can use these points, or a polynomial fit of thrust vs. power and mass flow rate vs. power, to optimize heliocentric trajectories that have varying solar power.

Another thruster parameter crucial to interplanetary SEP missions is the maximum throughput of each engine. This can be expressed in kilograms, hours, or total impulse. It is necessary to have thrusters that can provide the high ΔV's required for the trip from GTO to Mars – as much as 10 km/s or more, where up to half of the wet mass could be propellant.

The key thruster parameters needed are:

- Maximum and minimum input power [kW]
- Thruster string (with PPU, gimbal, etc.) mass [kg]
- Thrust vs. Power curve [N]
- Mass flow rate (or Isp) vs. power curve [g/s or sec]
- Maximum throughput [kg, hrs, or N/s]

It is important to select thrusters that are sufficiently sized for the mass and ΔV of the desired mission. We have found that a good rule-of-thumb is to select engines and power levels to give initial acceleration levels of 0.15 – 0.3 mm/s^2. This can be achieved via one thruster, or multiple smaller ones. As mentioned, it is also critical to use engines with enough throughput capability. For ESPA-class missions to Mars, this generally means > 100 kg of xenon (Xe). It is possible to carry extra "spare" engines to cover the requirement, but the mass penalty can be high.

Table 1 - Table of SEP hall thrusters considered for this mission concept.

Engine	Pmax [W]	Pmin [W]	Thrust [mN]	Isp [sec]	Est. Life [kg]
MaSMi (JPL) [1]	900	150	45	1733	150+
SPT100 (Fakal)	1560	720	89	1562	120
PPS 1350-G (Safran)	1560	590	89	1650	140
BHT-600 (Busek)	887	229	46	1683	30

Table 1 lists the hall thrusters that were found suitable for a small satellite mission from GTO to Mars. The MaSMi thruster [2] is in development at JPL and uses magnetic shielding to eliminate erosion and greatly extend estimated lifetime. The SPT100 and PPS-1350 are both based on a similar design, are commercially available, and have a long flight heritage. The BHT-600 is similar in size and performance to the MaSMi thruster, but was not designed to have the throughput necessary for this type of mission. Also note that the SPT and PPS thrusters take roughly twice the power and produce twice the thrust as the MaSMi and BHT.

In addition to the appropriate thruster, it is also essential that it be adequately powered. This requires large solar arrays. In general, lightweight flexible arrays will provide the desired power level and mass savings. A reasonable starting target would be in the range of 7 – 10 kg/kW. The actual optimal power level cannot be determined until a full mission-modeling tool is in place. For missions that remain near Earth, it is practical to start with an array that fully powers the thrusters and spacecraft systems, with adequate margins.

For missions to Mars, however, the input power can be decreased by 60% or more as solar distance increases. It is tempting to select a power level that keeps the engines fully powered throughout the mission, but this would mean much power is thrown away near Earth. This is where a tool that can estimate the propellant cost to carry additional power and determine the sweet spot is essential. There is also a trade between increasing power and decreasing time-of-flight. A good guess for a starting power level is to keep the thruster 60-80% of max power at Mars. Another way to get a rough estimate, which accounts for typical thruster performance, is to start with about 5 – 10 W/kg of estimated wet mass. It is acceptable to use even lower powers (at the cost of increased TOF), but keep in mind the minimum power level to keep the engine above its lowest throttle point.

Spiraling from GTO to Escape

For a spacecraft in GTO to depart from the Earth's vicinity, it must raise its energy to escape ($C_3 = 0\ \mathrm{km}^2/s^2$), at which point the spacecraft is only weakly bound to the Earth and can begin thrusting through heliocentric space en route to Mars. We use this escape condition as an interface point between a parametric model of low-thrust spirals from GTO and a database of heliocentric Earth-to-Mars transfers.

The parametric model is constructed using a curve fit of transfers generated using an early version of the Tycho tool [9], which optimizes the thrust profile over the many revolutions of a spiral. For a minimum-time transfer, the engine runs continuously while the thrust direction is varied. For longer transfer times, coast arcs are added each revolution in the least efficient locations, so as to reduce the propellant required to escape. This leads to a Pareto front of optimal transfers as a function of the time, t_{esc}, spent spiraling about the Earth.

Rather than compute spirals for each spacecraft mass, power level, and engine configuration, we normalize the results to obtain a single curve valid for preliminary design. Since there is an inverse relationship between acceleration and spiral time, we introduce a normalized parameter

$$u = (T/m_0) \cdot t_{\mathrm{esc}} \cdot (1 - \rho_{\mathrm{ecl}})$$

where T is the thrust in the Earth vicinity (1 AU), m_0 is the initial spacecraft mass, and ρ_{ecl} is the eclipse fraction over the transfer during which thrusting is not possible. The eclipse fraction depends on the Sun geometry over the dates of the transfer, but may be neglected in early analyses. Alternatively, a value of 0.13 can be used as a conservative estimate. In essence, u is the initial acceleration times the TOF, and modified by the fraction in eclipse. For example, if we have an acceleration of 0.3 mm/s^2, a TOF of 200 days, and an eclipse fraction of 0.13, the x-value is 4.51 km/s, yielding a ΔV of 3.2 km/s.

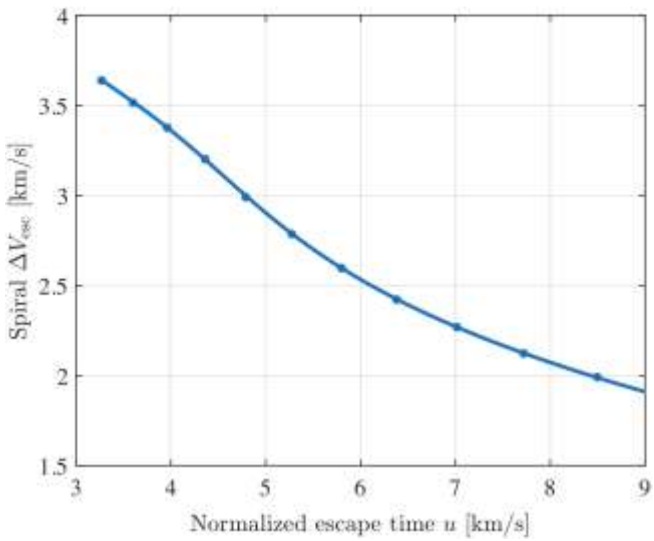

Figure 3 - Parametric model for SEP ΔV required to escape from GTO. u is computed as the initial acceleration times the TOF, and modified by the fraction in eclipse (typically 8-13%).

Using the normalized escape time, we approximate the ΔV required from the engine to escape using a rational polynomial,

$$\Delta V_{\mathrm{esc}}(u) = \frac{p_1 u^2 + p_2 u + p_3}{u^3 + q_1 u^2 + q_2 u + q_3},$$

which expects u and ΔV_{esc} to have units of km/s, and where $p_1 = 25.34$, $p_2 = -208.8$, $p_3 = 500.4$, $q_1 = -3.909$, $q_2 = -15.91$, and $q_3 = 83.15$. Note that we require $u \geq u_{\mathrm{min}}$, where $u_{\mathrm{min}} = 3.3$ km/s corresponds to a minimum-time transfer from GTO to escape.[2] The polynomial is fit with data points up to about $u = 8.5$ km/s corresponding to a transfer taking about 2.5 times longer than the minimum but only requiring about half the spiral propellant. This parametric model is plotted in Figure 3. Combined with a database of trajectories from Earth to Mars, the curve can be evaluated at a sequence of spiral times to determine the optimal time split between the phases for a given total transfer time.

Trajectory Database

The remaining two phases – heliocentric transfer and spiral down to Mars orbit – are simulated using a rapid, medium-fidelity low-thrust optimizer called MALTO [10]. This tool can quickly calculate trajectories from Earth to Mars under a variety of conditions and constraints. In this case, they start from C3= 0 km^2/s^2 (Earth escape at the sphere of influence) with a given wet mass and an acceptable range of starting dates. A TOF from Earth to Mars rendezvous (V_∞ = 0 km/s) is specified and the trajectory is calculated. MALTO also has the capability to add a circular capture spiral down to a desired orbit using the methods of Melbourne and Sauer [11]. This method analytically approximates the propellant mass and time necessary to complete the transfer, and is part of the optimization process.

As with most optimizers, the objective function is user specified. Most typically, this means minimizing propellant requirements by prescribing a fixed starting mass and maximizing the ending mass, or giving the ending mass and minimizing the starting mass. The optimizer is given free variables related to dates or date ranges, thruster curves, thrust vectors, etc. There are a number of ways to create a trajectory database that spans the possible mission trade space [12]. In this case, the database is created by sweeping through thruster types and quantities, power levels, mass levels, and flight durations. MALTO is used to create tens of thousands of trajectories over the full range of combinations.

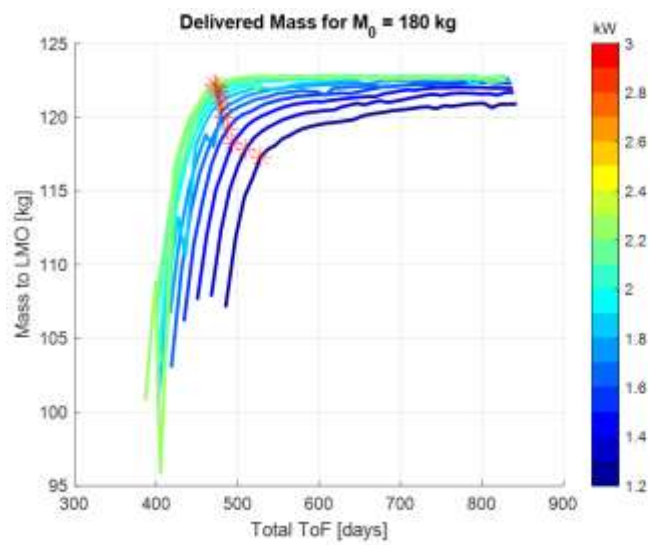

Figure 4 – Resulting mass in Mars orbit from a parametric sweep of power levels (1.2 – 3.0 kW) and TOF (350 – 850 days). Trajectories start at Earth escape with 180 kg, use 1 MaSMi thruster, and end at Areostationary orbit. This initial mas roughly correlates to 220 kg at GTO separation. The red asterisks are algorithmically selected “knee” points for each power level.

Figure 4 shows the results of a parametric sweep of power and TOF for trajectories starting with 180 kg at Earth escape and ending in Areostationary orbit. This plot is the result of ~1000 optimized trajectories, from 0.1 kW steps in power and 10-day steps in TOF. The key characteristic to note is that

[2] The minimum-time ΔV_{esc} is about 3.6 km/s, which is larger than u_{min} since the acceleration increases as the spacecraft expends propellant. The curve fit was generated using the MaSMi engine from Table 1, which is slightly conservative in terms of ΔV_{esc} when applied to the other engines since the MaSMi engine will have less acceleration available later in the spiral for a given initial acceleration.

the mass delivered to the final orbit increases sharply with TOF at first, and then quickly levels out to an asymptote as ideal transfers are achieved. The color scale also shows an increase in delivered mass with increasing power, but only to a point. Powers above 2.4 kW (yellows and reds) are not even visible because they coincide with 2.2 kW on the upper edge. This illustrates that one MaSMi thruster (P_{max} = 900 W) cannot use all of the power over most of the trajectory and most is wasted.

A third dimension, initial mass, is also swept from 180 kg to 400 kg, in steps of 20 kg, creating 13 plots such as the one in Figure 4, bracketing our expected trade space. (Note that these starting masses are the wet mass *after* the spiral from GTO to escape). These sweeps are then repeated with 1 or 2 engines for each of the engines considered. Altogether, tens of thousands of trajectories are created, which represents many hours of computational time.

The key characteristics of these trajectories are tabulated in a searchable database. Since the TOF vs. mass curves exhibit a strong "knee", it is possible to find these points algorithmically and store them, thus reducing the number of degrees of freedom. There is now just one trajectory for a given thruster, starting mass, and power level. Key stats on each one, such as Xe mass and TOF, are collected in small, two-dimensional tables that can be readily displayed and searched.

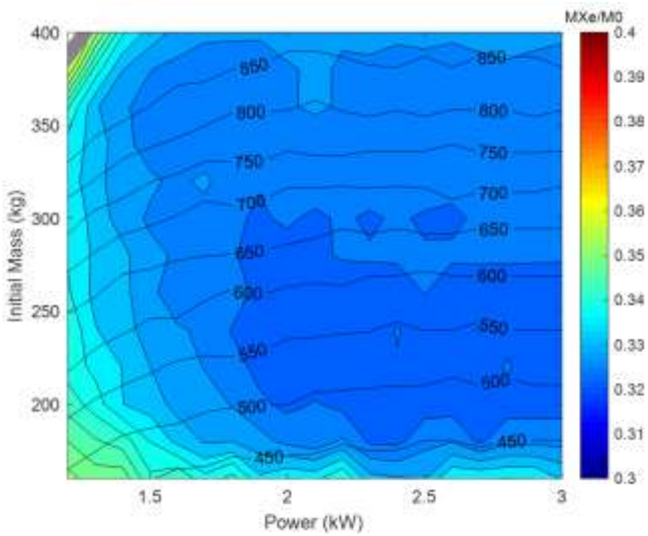

Figure 5 - Contour plot of propellant mass fractions (M_{Xe}/M_0) vs. power and initial mass from Earth escape to Areostationary orbit for 1x MaSMi engine. Black contour lines show the corresponding total TOF [days] for the transfer.

The "knee-point" data table for a given engine can be plotted as a contour plot such as those shown in Figure 5 and Figure 6. Over the range of mass and power levels surveyed, the ratio between Xe mass and initial wet mass ranges from 30 – 40%. The propellant fraction tends to decrease with increasing power. The TOF, shown by labeled black contours, increases notably with increasing initial mass. This is to be expected as the fixed maximum thrust leads to lower and lower accelerations. We also note that the contours are more or less smooth (minus some grid granularity), which means that they can be interpolated in two dimensions during the design process without the need to run new trajectories.

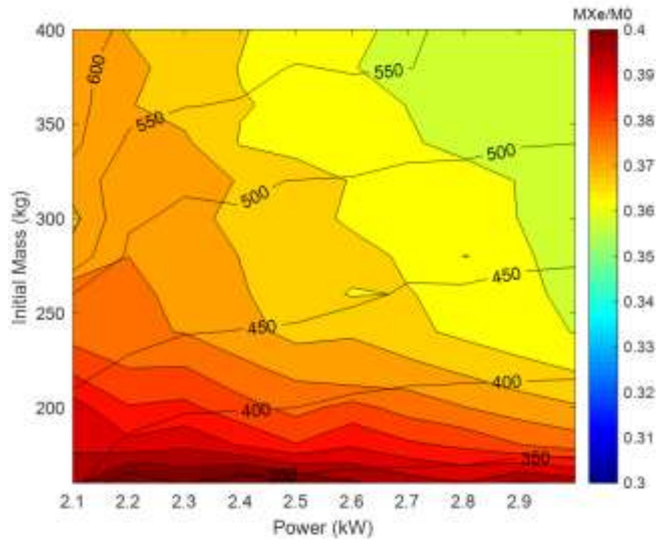

Figure 6 - Contour plot of propellant mass fractions (M_{Xe}/M_0) 1x SPT100 engine, similar to Figure 5. Note that the color scales between the plots are the same, illustrating the larger propellant use for the SPT100 with its lower I_{sp}. However, the SPT100 does have faster TOFs [days] for the same mass and power levels.

Now that the vast database has been reduced to just the "knee" points, it is possible to analyze some trends. First, let us do a simple analysis to determine the optimal starting power for the case in Figure 4 – that of one MaSMi thruster and 180 kg. The star points appear to achieve increasing final mass with increasing power, but if we subtract off the estimated mass of the additional arrays to provide that power, there will be a point at which additional power is not worth the additional performance.

Let us start by assuming an array mass factor of 7 kg per additional kW of power beyond 1.2 kW. The blue line in Figure 7 shows the adjusted final mass with the array mass removed. We note that adding solar array mass just barely pays for itself from 1.2 – 1.7 kW, then is not beneficial beyond that. The red line on the plot shows the associated TOF from Earth escape to Areostationary orbit. If 1.2 kW is selected as optimal, the TOF would be 525 days. Adding slightly more power can decrease the time by 50 to 75 days.

Making use of the preliminarily optimized power levels with their associated TOF, and by using the parametric model of the GTO escape spiral, it is possible to construct some basic missions. Table 2 shows a collection of parameters for missions using various engines. Each starts with a wet mass of 220 kg (standard ESPA) in GTO orbit. From there, a ~8-month spiral takes it to escape and the Xe use is calculated.

The resulting mass is used as a starting point for a trajectory in the MALTO database of knee points (sometimes interpolating between nearby points), which gives propellant usage and TOF. All of this yields a total duration and a dry mass in Mars orbit.

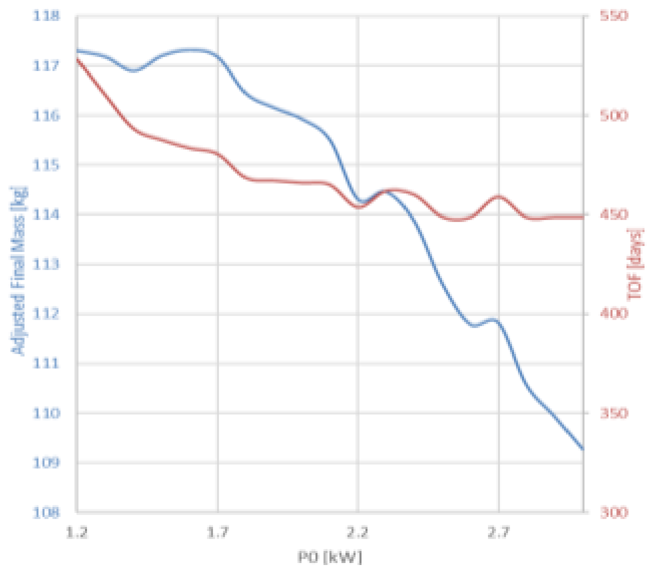

Figure 7 - Determining an optimal power level for 1xMaSMi thruster and 180 kg initial mass. An array mass factor of 7 kg/kW is removed from the final mass to create the adjusted final mass vs. power (blue curve). The red curve shows the associated TOF from Earth escape to Areostationary orbit.

The last column of Table 2 is a simple attempt at making a better "apples-to-apples" comparison of the useful mass delivered to orbit across engine sets. This is done by removing estimates of the mass for systems that vary amongst the architectures. These include the solar arrays, the thruster(s) and PPU(s), and the propellant tank. (More detailed subsystem models are described in Section 3.) After they are removed, we see that the MaSMi thrusters deliver more mass but with a slightly greater TOF, as is further illustrated in Figure 8. This is primarily due to the slightly higher I_{sp} and correspondingly lower thrust. The lower mass and power range is also better suited to a mission of this size.

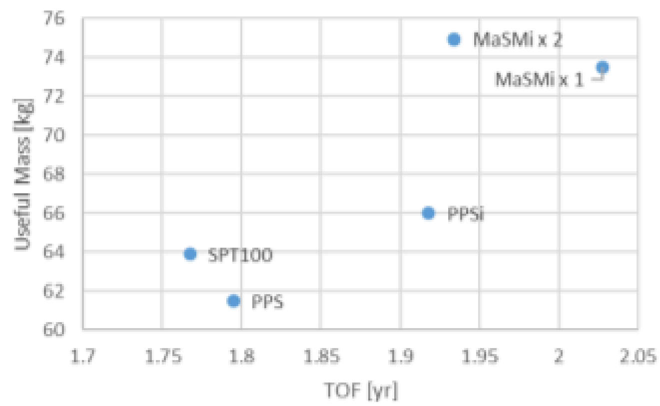

Figure 8 - Useful delivered mass vs. total TOF for various engines starting from 220 kg at GTO and ending in Areostationary orbit.

3. SUBSYSTEM MODELS

Since the trajectory flown is so highly dependent on the dry mass of the spacecraft, it is necessary to have a reasonably accurate model of spacecraft subsystems that are both fixed and vary with trajectory parameters. Early in the design process it is sufficient to start with an approximate allocation for fixed subsystems. These can include payload, telecom, command and data handling, thermal, etc. Reference texts [13] and similar mission can give good initial estimates for these. Next, we use simple models, databases, or parametrics to estimate the masses of subsystems affected by the propulsion system and trajectory. These include the solar arrays, thruster masses, PPUs, propellant tank, spacecraft structure, and cabling. In this section, we present some basic models for such systems, along with a description of how they can be used alongside the trajectory database to create a mission-sizing tool.

Propulsion System

To first order, the propulsion system mass is largely estimated by a database of masses associated with the SEP thrusters of interested, such as the one shown in Table 1. Mass numbers for the thruster, PPU, and gimbal are multiplied by the number of thrusters used, along with estimates for a Xe flow control (XFC) system and other lines and fittings. It is possible to have spare thrusters for

Table 2 - Optimized power levels and associated mission parameters for 220 kg wet mass at GTO.

Engine Set	Opt. P_0 [kW]	P_0 +15% [kW]	GTO Xe [kg]	GTO TOF [d]	Helio Xe [kg]	Helio TOF [d]	Dry Mass [kg]	Total Xe [kg]	Total TOF [yr]	Useful [kg]
MaSMi x 1	1.4	1.61	40	247	62	493	118	102	2.03	73.5
MaSMi x 2	1.7	1.96	26	250	64	456	130	90	1.93	74.9
PPS1350 Hi I_{sp}	1.9	2.19	28	240	69	460	123	97	1.92	66.0
PPS1350	2.0	2.30	28	240	73	415	119	101	1.79	61.5
SPT100	2.2	2.53	30	240	74	405	116	104	1.77	63.9

throughput or thrusters sharing a PPU, but these must be evaluated on a case-by-case basis.

The next question to answer is that of the method of attitude control. With two or more SEP thrusters on gimbals, it may be possible to use the EP system alone for reaction wheel desaturation and large body rolls. If a single engine is used or finer control is needed then a small reaction control system (RCS) should be used, adding 10+ kg for a basic system. On the plus side, an onboard RCS could obviate the need for large-travel gimbals on one or more of the EP thrusters.

The scalable portion of the propulsion system is the Xe tank. As the trajectory changes, so does the mass of the propellant needed. A good rule-of-thumb is that the tank will weigh between 5-10% of the propellant. Since Xe is stored as a supercritical pressurized gas, the amount of Xe per unit volume is not fixed. Values range from 1-2 kg per liter, with optimal tankage fractions around 1.4 kg/L [14].

Power System

Estimates of the power system mass largely depend on the structural technology of the solar arrays. The efficiency and packing density of the cells are important, but most arrays are similar enough in this regard to not be a distinguisher in mass. The solar cells are typically deployed using either rigid panels or a flexible substrate (e.g. UltraFlex or Roll Out Solar Arrays (ROSA)). The mass of the solar arrays can be estimated either by using a simple solar array mass factor (slope in kg/kW), or linear coefficients (slope plus offset). Table 3**Error! Reference source not found.** shows some typical ranges for the two types of arrays. In addition to the estimate of solar array mass, the power subsystem can also include a battery and other power related equipment. These items can be included either in the fixed masses of the spacecraft or the solar array offset.

Table 3 - Mass estimation coefficients for solar arrays. Use either the Mass Factor multiplier or the slope and offset to estimate a rough mass for solar arrays. A value from the middle of the range should provide a reasonable first guess.

Array Structure	Mass Factor [kg/kW]	*or*	Offset [kg]	Slope [kg/kW]
Rigid	11 - 15		10 - 15	7 - 11
Flexible	7 - 9		10 - 15	4 - 7

Structures and Cabling

At early stages of development, structural mass is typically estimated using parametric estimating relationships based on the masses of the other subsystems. The same is true of cabling and harness. Again, reference texts (such as [13]) can give estimates. What is important is that the spacecraft structure grows or shrinks to accommodate changes in total mass supported. Primary structure mass is often approximated by a fixed percentage of the wet mass, dry mass, or some combination of both. Cabling is typically 8-10% of the dry mass.

The Mission Sizing Tool

Once these models are determined, it is possible to represent their inputs, outputs, and relationships in a simple tool such as a spreadsheet. The simplified trajectory database from the previous section can also be linked, with a flow similar to that of **Error! Reference source not found.**. The total spacecraft dry mass, along with other parameters such as power and TOF, determine the trajectories that can be chosen from the database. The trajectories then determine the propellant mass needed which in turn drives the dry mass of the system using the models previously described.

The process flow can be run in either direction. The trajectory-centric way is to start with a wet mass at GTO, then find the trajectories, power, TOF, etc. that maximize the useful mass in orbit. This mass can be considered an allocation to the subsystems. If suitable components can be found to fit, then the solution is converged. The spacecraft-centric way, on the other hand, is to start from a bottoms-up approach with subsystem mass estimates. The mass equipment list (MEL) is built up along with a guess at propellant mass. The wet mass then goes into the trajectory database. The design is converged when a suitable trajectory meets the mass, power, and TOF constraints.

In addition to estimating masses, it also critical to add suitable margins to each subsystem along with power level and propellant mass. This can be a fixed percentage added to individual systems or to the spacecraft in general. The size of the margins is dependent on the confidence and maturity of the design in question. There are many references to find margin policies for early concept studies [13][15].

Figure 9 is a screen shot of a mission sizing tool set up to design missions in the spacecraft-centric way. User inputs are given on the power, type and number of thrusters, margins, fixed subsystem masses, etc. A guess at propellant mass allows the other subsystems to be sized and a wet mass is calculated. The wet mass, power level, thruster, and TOF constraint then specify GTO spiral parameters as described in Section 2. The remaining wet mass is then interpolated on the corresponding thruster trajectory table for the given power, providing the details on the heliocentric transfer and spiral down. The new total propellant number goes back into the spacecraft model and the process is iterated to convergence.

Main Inputs		
Power BOL	2.18	kW
S/C Power	271.5	W
Power Marg.	15%	
Thruster	MaSMi	
#	2	
GTO Strategy	TOF	
	300	days
Xe Unusable	5%	
System Margin	0.0%	

Subsystems (includes [illegible] margin)		
Payload	18.9	kg
CDH	4.1	kg
GNC	8.7	kg
RCS	0	kg
Telecom	21.4	kg
Thermal	7.2	kg
Array Tech	225	W/kg
Offset	12.65	kg
Pow. Sys. Marg.	8.0%	
Battery	5.92	kg
Power Other	11.8	kg

Variable Subsystems		
Tank Mass	9.1	kg
Prop. Sys. Mar.	25%	
Thr. Gimbal	2	kg

Outputs	Converge	
Mwet	332.8	kg
Mdry	183.2	kg
MXe	149.6	kg
MXe,used	142.5	kg
Minert	7.1	kg

State	Mass	TOF	Cum DV
Launch	332.8	0	0
Earth SOI	284.5	300	2664
Mars SOI	202.7	763	8371
Orbit	191.3	828	9353

MEL		
Payload	18.9	kg
CDH	4.1	kg
GNC	8.7	kg
RCS	0.0	kg
Telecom	21.4	kg
Thermal	7.2	kg
Propulsion	34.4	kg
Power	43.5	kg
Structures	34.2	kg
Harness	11.0	kg
Total SS	183.2	kg
System Margin	0.0	kg
Total Dry	183.2	kg

TOF			
Spiral Out	300	10.0	0.82
Helio	463	15.4	1.27
Spiral Down	65	2.2	0.18
Total	828	27.6	2.27
	days	months	years

DVs			
Spiral Out	2664	m/s	
Helio	5707	m/s	
Spiral Down	982	m/s	[illegible]
Total	9353	m/s	1.3365
			[illegible]
Acc. Init.	0.249	mm/s	3.50
			[illegible]
P0 @ E	1.660	kW	
P Eng @M	0.460	kW	175

Xenon		
Spiral Out	48.3	kg
Helio	81.8	kg
Spiral Down	11.4	kg
Mission Xe	1.0	kg
Total	142.5	kg

Figure 9 - Screen shot of example mission sizing tool for a GTO → Mars orbiter. The left side cells in yellow are user inputs for masses and other mission parameters. They are linked via equations on other sheets to the basic system models and trajectory database tables. Output masses, ΔV's, and TOFs are given on the right for the converged solution.

This tool allows the user to quickly vary input parameters to gauge their sensitivity and effect on the outcome. A much more detailed version of this type of tool has been developed at JPL for the study of Mars Sample Return architectures - called the Mars ORbiter Tool (MORT) [6]. It has evolved over the years to incorporate vast databases of Earth-to-Mars and Mars-to-Earth trajectories [16], spiral [9] and aerobraking models, rendezvous [17][18] and orbit change maneuvers, numerous launch vehicles and thrusters, and hybrid trajectories [19]. It also has much more detailed subsystem models, often to the component level, to keep track of more parameters and assure mission optimality. It has been used to survey millions of potential architectures and has been invaluable in understanding the complexity of the sample return problem with low-thrust. The simple tool presented in this work is affectionately dubbed "MORT Jr.".

4. EXAMPLE: MARS AREOSTATIONARY ORBITER

As an exercise, a mission to areostationary orbit was developed using the MORT Jr. tool. The desire was to start from an ESPA Grande (465 kg) rideshare in GTO and provide science and telecom relay from areostationary orbit. The science package would include a multispectral imager and spectrometer for a payload weighing 19 kg. A powerful telecom package consisting of a 1-meter Ka band high-gain antenna (HGA), an articulated medium-gain antenna (MGA), and two X-band low-gain antennas (LGA) would provide data relay for surface assets as well as collected science data [20]. The basic notional configuration is shown in Figure 10.

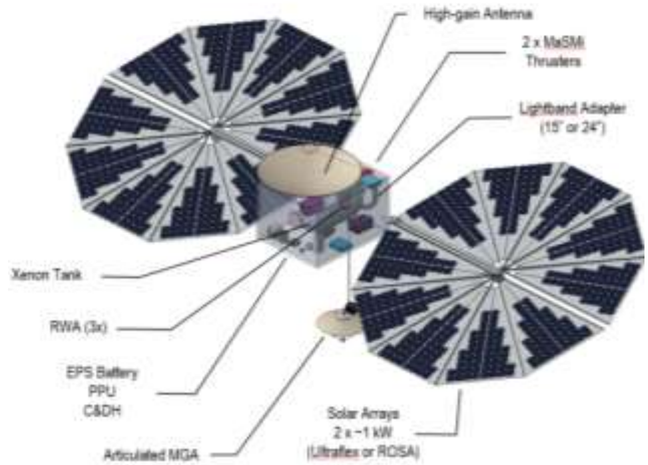

Figure 10 - Conceptual Areostationary Orbiter

Initial estimates of ΔV requirements meant that the propellant mass fraction was likely to just under 50%, leading to a Xe mass of up to 200 kg or more. For these masses, it was determined that 2 MaSMi engines and > 2 kW would be needed in order to provide sufficient acceleration to keep TOF reasonable. As a margin strategy, 15% was added to the required EP power, 5% of the Xe was deemed unusable, and 30% mass margin was carried on all elements.

After initial mass estimates were made for fixed mass subsystems, the MORT Jr. tool was used to find an optimized trajectory that met mission constraints and led to a converged mission. The optimized power level was found to be 2.2 kW at Earth, with margins. Figure 11 gives the outputs for the mission design. The mission fits on the ESPA Grande with > 50 kg to spare and delivers a 220 kg orbiter to Mars. The trip takes 2.4 years, with 10 months being dedicated to depart Earth and almost a year and a half in heliocentric transfer. The whole mission requires 9.8 km/s of ΔV.

Outputs			DVs		
Mwet	**409.3**	kg	Spiral Out	3110	m/s
Mdry	219.2	kg	Helio	5681	m/s
MXe	190.1	kg	Spiral Down	991	m/s
			Total	**9782**	m/s
Xenon			**TOF**		
Spiral Out	68.4	kg	Spiral Out	0.82	years
Helio	97.8	kg	Helio	1.39	years
Spiral Down	13.8	kg	Spiral Down	0.21	years
Minert	10.1	kg	**Total**	**2.43**	years

Figure 11 - Output for Mars Areostationary Orbiter. The wet mass is 409 kg with a ΔV of nearly 10 km/s and a total transfer time of 2.4 years.

5. CONCLUSIONS

In this paper, we describe the process to parameterize key elements of a low-thrust mission from GTO to Mars. It is essential to develop tools that can simultaneously optimize the spacecraft dry mass and the trajectory design together. Some subsystem masses are basically fixed, while others, such as power and propulsion, vary with trajectory parameters. There are many tradeable factors when optimizing low-thrust trajectories – such as power, time-of-flight, mass, etc. A mission designer must be able to rapidly explore the trade space in order to find the desired balance that meets mission requirements.

In this paper we give a number of rules-of-thumb to begin the estimation process. SEP missions that are underpowered poorly designed often appear to have subpar performance at best and infeasible at worst. This is true even if the mission trajectories and the spacecraft subsystems are designed and optimized with high fidelity. For interplanetary missions, especially those going to Mars and requiring many km/s of ΔV, a good starting point is 5 W/kg of dry mass.

If subsystem and trajectory models are judiciously developed, it is possible to assemble a tool as basic as a spreadsheet in order to carry out the simultaneous optimization of spacecraft and trajectory. Parameterizing GTO escape spirals and reducing trajectory table dimensions help keep the tool quick and efficient. The MORT Jr. tool can draw from the results of 10's of thousands of optimized trajectories to converge on a mission architecture in under a second, allowing mission designers the ability to explore many dimensions of the trade space. Once a promising candidate is selected, a more detailed study can follow.

ACKNOWLEDGEMENTS

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 information about a Mars Areostationary Orbiter in this document is pre-decisional, and is provided for planning and discussion only.

REFERENCES

[1] R.W. Conversano, D.M. Goebel, R.R. Hofer, and N. Arora, "Performance Enhancement of a Long-Life, Low-Power Hall Thruster for Deep-Space Smallsats," IEEE Aerospace Conference, Big Sky, MT, 2017.

[2] R.W. Conversano, D.M. Goebel, I.G. Mikellides, and R.R. Hofer, "Performance Analysis of a Low-Power Magnetically Shielded Hall Thruster: Computational Modeling," *Journal of Propulsion and Power*, Vol. 33, No. 4, July-August 2017.

[3] R.E. Lock, C.D. Edwards, A.K. Nicholas, R.C. Woolley, and D.J. Bell, "Small Areostationary Telecommunications Orbiter Concepts for Mars in the 2020s," IEEE Aerospace Conference, Big Sky, MT, Mar. 2016.

[4] Moog ESPA Payload Adapters Product Data Sheet. http://www.moog.com/content/dam/moog/literature/Space_Defense/spaceliterature/structures/SAIS_ESPA_Ring_Rev_0712.pdf Recovered Oct. 1, 2018.

[5] R.C. Woolley and A.K. Nicholas, "SEP Mission Design Space for Mars Orbiters," AIAA/AAS Astrodynamics Specialist Conference, AAS Paper 15-632, Vail, CO, Aug. 2015.

[6] A.K. Nicholas, R.C. Woolley, A. Didion, F. Laipert, Z.P. Olikara, R. Webb, and R.E. Lock, "Simultaneous Optimization of Spacecraft and Trajectory Design for Interplanetary Mission Utilizing Solar Electric Propulsion," AAS Astrodynamics Specialist Conference. Maui, HI, Jan. 2019.

[7] P.A. Penzo, "Planetary Missions from GTO using Earth and Moon Gravity Assists," AIAA/AAS Astrodynamics Specialist Conference, Paper AAS 98-4393, Boston, MA, 10-12 August, 1998.

[8] W.R. Gershman, P.A. Penzo, and A.B. Wiercigroch, "From GTO to the Planets", 3rd IAA International Conference on Low-Cost Planetary Missions, Paper IAA198-0903, Pasadena, CA, April 27-May 1,1998.

[9] Z.P. Olikara, "Framework for Optimizing Many-Revolution Low-Thrust Transfers," AAS/AIAA Astrodynamics Specialist Conference, AAS Paper 18-332, Snowbird, UT, Aug. 2018.

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

[11] W.G. Melbourne and C.G. Sauer, "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.

[12] 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.

[13] J. R. Wertz, and W. J. Larson. *Space Mission Analysis and Design, Space Technology Library*. Microcosm Press and Kluwer Academic Publishers, El Segundo, CA, USA, 1999.

[14] R.P. Welle, “Propellant Storage Considerations for Electric Propulsion,” 22nd International Electric Propulsion Conference, Paper 91-107, 1991.

[15] D. P. Thunnisen. "Method for Determining Margins in Conceptual Design", Journal of Spacecraft and Rockets, Vol. 41, No. 1 (2004), pp. 85-92.

[16] R.C. Woolley, F. Laipert, A.K. Nicholas, and Z.P. Olikara, “Low-Thrust Trajectory Bacon Plots for Mars Mission Design,” AAS Astrodynamics Specialist Conference. Maui, HI, Jan. 2019.

[17] Z.P. Olikara and A.K. Nicholas, “Chemical and SEP Orbit Matching for Mars Sample Rendezvous,” AAS Astrodynamics Specialist Conference. Maui, HI, Jan. 2019.

[18] A. M. Didion, A. K. Nicholas, J. E. Riedel, R. J. Haw, and R. C. Woolley, “Methods for passive optical detection and relative navigation for rendezvous with a non-cooperative object at Mars.” Paper AAS 18-288, AAS/AIAA Astrodynamics Specialist Conference, August 2018.

[19] F. Laipert, A.K. Nicholas, Z.P. Olikara, R.C. Woolley, and R.E. Lock, “Hybrid Chemical-Electric Trajectories for a Mars Sample Return Orbiter,” AAS Astrodynamics Specialist Conference. Maui, HI, Jan. 2019.

[20] N. Barba, T. Komarek, R.C. Woolley, L. Giersch, V. Stamenkovic, M. Gallagher, and C. Edwards, “Mars Small Spacecraft Studies: Overview,” *IEEE Aerospace Conference*, Big Sky, MT, Mar. 2019.

BIOGRAPHY

Ryan Woolley *received a B.S. in Physics-Astronomy from Brigham Young University, a M.S. in Astronautical Engineering from USC, and a Ph.D. in Aerospace Engineering from the University of Colorado. He has been with JPL since 2005. Ryan began his tenure as a systems engineer in the Mars Mission Concepts group and has since transferred to the Inner Planets Mission Analysis group. He has worked on nearly all aspects of the Mars Sample Return campaign and has developed many tools to design evaluate mission designs and architectures.*

Zubin Olikara *received bachelors and master's degrees in Aeronautics & Astronautics from Purdue University, and a Ph.D. in Aerospace Engineering Sciences from the University of Colorado. He now works at the NASA Jet Propulsion Laboratory in the Inner Planet Mission Analysis group supporting Mars Sample Return architecture studies.*