Analysis of System Margins on Missions Utilizing Solar Electric Propulsion

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NASA's Jet Propulsion Laboratory has conducted a study focused on the analysis of appropriate margins for deep space missions using solar electric propulsion (SEP). The purpose of this study is to understand the links between disparate system margins (power, mass, thermal, etc.) and their impact on overall mission performance and robustness. It is determined that the various sources of uncertainty and risk associated with electric propulsion mission design can be summarized into three relatively independent parameters 1) EP Power Margin, 2) Propellant Margin and 3) Duty Cycle Margin. The overall relationship between these parameters and other major sources of uncertainty is presented. A detailed trajectory analysis is conducted to examine the impact that various assumptions related to power, duty cycle, destination, and thruster performance including missed thrust periods have on overall performance. Recommendations are presented for system margins for deep space missions utilizing solar electric propulsion.

Nomenclature

CBE	=	current best estimate
ΔV	=	delta-velocity, chemical reference mission (m/s)
$M_{delivered}$	=	final delivered mass (kg)
M_o	=	initial mass (kg)
P	=	PPU input power (W)
$c_{electric}$	=	effective exhaust velocity, electric propulsion (m/s)
c_{chem}	=	effective exhaust velocity, chemical propulsion (m/s)
η_p	=	efficiency of electric propulsion system (P_{jet}/P)
t	=	total flight time (s)

I. O Introduction

Electric propulsion (EP) has been in development for more than 40 years and is now the primary source of propulsion on more than 180 commercial spacecraft now operating in space.[1] However, because the systems in use today were developed independently, spacecraft operators and manufacturers have adopted different and sometimes contradictory criteria for acceptable design margins for missions using electric propulsion in areas such as power, mass, and operational duty cycle. Today, as the use of EP expands into the solar system with missions such as Deep Space 1, Hayabusa, and Dawn, the acceptable design criteria for deep space missions have been defined on a mission specific, ad hoc basis. This has made it difficult to objectively evaluate the adequacy of and risks associated with proposed future deep space missions utilizing electric propulsion. Recognizing that there are numerous deep space EP missions either proposed or in development, NASA's Jet Propulsion Laboratory has conducted a study focused on the analysis of appropriate margins for deep space missions using solar electric propulsion (SEP). The purpose of this study is to understand the links between disparate system margins (power, mass, thermal, etc.) and their impact on overall mission performance and robustness. The results of this study show the important,

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interlocking relationship between subsystems on SEP missions. This study is part of an effort that will ultimately define JPL's standards for the evaluation of margins on proposed SEP missions.

II. OOverview

Traditionally, the organizations that build chemical propulsion systems are organized and managed as a "subsystem," one of several subsystems that are brought together to form the space vehicle. Propulsion engineering is treated as a distinct intellectual discipline, and systems engineers manage the interfaces between the propulsion subsystem and other elements of the spacecraft. By comparison, the organizations that build electric propulsion (EP) systems are inherently multidisciplinary, combining several areas of expertise that are traditionally organized and managed within separate subsystems. For example, most EP systems require the manufacturing of space qualified high power/high voltage avionics to drive the thrusters. More generally, electric propulsion systems interact strongly with multiple disciplines including:

1) power
2) propulsion
3) mechanisms
4) thermal
5) space environmental interactions (plasma physics)
6) low-thrust mission analysis
7) guidance and control

In the design phase, EP development programs naturally concentrate on propulsion centric areas of development, notably thrusters and life modeling and testing. In flight, however, the vast majority of anomalies have occurred in other elements of the subsystem, notably in the areas of thermal management, mechanisms, and power processing. Because of the many subsystems involved, a broad multi-disciplinary systems engineering approach is key to the successful design and flight of electric propulsion missions.

The end-to-end design of SEP missions leads to interactions with additional disciplines, notably in the areas of spacecraft operations and system margin management. Recent work on proposed deep space missions using SEP has shown that the links between trajectory design, power, mass, and operational duty cycle are strong, but not well modeled or understood (see Figure 1). The purpose of this study is to characterize these system level interactions and to understand the links between disparate systems margins. This will help mission designers determine the impact each margin element has on overall mission performance and robustness.

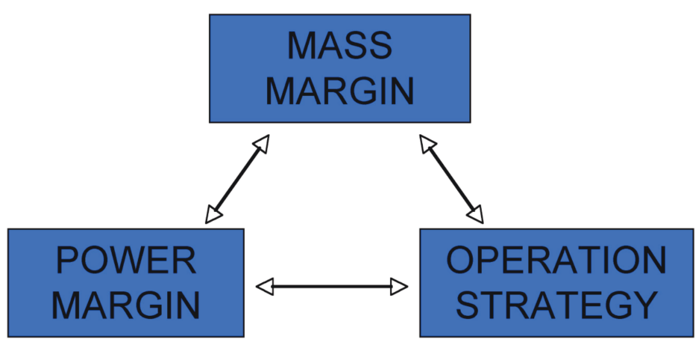

Figure 1: Electric Propulsion Missions are Inherently Multi-Disciplinary with Interactions occurring Between Multiple Elements of the Overall System

Figure 2 illustrates the principal elements of an EP system. Each element contributes various sources of uncertainty that affect the overall spacecraft design. In this study, we examine the interactions between system margins by:

- Identifying the sources of uncertainty and relevant parameters for EP systems
- Defining a framework for evaluating each parameter and establishing the governing interactions between parameters
- Evaluate the sensitivity of parameters and defining typical values

This study addresses an array of system-level interactions using data from actual trajectories, mission studies, and flight projects. While this data set is not inclusive of all destinations or assumptions, it presents a set of information that can be used as a reference in the development of system margins for most deep space missions using Solar Electric Propulsion.

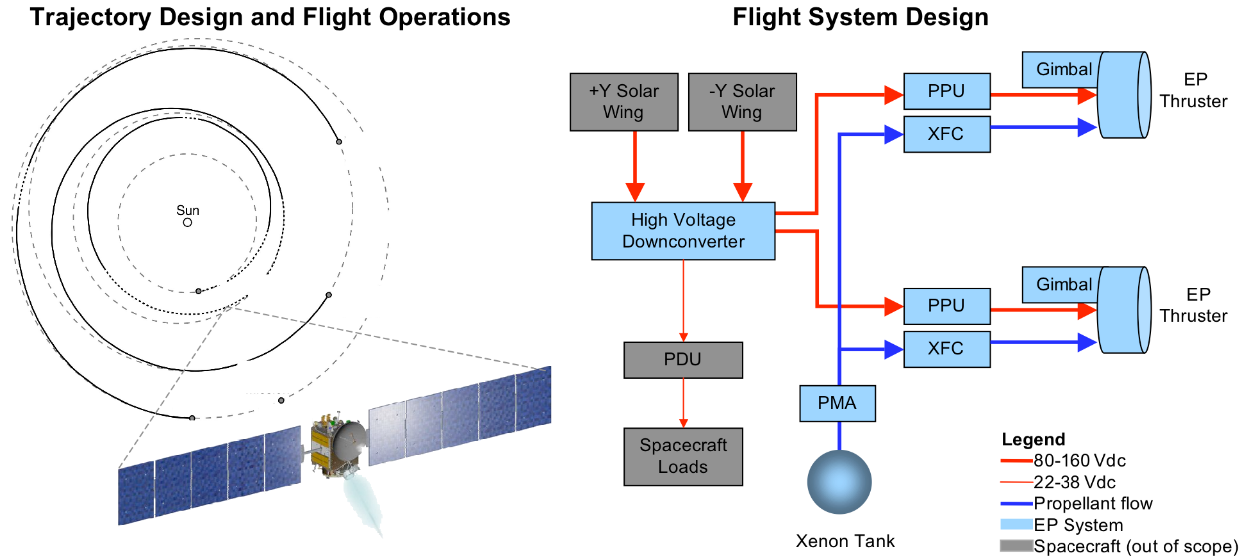

Figure 2: EP is Characterized by the Interaction of Trajectory Design, Operations, and Flight System Design

III. Analysis

A. Parameter Definition and Sources of Uncertainty

The design of low-thrust mission architectures is a challenging problem requiring simultaneous optimization of multiple design parameters, including flight time, power, duty cycle, thrust, specific impulse, and propellant mass. There is an inherent interaction between parameters, and it is often possible to trade between parameters (i.e. power and flight time) without affecting other parameters (i.e. delivered mass). Table 1 identifies typical sources of uncertainty and performance parameters for SEP missions. The table lists contributions from four areas traditionally considered separate "subsystems": mission design and navigation, mission planning and operation, power, and propulsion. To the overall mission, the "subsystem" associated with each element is not particularly important, but this categorization does illustrate the multi-disciplinary nature of EP mission architectures.

Table 1: Sources of Uncertainty in Electric Propulsion

Mission Design and Navigation
Trajectory modeling, trajectory execution errors, environmental effects, pointing error, and mission-specific encounters
Mission Planning and Operation
Post-launch checkout & testing, communications & tracking, uploads & regular maintenance, and number of restarts
Power Subsystem
Solar array power, high-voltage down-converter efficiency, power dissipation, and PPU input power uncertainty
Propulsion Subsystem
Thruster lifetime throughput, fill error, commissioning, leakage, residuals, plume impingement, thrust uncertainty, and propellant use uncertainty

To manage the uncertainties introduced in Table 1, a set of parameters is required to characterize and manage corresponding system interactions. These parameters and their associated margins are managed at the system level and can mitigate unexpected trajectory and design changes that would otherwise ripple through flight system or operational design. The principle parameters are:

- **Neutral Mass (kg):** The combined mass of the hardware and non-EP propellant (usually hydrazine). [2]
- **EP Power (W):** The maximum power used by the EP thrusters during the trajectory, usually specified at a reference solar range of 1 AU.
- **EP Propellant (kg):** The propellant used by the EP system for the successful completion of the mission.

- **Tank Capacity (kg):** The total capacity of the propellant tank.
- **Thrust (N)**: The thrust provided by the EP thrusters for propulsion, which varies depending on the thruster operating range, trajectory requirements, and available power and propellant.
- **Thruster Throughput (kg):** The usable propellant throughput of the thrusters throughout the mission lifetime.
- **Duty Cycle (%):** The actual operating time of the thrusters assumed during trajectory thrust arcs.
- **Flight Time (days):** The amount of time required to reach the destination(s).

The above parameters define the electric propulsion system and span the performance envelope necessary to ensure a successful mission. By addressing the margin requirements on each parameter, a mission can improve its robustness to the inevitable design changes that accompany project development. Although design margins can be independently applied to each parameter, it is useful to summarize all of these uncertainties into three relatively independent margins that govern trajectory performance.

- **EP Power Margin**
- **Propellant Margin**
- **Duty Cycle Margin**

The overall relationship between these three parameters and other major sources of uncertainty is shown in Figure 3. The performance margins needed to compensate for thrust uncertainty and missed thrust periods can be incorporated into the three major margin elements listed above: power, propellant, and duty cycle. This reduces the number of dependent parameters and also prevents "double-booking" of margins within each subsystem. In this study, flight time is treated as a separate independent parameter that is allowed to vary freely within a limited range (~5%). No effort was made to define the allowable flight time "margin" or to impose a strict limit on the flight time. This approach is sufficient for missions where there is a weak dependence between arrival time and overall mission performance. Missions where there are very strict arrival time requirements (for example, a Mars Sample return vehicle that is targeting a rendezvous with a lander delivered using chemical propulsion) would need to actively manage flight time margin as well as the other three margin elements.

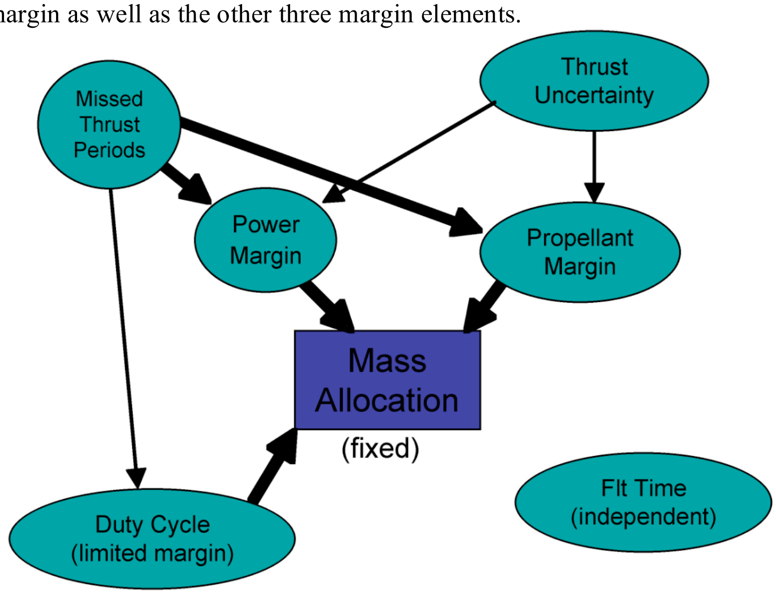

Figure 3: Multiple Error Sources can be Summarized into Three Relatively Independent Margin Parameters

EP power margin is expressed as a fraction of total EP power and includes all factors that directly affect the PPU input power including PPU regulation and efficiency errors. It also includes the uncertainty in the PPU power required to deliver the requested thrust level, which includes both PPU and thruster performance uncertainties. Propellant Margin is expressed as a fraction of the total nominal propellant load and includes factors that directly affect propellant mass. This includes propulsion tank ullage, flow rate regulation uncertainty, and dispersions in specific impulse performance. Duty Cycle Margin is expressed as a fraction of total thrust time and includes all factors that affect duty cycle or thrust level. Duty cycle calculations do not include pre-planned optimal coast periods inserted for trajectory optimization. Duty cycle calculations do include deep space network (DSN) "passes" for communications and tracking, non–optimal coast periods required for spacecraft maintenance and non-optimal coast periods required for contingencies. Duty cycle also includes dispersions in thrust magnitude and pointing uncertainties.

B. Trajectory Analysis

The design of a successful mission requires an understanding of how spacecraft mass (i.e. available dry mass and required propellant) is affected by a set of mission drivers. Traditionally, with chemical propulsion the mass is driven by launch date, flight time, launch vehicle, and specific impulse of the engine. With electric propulsion, the design parameters also include the available power to the engine, the engine throttle curve, the number of engines, duty cycle, forced coasting, and robustness to periods of missed thrust. This extended set of mission drivers creates a more complicated design space, but a systematic search over parameters specific to EP missions can lead to a more robust design. For this study, a trajectory database comprised of sixteen distinct destination-thruster combinations and more than 5,000 individual trajectories was constructed to characterize the relationships between launch mass, dry mass, propellant mass, launch date, flight time, spacecraft power, launch vehicle, EP engine, and duty cycle. A set of baseline missions was selected (comet rendezvous, asteroid sample return, Saturn flyby, and comet sample return) and different Hall (BPT-4000) and ion (NSTAR, 25 cm XIPS, and NEXT) thrusters were utilized. EP power and duty cycle were varied around chosen baseline levels to create the database of trajectories.

A mission using electric propulsion should be robust to contingencies or faults that cause unplanned loss of thrust. The effect of incorporating missed-thrust contingencies into EP trajectories was also explored by calculating the dry mass as a function of thrust-out time, power, and duty cycle. The dry mass of a trajectory that may lose thrust at any time was compared to the performance of trajectories without missed thrust at lower duty cycles. The results provide insight into how the design parameters specific to electric propulsion can be combined to create efficient, robust missions.

1. Baseline Mission Development

The trajectory selection process (shown in Figure 4) begins with the design of several baseline missions that represent a variety of destinations. The baseline set includes a rendezvous with Tempel 1, a near-Earth asteroid (1989ML) sample return, a Saturn flyby, and a Tuttle-Giacobini-Kresak sample return with 25 cm XIPS, BPT-4000, NSTAR, and NEXT thrusters as the primary propulsion. The missions are selected by varying launch vehicles, power levels and number of thrusters and comparing the delivered "dry mass" (or neutral mass) to previous missions and studies. Typically the smallest (or least expensive) launch vehicle that provides a neutral mass close to a desired neutral mass is chosen first. Different power levels are then sampled to adjust the dry mass to an acceptable value. The number of engines depends on the desired acceleration and is correlated to launch mass and available power over the trajectory.

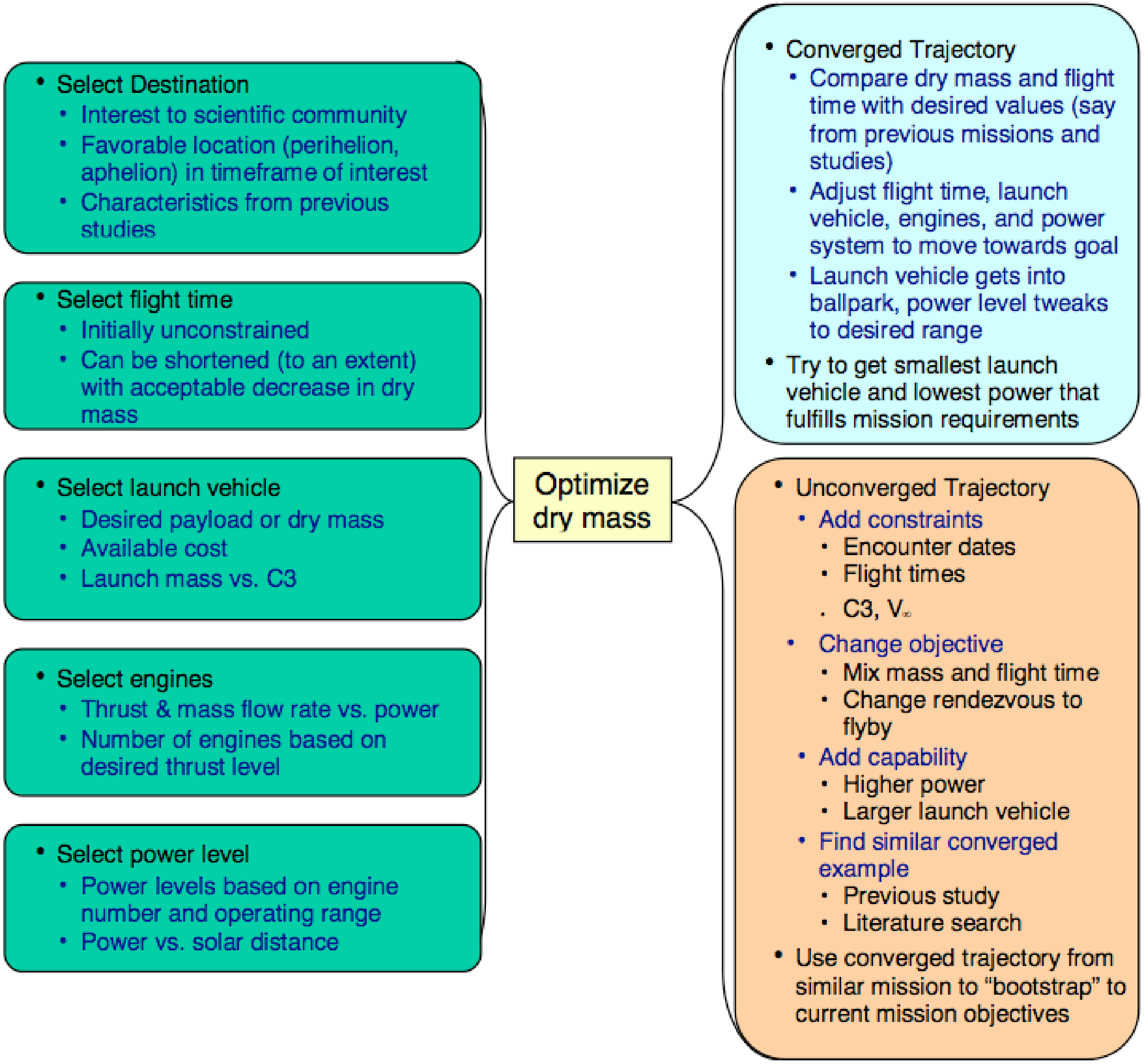

Figure 4: Baseline Trajectory Selection Process

All of the low thrust trajectories are optimized using the Mission Analysis Low-Thrust Optimizer (MALTO). MALTO was designed for preliminary design of electric propulsion missions, and provides reasonable accuracy relative to detailed design tools (which have higher fidelity with slower run times).[2] The key aspect of MALTO is that it approximates continuous thrusting with a series of impulsive ΔV connected by conic arcs as illustrated in Figure 5. MALTO can optimize the initial spacecraft mass, final spacecraft mass, flight time, or a weighted combination of mass and time. The optimization variables include encounter (launch, rendezvous, flyby, or non-body control point) times, spacecraft mass at encounters, stay times at rendezvous bodies, arrival and departure $\mathbf{V}_\infty$, states at non-body control points, power, and the thrust profile. The trajectory may be molded by placing constraints on flight times, propellant consumption, power levels, launch and arrival $\mathbf{V}_\infty$ magnitudes, distance from the sun and flyby conditions (altitude/radius, b-plane angle).

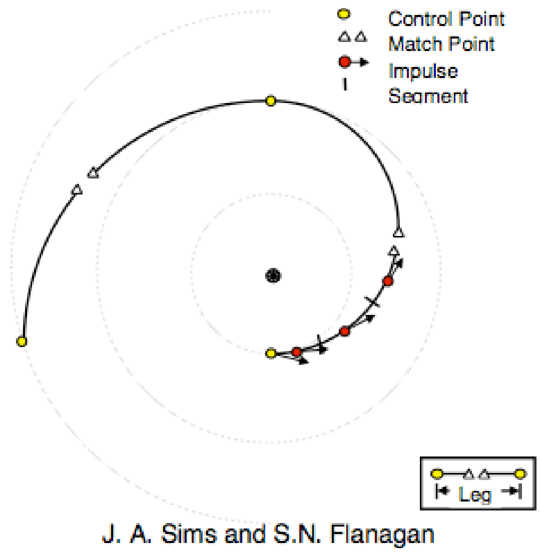

Figure 5: Schematic of MALTO Trajectory Model[2]

For this study, the trajectories are optimized for maximum final (neutral) mass. The comet rendezvous mission has a 20% weighting on the flight time, which keeps the optimizer from extending the rendezvous date by many months for a modest increase in dry mass. The Saturn flyby flight time is constrained to 8 years. The flight time for the asteroid and comet sample return missions are unconstrained. The arrival V_∞ is unconstrained for the asteroid sample return and for Saturn flyby missions, and is constrained to 9 km/s for the comet sample return. The initial mass as a function of C_3 for the launch vehicles are derived from data on the KSC expendable launch vehicle performance estimation website.[3] Launch declinations are unconstrained, except for the Saturn flyby mission which keeps the declination within ±30º. No launch vehicle margin or adapter mass included in optimization. Typical throttle curves were used for the NSTAR, XIPS, NEXT, and BPT-4000 thrusters.[4],[5],[6] The minimum number of engines that supports the available power is used at a given time, which typically runs the engines at the highest efficiency, but shortens lifetime. The effect on running the maximum number of engines for longer engine life is not examined. The duty cycle for baseline missions is 90%.

Xenon tankage and electric propulsion system mass are not included in the optimization; but are subtracted from the delivered mass after the fact. A 30% contingency is added to the power and propulsion mass to determine net mass [i.e. Net mass = Neutral Mass – (Power and Propulsion system mass)/(1-0.3)]. The power system uses triple-junction gallium arsenide arrays, and the power generated as a function of distance from the sun is found in the MALTO database. Radiation and time dependent losses were all subtracted at the beginning of life for the trajectory, a conservative assumption. For the Saturn mission, the spacecraft (non EP) power is 600 W with RTGs as the power source. All other missions have a spacecraft power of 500 W from the solar arrays. There are no limits on propellant throughput, which could also affect engine life. Finally, no forced initial coast or coast before flybys are included in the baseline missions.

Table 2: Baseline Trajectories (90% Duty Cycle)

Mission	Engine	LV	P0, kW	Max. # of Engines Thrusting	# of Engines Life	Initial mass, kg	Neutral mass, kg	Xenon mass, kg	TOF, days	Propulsion & Power, kg	Net Mass, kg
Tempel 1 Rendezvous	XIPS	A401	10	2	2	1212	951	261	1071	316	500
	BPT	A401	8.5	2	3	1661	1028	633	1097	316	577
	NSTAR	A401	10	2	2	1250	967	283	1057	295	545
	NEXT	A401	11	1	1	1258	1014	244	1128	314	565
1989ML Sample Return	XIPS	D2925H	2.5	1	1	973	867	106	1108	172	622
	BPT	D2925H	3	1	2	1211	931	280	1112	189	661
	NSTAR	D2925H	2.6	1	1	1008	891	117	1108	160	662
	NEXT	D2925H	2.7	1	1	1001	865	136	1111	213	560
Saturn Flyby	XIPS	A551	10	3	4	5215	4552	663	2922	664	3604
	BPT	A551	10	3	5	5617	4440	1177	2922	677	3473
	NSTAR	A551	10	4	5	5263	4501	762	2922	675	3536
	NEXT	A551	11	2	2	5262	4682	580	2922	650	3753
TGK Sample Return	XIPS	A511	20	4	4	2248	1623	625	2944	554	832
	BPT	A521	20	4	5	2837	1609	1228	2947	565	801
	NSTAR	A521	20	5	5	2345	1654	691	2940	539	884
	NEXT	A511	20	3	2	2164	1625	539	2940	572	808

Trajectory Envelope

Following the completion of the sixteen baseline missions, a power and duty cycle tradespace was created about each baseline trajectory. The power was varied plus or minus 20% and the duty cycle was varied between 80-100%. The resulting database includes additional information on the destinations, thrusters, launch vehicles, and assumptions used in this study, along with launch dates, flight times, number of thrusters, and several mass numbers. The mass information includes propellant mass and component-level breakouts for power and propulsion subsystems. These numbers can be used to determine the net mass, or the neutral mass without the power and propulsion subsystem masses.

Analysis of Trends

Using the trajectory database, a variety of trends may be identified and evaluated. The trajectory data set is presented in Figure 5, where delivered mass is plotted with respect to EP power levels for duty cycles ranging from 80 to 100%. The four missions represent four distinct types of EP trajectories. The Near-Earth Asteroid (NEA) Sample Return and Comet Rendezvous Missions are sized for a Discovery class opportunity with a baseline delivered mass in the vicinity of 1,000 kg. The Saturn Mission is a flagship mission with an estimated delivered mass of 4,500 kg, which allows sufficient mass for either a flyby or rendezvous, depending on the mission requirements. The Comet Sample Return Mission is consistent with a New Frontiers class opportunity with a delivered mass near 1,500 kg. It should be noted that on several missions, the BPT-4000 Hall thruster is able to deliver nearly as much mass as the ion thrusters despite operating at a lower specific impulse. This occurs because the high thrust delivered by the Hall thruster compensates for the lower specific impulse by increasing the efficiency of the thrust arcs and/or lowering the velocity of the launch vehicle at separation. This performance characteristic is typical and has been seen in previous studies of BPT-4000 performance on deep space missions.[6]

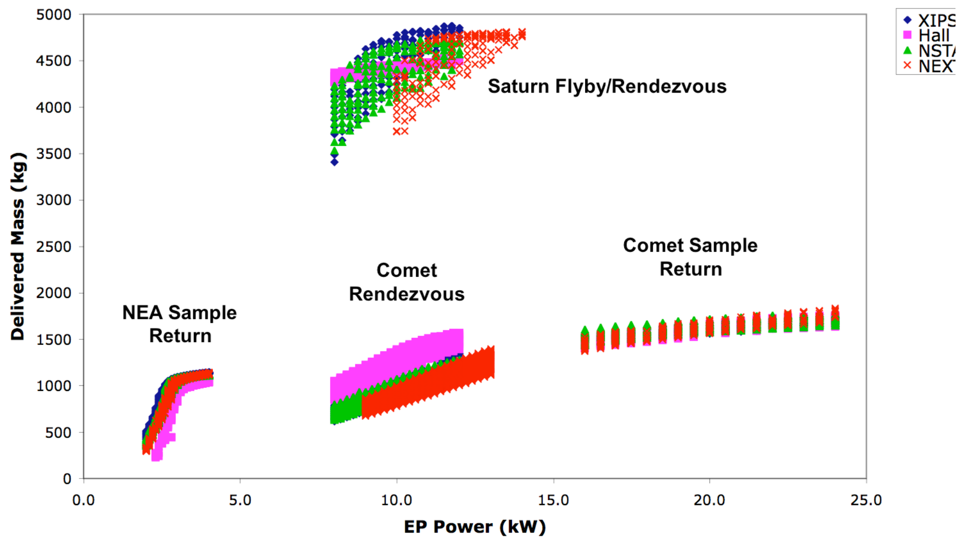

Figure 6: Trajectory Database Overview: EP Power (kW) versus Delivered Mass (kg)

In Figure 5, the plots are characterized by groupings of points that vary in both mass and power as a function of mission and duty cycle. In previous work, large databases of low thrust trajectories have been analyzed analytically identify governing variables and fundamental relationships.[7] The same technique is used here to identify the basic relationship between power and duty cycle.

The rocket equation for combined chemical-electric missions has been derived analytically under the specific conditions of constant power, constant thrust and fixed specific impulse operation and is given by:[8]

$$e^{-\Delta v_{chem}/c_{chem}} = \frac{m_{delivered}}{m_o}\left(\frac{\eta_p P t}{\frac{1}{2} m_{delivered} c_{electric}^2} + 1\right)^{\left(1 - \frac{\eta_v c_{electric}}{c_{chem}}\right)} \quad \text{(eq 1)}$$

Deep space SEP missions have variable power, generally do not thrust constantly and generally use variable specific impulse. Nevertheless, the basic physics of equation 1 still applies. The numerator in the right most term is the total kinetic energy delivered by the electric propulsion system. The overall delivered mass is largely a function of total integrated electrical energy available for propulsion. Varying any of the three quantities in the numerator: the propulsive efficiency, the power, or the total thrust time, has exactly the same impact on the delivered mass. For a deep space mission, equation 1 implies that variations in duty cycle (thrust time) should have the same effect as a reduction in power. This hypothesis can be tested by plotting delivered mass as a function of the product of the duty cycle and power level, which results in Figure 7.

When plotted in this manner, the clusters of data seen in Figure 6 are generally replaced by the thin lines of data shown in Figure 7. The implication is that, to some degree, EP power and duty cycle are interchangeable. This does not occur in all cases, but often the trend is that performance may be increased by either increasing duty cycle or power by the same factor. From a system margin point of view, it can be said that:

• In many cases, duty cycle margin and power margin have equivalent impacts on delivered mass and are therefore interchangeable.

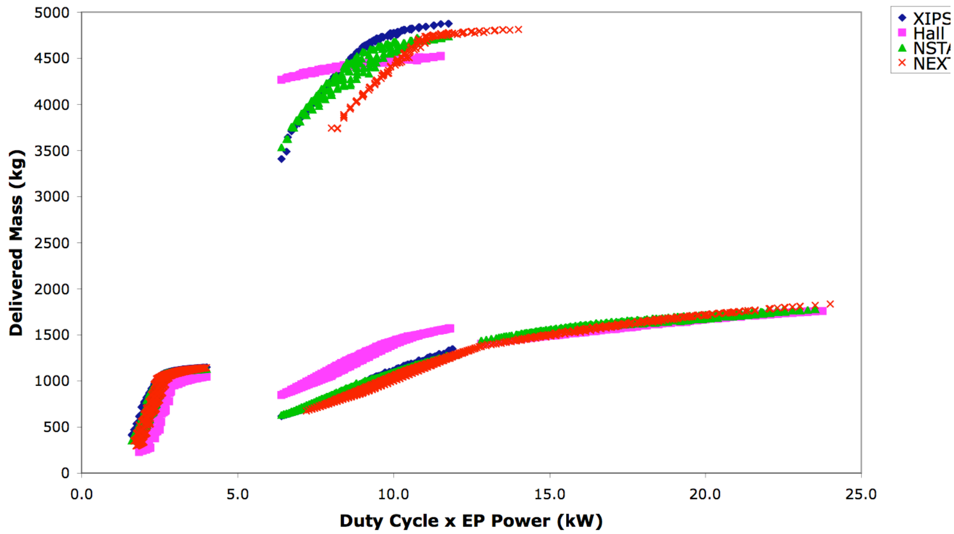

Figure 7: Duty Cycle and EP Power Relationship

2. Missed-thrust Analysis

Should a complication occur while the spacecraft is on its way to the target, it may take several days to a few weeks to fix the problem. Consequently, a robust mission design should have the capability to miss thrust at any point on the trajectory (as unexpected problems can occur at unexpected times), and still meet the mission requirements. The objective is to design a trajectory that maximizes dry (neutral) mass while missing thrust at any given time. In this study, we approached this problem by applying forced coasts at sensitive points in the trajectory (e.g. near perihelion) and optimizing the nominal trajectory around these points. A "forced coast" is a planned event that is included in the trajectory optimization, while a "missed thrust" is unplanned event on a previously optimized trajectory. The entire trajectory can become resilient to lost thrust by designing around a few critical points.

For single leg trajectories (such as the comet rendezvous mission), a single "rolling forced coast" approach works well. The trajectory is first optimized around a strategically placed forced coast. Then as the spacecraft approaches this initial forced coast point, the coast is pushed back and the spacecraft thrusts towards the new coast point. Nominally, the spacecraft never reaches a forced coast point, as the trajectory is reoptimized for each new forced coast time, but a nearby coasting solution is always available should the spacecraft lose thrust. This method is current used on the Dawn mission to 28 days of missed thrust margin at most points in the trajectory.

A schematic of the rolling coast process and design variables is provided in Figure 8. For the comet rendezvous mission, the time of the initial forced coast T_0 was optimized using a Nelder-Mead simplex algorithm. The coast duration T_{FC} was fixed to the missed thrust constraint (7, 14, or 28 days), the increment to roll coast T_{roll} was 28 days, and the advance time $T_{adv.}$ was 0 days (i.e. the trajectory was reoptimized at the beginning of each forced coast).

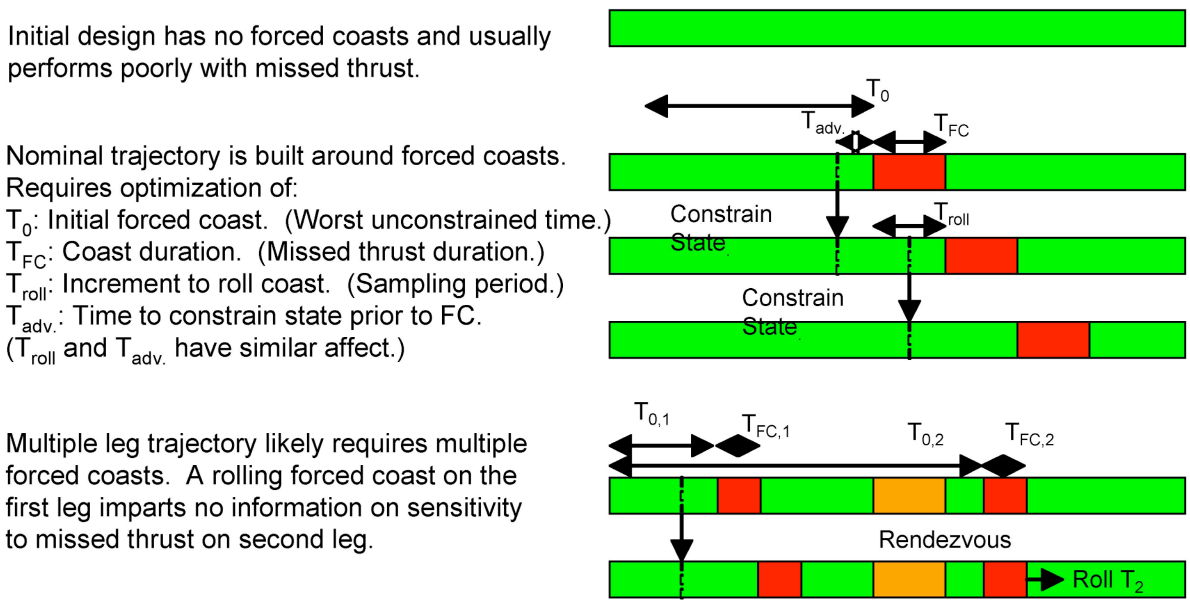

Figure 8: Schematic for Forced Coast Optimization and Rolling Process

It was found that the dry mass constraint on the nominal trajectory can have unexpected effects on the dry mass with missed thrust. For example, consider a trajectory designed to maximize delivered mass that can deliver a dry mass of 1200 kg in the nominal no missed thrust scenario. When the missed thrust analysis is run starting from this nominal trajectory, it is found that the trajectory can only deliver 1000 kg if a missed thrust period occurs at a critical point in the trajectory. In this case, the mission should be designed with a dry mass allocation of 1000 kg because it must account for any unexpected loss of thrust. Intuitively, one would expect that an increase in nominal mass would provide more mass with missed thrust as well. However, what was actually found was that in some cases, if the nominal mass on the no missed thrust trajectory (the nominal trajectory) is constrained to 1100 kg, the corresponding missed thrust analysis generates a missed thrust trajectory that can also deliver 1100 kg, an increase of 100 kg from the previous case. Essentially, the nominal trajectory is constrained to perform more like a missed-thrust trajectory, providing better performance with missed thrust at the expense of performance when thrust is always available.

This trend is detailed in Figure 9 for different combinations of T_{roll} and $T_{adv.}$ for a Tempel 1 rendezvous mission with 9.5 kW at 1 AU and 92% duty cycle. The maximum delivered dry mass for this mission in the no missed thrust scenario is about 1150 kg. The corresponding missed thrust trajectories perform poorly. However, if the nominal mass is constrained (going from right to left in Figure 9), then the missed thrust mass improves to the point where it matches the nominal mass. The time between the constrained state and the new forced coast (T_{adv} + T_{roll}.) affects the point where nominal and missed thrust masses meet. Longer times (sum of 42 days) allow the dry mass to go up to 1100 kg, whereas shorter times (sum of 14 days) only support 1050 kg with missed thrust. However, more time between the constrained state and the next forced coast can cause the trajectory to wander from the previous coasting solution, defeating the purpose of rolling the forced coast (to maintain a nearby coasting solution). When designing trajectories for missed thrust, we suggest optimizing the nominal dry mass and the time of the initial forced coast, and using engineering judgment to choose T_{FC}, T_{roll}, and $T_{adv.}$ so that the trajectory remains resilient to missed thrust between forced coast solutions.

Missions with multiple legs (e.g. a sample return, which has a leg to the target and a leg back to Earth), become more complicated as multiple forced coasts may be required to maximize mass. For example, we found that the asteroid sample return mission was very susceptible to missed thrust on the return leg. But placing an initial forced coast on this leg resulted in a trajectory that performed very poorly when thrust was lost on the first leg. So two forced coasts were required to maintain performance on each leg. The rules for rolling coasts are not straightforward in this case. Should both coasts roll simultaneously, or should the first coast roll up to the second and disappear at some point? Further, should each coast have the same duration, or should the coast duration change during the rolling process? Further investigation of such questions is needed to produce effective methods to build missed-thrust margin into certain types of electric propulsion missions.

Thus far, the asteroid sample return mission has been analyzed for missed thrust with two fixed forced coasts (no rolling). The time T_0 and duration T_{FC} of each coast are optimization variables for trajectories designed to maximize dry mass. The optimization of nominal dry mass has not yet been applied to this mission.

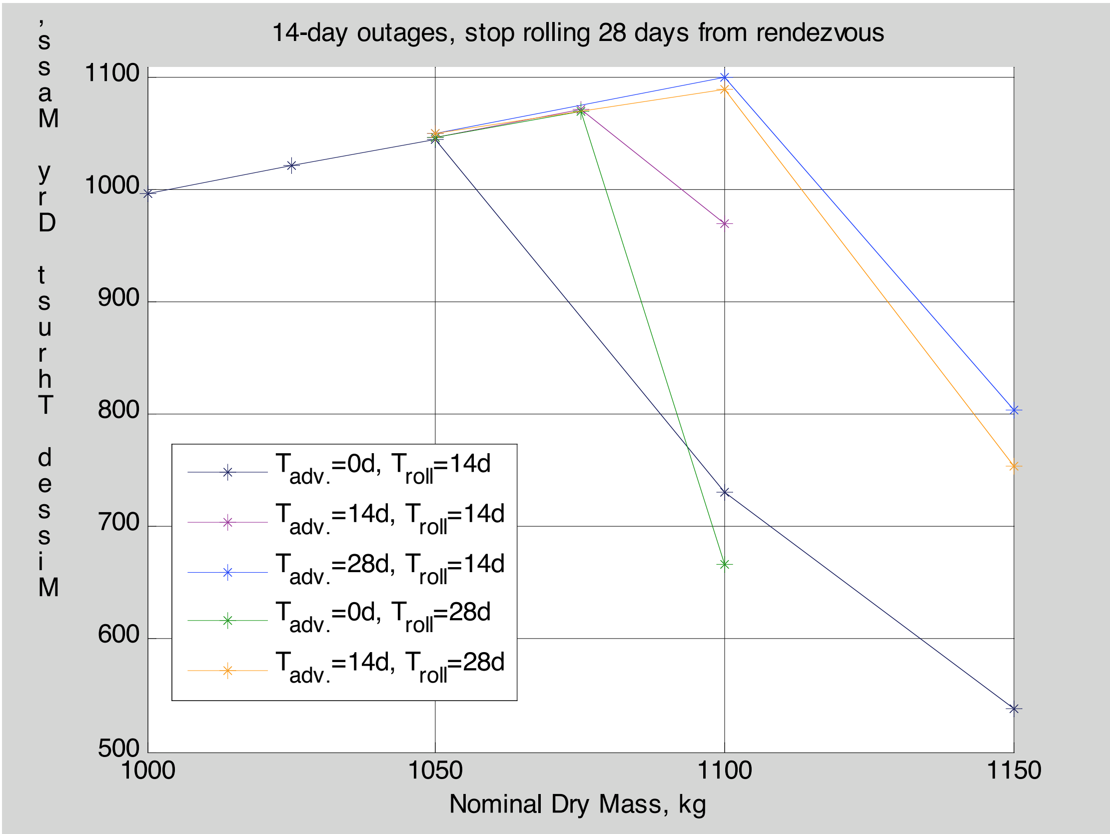

Figure 9: Sensitivity of mass with missed thrust to constrained nominal mass for comet rendezvous mission

3. Missed-thrust Database and Comparison to Nominal Mission Database

To understand the necessary margins, trajectories were derived using the missed-thrust assumptions and procedures in the previous section and compared to missions from the database of nominal missions to evaluate appropriate design margins for mass and power. In Table 4, an overview is presented of the trajectories evaluated that provide 7 to 28-day missed-thrust coverage at any point during the mission (100% coverage). These trajectories consider several durations of missed-thrust, power levels, and duty cycle to determine the impact on the original trajectory design. As shown in Table 4, trajectories that include both a loss of thrust and a higher duty cycle (92%-98%) than the baseline (90% from Table 2) typically result in lower delivery masses. (The mass delta in Table 3 is the difference between the baseline mass and the missed thrust mass.) There are two exceptions where the increased duty cycle (that is, 95% versus 90%) more than compensates for missed thrust period.

Table 3: Trajectories that Assume One 7 to 28-day Missed Thrust Period (100% coverage)

						Comparison of 14 Day Missed Thrust Trajectory with Baseline (see Table 2)	
Destination	**Thruster**	**Missed-Thrust**	**EP Power (kW)**	**Duty Cycle**	**# of Trajectories**	**14 Day MT 95% Duty Cycle Delivered Mass***	**Mass Δ From Baseline**
Near-Earth Asteroid Sample Return	BPT-4000	7/14/28	2.9-3.3	92-98%	22	766 kg	-165 kg
	NSTAR	7/14/28	2.5-2.9	92-98%	27	857 kg	-34 kg
	XIPS	7/14/28	2.3-2.8	92-98%	27	823 kg	-44 kg
	NEXT	7/14/28	2.6-3.0	92-98%	23	831 kg	-34 kg
Comet Rendezvous	BPT-4000	7/14/28	8.1-9.5	92-98%	22	958 kg	-70 kg
	NSTAR	7/14/28	9.5-11.0	92-98%	22	982 kg	**+15 kg**
	XIPS	7/14/28	9.5-11.0	92-98%	22	959 kg	**+8 kg**
	NEXT	7/14/28	10.5-12.0	92-98%	21	968 kg	-46 kg

*Power level for missed thrust mission is same power as the baseline mission (see Table 2)

The results in Table 3 show that the requirement to compensate for missed thrust periods can result in significant mass performance penalties, in some cases over 100 kg. The potential for such performance penalties show the importance of carrying out a detailed missed thrust analysis or carrying adequate margin to accommodate missed thrust cases. Where Table 4 presents trajectories that are robust to the loss of thrust at any point during the mission, Table 4 presents trajectories that are robust to missed thrust periods over 85% of the mission timeline. For the remaining 15% of the mission, it is the responsibility of the mission operations team to maintain the continuous operation of the EP thruster. This period of time is analogous to a "critical event" on a traditional chemical propulsion mission; it is a period of time when the spacecraft must maintain near continuous operation of the EP system to meet the mission's overall performance requirements. While the reality is that most missions will require neither 100% nor 85% coverage, the trajectories in Tables 10 and 11 bound the most likely range of assumptions.

Table 4: Trajectories that Assume One 7 to 28-day Missed-thrust period (85% coverage)

						Comparison of 14 Day Missed Thrust Trajectory with Baseline (see Table 2)	
Destination	**Thruster**	**Missed-Thrust**	**EP Power (kW)**	**Duty Cycle**	**# of Trajectories**	**14 Day MT 95% Duty Cycle Delivered Mass***	**Mass Δ From Baseline**
Near-Earth Asteroid Sample Return	BPT-4000	7/14/28	2.9-3.3	92-98%	22	881 kg	-50 kg
	XIPS	7/14/28	2.3-2.8	92-98%	27	847 kg	-20 kg
Comet Rendezvous	BPT-4000	7/14/28	8.1-9.5	92-98%	22	963 kg	-65 kg
	NSTAR	7/14/28	9.5-11.0	92-98%	22	1,002 kg	**+35 kg**
	XIPS	7/14/28	9.5-11.0	92-98%	22	962 kg	**+11 kg**
	NEXT	7/14/28	10.5-12.0	92-98%	21	973 kg	-41 kg

*Power level for missed thrust mission is same power as the baseline mission (see Table 2)

4. Missed-thrust Results and Analysis

A detailed missed thrust analysis is a relatively complex, and early in the mission development process it is often not practical to conduct a missed thrust analysis for each trajectory under consideration. Instead, it is desirable to carry some margin (duty cycle, power and/or propellant) to compensate for lack of a detailed missed-thrust analysis. By comparing trajectories from the missed-thrust database with trajectories from the nominal mission database, it is possible to calculate the amount of power, duty cycle, and flight time margin that is equivalent to a given level missed thrust duration and robustness. This section describes the results of an analysis examining the correlation between missed thrust robustness and equivalent system margins.

For each missed thrust trajectory, the mass delta between the equivalent non-missed thrust baseline mission and the missed thrust mission was calculated. In cases where the mass delta was negative (a missed thrust mass penalty), the nominal mission database was examined to identify a trajectory with the same mass as the missed thrust trajectory, but at a lower duty cycle. The difference between the missed thrust duty cycle and the nominal duty cycle provides a measure of the margin required to make up for the mass penalty. While duty cycle is the chosen parameter to examine missed thrust margin, increased power and/or propellant consumption can also improve neutral mass (see Figure 10). In some cases, both additional power and propellant are required to compensate for the missed thrust periods. The results are non-uniform and must be treated statistically, as some trajectories require relatively large power/propellant margins to offset the missed thrust mass penalty, and others require no margin at all. As stated previously, flight time was treated as a free parameter and was allowed to vary within approximately 5% of the original flight time.

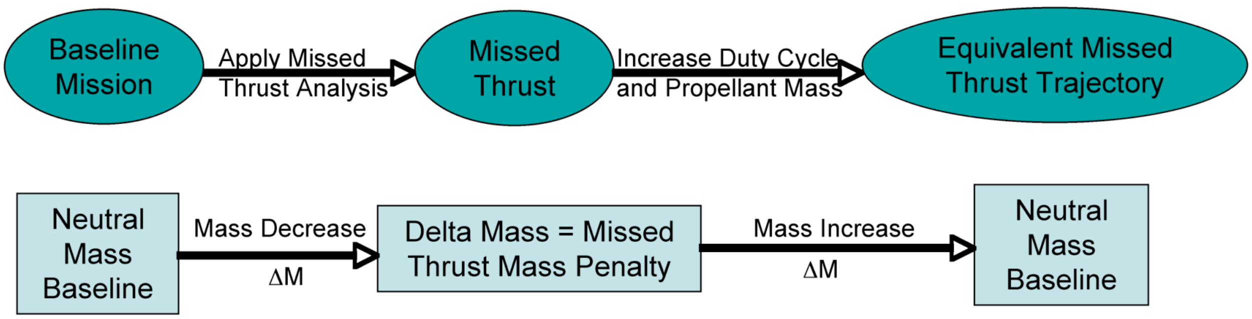

Figure 10: Increased Duty Cycle and/or Propellant Margin Can Compensate for Missed Thrust Periods

Figure 11 shows data from 300 missed-thrust trajectories plotted as cumulative distribution function of the duty cycle required to compensate for the missed-thrust penalty. In other words, it shows the duty cycle margin calculated for each of the missed-thrust trajectories to ensure that the desired baseline mass can be delivered (see Figure 10). Propellant and/or flight time increases were also calculated for each of these trajectories, and must be imposed in addition to the duty cycle margin to deliver the desired baseline mass. The plot below shows the probability of a given duty cycle margin meeting the mass requirements of all of the trajectories for that case. When 100% is reached, it means that that level of duty cycle margin ensures that sufficient mass is delivered for all of the missed-thrust trajectories.

Each line shown in Figure 11 is based on approximately 50 trajectories, varying destination, thruster type, power level, and actual duty cycle during operations. Based on these results, the dotted red line shows that a 10% duty cycle margin is required to ensure the trajectory is robust either to seven-day outages at any point during the mission or fourteen-day outages over 85% of the mission.

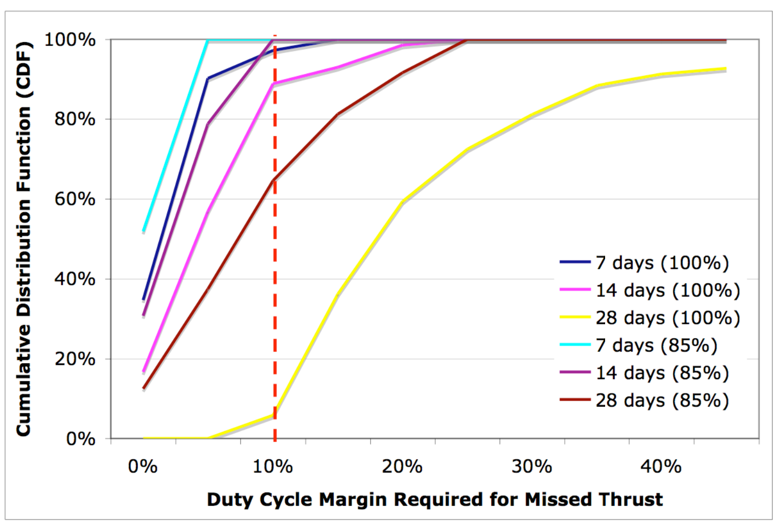

Figure 11: Duty Cycle and Power Missed-thrust Margin

Similarly, Figure 12 shows the propellant margin required to compensate for different types of thrust outages. For example, the first dashed red line shows that a 4% propellant margin ensures robustness to 14-day outages at any point during the mission (and also to less stressing cases). The second red line shows that a 20% propellant margin ensures robustness to 28-day outages during 85% of the mission.

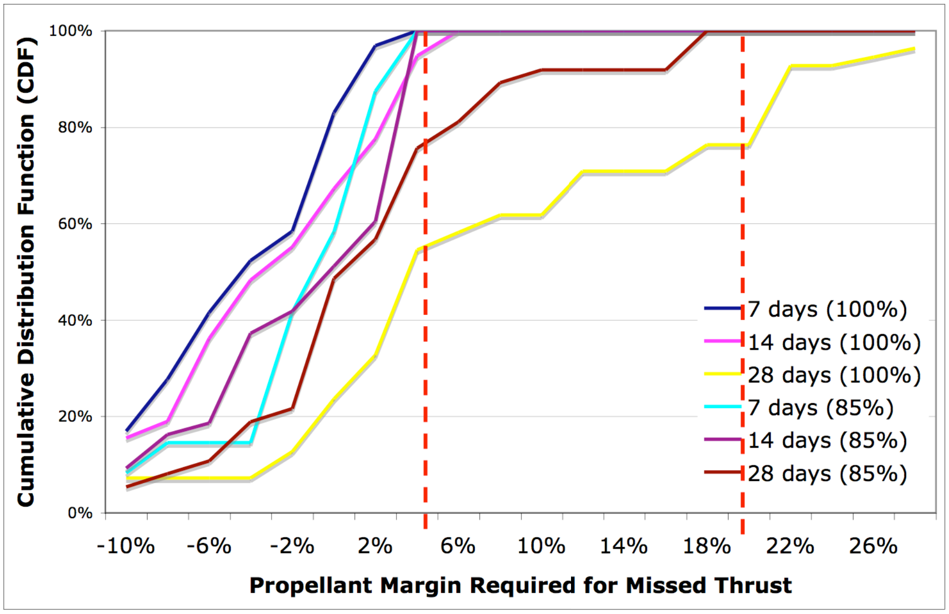

Figure 12: Propellant Missed-thrust Margin

For reference, Figure 13 shows the same plot as a function of flight time. Flight time variations of up to 10%, but generally 5% or less, were seen in the baseline mission equivalent missed thrust trajectories.

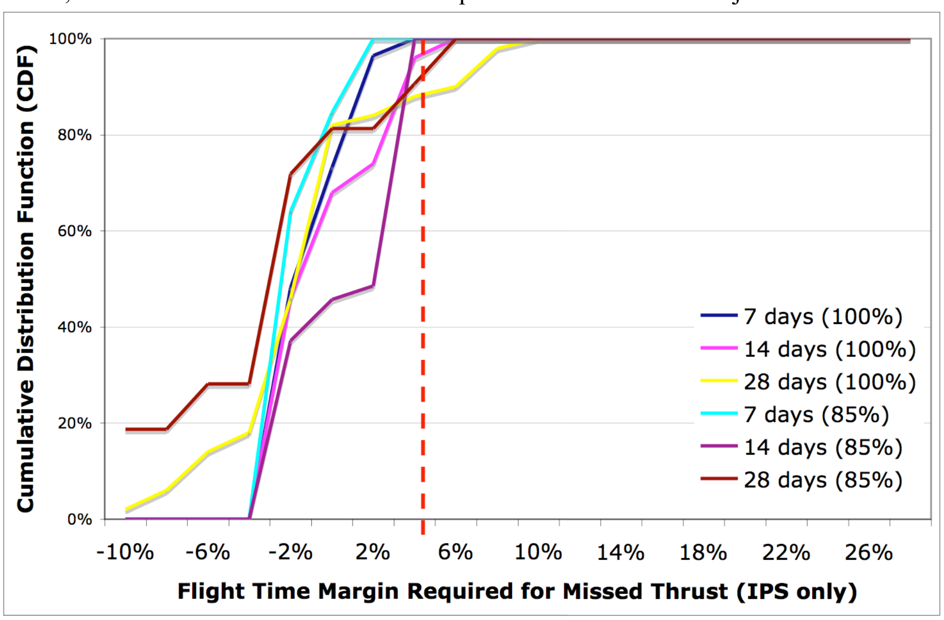

Figure 13: Flight Time Missed-thrust Margin

A summary of these results is shown in Table 5, which lists the system margins needed to compensate for missed thrust vs. the number of days of lost thrust and the percentage of mission coverage. Each of these margins is based on more than 40 different trajectories (including six destination-thruster combinations). The detailed results show high variances across the different combinations. Therefore, while these results should provide an initial guideline, additional mission-specific analysis is needed early in the project lifecycle to ensure adequate mission performance.

Table 5: System Margins Required to Compensate for Missed Thrust

EP Margin	85% Coverage			100% Coverage		
	7 days	14 days	28 days	7 days	14 days	28 days
Power or Duty Cycle (%)	5%	**10%**	24%	13%	20%	> 50%
Propellant (%)	4%	**4%**	18%	4%	6%	30%
Flight Time (%)	2%	4%	6%	4%	6%	9%

Given these results, the boldface values (14 days at 85% coverage) seem like reasonable, minimum margins that would ensure robustness to loss of thrust for at least 14 days throughout 85% of the thrust timeline. Based on our previous finding that power and duty cycle margin are equivalent, the power/duty margin can be split between the power and duty cycle margin budgets. Again, it should be noted that flight time is treated as a separate independent parameter that is allowed to vary freely within a limited range (~5%). This approach is sufficient for missions where there is a weak dependence between arrival time and overall mission performance, but missions where there are very strict arrival time requirements may need to actively manage flight time margin as well as the other margin elements.

C. Duty Cycle Margins

Thruster Duty Cycle is expressed as a fraction of total thrust time and includes all factors that effect duty cycle or thrust level. During trajectory development, when details of the mission operations plan are not well understood, duty cycle encompasses a variety of planned and unplanned non-operational periods including:

- Deep Space Network communications and tracking passes
- Spacecraft maintenance and check-outs (when not part of optimal coast periods)
- Fault Protection and Contingency responses resulting in loss of thrust

In addition, duty cycle can encompass the following additional factors:

- Thrust magnitude uncertainties
- Thrust vector pointing uncertainties
- Solar radiation pressure
- Operational errors
- Trajectory execution errors

Duty cycle does not encompass the pre-planned, optimal coast periods that are required for trajectory optimization purposes. As the spacecraft configuration and mission operations plan mature, many of the factors initially included in duty cycle can be calculated directly and, in some cases, removed from the duty cycle budget and modeled using monte-carlo methods or as part of the trajectory plan.

It should be noted that thruster duty cycle is generally modeled in trajectory optimizers by simulating continuous operation at a lower than nominal thrust level. A 90% duty cycle, for example, is often modeled as a system operating continuously at only 90% of its nominal thrust level.

The throttle profiles used in the trajectory optimizers should be used to model operation at nominal (CBE) thrust levels and uncertainties in thrust magnitude can be incorporated into the duty cycle. Duty cycle is defined as the fraction of time the system is generates usable thrust during a thrust arc, and duty cycle margin as the remaining time, or:

$$\text{Duty Cycle} = \frac{\text{Useful operating time}}{\text{Thrust Arc Duration}} \times 100$$

$$\text{Duty Cycle Margin} = \left[1 - \left(\frac{\text{Useful operating time}}{\text{Thrust Arc Duration}}\right)\right] \times 100$$

Based on this definition, typical values for duty cycle margins are shown in Table 10.

Table 6: Typical Thruster Duty Cycle Margins

Duty Cycle Margin	Typical Values	Comments
Planned Shutdowns		
Post-launch checkout and testing*	Mission Specific	Typically modeled directly, not included in margin
Encounters / Eclipses	Mission Specific	Typically modeled directly, not included in margin
Communications and Tracking	5%	1 weekly DSN pass
Uploads, regular maintenance	0%-4%	Depends on # of coast periods
Fault Detection and Recovery	5%	Also assumes 5% power margin (see Table 5)
Inefficiencies		
Thrust Underperformance	0%	Accommodated in power and propellant margins (see Figure 3)
Pointing Error	Small	Minimal impact
Plume Impingement	Small	Minimal impact
Environment Effects	Small	Minimal impact
Trajectory Execution Errors	Small	Minimal impact
Total Duty Cycle Margin	**10% to 14%**	

For reference, Table 7 shows duty cycle values used on the Dawn mission. The following sections describe each of the items in the duty cycle budget in detail.

1. Planned shutdowns

Planned shutdowns are divided into two categories: deterministic items that should generally be modeled using forced coast periods and recurring shutdowns that can sometimes be modeled as a reductions in duty cycle.

- Post-launch checkout and test can be modeled as a forced coast period corresponding to the planned checkout period for the mission. This period is mission specific, though the Dawn Mission allowed a 40 day nominal coast period with an additional 40 days of forced coast for contingencies (a total of 80 days). Shorter periods could be appropriate based on mission specific requirements.
- Encounters and eclipses are typically modeled deterministically by the low thrust optimizer used for mission design. Eclipses are relatively rare during cruise for deep space missions.
- Communications and tracking time can be included in duty cycle margin. A 5% duty cycle allocation corresponds to one 8-hour DSN pass per week in cruise. This element of duty cycle margin should be calculated based on the actual operations plan.
- Regular maintenance periods for software uploads and other non-thrusting operations consist of periods of several days occurring relatively infrequently during cruise. Often, these maintenance operations can be incorporated into existing optimal coast arc periods, so they have no impact on duty cycle during thrusting. When optimal coast arc periods of 7 to 20 days are incorporated into the baseline trajectory, the duty cycle margin allocated to this item can be 0%, but if optimal coast arcs occur rarely (fewer than once per year), additional duty cycle margin should be applied, typically between 1% and 4%, to account for maintenance periods.

Table 7: Duty Cycle Assumptions on the Dawn Mission

	Dawn
Planned Shutdowns	
Post-launch checkout and testing	• 40 days forced coast for s/c testing •An additional 40 days of forced coast to resolve any issues detected during checkout (total of 80 days)
Eclipses	N/A
Communications and Tracking	• One DSN pass per week in cruise (95% duty cycle) • Number of DSN passes rises to 2, 3, or 4 times per week on approach to the target bodies, bringing the duty cycle as low as 75% during these periods.
Upload, regular maintenance	• At least 7 days of coast period every 180 days for spacecraft maintenance, including thruster calibration. • Additional periods of 7 to 18 days for flight software uploads. These periods are spaced out by a year or more.
Fault Detection and Recovery	• Up to 28 days of unplanned thrust outage at nearly any point in the trajectory. Includes up to 21 days of “rolling coast” period in the planning to accommodate this. • Rolling coast: under normal conditions, spacecraft will thrust through this period. If there is an anomaly, spacecraft can shut down for 21 days and still make its nominal target

2. Fault Detection and Recovery

While the previous section described the trajectory analysis to account for missed-thrust, this section considers the flight system capability with respect to fault detection and recovery. In the event of an EP or flight system fault, there are several possible outcomes, including entering safe mode and/or loss of thrust operation. In this context, faults that do not result in the loss of EP functionality are ignored, as they do not directly impact the trajectory design. When a fault occurs that leads to the loss of thrust, this impacts the mission performance. In this context, it is important to both detect the fault and recover efficiently to reduce the performance impact. This can be accomplished either autonomously onboard or with operator intervention. Each of these methods involves different design and cost assumptions. Given weekly DSN passes, it is assumed that the fault would be identified within seven days. Assuming a second seven days period to recover from the fault, this is consistent with a 14-day minimum requirement for missed-thrust robustness, or 10% power/duty cycle margin and 4% propellant margin. As a comparison, Dawn is planning to support up to 28 days of unplanned thrust outage, using 21 days of “rolling coast” period to accommodate this (see Table 7).

It should be noted that the relationship between missed-thrust duration and the relevant margins is mission dependent. Long duration, rendezvous only missions (like Dawn) are much less sensitive to long outages than short duration, sample return missions. There is an inherent tradeoff between acceptable days off and the level of fault protection available on the spacecraft. The shorter the acceptable outage time, the higher the scope and cost of fault protection.

3. Pointing Error

Pointing error accounts for the total impact that thrust vector pointing errors due to gimbal errors, alignment errors, spacecraft pointing errors, and other sources have on the thrust vector. For three axis stabilized spacecraft, these errors are relatively small. The thrust penalty is approximately proportional to the cosine of the error in the pointing angle. For a 1.5 degree (three sigma) error, the cosine loss is only 0.034%, which is negligible compared to other elements in the duty cycle budget.

4. Environmental Effects

For spacecraft operating in deep space, the dominant interaction is solar radiation pressure. The effects of radiation pressure are generally very small, and can usually be neglected.

5. Trajectory Execution Errors

For spacecraft using electric propulsion on deep space missions, variations in thruster on and off times on the order of minutes generally have a negligible impact on the trajectory. The impact of such variations is small compared to other elements in the duty cycle budget, so these errors can generally be neglected.

D. Power Margins

The electrical power used for electric propulsion and spacecraft loads is generated by solar panels in the electrical power subsystem (EPS). The EPS is responsible for power generation, distribution, regulation, and energy storage.

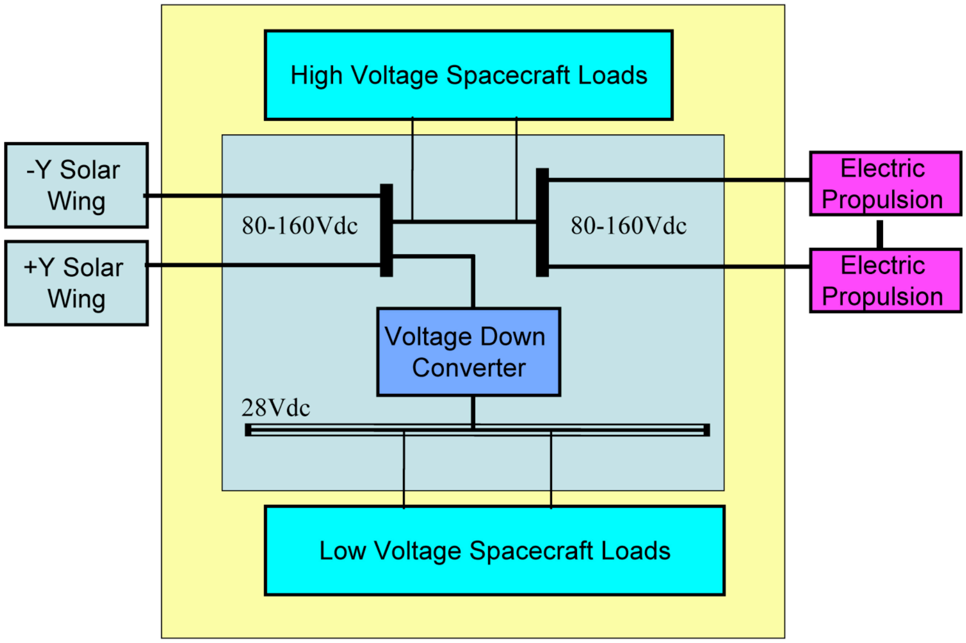

Figure 14: Generic EPS Block Diagram

As the single largest electrical load, the EP power processing unit, (PPU) input power requirements drive the design of the EPS. The EP PPU power input typically operates at a relatively high voltage. JPL has historically used a variable input PPU voltage (80-140Vdc). The remaining loads are operated off one of the spacecraft electrical buses. With the power source on the high-voltage bus, the EPS must include a down-converter to power all spacecraft loads on the low-voltage bus. Energy storage for periods without solar power is provided by a battery system, which can be attached directly to the low voltage spacecraft bus or via a boost regulator to the high voltage bus.

The EPS loads are composed of spacecraft loads and EP loads. The EPS has to carry margins against the sum of both loads. The total load requirement must also include any losses associated with distribution and conversion losses. The solar array capability at beginning of life (BOL) must be adjusted to reflect environmentally induced losses and range effects. In the process of doing trajectory analysis, the loads and source capability are balanced against EP throttle setting, to produce a trajectory. When a trajectory can be achieved, with all margins and losses included, a mission design is successful. A summary diagram of the overall process is provided as Figure 15, shown below.

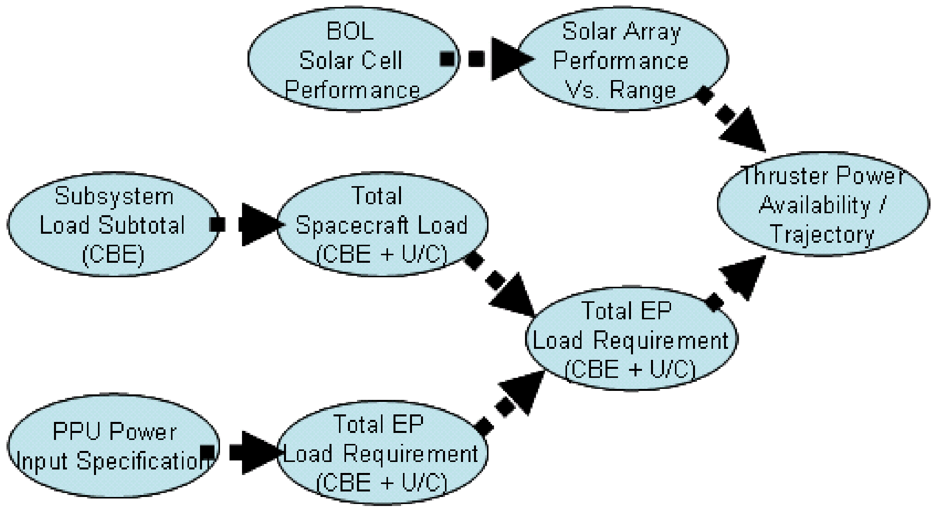

Figure 15: Overall EPS Margins Flowchart

6. EPS Loads

Total loads include both EP loads and Spacecraft loads. The spacecraft loads are roughly constant with solar range. Due to throttling of the EP system to obtain maximum thrust, EP loads vary with solar array power availability and solar range. The EP loads are the majority loads at near 1AU operations, with the ratio of EP loads to spacecraft loads often 10:1. At extreme solar distances, it is possible for EP loads to become roughly as small as spacecraft loads. Standard spacecraft load margins apply to the spacecraft loads. They dictate that the sum of all CBE loads, plus appropriate uncertainty against those values be applied throughout the development cycle. Down conversion losses also apply to loads on the low-voltage spacecraft bus. Distribution losses (ohmic losses in the harness) must also be applied to all loads. Typical distribution losses are 4% for low voltage loads and 1%-2% for high voltage loads. **Figure 16** provides a diagram of the spacecraft and EP load margin process.

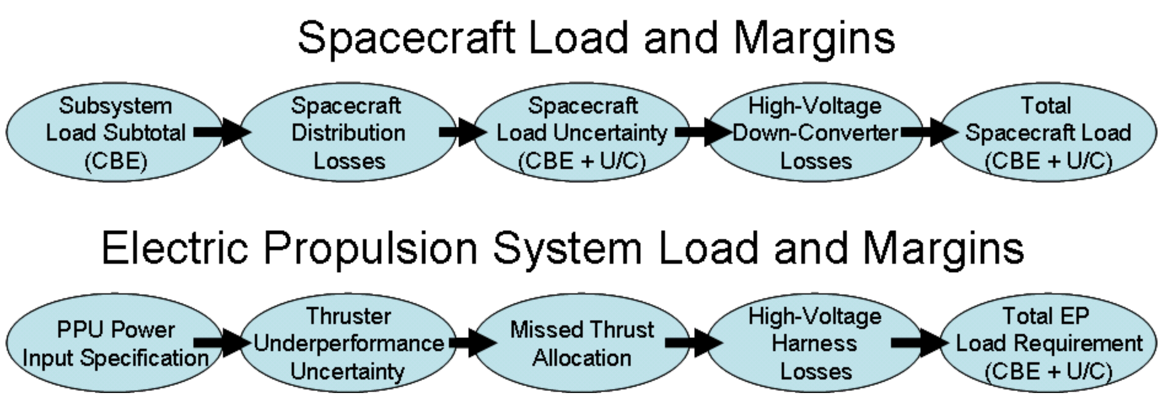

Figure 16: Overall EPS Margins Flowchart

Down-conversion Losses

A characteristic of the generic EPS architecture shown in Figure 14 is the presence of down-conversion losses associated with making 28V power from the high voltage bus via the High Voltage Down-Converter. These losses generally scale with spacecraft power consumption. A typical value for down-converter efficiency is 92%, but this is a peak value, and varies with loading and topology.

For EPS architectures where the battery is on the low voltage bus, the down conversion losses would apply to any mission phase operated off the solar array, but not those operated off battery, such as a solar occultation. This nominal 8% increase in spacecraft loads does not typically occur in non-SEP missions, and needs to be added to the sum of the spacecraft loads

7. Power Source Margins

The discussion of power source margins in this section focuses on the solar arrays, the primary power source for missions utilizing solar electric propulsion. Figure 17 provides an overview of the solar array margins and losses process.

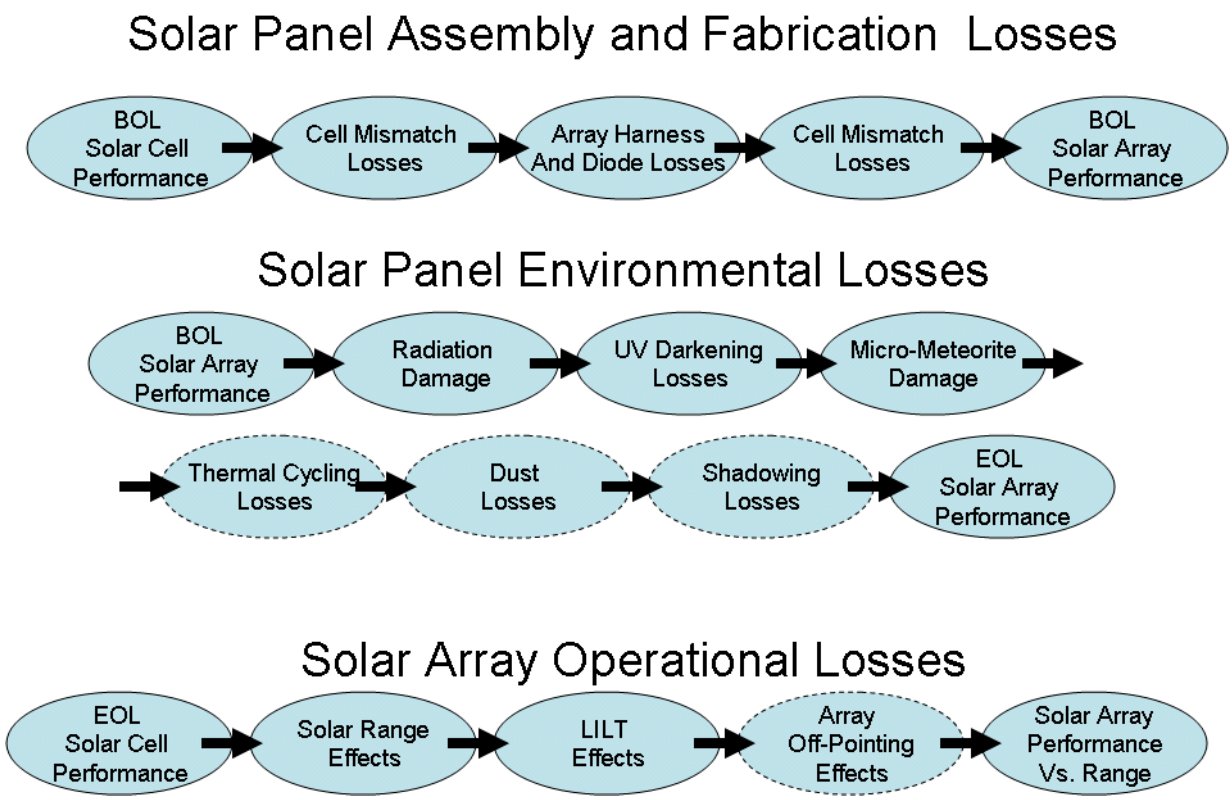

Figure 17***: Solar Array Margins and Losses***

Solar Array Margins

The solar array's power production must be large enough to support the spacecraft loads through all mission phases, including a closure of the trajectory analysis with appropriate margins. Solar array performance estimates must include fabrication, assembly, and operation losses, as well as uncertainty and margin. The apportionment of losses and uncertainties vary by mission, and a discussion of typical loss factors is included below..

Solar Array Losses

Solar array output losses can be broken up into four categories: 1) fabrication and assembly losses, 2) operational losses encounter due to environments, radiation being the predominant factor, 3) LILT effects (low-intensity, low temperature), and 4) solar range effects.

Solar array performance loss models are constructed starting from cell-level performance, which is scaled up according to the array design and the assembly and fabrication losses to give array level power at BOL. Array level power at EOL is calculated by adding the environment induced loses, such as radiation and UV darkening of the coverglass adhesive. The array power can be expressed at EOL array power at 1AU, AM0 (air mass zero) and 28°C, (standard temperature). For a SEP mission, this EOL, 1AU, AM0, 28°C value is input into the trajectory analysis program. LILT and range effects are then added during the iterative thrusting analysis.

Cell Level Environmental Testing

Environmental loss testing is done at the cell or cell-interconnect-coverglass unit (CIC) level. These CIC tests typically include radiation exposure and low-intensity low-temperature (LILT) testing. These cell level performance effects can then be scaled up to predict panel level performance. Array level performance includes a large number of factors that must be combined analytically, as no single environmental test can simulate the flight conditions.

Fabrication Losses

Fabrication losses are the first loss contributing to BOL panel level performance. Mismatch between cells in a series string, and between series strings drops the array level performance by approximately 4% over bare cell levels. The harness losses add another 4% to the losses at the panel level. This 4% voltage drop associated the array harness typically includes the diode drops, as they are most often found physically on the array itself. Alternatively, it may be a separate dissipation value associated with a different piece of hardware, such as a diode assembly. In the case of Dawn, where heritage considerations forced the adoption of slip ring assemblies, the diodes were split between the array and the fuse board in the High Voltage Down-converter assembly. These losses apply to all solar panels, and are not highly variable across missions, except for the architecture specific bookkeeping noted above.

Operational Losses

Operational losses are mission specific, and thus are much more variable. The operational losses included in derating panel performance include: radiation effects on cells, LILT effects, array shadowing losses (if applicable), dust losses (if applicable), micro-meteorite damage losses, and thermal cycling losses. The radiation and LILT losses are the largest factors to be considered. Dust loss, micro-meteorite damage, thermal cycling loss, and array shadowing are often very small terms for SEP missions, and are ignored in this simplified treatment.

Radiation Losses

Radiation causes damage to solar cells, with efficiency losses occurring randomly as solar flares damage the cells. Because ionizing radiation intensity is reduced as solar range increases, varying strongly as $1/R^2$ and $1/R^3$, it is sufficient to book keep all radiation damage as though it occurred at the beginning of the mission. For this study, all radiation losses were booked at beginning of life (BOL). This is a conservative approach that is easily modeled in trajectory optimization tools, but may not be appropriate for some types of SEP missions. Examples of such mission include:

a. A spacecraft that spends significant time in radiation belts late in the mission, such as a SEP mission to Jupiter

b. A sample return mission, where power to begin the journey home could require an intermediate total integrated radiation dose (TID) estimate to support the return leg

c. Spiral out through the predictable MEO trapped radiation belts.

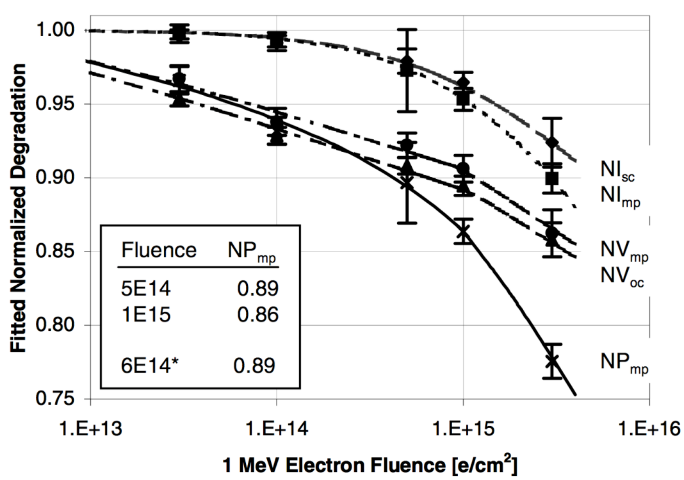

Figure 18: Triple-junction Cell Response to TID in 1 MeV Electron Fluence[9]

An added complexity for the solar panel is that damage coefficients are expressed in 1 MeV equivalent electrons, unlike electronic parts, which use Rads. Figure 18 shows the effect that radiation has on triple junction cell performance as a function of total radiation dose, expressed as 1 MeV equivalent electrons.

Solar Range Effects and LILT Losses

As a spacecraft travels away from the sun, the solar radiation falling incident on solar panels decreases according the inverse square of the distance, $1/R^2$, where R is solar distance in AU. This affects the solar panel performance in a several important of ways. Beyond the simple $1/R^2$ reduction in incident photons and the resulting decrease in cell current, the panels operate at colder temperatures at greater solar range. The colder cells operate at higher voltage, changing the operating point of the array, as shown in Figure 19. For a peak power tracking system, this is a benefit that can be beneficial to array power at range.

The low-intensity light creates also an adverse effect on solar cell performance, analogous to dark current in CCD's, or leakage current in MOSFETs. The parasitic currents present in the cells junctions are most often swamped by cell current at 1AU where cells are typically designed, tested and operated. The leakage only becomes appreciable when the intensity level falls, making cell current drop off faster then $1/R^2$ would predict. These two effects are typically combined into the power vs. range curve coefficients used for trajectory analysis purposes.

The modeled results shown Figure 19 were developed using a radiative heat transfer model and a simple linear temperature coefficient for solar array cell voltage. This is a slightly more detailed approach than used in trajectory modeling.

During the detailed design phase, analysis of solar array power at critical points of the mission timeline is required. These might include full throttle operation near earth, partial throttle operation at several ranges, including maximum range, and power available during encounters or other mission specific operational milestones. During this analysis, it is good to look at any potential thermal mismatch across the various panels off the solar array. Variation in temperature would result in some panels operating off-peak, and some small additional operational losses.

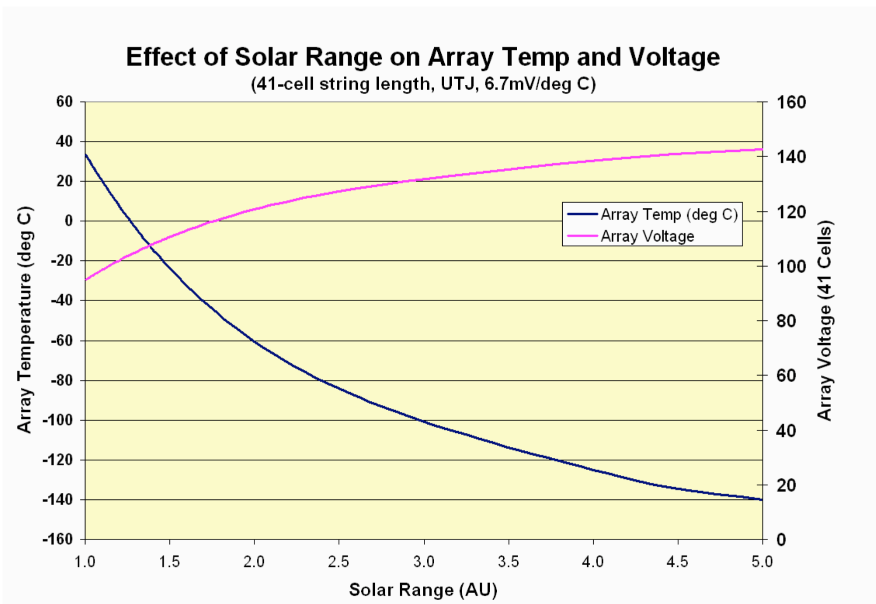

Figure 19: Model of Solar Array Operating Point versus Range

Mission Specific Cell Screening

An issue of solar array performance matching surfaced during the Dawn program. Solar cells are typically flow and tested at 1AU conditions. But like any semiconductor device, there are potential leakage currents at junctions. The variability of leakage currents does not typically manifest itself in detectable cell level variations at 1AU with 135 mW/cm^2 of illumination. When taken out to LILT conditions, a lot of cells may show significant variation in the shape of the IV curve, creating unexpected cell matching issues in the array. These variations would invalidate the margins approach by creating array performance shortfalls versus pre-launch predictions. To avoid this problem, cell level screening under mission-like LILT conditions should be imposed by the cell/array vendor.

8. Power Margins

A SEP mission presents an unusual set of circumstances, where the source in declining at a rate that is uncertain, primarily though predictable range effects and unpredictable solar flare induced radiation damage events. The EP system largely makes use of the power available to thrust as often as practical, with unscheduled interruptions providing uncertainty in the load and trajectory. To deal with the uncertainties in this analysis, a standardized bounding case is used in this study. It is assumed that the spacecraft is operating in a cruise thrusting mode with full throttle settings at 1 AU solar range, with EOL panel degradation. This EOL panel degradation accounts for the possibility that a solar flare event will occur shortly after launch. This is a conservative approach that may not be appropriate for all mission scenarios.

The sum of the expected loss and uncertainty in the loss is subtracted from the reference design value, which is fixed at 1.00 for BOL. The final product off the successive losses plus uncertainties is the reference design for EOL. The meaning of the 0.815 EOL reference design factor, is that the array would produce 81.5% of BOL power under the same reference conditions, (1AU, AM0, 28°C).

- **Power Margins Summary**

Power margins should be kept separately for Electric Propulsion and Spacecraft power. Spacecraft power and solar array margins are mission and vendor specific, but should account for the various degradation factors discussed above. EP power is based on trajectory uncertainty (prior to running a missed-thrust analysis), PPU dissipation, and thruster performance, as shown in the following table.

Table 8: Power Margins

	Margin	Rationale
EP Power		
Uncertainty	5% of thruster power	Based on missed-thrust analysis
PPU Dissipation	2% of PPU power	
Thruster Performance	3% of thruster power	

E. Propellant Margins

1. Thruster Performance

Propellant margins required for thruster underperformance are dependent on the thruster type and overall methodology for booking margins. Given the methodology assumed for this analysis, that power and trajectory (thrust) underperformance are equivalent, only a specific impulse underperformance needs to be considered. For example, if there is a thrust underperformance, but the specific impulse is unaffected, only the power/trajectory margin is affected as more thrust time or more thrust power will be required. This combination of power/thrust margins eliminates the largest component of electric thruster performance uncertainty, the delivered beam current uncertainties resulting from tolerances in power component specifications. Such beam current uncertainties create errors that effectively balance each other out when the power and thrust margins are combine. For example, a lower than expected beam current will result in lower than expect thrust; but also a lower than expected power draw. This results in no impact to the margin budget for linear dependencies between beam current and thrust. Other power processing issues, such as greater than expected PPU inefficiencies, should be clearly booked in the power margin, not the propellant margin.

Hall thrusters and ion thrusters have different propellant margin requirements due to fundamental differences in the way the propellant flow is regulated and the way in which the propellant is ionized and accelerated. Ion thrusters have a separate ionization and acceleration stage. This allows for very accurate performance control; but propellant must be controlled separately and excess delivered propellant is considered wasted. Hall thrusters have a combine ionization and acceleration stage. This provides poor performance control; but propellant can be controlled by close loop control with the discharge current so no separate propellant control is required.

Specific Impulse uncertainties that affect propellant margins can be broken down into three primary categories; digital throttling uncertainties, mass flow rate uncertainties, and thrust uncertainties. The digital throttling error is the uncertainty between discrete throttling points and the desired analog performance curve used by trajectory optimizers. With closed loop flow control, hall thrusters provide a continuous throttling curve over the entire power range; thus this error is negligible. For ion thrusters, the flow control system requires discretized throttling points with an uncertainty with respect to the continuous throttling curve of about 2%. The mass flow rate error for hall thrusters is zero due to the closed loop flow control and 3% for ion thruster due to uncertainties in the delivered mass flow rate for existing propellant feed systems. Due to precise control of beam acceleration, the ion thruster thrust uncertainty is negligible. Due to discharge and acceleration loss uncertainties in hall thrusters, the acceleration error is typically close to 4%. The root sum square of these uncertainties for both ion and Hall thrusters is approximately 4%; thus, given the linear dependency of propellant uncertainties on the specific impulse, a propellant uncertainty of 4% accounts for thruster uncertainties.

2. Commissioning & Thruster Starts

The thruster commissioning propellant consists of the propellant used before all thrusters are ready to begin their standard mission activities. These commissioning activities consist primarily of cathode conditioning and thruster calibration. Cathode conditioning typically consists of a four hour period where the cathode(s) (one for a hall thruster, two for an ion thruster) are heated with propellant flow to drive off pre launch contamination; but usually no propellant flow is required. Typically only one such cycle is necessary; however, if there are problems starting the cathode later in life, it is likely that the conditioning sequence may be repeated. Thruster calibration consists of operating the thruster long enough to calibrate the engine performance in space. This activity typically consists of a single 20 hour firing where the thruster is operated at a few points over the entire throttling range. Propellant used over the commissioning sequence is highly thruster dependent. Typical values consumed for each thruster of interest are shown in Table 20.

Table 9: Propellant Consumed for Commissioning and Start-ups per Individual Thruster

Mode	BPT (kg)	NEXT (kg)	NSTAR (kg)	XIPS (kg)
Conditioning (1 cycle)	0.000	0.00	0.000	0.000
Calibration (1 cycle)	1.464	0.371	0.276	0.307
Start Up (500 cycles)	0.170	0.172	3.834	0.142
Turn Off (500 cycles)	0.006	0.009	0.009	0.009

Propellant must also be budgeted for thruster start up and turn off events that occur over the course of mission life (Table 20). During thruster start ups, propellant flow must be initiated for a period of between several seconds or a few minutes (depending on the specific thruster/flow system configuration) to establish stable flow before the thruster is started. Note that small attitude disturbances will occur in this period of time as the thrust from the flow alone will be on the order of a few percent of nominal thrust. Once the thruster is turned off, additional propellant, downstream of the thruster solenoid valves, is lost when the burn is terminated. The number of such on/off cycles is mission dependent; but typical values range around 100 cycles per year.

3. Leakage

The leakage budget consists of the xenon lost through external valve leakage and internal valve leakage that is lost to space without being used by a thruster. Calculation of the contribution of external leakage is straightforward. The equivalent xenon mass flow from the maximum external leakage of each flow component is multiplied by the mission life and then summed over all the contributing components (Table 18). The contribution of internal leakage, which dominates the total leakage contribution, is much more complicated and dependent on the specifics of the mission. This analysis makes the following assumptions about the internal leak rate contribution: the internal leakage to space is dominated by the downstream solenoid valves (not the upstream latch valves), the internal leak rate has been assumed to be less than the typical specification maximum of 3 scc/hr, and all the solenoid valves leak over the entire mission life. With these assumptions, the total internal leakage to space is summarized in Table 21.

Table 10: Contribution of Xenon Leakage from Each Flow Component (six downstream flow paths assumed equivalent to 3 Hall thrusters or 2 ion thrusters)

Component	Volumetric Rate (scc/s)	Number	Years	Total (kg)
Latch Valve, External	10^{-6}	2	10	0.0006
Regulator, External	10^{-6}	2	10	0.0006
Fill/Drain Valve, External	10^{-5}	2	10	0.0060
Throttling Valve, External	10^{-6}	6	10	0.0018
Solenoid Valve, External	10^{-6}	6	10	0.0018
Solenoid Valve, Internal	10^{-4}	6	10	0.1807

4. Fill Error

The propellant fill error is the uncertainty of xenon loaded into the spacecraft during launch base activities. Although there are usually PVT (pressure/volume/temperature) and mass flow rate book keeping checks, the most accurate measurement of propellant loading is the change in spacecraft mass measured by a scale. The scale uncertainty is typically driven by the scale resolution which is normally about ± 0.25 kg; thus 0.25 kg should be booked for the fill error.

5. Residuals

Residuals are comprised of xenon mass remaining in the tank and flow system that are unusable at the end of EP system life. These residuals can be calculated from the volume of the flow system and the xenon density at which the pressure is insufficient to allow adequate xenon flow for proper thruster operation. Such residuals are usually proportional to the xenon tank volume; because the xenon tank volume comprises most of the flow system volume. The minimum pressure, used to define the residual xenon density, can vary depending on the method of xenon extraction. With conventional xenon flow regulation systems, the minimum pressure required to insure stable flow operation is between 50 to 100 psi (3.45 to 6.89 bar). The temperature at end of life typically ranges between the xenon critical temperature (17º C) and about 60º C. The resulting density range at end of life is from about 16.6 g/l (3.45 bar at 60º C) to 39.1 g/l (6.89 bar at 17º C) [reference NIST software].

To define the residual load as a proportion of maximum xenon tank capability, the maximum xenon density must be defined. This maximum loaded xenon density is a function of the maximum rated tank pressure at the maximum tank temperature. Maximum conventional xenon tank pressures range from about 1310 psi to 2700 psi (90 bar to 186 bar). Within the 17 º C to 60 º C temperature range, this results in maximum xenon tank densities of between 748 g/l (90 bar at 60º C) and 2155 g/l (186 bar at 17º C). As a result, reasonable estimates for xenon residuals as a function of maximum xenon load range from 0.8% to 5.2%; but typical values are around 1.5%.

For propellant budgeting prior to spacecraft launch, an EP system specific calculation of unusable propellant must be made from the uncertainty and residual sources mentioned above. For preliminary analysis a sum total of 6% should be adequate to account for propellant residuals and uncertainties.

IV.O Conclusion

Recognizing that there are numerous deep space EP missions either proposed or in development, NASA's Jet Propulsion Laboratory has conducted a study focused on the analysis of appropriate margins for deep space missions using solar electric propulsion (SEP). The purpose of this study is to understand the links between disparate system margins (power, mass, thermal, etc.) and their impact on overall mission performance and robustness. The results show the important, interlocking relationship between subsystems on SEP missions. This study is part of an effort that will ultimately define JPL's standards for the evaluation of margins on proposed SEP missions.

In this study, we examined the interactions between system margins by:

- Identifying the sources of uncertainty and relevant parameters for EP systems
- Defining a framework for evaluating each parameter and establishing the governing interactions between parameters
- Evaluating the sensitivity of parameters and defining typical values

In the course of this study, it was determined that the various sources of uncertainty and risk associated with electric propulsion mission design can be summarized into three relatively independent parameters:

- **EP Power Margin**
- **Propellant Margin**
- **Duty Cycle Margin**

The overall relationship between these parameters and other major sources of uncertainty is shown in Figure 3. Note that in this study, flight time was treated as a separate independent parameter that is allowed to vary freely within a limited range (~5%). No effort was made to define the allowable flight time "margin" or to impose a strict limit on the flight time. This approach is sufficient for missions where there is a weak dependence between arrival time and overall mission performance. Missions where there are very strict arrival time requirements (for example, a Mars Sample return vehicle that is targeting a rendezvous with a lander delivered using chemical propulsion) would need to actively manage flight time margin as well as the other three margin elements.

It is recommended that trajectory analysis be carried out using thruster performance curves based on nominal performance, not worst case performance, and that adequate margin be incorporated into the three elements above to accommodate the worst case performance case. This philosophy was used to calculate the typical and recommended margin values given below.

A detailed trajectory analysis was conducted to examine the sensitivity of the trajectory to various assumptions related to power, duty cycle, destination, and thruster performance. Based on this analysis, it was shown that:

- In many cases, duty cycle margin and power margin have equivalent impacts on delivered mass and are therefore interchangeable.

One parameter of interest to mission designers is the sensitivity of the trajectory to missed thrust periods. Missed thrust periods can occur due to spacecraft faults or planning errors. A detailed analysis was also conducted of the impact that missed thrust periods have on overall performance. It was found that:

- The impacts of missed thrust periods can be significant (over 100 kg in some cases), and it is important to account for missed thrust when developing deep space low thrust missions.

Missed thrust periods can be accounted for through detailed analysis, or by carrying enough power, propellant, and duty cycle margin to compensate for likely missed thrust periods. By comparing trajectories from a missed-thrust database with trajectories from a nominal mission database, we calculated the amount of power, duty cycle, and flight time margin that is equivalent to a given level missed thrust duration and robustness. These margins for missed-thrust were included in the overall margins shown in Table 11.

Table 11: Recommended System Margins for Deep Space Missions using Solar Electric Propulsion*

SEP Technical Parameters	Margins	Comments
Neutral Mass Allocation	-	Following initial trajectory optimization, mass allocation is considered fixed without additional SEP-related margin (Figure 3). Mass budget should carry standard margins against this allocation.
EP Power	10%	EP power margin should be maintained *separately* from non-SEP power margin. Includes missed-thrust, thrust performance, and harness losses. (Table 8)
Propellant	9-13%	Includes missed-thrust, thrust performance, commissioning and thruster starts, leakage and fill error, and residuals
Thrust	0%	Thrust performance uncertainty accounted for by increased power and propellant margins.
Duty Cycle	10-14%	Recommended duty cycle range is 86-95%, which includes missed-thrust, uploads & maintenance, and communications & tracking. (Table 6.)
Flight Time	-	In this study, flight time was allowed to vary as much as 5%, independently of other parameters.

*Note 1: Margins shown in this table include accommodation of missed-thrust periods. A detailed missed thrust analysis can replace several subelements, resulting in lower system margins (see Tables 5,6,8)

**Note 2: Margins shown in this table assume margin subelements are directly summed. In some cases, it may be acceptable to RSS subelements, resulting in lower overall system margins.

Table 11 provides a summary of the margins presented in this paper, which are primarily reflected through power, propellant, and duty margins. The margins shown here are generally appropriate before a detailed missed-thrust analysis is complete. Once a detailed missed-thrust analysis is complete, the missed-thrust components of the trajectory analysis may be relaxed, reducing the margins shown in Table 11.

This study addressed an array of system-level interactions using data from actual trajectories, mission studies, and flight projects. While this data set is not inclusive of all destinations or assumptions, it presents a set of information that can be used as a reference in the development of system margins for most Deep Space missions using Solar Electric Propulsion.

Appendix

Figures that will not be included in the final paper, but may be used in the presentation. Please approve for public release.

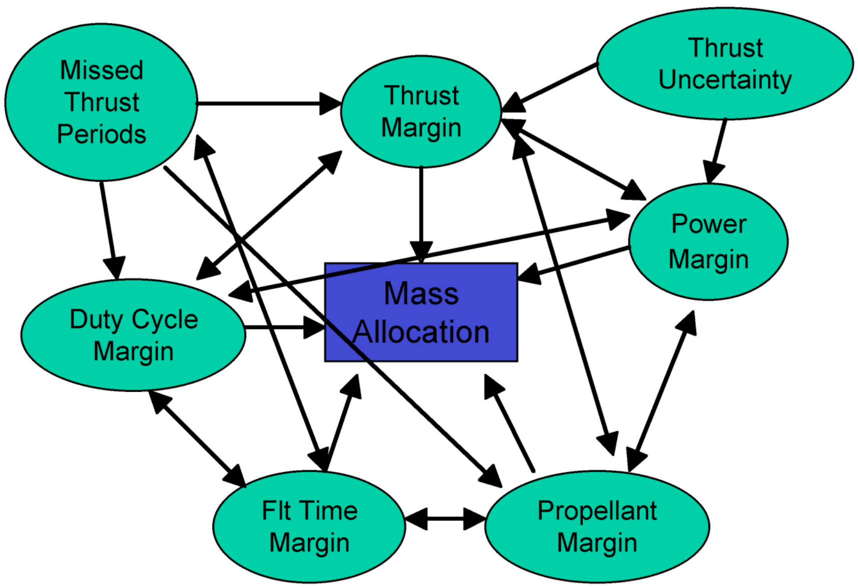

Table showing allocation of error elements to three major margin elements

Duty Cycle Margin	EP Power Margin	Propellant Margin
	Thrust Magnitude Uncertainty	
Missed Thrust Periods		
Communications and Tracking Periods	PPU Dissipation	Commissioning and Startups
Spacecraft Maintenance and check-outs		Leakage
Fault Protection and Contingency Responses		Fill Errors
Thrust Vector Pointing Uncertainty		Residuals
Environmental Effects		Propellant Use Uncertainties
Operational Errors		
Trajectory Execution Errors		
Plume Impingement		

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