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Design of the First Interplanetary Solar Electric Propulsion Mission

Marc D. Rayman* and Steven N. Williams†
Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California 91109

Deep Space 1 was the first interplanetary mission to be propelled by solar electric propulsion. The detailed design, development, and analysis of its trajectory have led to important new insights into the design of low-thrust trajectories. Tying the testing of solar electric propulsion technology to an operational mission has allowed the identification of trajectory design issues that were not considered in concept studies, such as constraints on the spacecraft attitude and periods of thrusting or coasting that are dictated by reasons other than trajectory considerations. Models of the spacecraft performance unimportant for trajectory analysis in missions using conventional chemical propulsion are intimately connected with the design of the trajectory when solar electric propulsion is employed. In addition, mass margin is not sufficient to assess a low-thrust mission. Unplanned thrust interruptions may result in a situation in which the propulsion system cannot provide sufficient impulse to compensate for the lost thrust in time to reach encounter targets. Mission margin (as distinct from mass margin) is a quantification of the mission's susceptibility to loss of thrust and is an important indicator of mission robustness for low-thrust trajectories.

Nomenclature

A_n	=	curve-fit coefficients for time-independent solar array power
B_n	=	curve-fit coefficients for time-dependent solar array power
C_3	=	square of hyperbolic excess velocity, km^2/s^2
D_n	=	curve-fit coefficient for spacecraft power
K_n	=	curve-fit coefficients for ion propulsion system thrust
L_n	=	curve-fit coefficients for ion propulsion system xenon mass-flow rate
$\dot{m}$	=	ion propulsion system xenon mass-flow rate, mg/s
P_{IPS}	=	power available to the ion propulsion system, kW
$P_{\mathrm{S/C}}$	=	power required by spacecraft (not available to the ion propulsion system), kW
$P_{\mathrm{SA}}(r)$	=	time-independent solar array power, kW
$P_{\mathrm{SA}}(r, t)$	=	time-dependent solar array power, kW
P_0	=	solar array power at 1 astronomical unit (AU) at the beginning of the mission, kW
r	=	heliocentric range, AU
T	=	ion propulsion system thrust magnitude, mN
t	=	time from launch, day
Δv	=	change in velocity, km/s

Introduction

IN 1994, NASA formed the New Millennium Program (NMP) to help reduce the cost and risk of its ambitious space and Earth science programs. The NMP conducts missions designed to evaluate high-risk advanced technologies so that subsequent flight projects will be spared the normal cost and risk connected with being the first user of a new technology. The principal criteria for selecting a technology for flight testing are that it should be considered high risk to the first user, that the reduction of the risk should require testing to occur in space, and that the new capability offered by the technology should be a significant advancement over lower risk alternatives.

Deep Space 1 (DS1) was the first mission of the NMP. Its payload comprised 12 technologies, and its mission was to exercise these systems rigorously. Rayman et al.[1] describe the mission, technologies, and results of the testing. One of the technologies was solar electric propulsion (SEP), implemented on DS1 as an ion propulsion system (IPS).[1,2] SEP is of great interest to mission designers for future missions, primarily those requiring large changes in orbital energy.[3–5] Indeed, the inclusion of SEP on the first flight of the NMP was among the highest priorities of the program's science advisory team because of the many important missions this technology enables.

The formalized primary objectives of testing SEP on DS1 were to 1) demonstrate flight operations using this technology and quantify its performance, 2) assess its interaction with the spacecraft, scientific instruments, and space environment, and 3) use it to propel the DS1 spacecraft onto a trajectory that would encounter an asteroid during the 11-month primary mission. The first two objectives were focused principally on the hardware and the operability of the entire system. The third was included to force a complete system-level validation of the capability of SEP to achieve a targeted interplanetary flight. One aspect of the challenge in such an undertaking was the trajectory design at a level of detail sufficient to accomplish the flight.

Although DS1 was not intended as a science mission, the methods developed and used to analyze the mission performance are directly relevant to future science missions that will use SEP. Before the DS1 development, many interplanetary missions using this technology had been studied, but always at the level of a conceptual design. In designing a trajectory for actual flight, many new detailed issues were discovered that distinguish low-thrust trajectories from the better studied and, thus, more familiar ballistic ones. For convenience, missions that rely on chemical propulsion are referred to here as ballistic missions. Although such missions may have large deterministic maneuvers or gravity-assist flybys, most of the time their trajectories can be well described by ballistic arcs, in contrast to missions that rely on SEP. Throughout the detailed mission design work, it was clear that design principles, rules, tools, and intuition applicable to conventional missions are not necessarily relevant, and may even be misleading, when applied to SEP missions. One of the significant results of the testing of the SEP technology on this flight project is the experience gained in designing, analyzing, and flying an interplanetary low-thrust trajectory. The purpose of this paper is

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*Project Manager and Chief Mission Engineer, Deep Space 1, 4800 Oak Grove Drive; mrayman@jpl.nasa.gov.

†Senior Engineer, Navigation and Mission Design Section, 4800 Oak Grove Drive; Steven.N.Williams@jpl.nasa.gov. Member AIAA.

to document the new insights and new techniques derived from this work and to aid future designers of trajectories that use SEP.

Mission Overview

During most of the development phases of DS1, the trajectory that was planned began with a launch in July 1998. The schedule was explicitly very ambitious, in part to keep the overall project cost low as well as to force the creation of new, faster methods of development. In addition to the requirements of testing technologies, the spacecraft would have had the opportunity to collect bonus science data during encounters with asteroid 3352 McAuliffe, Mars, and comet 76P/West–Kohoutek–Ikemura in two years of flight.[6] The trajectory was studied in great detail in preparation for flying this mission.

In April 1998 it was decided that the spacecraft would not be ready for launch in July, and the scheduled launch was moved to October, taking a slot vacated by another mission that was postponed to the subsequent year. The new launch period was defined exclusively by the availability of launch facilities during an unusually dense period of activity at the Cape Canaveral Air Force Station and the Kennedy Space Center. Requirements on the new trajectory were that it allow an asteroid to be reached within 11 months of launch but no earlier than 6 months after launch (to allow the technology testing, which was of higher priority, to be completed first) and that it require no changes in the spacecraft or launch vehicle, which had already been prepared for the mission described earlier. This second requirement imposed a significant constraint on the launch, because it forced spacecraft separation from the launch vehicle to occur in Earth's umbra with enough time before exiting the shadow for the spacecraft to stabilize and turn itself so that the transparent cover over the camera would not admit sunlight into the sensitive optics. (The cover had been selected for the earlier mission in which the separation happened to occur in the shadow. With a transparent cover, the failure of the deployment mechanism would not have completely precluded the testing of the camera technologies.) Constraining separation to occur in the shadow led to an extremely narrow range of right ascensions of the launch asymptote and, thus, greatly limited potential destinations. Furthermore, periods of ballistic coasting had to be included in the trajectory design to accommodate technology experiments incompatible with IPS thrusting. Other constraints on the launch, including the use of a single value for C_3 and a single value for the declination of the launch asymptote throughout the launch period, were agreed to as simplifications and, thus, cost-saving measures for the launch service and were acceptable because of the flexibility afforded by the IPS. To provide the opportunity for bonus science data acquisition at the asteroid, there were specific constraints on the approach phase angle (to allow imaging) as well as the estimated minimum time before closest approach that the body would first be detectable with the onboard camera. That time interval was a function of the estimated size and albedo of the body, the approach phase, and the relative speed. Finally, there was a desire to include a comet as a possible target if an extended mission were approved; it would have had to be reachable with the 81.5 kg of xenon propellant for the IPS that could be loaded for launch.

The trajectory used by DS1 was obtained by solving a constrained mass-optimization problem, and in this paper, an optimal trajectory is one for which propellant consumption has been minimized. Constraints included encounters with several targets, time intervals during which thrusting was not permitted, and time intervals during which thrusting in a prescribed direction and magnitude were required. Free parameters in the optimization included the time of each encounter, the thrust vector as a function of time, and the timing of coasting arcs. Therefore, some of the coasting during the mission was the result of enforcing a constraint, whereas at other times coasting was optimal.

The mission that was selected for DS1 included several periods of coasting (some optimal and some imposed) before attaining a trajectory that would encounter 9969 Braille (known at that time as 1992 KD) in July 1999. The potential mission extension included an encounter with comet 19P/Borrelly in September 2001. A plot of the trajectory that was flown is shown in Fig. 1. Periods of deterministic thrusting appear as heavy solid lines, and coasting periods are dashed lines. For clarity, the orbit of Braille is not shown.

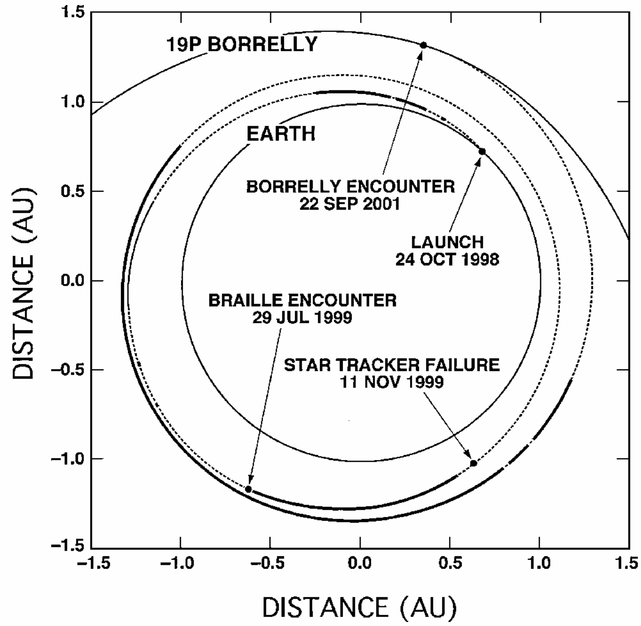

Fig. 1 DS1 trajectory.

The primary mission of DS1 began with launch on the first Delta 7326-9.5 from the Cape Canaveral Air Force Station's Space Launch Complex-17A at 12:08:00.502 universal time, coordinated (UTC) on 24 October 1998. DS1 comprised a 373.7-kg dry spacecraft, 31.1 kg of hydrazine, and 81.5 kg of xenon. (The sum of the mass of the dry spacecraft and the mass of the hydrazine is defined to be the neutral mass. The neutral mass and the xenon mass are the critical masses for trajectory analysis.) The launch vehicle delivered the spacecraft to a trajectory with a $C_3 = 2.99\ \mathrm{km^2/s^2}$. In fact, the Delta 7326 could have imparted significantly more energy to a payload of DS1's mass, but the additional performance was unnecessary for this mission. Therefore, the launch vehicle carried a secondary payload, SEDSAT-1, which was attached to the second stage and was delivered to its targeted low Earth orbit.

Technology testing began immediately after launch and continued through July 1999. Some basic spacecraft verification activities and technology tests had to be conducted before the IPS was operated. The new solar concentrator arrays, for example, had to be characterized so that achievable IPS throttle levels could be determined. The mission design included a period of coasting at the beginning of the mission as well as later coasting periods for technology testing that would have been incompatible with the optimal thrust attitude (e.g., for long-duration telecommunications tests, which required the unarticulated high-gain antenna to be pointed to Earth) or the presence of the xenon plasma (for the initial tests of an ion and electron spectrometer, although later tests showed that operation of the spectrometer with the IPS thrusting was acceptable).

In part to protect the mission from unexpected loss of thrust time, the trajectory was designed to include about 90 days of coasting before the encounter with Braille, although other technology experiments that required short intervals of IPS thrusting during that time had been planned and already were accounted for in the trajectory design. Some of that thrusting was in specially defined attitudes, for example, along the Earth–spacecraft line to measure Doppler shifts as part of the IPS testing. The need to execute nonoptimal thrusting arose on many occasions for a variety of reasons during the mission, and the capability to account for that in the overall trajectory optimization proved important. Some trajectory correction maneuvers were executed during the coasting period to the asteroid, using either the IPS or the hydrazine-based reaction control system (RCS). The RCS maneuvers were used as a part of the testing of the spacecraft's experimental autonomous optical navigation system[7] and to save time during the final 2 days before the encounter. At closest approach to Braille, at 04:46 UTC on 29 July 1999, the spacecraft passed 28 km from the center of the asteroid at a relative speed of 15.5 km/s.

Once the spacecraft was in-flight and key subsystem performances were quantified, a second target was added to the optional extended mission. An encounter with comet 107P/Wilson–Harrington in January 2001 was shown to be achievable with acceptable margins. Thrusting to accomplish the encounters with the two comets resumed 1.5 days after the encounter with Braille, although the extended mission had not yet been approved. The thrusting was constrained to occur with the high-gain antenna pointed to Earth for 10 days to complete the return of data from the asteroid encounter and to conduct other spacecraft activities. Trajectory analysis had shown that thrusting in this attitude was better than coasting, as measured both by mass margin and mission margin (to be discussed).

Having met or exceeded all of its success criteria, the primary mission concluded on 18 September 1999, by which time NASA had approved the extended mission. In fact, on the chance that the extension would be approved, IPS thrusting through most of the primary mission had targeted not only Braille but also the comets.

A period of optimal coasting from late October to December 1999 was used for a variety of spacecraft calibration activities as well as the acquisition of infrared spectra of Mars. During this hiatus in thrusting, the spacecraft's sole star tracker, required for the attitude control system's celestial reference, failed. Previous analysis had shown that thrusting would have to resume by late January 2000 to retain both comet encounters, but the severity of the loss was too great to rescue the spacecraft that quickly. Indeed, the loss of the star tracker had been considered a mission-terminating failure; nevertheless, the operations team successfully completed a complex recovery described by Rayman and Varghese.[8] The target of the original extended mission was retained, and after the recovery was completed in June 2000, thrusting to comet Borrelly resumed. At closest approach, at 22:30 UTC on 22 September 2001, the spacecraft passed 2171 km from the comet's nucleus at a relative speed of 16.5 km/s.

Part of the recovery included using the unarticulated science camera as an attitude sensor in place of the star tracker. Except for brief periods during which the attitude control system (ACS) could rely on gyros, the camera had to remain pointed at an isolated bright star. This new attitude constraint substantially altered the trajectory design because, instead of an essentially continuously variable thrust direction, the spacecraft could thrust only in selected discrete inertially fixed directions. For the DS1 trajectory, the penalty for making this change was extremely small.

The unplanned long coasting period and the complex operations during the recovery consumed a significant amount of the spacecraft's hydrazine. By the end of June 2000, the remaining supply was too low for normal operations. The ACS could use the IPS to control two axes of spacecraft attitude whenever the IPS was thrusting; thus, subsequent hydrazine consumption was reduced by thrusting with the IPS for most of the remainder of the mission. When deterministic thrusting was not required, the IPS was operated near its lowest throttle level (a mode referred to here as impulse power) to give the ACS sufficient control authority but to cause minimal, although not negligible, disturbances to the trajectory. Despite the innovations devised to account for the aged and debilitated spacecraft, most of the fundamental trajectory design challenges remained the same and were accommodated using the techniques developed earlier in the mission.

SEP Models and Trajectory Software

One of the reasons that SEP mission design is unique is that, unlike ballistic trajectories, SEP trajectory optimization is coupled to the performance of the propulsion system and the availability of power on the spacecraft. For example, the propulsion system is designed to be throttled; that is, it operates over a range of input powers, such that the performance is a function of the power input to the propulsion system. With 525 W delivered to the IPS, the thrust is 19 mN and the specific impulse is 1900 s; with 2500 W, the corresponding values are 92 mN and 3100 s. As the spacecraft recedes from the sun, the power from the solar array is reduced. For DS1, the throttle levels are stored onboard the spacecraft as 112 discrete settings in a throttle table. Each setting has its own set of currents, voltages, and xenon flow rates to produce the desired conditions in the thruster. The accuracy of the trajectory design may be limited by the quality of the models of solar array power production, spacecraft power consumption, and propulsion system performance.

The trajectory design on DS1 used a computer code named SEPTOP.[9] This program has been used for many years to do preliminary design studies at the Jet Propulsion Laboratory (JPL). SEPTOP uses a calculus-of-variations approach to perform a constrained optimization. The program maximizes neutral mass (which is equivalent to minimizing propellant), optimizing encounter times and the thrust direction as a function of time. On DS1, SEPTOP was used for a variety of planning studies during the mission, as well as providing the starting conditions for a second program,[10] NAVTRAJ, that contains higher fidelity models of small forces (N-body gravitation and solar pressure) on the spacecraft. By the use of its superior models of solar system forces, NAVTRAJ makes adjustments to the thrust profile necessary to retarget the trajectory to the desired encounter bodies. NAVTRAJ provided trajectory solutions that were uploaded directly to the spacecraft without further processing. The two programs have essentially the same models of spacecraft power and propulsion systems. The primary difference is that SEPTOP designs an optimal trajectory and NAVTRAJ then retargets the trajectory using the higher fidelity small forces models. The combination of the successful navigation of the spacecraft and NAVTRAJ's inheritance from previously validated software at JPL allows NAVTRAJ to be considered the standard of accuracy against which to compare a SEPTOP trajectory.

It is significant that for this mission it was not necessary to include a high-fidelity solar system model in the design solution, as is typically done with ballistic trajectories. With SEP, the more accurate gravitational model is not as important at the optimization stage of design because small errors can be corrected by the retargeting program through small changes to the thrust profile all along the trajectory. Figure 2 shows a typical example of the corrections NAVTRAJ made to a SEPTOP solution. This example begins immediately after the Braille encounter when a Wilson–Harrington encounter was still planned. NAVTRAJ has the freedom to change the direction and duration of the thrust vector during periods of deterministic thrusting (referred to as thrust arcs). Most of the corrections are less than 0.5 deg, and even the largest changes of up to 2.6 deg are still small. The coasting arcs (or gaps) in Fig. 2 are essentially the same as determined by the two programs. The differences are generally only hours and never more than a few days. Also note that the total duration of thrusting was changed by only 1.2 out of 466 days. Moreover, for some trajectories, errors of only a few tens of watts in estimating power available to the IPS can produce larger propagation errors than the small errors introduced by the simpler gravitational model.

As the mission developed beyond the stage of a concept study, new requirements arose, which generally resulted in new capabilities being added to SEPTOP. One of the new requirements was the capability to make regular discrete adjustments to the neutral mass to simulate the expenditure of hydrazine. The ability to restrict the thrust direction with respect to the sun–spacecraft line was added to accommodate the requirement to protect sensitive body-mounted instruments and the need to point radiators away from the sun. In addition, it became necessary to require coasting for different periods during the mission, as well as to force thrusting in specific (nonoptimal) directions. Indeed, one of the ways DS1 maintained mission margin after the failure of the star tracker was by thrusting at high throttle levels during telecommunications whenever that put the spacecraft within about 30 deg of the optimal thrust attitude. This was shown to be more beneficial than thrusting at impulse power in such attitudes. (Recall that, to conserve hydrazine, coasting was not an option.)

There are a number of important spacecraft models and trajectory constraints used in the design of the DS1 trajectory. The solar array performance is a function both of distance from the sun and time. It is formulated as a time-independent part, as shown in Eq. (1), and a time-dependent part, as shown in Eq. (2):

$$P_{\mathrm{SA}}(r) = \left(\frac{P_0}{r^2}\right)\frac{A_1 + A_2/r + A_3\big/r^2}{1 + A_4 r + A_5 r^2} \tag{1}$$

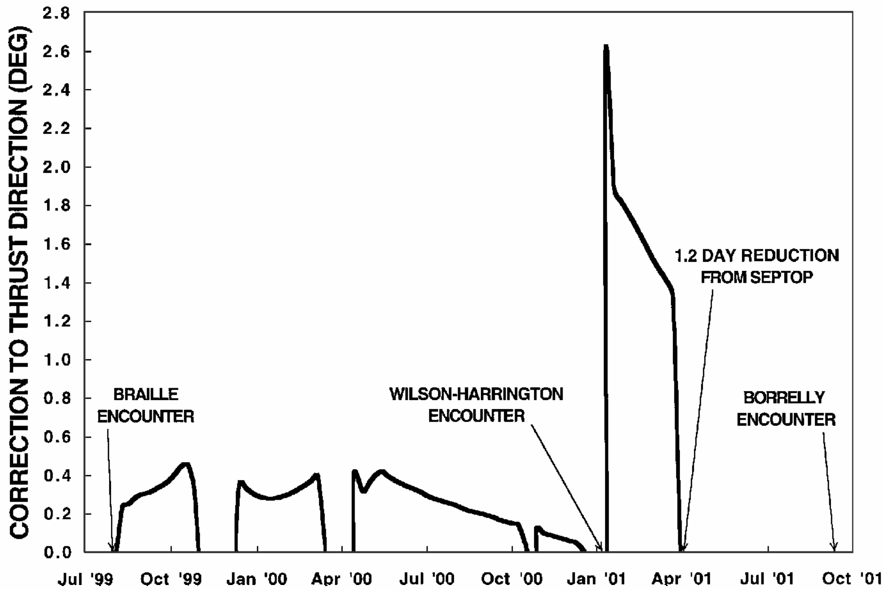

Fig. 2 NAVTRAJ corrections to SEPTOP after the Braille encounter.

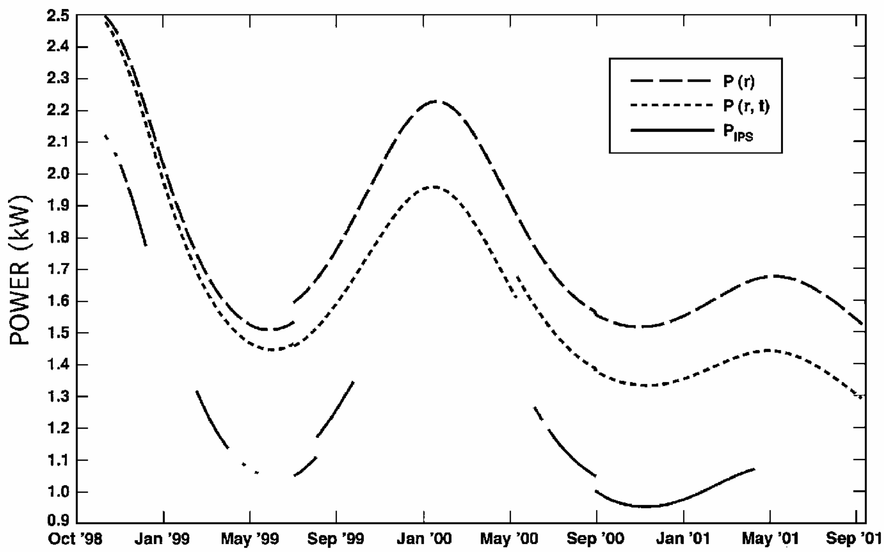

Fig. 3 Power models.

$$P_{\mathrm{SA}}(r,t) = P_{\mathrm{SA}}(r)\left[B_1 + B_2 e^{B_3 t} + B_4 t\right] \tag{2}$$

The solar array performance degrades with time because of phenomena such as the destructive effects of radiation on the semiconductor material of the solar cells and decreased optical transmission of concentrator lenses from solar ultraviolet light and contamination from spacecraft outgassing. The spacecraft power model represents power needed for all systems except IPS:

$$P_{\mathrm{S/C}} = D_1 + D_2/r^2 \tag{3}$$

Obtaining good estimates of this function for the DS1 spacecraft proved rather difficult, in part because of the complexity of the spacecraft power consumption's dependence on attitude (which affects the thermal conditions). Figure 3 shows how power changes as a function of time during the mission. In Fig. 3, the highest curve shows the time-independent solar array performance, which varies with heliocentric range. The middle curve includes the effect of time degradation on the array. As expected, these two curves diverge as the mission proceeds (time increases). Finally, spacecraft power consumption is subtracted from the array output to get power available to the IPS [Eq. (4)], which is shown in the lowest curve in Fig. 3:

$$P_{\mathrm{IPS}} = P_{\mathrm{SA}}(r,t) - P_{\mathrm{S/C}} \tag{4}$$

Because P_{IPS} is only important during thrusting, it is not computed during coasting phases, which accounts for the gaps in the lowest curve in Fig. 3. Note also the discontinuities in several places on these curves representing in-flight updates to these models.

The trajectory optimization software requires a model of the throttle table, expressed as the power dependence of thrust and mass-flow rate, as shown in Figs. 4a and 4b. Each discrete point in the throttle table is shown as a plus (+). These points would provide the most accurate description of propulsion system performance, though using discrete points was impractical because it greatly slowed software execution. Selected test cases, however, showed that approximating the table values with polynomials produced very similar results. The curves in Figs. 4a and 4b are linear regression fits to data in the throttle table using fourth-order polynomials:

$$T = K_1 P_{\mathrm{IPS}}^4 + K_2 P_{\mathrm{IPS}}^3 + K_3 P_{\mathrm{IPS}}^2 + K_4 P_{\mathrm{IPS}} + K_5 \tag{5}$$

$$\dot{m} = L_1 P_{\mathrm{IPS}}^4 + L_2 P_{\mathrm{IPS}}^3 + L_3 P_{\mathrm{IPS}}^2 + L_4 P_{\mathrm{IPS}} + L_5 \tag{6}$$

As the mission progressed, least-squares fits to the throttle table of the same functional form were used that weighted more heavily those throttle states that would be used the most. Because the remaining trajectory used smaller segments of the throttle table, the weighted fits, such as the ones in Figs. 4a and 4b, provided a good representation of only a portion of the table.

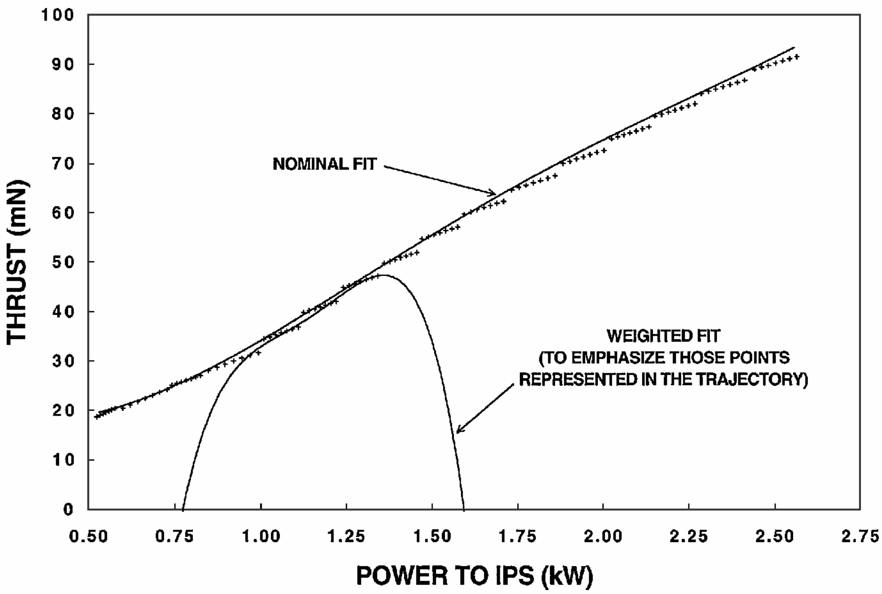

Fig. 4a Thrust model.

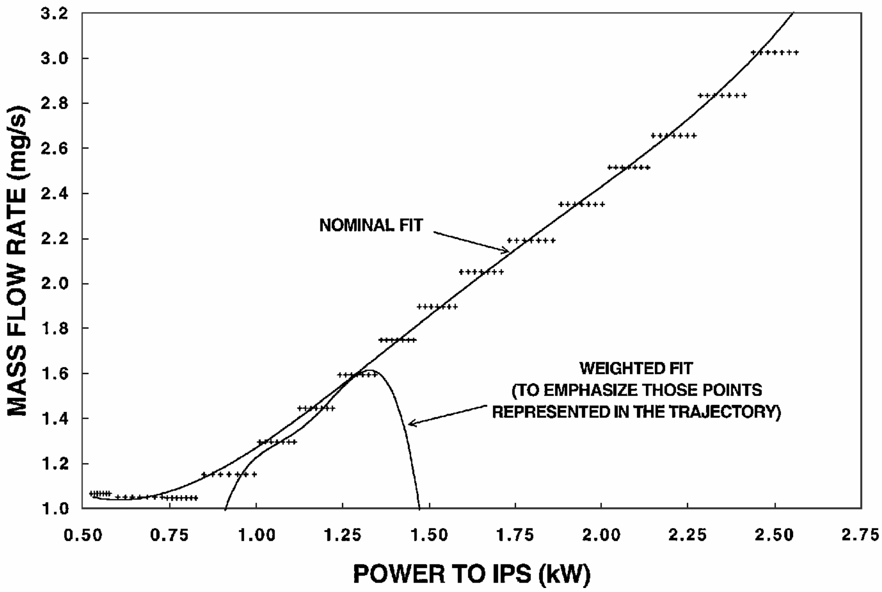

Fig. 4b Mass-flow rate model.

There are two other models that affect the trajectory: mass drops and duty cycle. As the trajectory is propagated into the future, it is important to have a reasonably accurate estimate of the total mass because this will affect the acceleration produced by the IPS. Hydrazine and small amounts of xenon are intermittently expended for attitude control and other purposes. This mass loss is modeled in the design software as periodic discrete reductions throughout the mission. Occasionally during the mission, more accurate hydrazine and xenon masses were deduced from spacecraft telemetry, confirming that our discrete estimates were maintaining values accurate to about 1 kg. The duty cycle is defined here to be the fraction of time during deterministic thrust periods that the IPS is thrusting. On DS1, the thruster was turned off for short periods (on the order of hours) each week, primarily for telecommunications and autonomous optical navigation data acquisition. Duty cycle is an approximation used to simplify the trajectory optimization approach. DS1 maintained a duty cycle of 92% before the loss of the star tracker. It would have been possible to model 13 h of coasting each week, but that would have been impractical because it would have required an unwieldy number of constraints for the optimizer. Instead, the appropriate thrust and mass-flow rate values were multiplied by 0.92 during the deterministic thrust arcs. In other words, to streamline the analysis, 100% of the thrust (and mass-flow rate) 92% of the time is approximated in the software as 92% of the thrust (and mass-flow rate) 100% of the time. This approach was validated by comparison with NAVTRAJ results; NAVTRAJ does use a specific coasting period each week. Figure 2 shows that the corrections imposed by NAVTRAJ are small.

Measuring Mission Margin

In conventional interplanetary missions, the most common way of measuring margin is with mass, or Δv (which is equivalent to mass through the rocket equation). For low-thrust missions, positive mass margin is necessary but not sufficient. Also, there are uncertainties in the mass margin due to uncertainties in the models, as described earlier. Figure 5 shows the DS1 mass margin sensitivity to various models shortly after launch, when the mass margin was about 9.4 kg. Note that the abscissa is different for each curve. Cone angle is defined to be the angle between the sun–spacecraft line and the IPS exhaust vector. The curve for the duration of the coasting arc before Braille illustrates an interesting feature: the duration of this coasting cannot be increased until the mass margin reaches zero. The baseline trajectory requires 90 days of preencounter coasting. When this is increased by an additional 66 days, the IPS can no longer provide enough impulse to reach Braille and satisfy the other constraints. For this case, the 9.4-kg mass margin clearly does not

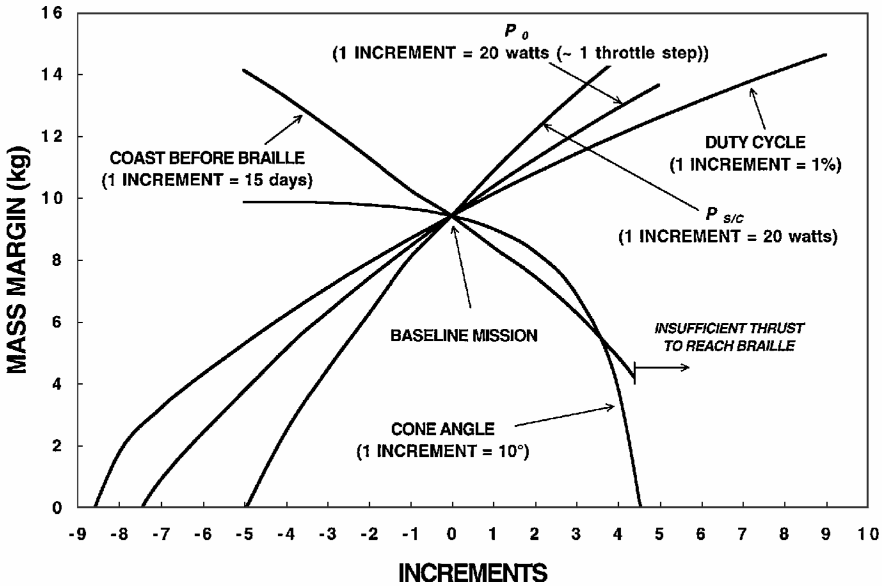

Fig. 5 Mass margin sensitivities.

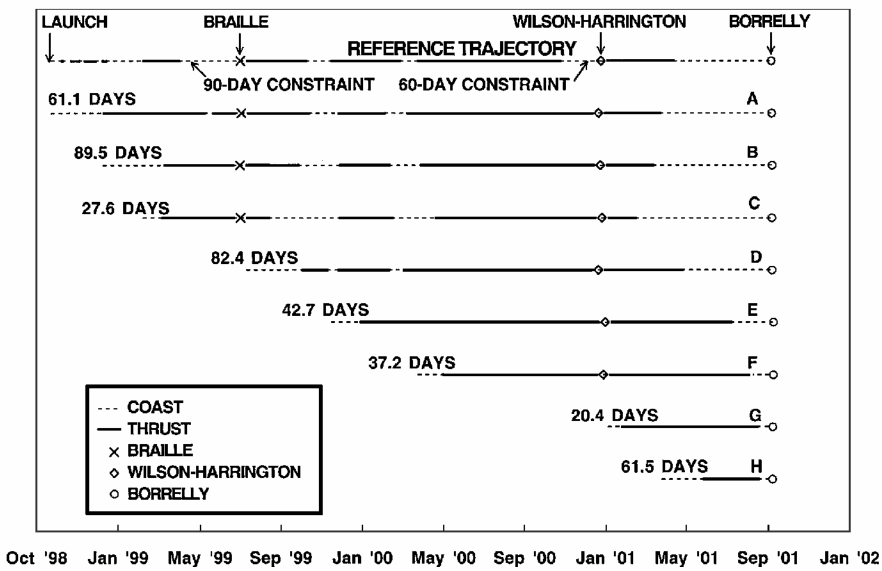

Fig. 6 Mission margin.

adequately describe the margin the mission has for reaching its target. This clearly demonstrates the need for a metric other than mass margin; the metric for the availability of thrust time is referred to here as mission margin. In general, the failure of mass margin to be a sufficient metric can occur if the nominal mission is interrupted for some reason and thrusting is missed. A new analysis is described here that provides a much better prediction of margin but is also more difficult and time consuming.

Figure 6 shows mission margin expressed as susceptibility to missed thrust at various times in the mission. Each line represents a timeline of thrusting and coasting. A reference trajectory is shown schematically at the top of the plot as a series of thrusting (solid lines) and coasting (dashed lines) phases. Note that the reference mission included a nonoptimal coasting period of 90 days before Braille and 60 days before Wilson–Harrington. These constraints could be relaxed in an emergency. The coasting before Borrelly is optimal. Each of the trajectories A–H begins with a state (position, velocity, and mass) from the reference trajectory. An unplanned interruption of thrusting at the initial point on each trajectory timeline is assumed, and then the question is posed: "How long can the spacecraft coast and still accomplish the planned encounters if the 90- and 60-day constraints are relaxed to 10 days?" When the less stringent constraints are used before each flyby, a new optimum trajectory is then found, making the initial coasting arc as long as possible. In each case, the length of this arc (given at the beginning of each line) represents the mission margin at the time in the mission at which the interruption occurred. Of course, in reality one also would have to consider dropping one or more of the encounters, which is exactly what DS1 did when the star tracker failed. A less drastic (and, in general, less effective) alternative would be to increase duty cycle (perhaps at the expense of telecommunications); DS1 accomplished this by thrusting while communicating.

Figure 6 shows that the spacecraft could have gone 61 days after launch without thrusting, case A, and still reached the targets. Notice that, if some of the early thrusting were accomplished, case B, that margin would grow to 89.5 days. Part of the reason this margin is so large is that much of the 89.5 days encompasses an optimal coasting arc from the reference trajectory in early 1999. If thrusting could not resume in March 1999, case C, only 27 days would be available to solve the problem before more substantive changes would be required. After the Braille encounter, case D, the margin jumps to almost three months, and then declines, cases E–G, until late in the mission, case H, when it increases again. The margin climbs late in the mission because, in the reference mission, by April 2001 the spacecraft is on a ballistic trajectory to intercept Borrelly. If thrust were interrupted shortly before this date, only a small additional impulse would be required to reach the target, and this could be performed later if necessary.

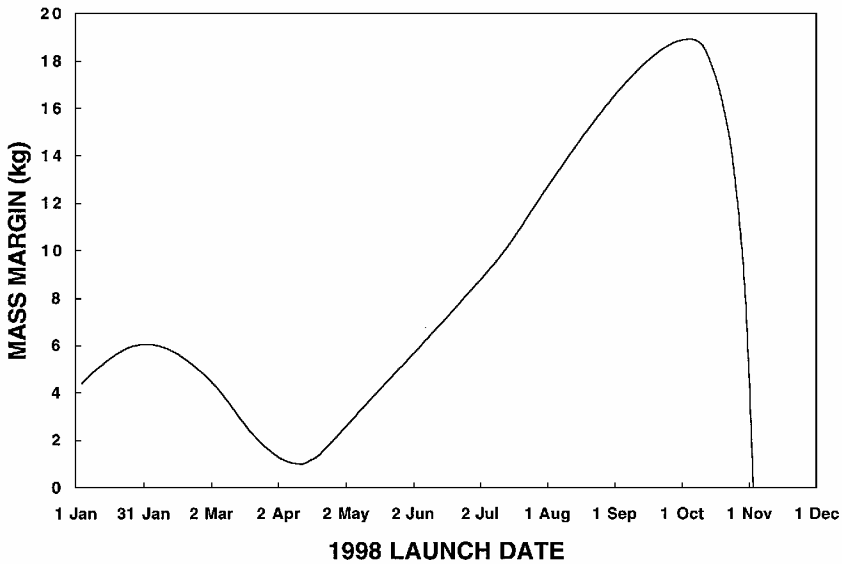

Fig. 7 Prelaunch mass margin estimate for the extended launch opportunity.

These results are not always intuitive, and they can be rather labor intensive to produce. This type of analysis was conducted both in prelaunch mission planning and several times during the mission, when new navigation information was incorporated or when new power models were received from subsystem engineers. If one of these times comes up unacceptably small, indicating inadequate mission margin, there may be ways to alleviate the problem. This might be done by strategically inserting coasting arcs in the reference trajectory; this can improve the margin in certain parts of the mission but reduce it in places where ample margin already exists. This strategy always reduces mass margin but may be a desirable trade in many instances and was used on DS1.

Of course, missions using chemical propulsion also have sensitivity to unexpected loss of thrust. In general, such missions have large impulsive maneuvers, and the result of missing such a maneuver, as has occurred on several interplanetary missions, can be extremely expensive. SEP missions tend not to have these short periods of high vulnerability.

SEP Mission Flexibility

One of the interesting features that occur sometimes in missions using SEP is shown in Fig. 7. DS1 was launched in October 1998, but Fig. 7 shows that there is positive mass margin for reaching all three targets with a launch on any date as early as January of that year or even before that. (Figure 7 is based on prelaunch analysis and does not reflect actual injection errors and other postlaunch results.) Note that the launch energy C_3 is fixed over this entire period (to match the launch vehicle performance and DS1 injected mass), although the declination and right ascension of the launch asymptotes are allowed to vary. (For convenience, to avoid incurring a launch vehicle performance penalty for this analysis, the declination of the launch asymptote was constrained to vary only between -28.5 and $+28.5$ deg.) Certainly allowing a variable C_3 would increase the mass performance. Launch periods for conventional interplanetary missions are typically only a few weeks long.

Conclusions

The capability offered by SEP, with its remarkably high specific impulse, clearly is important for future missions. The verification on DS1 that the system operated as predicted means that future missions can consider using this technology without increasing the cost and risk that the first user otherwise would face. The extensive analyses performed for DS1 should provide guidance to these projects in how to design and analyze their trajectories. The differences between ballistic and low-thrust trajectories amount to much more than simply the difference in mission performance: different methods must be applied to design and evaluate SEP missions. Because the trajectory design and spacecraft performance are coupled, a new set of models is needed for mission design. An understanding of the sensitivity to these models is an important element of the design and operation of the mission. The concept of mission margin has been introduced, and it may prove to be a useful metric for evaluating and improving the robustness of missions that use SEP.

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C. A. Kluever
Associate Editor