PREPARING FOR THE DAWN MISSION TO VESTA AND CERES

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The Dawn project is progressing toward its 2006 launch on a mission to orbit main belt asteroids (4) Vesta and (1) Ceres. Designed to provide insights into important questions about the evolution of the solar system, Dawn will spend more than 0.5 years in orbit about each of these bodies. This challenging mission is enabled by an ion propulsion system. In contrast to missions that use conventional chemical propulsion, the use of this system couples flight system mass and power , thereby requiring different methods of managing these and other technical resources. Now that the project is nearing launch, the refinement of resource estimates allows the identification of excess margin, which is being applied in novel ways to increase the scientific potential of the mission. The unusual relationship of the margins is described, and progress in preparing for the mission is presented.

INTRODUCTION

The ninth project in the National Aeronautics and Space Administration's Discovery Program, Dawn is designed to provide a major advance in the understanding of the nature of the solar system as planets were coalescing. The project is designed to conduct extensive investigations of the two most massive asteroids, (4) Vesta and (1) Ceres, with a single spacecraft that orbits both bodies. Because Vesta and Ceres are intact remnants of the epoch in which planets developed, studies of their properties are expected to elucidate important questions about the physical and chemical conditions and processes that prevailed in this early stage of the solar system's formation. The compositional gradient in the early solar nebula impressed itself upon these extremely large asteroids, providing a record that is recoverable now. Dawn will contribute panchromatic and multispectral imagery; visible, infrared, γ-ray, and neutron spectrometry; and gravimetry to the effort to understand these bodies and thus the dawn of the solar system. Dawn will be the first spacecraft to orbit a main belt asteroid and the first to orbit two target bodies after leaving Earth. A thorough description of the project has been presented elsewhere,[1,2] and an overview is included here.

The Dawn project is a product of collaborations by many institutions. Dawn's principal investigator, from the University of California, Los Angeles has overall responsibility for the project as well as direct management of the science team and the Dawn Science Center. He has assigned project management to the Jet Propulsion Laboratory (JPL). JPL also is responsible for management of the spacecraft and science payload development, safety and mission assurance, project systems engineering, mission design and navigation development, mission operations system development, and mission operations.

Orbital Sciences Corporation provided most of the spacecraft and is in charge of the overall assembly, system-level tests, and launch operations. JPL delivered the ion propulsion system and major elements of the electrical power system and telecommunications system

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to Orbital. As the first deep space project for Orbital, Dawn provides the company with vital experience that is expected to help it participate in more such missions.

The Los Alamos National Laboratory, funded by the Dawn project, delivered the Gamma Ray and Neutron Detector (GRaND). The other two science instruments are contributed to Dawn. Italy is donating the Visible and Infrared (VIR) mapping spectrometer. ("Vir" is Latin for "man.") It is provided through Agenzia Spaziale Italiana (Italian Space Agency), and it was designed, built, and tested at Galileo Avionica. The Framing Camera is donated by Germany. The Max-Planck-Institut für Sonnensystem-forschung (Max Planck Institute for Solar System Research) was responsible for the design, fabrication, and testing in cooperation with the Institut für Planetenforschung (Institute for Planetary Research) of the Deutsches Zentrum für Luft- und Raumfahrt (German Aerospace Center) and the Institut für Datentechnik und Kommunikationsnetze (Institute for Computer and Communication Network Engineering) of the Technischen Universität Braunschweig (Technical University of Braunschweig).

The Dawn flight system comprises the spacecraft and the payload. The design of the system and subsystems was described by Rayman *et al.*[1]

SCIENTIFIC MOTIVATION

Vesta and Ceres are the two most massive asteroids, and they are far larger than the other asteroids visited by spacecraft. Indeed, they are so large that they are often referred to as protoplanets. Some of their essential characteristics are in Table 1. For comparison, the longest axis of the irregular (253) Mathilde, the largest asteroid yet encountered by a spacecraft, is less than 12% of Vesta's equatorial diameter.

Vesta displays evidence of a hot and dry history. Spectra suggest the presence of pyroxene-bearing lava flows on the mineralogically heterogeneous surface. Vesta's shape is dominated by what appears to be an impact crater centered near the southern pole 460 km in diameter, or about 80% of the body's equatorial diameter.

Parameter	Vesta	Ceres
Principal radii (km)	289 × 280 × 229	487 × 487 × 455
Bulk density (g/cm^3)	3.7	2.1
Rotational period (hours)	5.34	9.08
Perihelion (AU)	2.15	2.55
Aphelion (AU)	2.57	2.99
Inclination (°)	7.1	10.6

Table 1. Physical and orbital characteristics of Vesta and Ceres. (AU is astronomical unit.)

Ceres, the largest body in the asteroid belt and only slightly farther from the Sun than Vesta, is quite different. Several lines of evidence indicate water has played a role in Ceres' history and may remain there now. For example, microwave observations seem to indicate the surface is largely composed of a material like clay, the formation of which probably requires water. Models demonstrate that it is possible for water ice to be preserved within Ceres for the age of the solar system.

Vesta, which is thought to have melted and differentiated, and Ceres, whose thermal evolution may have been arrested by its substantial inventory of water ice, seem to straddle the boundary between the rocky bodies of the inner solar system and the icy bodies of the outer solar system. Understanding the differences between these two intriguing complementary protoplanets that formed and evolved so close to each other provides the motivation for Dawn's mission of comparative planetology.

TECHNICAL MARGINS COUPLING

Any mission to orbit both Vesta and Ceres that relied on conventional chemical propulsion and other technologies already tested in operational space flights would be far beyond the scope of the Discovery program; indeed, even a rendezvous with only one of these bodies would not be affordable. Dawn's use of an ion propulsion system (IPS)[1,3] enables the compelling scientific mission to be undertaken within the constraints of available

resources.

The utilization of an IPS has many implications for the engineering of the mission apart from the significant increase in overall capability. It leads to a mission design and operational strategies that can be quite different from those for conventional interplanetary missions. Some facets of these topics have been explored elsewhere.[4] We present here a discussion of an important effect on the system design that allows a flexibility unavailable to other mission types.

The integration of the IPS into the design causes a coupling of resource margins that is different for missions which rely on conventional chemical propulsion. An understanding of the nature and consequences of this coupling is essential not only to the effective design and development of a system using an IPS but also to gaining the maximum benefit from exploiting this technology.

The IPS will be employed for essentially all post-launch trajectory control, including interplanetary transfers, trajectory correction maneuvers (if needed during long periods of coasting), Vesta and Ceres rendezvous and orbit insertion, transfers between science orbits, orbit maintenance, and escape from Vesta. (A maneuver of up to 10 m/s with the hydrazine-based reaction control system is available as a contingency for certain cases.) The Δv from the IPS is about 11 km/s, which is comparable to the Δv provided by the entire Delta 7925H-9.5 launch vehicle. With such a significant fraction of the total Δv from launch pad to end of mission being provided by the IPS, the maximum mass that can be accommodated by the mission depends strongly upon the performance of the IPS.

Some of the IPS components are on the spacecraft's low voltage (22 V to 35 V) bus, but the majority of the power drawn by the IPS is delivered by the electrical power system to the IPS' power processing unit (PPU) at high voltage (80 V to 140 V).

The IPS operating regime is spanned by 112 discrete throttle levels, ordered according to PPU power consumption, with each one specified by thruster electrical parameters and flow rates for the xenon propellant. The performance profile of the IPS is not constant over its operating range. (Indeed, the performance variation throughout the operating regime is nonlinear in thrust and not monotonic in specific impulse.) The extremes of the IPS operating range are shown in Table 2, from which it is clear that greater power translates into both greater thrust and greater specific impulse. That characteristic of the IPS is responsible for many of the unique consequences of using the IPS. The availability of greater power means a shorter time and a lower expenditure of propellant to accomplish a given Δv. This dependence of the thrust and specific impulse upon power delivered to the PPU couples the allowable mass to the available power.

PPU input power (kW)	Thrust (mN)	Specific impulse (s)
2.5	91	3000
0.5	19	1700

Table 2. IPS characteristics for extrema in PPU input power.

Neutral mass is defined to be the flight system's dry mass plus the mass of the hydrazine. At launch, this is equivalent to the total flight system wet mass injected by the launch vehicle minus the mass of the Xe. The neutral mass decreases during the mission because hydrazine is expended. Of greatest interest for the discussion here is the neutral mass at launch.

The thrust from the IPS is low, so thrusting is required for very long times. During its 9.5 year mission, Dawn will thrust for nearly 6 years. Therefore, while positive margins for neutral mass and power are necessary, they are not sufficient, as there also must be enough time to accomplish the required thrusting. The *missed-thrust margin* (formerly known as *mission margin*) is defined to be the duration of the unexpected missed thrust that can be accommodated at a specified time in the mission. The missed-thrust margin cannot be expressed as a single number; rather, its full description requires specifying the margin as a function of time in the mission.

Because of their coupling, the margins for mass, power, and missed thrust cannot be assessed or managed independently. Before exploring the implications of this coupling, we define some additional terms. The current best estimate (CBE) for any resource (in particular, neutral mass or power) is the most probable value for the resource at the time it will be utilized, based upon the present knowledge of the system with no allowance for design changes or uncertainty in the estimate. The uncertainty is the amount by which the CBE could change because of the incomplete knowledge at the current stage of design. Finally, the margin is the fraction of the resource capacity that is not consumed by the CBE. For example, the neutral mass margin, M_m, is

$$M_m = 1 - (m_{CBE}/m_{max}) \qquad (1)$$

where m_{CBE} is the CBE for the neutral mass and m_{max} is the maximum allowable neutral mass. (Note that missed-thrust margin is expressed in units of time. While it could be treated dimensionlessly, as neutral mass and power margins are, it is less convenient to do so. Once a loss of thrusting has occurred, the arrival date at the target may be altered significantly, so the resource capacity of remaining thrusting time is not a constant.)

The principal trajectory design tool used for Dawn, Mystic,[5] yields m_{max} at launch subject to the power available to the PPU and other constraints, some of which are discussed below. This neutral mass limit is used to establish mass allocations to the flight subsystems and is termed the launch neutral mass allocation.

Because the power available to the PPU is an input to the trajectory design, we can choose the power margin to apply and be assured that the resulting mission profile and launch neutral mass allocation will be consistent with that margin.

For convenience, we decompose the power into three separate terms: the solar array output (P_s); the sum of all the flight system loads (including losses) during IPS thrusting except the power consumed by the PPU (P_l); and the power consumed to the PPU (P_{PPU}).

It is important to recognize the distinction between P_{PPU} and the power available to the PPU. They are not necessarily the same because of the finite PPU operating range, as indicated in Table 2. Whenever IPS thrusting is desired, it is optimal to thrust with as much power as possible. Therefore P_{PPU} is either the power that corresponds to the highest IPS throttle level or the power that is available to the PPU, whichever is smaller. (We ignore here the case in which there is insufficient power available to attain the minimum PPU input power, because that does not occur in the Dawn mission.)

We add the superscript m to solar array and non-PPU power terms to indicate that margin has been applied; without the superscript, each represents a CBE. Thus the power available to the PPU is $P_s^m - P_l^m$. In other words, P_{PPU} is either 2.5 kW or $P_s^m - P_l^m$, whichever is smaller. (No explicit margin is applied to P_{PPU} because doing so would amount to adding margin to the margin already accounted for in the solar array and the non-PPU loads. Uncertainties in the performance of the PPU and other IPS components are covered by the Xe margin.) The system power margin, M_P, is

$$M_P = 1 - (P_{PPU} + P_l)/P_s \qquad (2)$$

To establish the margins applied to P_s and P_l, we quantify two separate methods and select whichever yields the more conservative (i.e., larger) margins. In the first procedure, we develop separate profiles for the minimum acceptable margins on P_s and P_l, based on time during the project's development lifecycle. These values are derived from JPL's studies of its institutional experiences in development of spacecraft and large instruments. In the second method, we separately estimate the uncertainty in P_s and P_l. P_s uncertainties are dominated by the uncertainty in predicting the performance of the array under low intensity solar illumination and low temperature. P_l uncertainties are treated as a simple sum of the individual uncertainties applied to component powers. Each uncertainty is obtained from a prescription that corresponds to a given level of component design maturity. The values we use are typical for those used on other programs at Orbital and are related to the International

Standards Organization's standards for mass uncertainty, which we also use (see below). To assure that the margin is sufficient to account for more than the uncertainty, the uncertainties for P_s and P_l are increased by 20% to achieve the margin value for this second method.

At different times during the project development, each of the two methods for computing power margin has yielded the preferred margin. At the present stage, the first method yields the greater margin at all heliocentric ranges. Following the margin policy, we apply sufficient margin to accommodate a reduction of 5% in P_s and an increase of 15% in P_l, so $P_s^m = 0.95P_s$ and $P_l^m = 1.15P_l$. Figure 1a shows P_s, P_l, and P_{PPU}, and Figure 1b shows the corresponding M_P.

Figure 1 requires some further explanation. The curves are fits to estimates of solar array and non-PPU loads. Because P_{PPU} is limited to 2.5 kW inside about 1.98 AU, it is insensitive to details of P_s and P_l. Therefore, the points used in developing fits for P_s and P_l at heliocentric ranges for which P_{PPU} = 2.5 kW are given low weight. The discontinuity in P_l at 2.5 AU reflects the expectation that by the time Dawn has passed 2.5 AU, 1 of the 3 IPS thrusters will no longer be operational because it will have exceeded its qualified Xe throughput, so heater power will not be required to keep 1 of the 2 inactive thrusters and the associated gimbal within their required allowable temperature ranges. Finally, it should be noted that this figure contains a simplification for the purposes of illustration. The model for P_s used in our analyses has terms that are dependent not only on heliocentric range but also on time. As the flight system's heliocentric range does not change monotically with time, the time dependence of P_s (and thus P_{PPU} and M_P) cannot be captured in these simple plots.

Now that we have the power models that are used as inputs to the trajectory analysis, we may follow the flow shown in Figure 2 for computing and balancing margins. We begin on the left, in which the power estimates are used to formulate an overall power margin, as above. That margin and the CBEs together allow the computation of the heliocentric range-dependent P_{PPU}, which is one of the inputs to the trajectory optimization tool.

While P_{PPU} is a constraint on the trajectory design, we have discussed it separately from the other constraints for clarity. There are many more constraints included in the trajectory optimization, and we describe some here. The IPS constrains the mission design not only through the thrust and specific impulse (both as functions of P_{PPU}) but also by the maximum mass of Xe that may be expended. The flexibility afforded by the IPS allows exceptionally long launch opportunities, so the launch period for Dawn is dictated not by technical capability but rather by other considerations, including project readiness, as discussed below. Therefore, the launch date is a constraint, not a result of the optimization. The injectable mass as a function of injection energy of the Delta 7925H-9.5 launch vehicle provides another constraint. (In some cases, the maximum neutral mass is achieved with an injection that delivers less energy than the maximum capability of the launch vehicle for a specified flight system wet mass.[1])

Operational considerations form another set of constraints. As the high gain antenna is fixed to the spacecraft body, we expect to interrupt thrusting for high rate telecommunications in each week when supportable rates through the high gain or low gain antennas are inadequate in the thrust attitude. The *duty cycle* is defined to be the time available for IPS thrusting at the maximum achievable throttle level in the optimal attitude when thrusting is desired. Dawn plans for a conservative mean duty cycle of 95%, allowing > 8 hours/week for telecommunications (and associated turns). In addition, to allow for less frequent activities that are not compatible with IPS thrusting at the maximum achievable throttle level in the optimal attitude (e.g., instrument calibrations that require special pointing), we force coast periods of 4 days every 6 months. (Of course, there are extended periods in which coasting is optimal, but this constraint ensures that even during long thrusting periods, time is available for these other activities.)

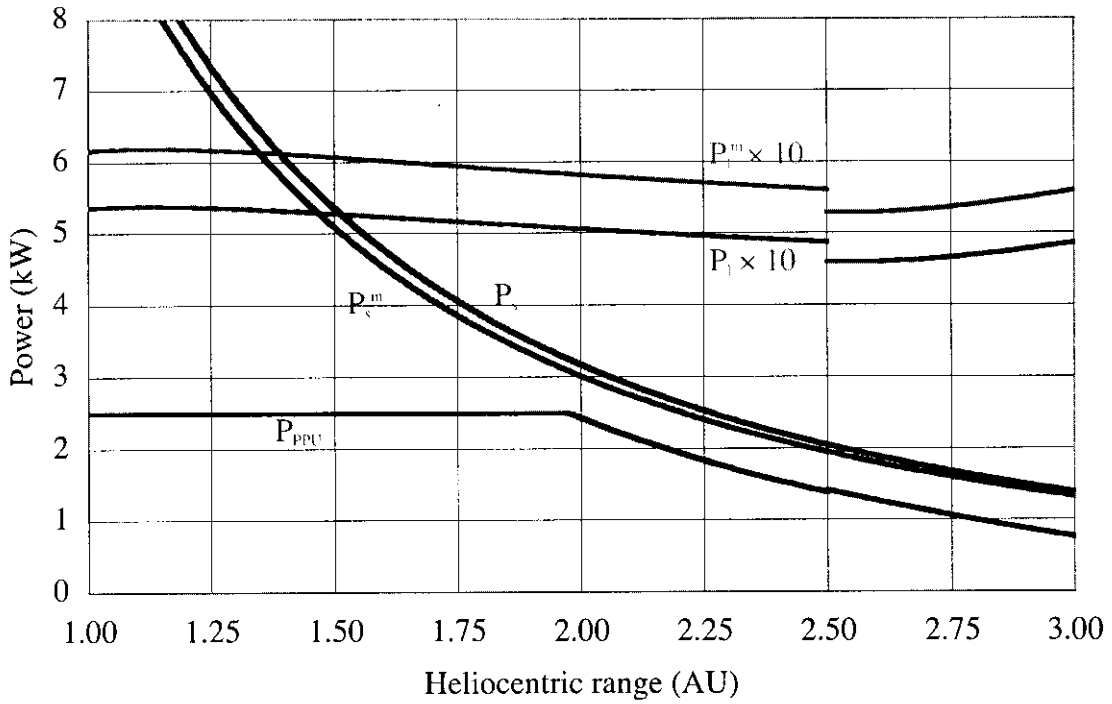

Figure 1a. Power models. Note that P_l and P_l^m are increased by a factor of 10 for clarity.

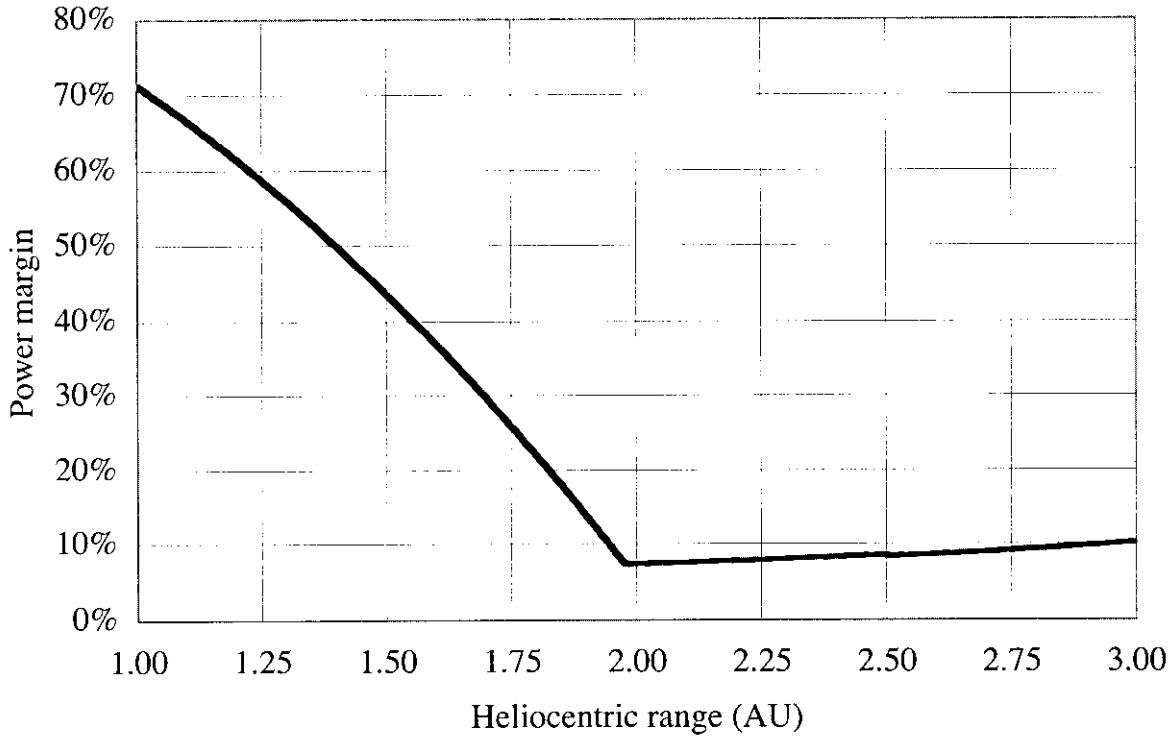

Figure 1b. System power margin (M_P) during IPS thrusting.

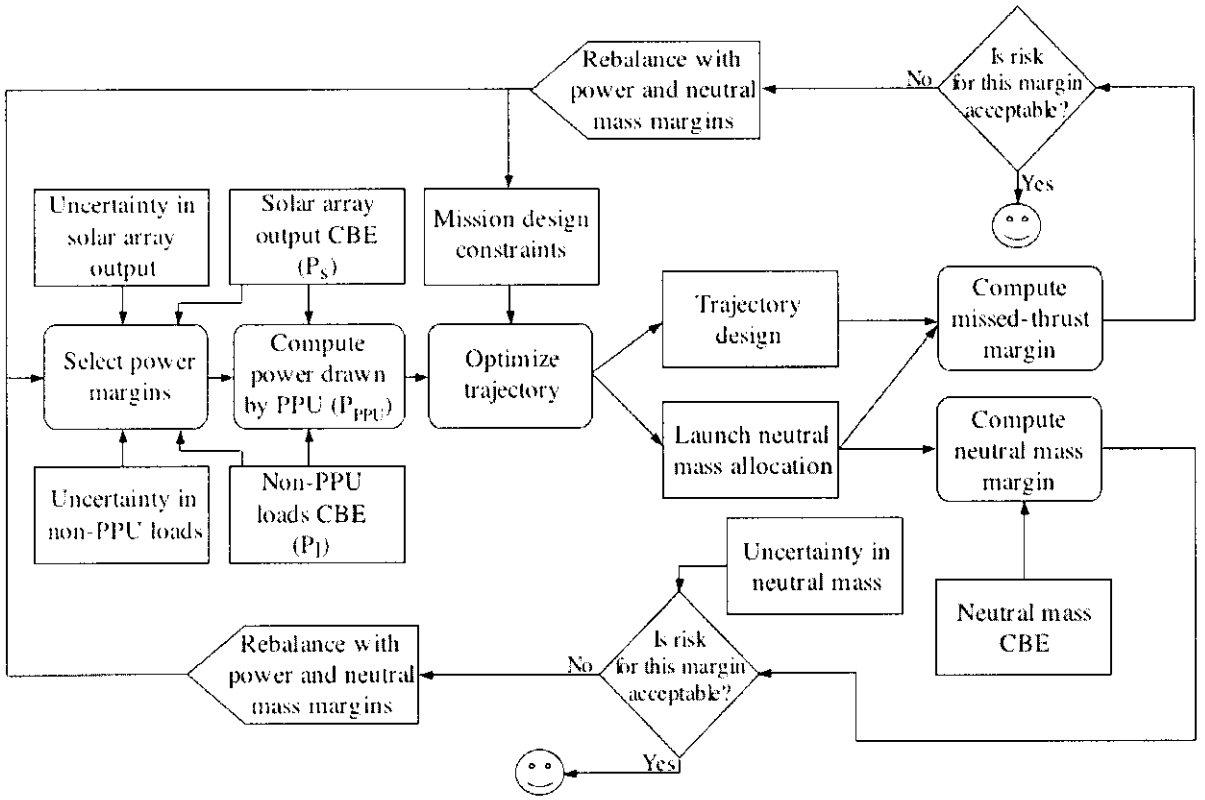

Figure 2. Margin balancing process. The description in the text begins with the solar array output and non-PPU loads CBEs and uncertainties.

Finally, the double rendezvous provides several constraints besides the obvious one that the trajectory must accomplish both rendezvous. The duration of the residence at Vesta is a special case of a forced coast period, because that time represents a period in which the IPS is not allowed to change the interplanetary trajectory. While there is an optimal time to arrive at Vesta, subject to all other constraints, the Ceres arrival date cannot be mathematically optimized. Postponing the arrival date for the final rendezvous in a mission using an IPS is certain to yield equal or greater margins on parameters of interest. Therefore, the Ceres arrival date is specified based on test cases to establish reasonable neutral mass margins, the financial budget for operations, and other considerations.

In addition to the Ceres arrival date, relaxing any of the mission design constraints will provide an equal or greater neutral mass margin for a fixed missed-thrust margin or will provide an equal or greater missed-thrust margin for a fixed neutral mass margin. These relations are essential to the management of the margins.

The mission design constraints and the description of P_{PPU} are combined in the trajectory optimization tool to generate a trajectory and a corresponding m_{max}. Both are consistent with the power margins that were used as inputs.

As described above, now the neutral mass margin may be computed. To assess that margin, we use a methodology similar to that for power margin in which we have one approach based on JPL historical experience for development of flight hardware and one based on uncertainties of project-specific component masses. At the present, in contrast to the power case, the more conservative method is the second, which yields a required margin of 5%. With m_{CBE} = 746 kg, the actual margin is about 8%.

The missed-thrust margin requirement is

that a well defined minimum science mission at both Vesta and Ceres be completed even with an unexpected missed thrust of 28 days at any time in the mission. The requirement includes the detail that during solar conjunction, the missed-thrust margin must be 28 days plus the duration of the conjunction period.

To determine the missed-thrust margin at a specified time, the position, velocity, and mass from the baseline trajectory are selected at that time and then propagated for 28 days of coasting. The trajectory is reoptimized at the end of that coast period, now with the constraints that correspond to the minimum science mission. (Note that, in general, substantially more than the minimum science mission will be possible, and in some cases the full mission will be possible even with an unexpected loss of thrust in excess of 28 days, but the analysis is so laborious that this approach is used during development. Once a final trajectory is selected, a more complete quantification of the missed-thrust margin will be made. In the meantime, demonstration of compliance with the requirement is sufficient.) This procedure is performed every 100 days along the thrusting portion of the interplanetary trajectory, although where the profile of the margin suggests the need, it is performed at finer intervals. Because this analysis uses the baseline trajectory, it rests on the conservative assumption that the actual neutral mass at launch is the maximum allowed neutral mass at launch. Of course, the analysis also is guaranteed to be consistent with the power margin.

If the computed neutral mass and missed-thrust margins are acceptable, no change is needed. If any of the margins is too small however, the margins may be rebalanced. The neutral mass margin and the missed-thrust margin may be adjusted by modifying the power margin or the mission constraints.

Following this approach, resources may be traded in a manner unavailable to missions that employ standard chemical propulsion. For example, during development a shortfall in power may be compensated by a reduction in neutral mass (or, if necessary, by acceptance of somewhat greater risk through a reduction in the required mass margin).

A particularly common case in developing the mission is that missed-thrust margin is determined empirically to be insufficient at some time in the mission. To improve that margin, a coasting period may be forced into the trajectory shortly after the sensitive time to act as a buffer against the missed thrust; following an unexpected loss of thrust, the prohibition against thrusting during the buffer period would be removed. The insertion of this forced coast will reduce the neutral mass margin, although experience on DS1 and Dawn has been that the neutral mass cost of raising the missed-thrust margin for a limited portion of the mission is small. Of course, in operations, if there were no missed thrust, we would not coast during a segment that was planned only to improve the missed-thrust margin; rather, we would reoptimize the trajectory without the enforcement of the coasting.

Because the technical margins increase (or, at worst, remain constant) with later Ceres arrival dates, the mass, power, and missed-thrust margins all may be increased by postponing the rendezvous date. This option remains available even for some in-flight problems in which there is an excessive interruption in thrusting, an unanticipated loss in power, or performance of the IPS that is not covered by other margins. Therefore, if the depletion of all technical margins proved insufficient for resolving some extraordinary anomaly, in some cases the postponement of the Ceres rendezvous, at the expense of the financial cost of mission operations, would provide the needed relief to rescue the scientific objectives.

As the flight system design matured and flight system integration began, the consumption of margins for engineering needs was less than had been anticipated, so Dawn has excess mass, power, and missed-thrust margins. The margins now are sufficiently large that the project is engaged in an assessment of several options for how to take the greatest advantage of them.

Just as targeting a later arrival at Ceres allows an improvement in technical margins, unnecessarily large margins may be expended to reach Ceres earlier. Until recently, the arrival at Ceres was planned to be in August 2015.[1]

Consumption of some of the excess would allow not only reaching Ceres in February 2015 but also increasing the residence at Vesta from 7 months to 8 months.

The earlier arrival at Ceres reduces the total cost of mission operations because of the shorter interplanetary cruise phase. With science operations in orbit around the asteroids about 1.6 times as expensive as in interplanetary cruise, the earlier arrival permits an increase in operations time at the asteroids of 3 months with no change in the total cost. Therefore, with the addition of 1 month at Vesta, we may extend the primary mission at Ceres from 5 months to 7 months, thus completing the primary mission in September 2015 instead of January 2016. (Note that a longer time at Ceres, as the final target of the primary mission, has minimal effect on the technical margins, requiring only very small amounts of Xe for orbit maintenance and hydrazine for desaturation of reaction wheels. Longer times at Vesta, however, reduce the margin for reaching Ceres.)

The investigation of the failure in 2005 of a flight spare Xe tank during a pressure test as part of the manufacturing flow led the project to reduce the maximum load from 450 kg to 425 kg. (The flight tank and a replacement flight spare tank each passed the same test several times.) This provided additional human safety margin during prelaunch operations as well as greater tank robustness in flight. Reducing the Xe mass modified a mission constraint, thereby altering the margins, but with surplus margin, no significant change in the overall mission plan was necessary.

The interplanetary trajectory with the MGA and the February 2015 Ceres arrival using the smaller Xe mass is shown in Figure 3. While the new timeline for the mission scenario described by Rayman *et al.*[1] is changed only in its details, those details translate into lower risk or greater science return at each body. As safe and efficient transfers between science orbits are complex and not yet fully understood, the availability of more time at the asteroids reduces the risk to the acquisition of the baseline science data. With more time, the surface coverage may be expanded for FC and VIR, and the number of observations through different FC filters may be increased. Additional time also will yield superior spatial resolution for GRaND and gravimetry as well as more opportunities to acquire bonus observations with FC and VIR in the low-altitude orbits.

In contrast to most interplanetary missions, which have launch periods of a few weeks, Dawn's effective launch period is extremely long because of the IPS. It is known that a launch any time between May 2006 and September 2007 yields adequate margins; while launches outside that range have not been explored, it is certain that the full launch opportunity is even longer. The 17 June opening of the official launch period was defined by an early estimate of project readiness. More recent schedule analysis has shown that, while that launch date remains achievable, there is a significant probability that it would require descoping some of the planned prelaunch testing. As a result, the project is evaluating shifting the opening of the scheduled launch period to July.

An option under investigation is to remove the Mars gravity assist (MGA) from the itinerary. This possibility was enabled by the health of the technical margins, as the MGA increases the margins at the expense of flight time. The launch opportunity for the mission without the MGA however, while still significantly longer than that for conventional missions, may not yield acceptable margins after July 2006. A decision on whether to eliminate the MGA from the mission will be made late in 2005.

PROJECT STATUS AND PLANS

Dawn commenced its assembly, test, and launch operations (ATLO) phase in January 2005 after having completed its ATLO readiness review the previous month. By early September 2005, the flight software had all the necessary functionality, and by the end of that month, all of the flight system hardware will have been delivered to Orbital.

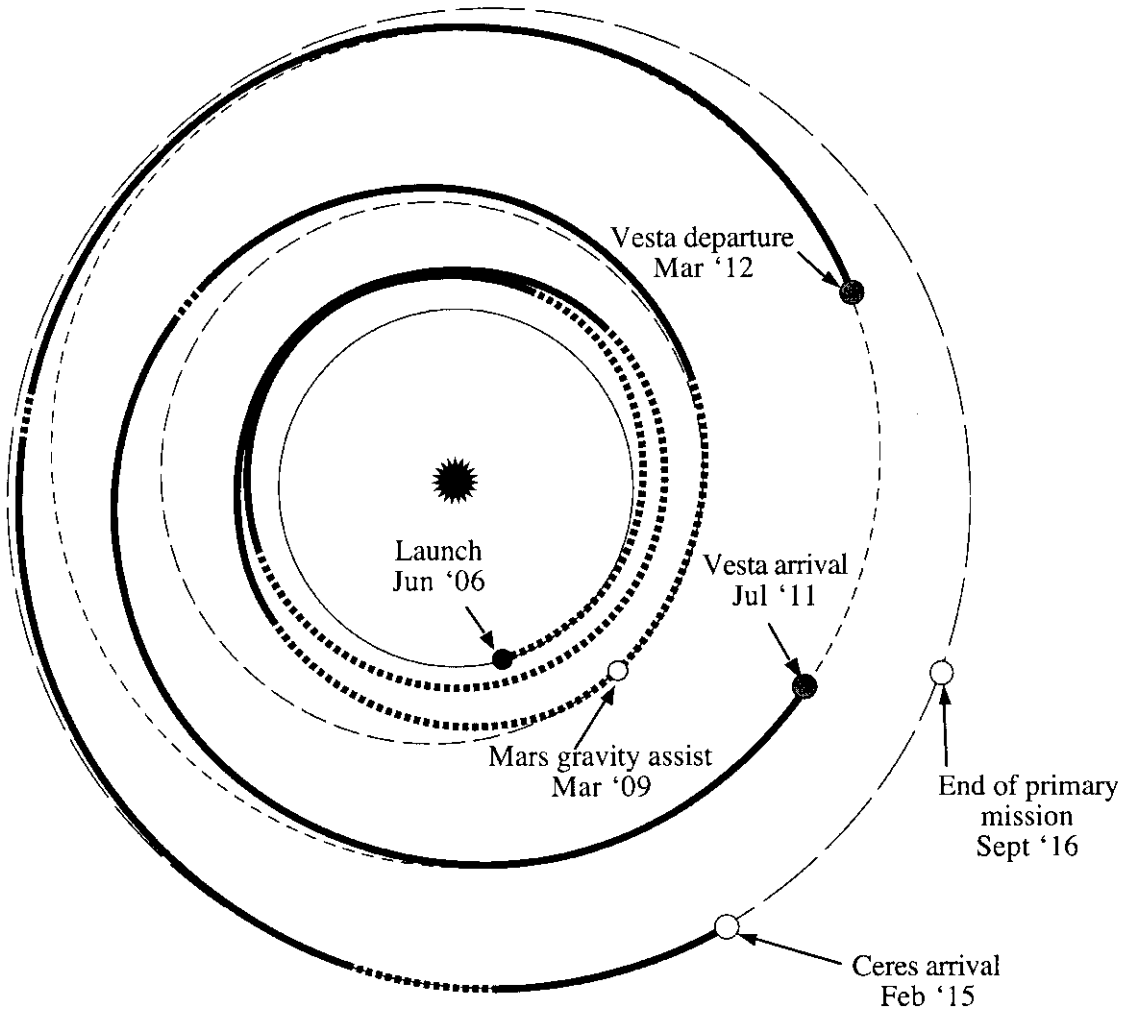

Figure 3. Interplanetary trajectory. The dotted portions denote periods of coasting, and the solid portions show when the IPS is thrusting. Coasting periods as short as 7 days are not shown.

Following integration and testing at Orbital, including system-level acoustic, vibration, and pyro shock, the flight system will be shipped to Goddard Space Flight Center (GSFC) in January 2006. GSFC's larger vacuum facilities and its propinquity to Orbital make it an excellent location for further testing. The majority of the time there will be devoted to system thermal vacuum testing, a portion of which will include operating the IPS.

Transportation from GSFC to Cape Canaveral is scheduled for May 2006. The spacecraft will receive its final loads of Xe and hydrazine there, and after a wet spin balance will begin integration with the launch vehicle. Launch will take place from Space Launch Complex 17-B at Cape Canaveral Air Force Station.

In the meantime, many tests that are not performed on the flight system will be conducted in testbeds. Dawn has 3 single-string testbeds, any 2 of which may be combined to emulate the spacecraft's dual-string configuration.

CONCLUSION

Now less than one year from launch, the Dawn project is continuing to take advantage of the flexibility of the mission design and of the management and expenditure of technical margins that the use of the IPS yields. The scientific rewards that Dawn plans to reap by revealing the properties of Vesta and Ceres could not be afforded without the technical performance of the IPS. These benefits will remain important through the completion of the project, both before launch and in operations.

ACKNOWLEDGMENTS

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