AAS 14-255

THE EVOLUTION OF DEEP SPACE NAVIGATION: 1999-2004*

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The exploration of the planets of the solar system using robotic vehicles has been underway since the early 1960s. During this time the navigational capabilities employed have increased greatly in accuracy, as required by the scientific objectives of the missions and as enabled by improvements in technology. This paper is the third in a chronological sequence dealing with the evolution of deep space navigation. The time interval covered extends from the 1999 launch of the Stardust spacecraft to comet P/Wild 2 through the arrival of the Cassini spacecraft at Saturn in 2004. The paper focuses on the observational techniques that have been used to obtain navigational information, propellant-efficient means for modifying spacecraft trajectories, and the computational methods that have been employed, tracing their evolution through eight planetary missions.

INTRODUCTION

Two previous papers have described the evolution of deep space navigation over the time interval 1962 to 1999.[1,2] The missions covered in the first of these ranged from the early Mariner missions to the inner planets to the Voyager mission to the outer planets. The second paper extended the previous paper by one decade. It covered the entirety of the Magellan, Mars Observer, Mars Pathfinder, Mars Climate Orbiter, and Mars Polar Lander missions, as well as the portions of the Pioneer Venus Orbiter, Galileo, Ulysses, Near Earth Asteroid Rendezvous, Mars Global Surveyor, Cassini, and Deep Space 1 missions that took place between 1989 and 1999.

The current paper extends the second paper by five years. It covers the portions of the Galileo, Near Earth Asteroid Rendezvous, Mars Global Surveyor, Cassini, Deep Space 1, Stardust, 2001 Mars Odyssey, and Mars Exploration Rover missions that took place between 1999 and sometime in 2004. As in the previous papers, attention is limited to those missions that involved travel well in excess of 1,500,000 km from the Earth and that were targeted to fly close to one or more distant natural bodies. Attention is focused primarily on planetary missions that were managed by the National Aeronautics and Space Administration (NASA).

EXPLORATION OF THE TERRESTRIAL PLANETS

Mars Global Surveyor

The interplanetary, aerobraking, and early mapping orbit phases of the Mars Global Surveyor (MGS) mission have been described in Ref. 2 and references listed therein.

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The mapping orbit design called for a constant ground-track walk (GTW), referenced to downward equator crossings, of 58.6 km eastward after a cycle of 88 orbits or 7 sols (Martian days). Had this been the case throughout the mapping orbit phase, uniform ground tracks would have been obtained with 3-km spacing at the equator, over 6917 orbits or 550 sols. After the transfer-to-mapping-orbit maneuver on 19 February 1999 and the start of the mapping mission on 9 March, the actual GTW rate was 27.8 km (over 88 orbits). Orbit trim maneuvers (OTMs) on 7 May and 10 June brought the mapping orbit closer to a "frozen" condition[2] and corrected the GTW rate. A third OTM on 11 August compensated for GTW drift since the prior OTMs. The GTW rate remained close to its desired value until around solar conjunction on 1 July 2000, after which the perturbing effects of angular momentum desaturations (AMDs) caused the orbit semi-major axis to increase and the GTW rate to decrease. Two subsequent OTMs during 2000 were considered but ultimately not executed. Over the course of the 695-day primary mapping mission, the ideal ground track spacing of 3 km could not be obtained precisely, due to orbital variations. Instead, 95.7% of the ground-track gaps were less than 9 km, with none greater than 18 km. 34.6% of the gaps were between 1.5 and 4.5 km, and 48.7 % were less than 1.5 km.[3]

As the primary mapping mission progressed, the evolving gravity field model became accurate enough to no longer represent a major error source in orbit determination. (A comparison of orbit determination results using various Martian gravity fields may be found in Ref. 4.) Velocity changes due to AMDs were the dominant error source, with thruster plume impingement on the high-gain antenna presenting a complication in modeling AMDs. Unanticipated events, such as the failure of a reaction wheel assembly, necessitating the transfer to another assembly, and the spacecraft's entry into a contingency mode caused the subsequent orbital effects of AMDs to change in unexpected ways.[3]

The primary mapping mission was completed on 1 February 2001, and several extended missions followed. On 16 August 2001, one of the spacecraft body axes was rotated so as to point 16° off of nadir, rather than at nadir. This new configuration substantially reduced the frequency of spin-axis AMDs, making it easier to predict the orbit of the spacecraft several months into the future.[5]

An orbit synchronization maneuver (OSM) was executed on 3 October 2003 to allow the spacecraft to be overhead, at a specific time with a 30-s tolerance, during the atmospheric entry, descent, and landing (EDL) phase of the Mars Exploration Rover-A (MER-A) mission on 4 January 2004, in order to provide UHF communication relay-link support. The 0.534-m/s OSM changed the orbital period by 3.34 s, resulting in an orbit-timing change of 62.3 minutes at the time of the overflight 1130 orbits later. Two later opportunities for OSMs were not needed because there were no subsequent drifts in orbit timing of significance. The desired relay configuration was achieved with a timing error of 8.5 s.[5]

After providing relay support to the MER-A spacecraft, an MGS OSM was executed on 4 January 2004 to provide similar EDL UHF relay support to the MER-B spacecraft on 25 January. This maneuver changed the orbital period by 1.31 s, resulting in an orbit-timing change of 5.4 minutes at the time of the overflight 251 orbits later. The desired relay configuration was achieved with a timing accuracy of 3.5 s.[5]

2001 Mars Odyssey

Interplanetary Phase. The Mars Odyssey spacecraft was launched on 7 April 2001. The injection aimpoint was biased about 450,000 km away from Mars, to ensure that the likelihood of an unplanned impact by either the Odyssey spacecraft or the launch vehicle upper stage was less than 10^{-4} (Refs. 6, 7).

One tracking pass per day by a 34-m Deep Space Network (DSN) antenna was typically used during much of the interplanetary orbit transfer, with continuous tracking used around critical events and during the last 50 days of cruise. The extreme negative declinations of the spacecraft limited DSN tracking to only the Canberra, Australia, complex during the first two months of cruise. The spacecraft later became visible from the Goldstone, California, and Madrid, Spain, complexes, but only at low elevation angles, making the resulting tracking data susceptible to uncalibrated ionospheric and tropospheric modeling errors. Traditional Doppler and ranging measurements (at X-band frequencies) were supplemented with 47 delta-differential one-way range (ΔDOR) measurements, almost all along the Canberra-Goldstone baseline, over the last four months of cruise. ΔDOR was included as an additional measurement data type, independent of line-of-sight measurements, to make the orbit determination process more robust to potential modeling issues than was the case with the Mars Climate Orbiter (discussed in Ref. 2). It was particularly useful in determining spacecraft position in the direction perpendicular to the ecliptic plane, which is least well determined by Doppler and range data and had a significant influence on encounter conditions. Although the ΔDOR technique had been used at times in the past, the aging hardware and software system was rebuilt in preparation for the 2001 Mars Odyssey mission.[6,7,8]

Spacecraft attitude control was provided principally by reaction wheel assemblies, which had to be desaturated periodically by firing pairs of 1-N reaction control subsystem (RCS) thrusters. These thrusters are not coupled, with the result that the AMDs produced net changes in spacecraft translational velocity. Although each such event produced a velocity change (ΔV) of less than 10 mm/s, the cumulative trajectory perturbation was on the order of 10,000 km. The principal external torque causing angular momentum accumulation by the reaction wheels was due to solar radiation pressure. Thus, the recording and downlinking of thruster telemetry data and careful analysis and calibration of these data were needed to meet the delivery accuracy requirements. Two in-flight calibration activities were carried out to fully characterize the magnitude and direction of the thrust vector for each RCS thruster pair, as reaction wheels were spun up and spun down. During the last two months of cruise, the spacecraft's attitude and solar array were adjusted to balance the solar radiation pressure torque relative to the spacecraft's center of mass. This significantly reduced the frequency of AMDs and the resulting spacecraft translational velocity changes.[6,7,8]

Quantities that were estimated during the orbit determination process include spacecraft position and velocity at a reference time (or epoch), velocity changes associated with trajectory-correction maneuvers (TCMs) and AMDs, solar radiation pressure, and (shortly before encounter) the Mars ephemeris. Stochastic range biases and spacecraft accelerations were also estimated. Ionospheric and tropospheric effects on radio signals, tracking station locations, Earth and Mars ephemerides and gravity parameters, Earth orientation parameters, quasar locations, spacecraft areas subject to solar radiation pressure, and future AMD ΔVs were "considered"[2] as error sources, rather than estimated. The nominal weightings of Doppler, range, and ΔDOR data were 0.1 mm/s (for a 60-s count time), 3 m, and 0.12 ns.[6]

Four TCMs were scheduled and executed on the way to Mars, all using 22-N TCM thrusters. TCM-1 (at launch plus 46 days) and TCM-2 (at launch plus 86 days) were designed to correct injection errors, aimpoint bias, and other trajectory errors, while satisfying planetary protection requirements. TCM-3 (at encounter minus 37 days) and TCM-4 (at encounter minus 12 days) corrected the remaining trajectory errors and targeted directly to the desired encounter conditions, in preparation for the Mars Orbit Insertion (MOI) burn. In general, the velocity change direction was constrained such that the medium-gain antenna could remain in communication with the Earth during the burn. Launch vehicle injection errors were coincidentally such that significantly less

ΔV was needed for the four TCMs than had been budgeted pre-flight. A fifth maneuver opportunity was available as a contingency option at either 24 or 6.5 hours prior to MOI, but was not needed.[6,7,8]

The driving navigation requirement on approach was to deliver the spacecraft to an altitude 300 km above the north pole of Mars (at the second periapsis) to an accuracy of ±25 km (3σ). In addition, the orbit inclination was to be controlled to 0.2° (3σ); and the closest approach time to 10 s (3σ). The Odyssey spacecraft was inserted into an 18.6-hour elliptical capture orbit about Mars on 24 October 2001 by means of a 1433-m/s MOI burn, using the spacecraft's 695-N main engine. The navigation enhancements and robustness measures incorporated into this mission paid off, as the achieved periapsis altitude was 0.7 km from the targeted altitude, with the orbit inclination and time of periapsis 0.04° and <1s from their targeted values.[6,7,8]

Aerobraking Phase. Once captured into orbit, the spacecraft used aerobraking in the Martian atmosphere to reduce orbital energy. Had the post-MOI orbit period exceeded 22 hours, a propulsive maneuver would have been performed to reduce it to 20 hours prior to the start of aerobraking; but the achieved 18.6-hour period made this unnecessary. During an initial "walk-in" segment, the periapsis altitude was gradually lowered from 292 km down to 111 km, by means of seven maneuvers, one every second orbit, to set up the "main" segment of aerobraking – a gradual descent reflecting the uncertainties in the correct target altitude for the onset of aerobraking.[9]

The dominant mission constraint throughout aerobraking was the heating of the solar panels during atmospheric passes, which limited the total drag that could be achieved with each pass. During the "main" segment, the peak heating rate was the limiting factor. During a subsequent "end game" segment, integrated heating rate was instead the limiting factor. While these heating considerations placed an upper bound on the drag effects that could be experienced on each orbit, a minimum amount of drag was nevertheless needed to limit the total number of aerobraking passes and the duration of the overall process. Aerobraking needed to be completed before the local true solar time (LTST) at the descending equator crossing, decreasing at about two minutes per day due to Mars' motion about the sun, caused the spacecraft to be in Mars' shadow for too long per orbit to maintain a satisfactory battery state of charge. Thus, on each drag pass, there were upper and lower limits on the acceptable heating effects, with considerable margin built into the upper limit to ensure spacecraft safety in the presence of unexpected atmospheric variations.[8,9,10]

The aerobraking process demanded around-the-clock operations and quick reactions to accommodate the ever changing Martian atmosphere. Variability of the atmosphere limited the ability to predict the atmospheric density that would be encountered on any upcoming pass, with observed atmospheric densities varying by 20-40% (1σ) from the predicted values. Although much had been learned from the MGS aerobraking experience,[2] the Odyssey aerobraking location (northern latitudes) and season (northern hemisphere winter) had not been encountered during the MGS mission, leaving substantial atmospheric modeling uncertainties. The estimation of atmospheric density characteristics from spacecraft accelerometer data is discussed in Ref. 11. The variations in subsequent periapsis times that resulted from unpredicted drag-induced changes in orbital period necessitated frequent updates to on-board sequences. Certain critical spacecraft events, such as slewing the spacecraft to the drag attitude and stowing the solar array, were keyed to the sequenced periapsis time. Compromises to spacecraft safety could result if these times were inaccurate.[8,9]

Drifts in argument of periapsis due to the oblateness of Mars alter the periapsis altitude, as the periapsis location moves through a range of Martian latitudes. Therefore, periodic aerobrake trim maneuvers (ABMs), performed at apoapsis, were needed to maintain the heating rate within the desired corridor. Only one such maneuver was allowed per day and only on the last apoapsis of a command sequence, to avoid corrupting sequence timing subsequently. Thus, decisions were made on a daily basis as to whether to execute a trim maneuver. Any such maneuver would have to result in satisfactory predicted atmospheric densities at future periapses prior to the next maneuver opportunity.[9,10]

Aerobraking latitude began at 68°, gradually increased to a maximum of 86°, and then gradually decreased to 23°. Thus, the ABMs were performed primarily to decrease periapsis altitude until the maximum latitude was achieved and to increase periapsis altitude thereafter, reflecting the oblate shape of Mars, although resonances with Mars' rotation period, higher-order gravity harmonics, and other factors came into play from time to time. When the periapsis altitude was naturally tending to decrease, due to decreasing periapsis latitude, commands were enabled on board the spacecraft to autonomously execute a maneuver to raise the periapsis out of the atmosphere if the spacecraft entered a safe mode. Fortunately, this protective measure, which would have expended propellant more desirably used for other purposes, never had to be executed.[9]

During aerobraking, the Navigation Team was required to predict the upcoming periapsis altitude to 1.5 km and the time of periapsis to 225 s, with the latter requirement being by far the more difficult of the two. Sequence timing adjustments were needed when the predicted time of an upcoming periapsis shifted by more than 225 s. Periapsis timing prediction was considered sufficiently accurate only for a single orbit until the orbit period had been reduced to six hours, after which accurate predictions more than one periapsis into the future were feasible. Thus, trajectory updates were sometimes delivered up to four times per day to accommodate atmospheric uncertainties and meet the timing requirement. Continuous DSN tracking coverage was provided during aerobraking.[8,9]

Once the predicted mean orbit lifetime of the spacecraft (defined as the time for the apoapsis altitude to decay to 300 km) had diminished to one day, the "walkout" segment began. Daily ABMs were done during this segment.[9]

After 332 consecutive drag passes over 76 days, the orbital period had been reduced to 1.9 hours, with the apoapsis altitude having decayed from 27,000 km to 503 km. The cumulative drag force had produced an equivalent ΔV of 1.08 km/s. To control the atmospheric heating rate, 33 ABMs were executed, expending a total of 46.6 m/s.[9]

On 11 January 2002, the aerobraking process was brought to an end with the first of five transition maneuvers, an aerobraking termination maneuver, which raised the periapsis altitude to 201 km. A second, larger transition maneuver was executed on 15 January, to raise the periapsis altitude to 419 km and adjust the orbit inclination to 93.1°. A third maneuver on 17 January established a frozen orbit by rotating the periapsis to the south pole, while reducing the apoapsis altitude to 450 km and the periapsis altitude to 387 km. Two small trim maneuvers were executed on 28 and 30 January.[8,9]

At the end of aerobraking, the LTST at the descending equator crossing was required to be between 2:00 and 4:10 PM, and preferably between 2:30 and 3:30 PM, so as to satisfy both power and science constraints. The achieved LTST was 3:04 PM.[9,10]

Mapping Orbit Phase. The primary science mission began on 19 February 2002, with a planned duration of 917 days. The near-circular frozen orbit, with a period of about 1.9 hours, was chosen to provide the observational geometries needed by the science instruments. In the baseline mission design, successive ground tracks were separated at the equator by approximately 28.8°; and the ground track pattern nearly repeated every two sols (25 orbits), with a 1° shift to the west. In addition, the ground track pattern nearly repeated every 30 sols (362 orbits), with a 1/4° shift to the east.[8,12]

Various perturbing effects could cause the orbit to wander from the frozen condition and the ground track to deviate from the desired pattern. AMDs were the largest such perturbing effect. Because of the configuration of the RCS thrusters, angular momentum desaturation could be carried out most efficiently in either of two orbital locations, which drifted slowly with time. The rate of accumulation of angular momentum was such that excessive reaction wheel speeds could be avoided by scheduling AMDs roughly once per day. With the thruster pairs not fully coupled, each daily AMD event resulted in between 2 and 5 mm/s of spacecraft ΔV. It was found possible to predict AMD ΔV effects to within 15% over one week, which resulted in significantly more accurate trajectory predictions than if no such modeling were performed. The fact that AMDs could be performed efficiently in either of two widely separated orbit locations allowed the selection to be such that GTW errors associated with 25- and 362-orbit near repetitions were satisfactorily controlled, with little need for dedicated orbit trim maneuvers.[12]

The navigation requirements during this mission phase were to provide short-term trajectory predictions and reconstructions to support the science observations and long-term predictions for sequence development. Accuracies of orbit predictions over a 21-day period (which were typically updated once per week) were required to be 20 km, 10 km, and 10 km or better in downtrack, crosstrack, and radial directions, respectively. Equator crossing times were to be predicted to within 6 s. This latter requirement turned out to be the most restrictive and was met 99% of the time through July 2003. These predicted equator crossing times were generally accurate to within 0.5 s over one week and 4 s over 20 days.[8,12]

Reconstructed trajectories were required to be accurate to within 1 km in each of three orthogonal directions. As of July 2003, 97% of reconstructed trajectories were accurate to within 25 m, 14 m, and 2.6 m in downtrack, crosstrack, and radial directions (as determined from differences between overlapping orbit solutions).[12]

Throughout the mapping mission, the local mean solar time at the descending equator crossing was required to be between 3:45 and 5:30 PM and the orbit altitude between 355 and 460 km.[12]

Quantities estimated in the orbit determination process included the spacecraft position and velocity at an initial (or epoch) time, a subset of the gravity field coefficients, the base density of the exponential atmospheric drag model, a solar radiation pressure force scale factor, sinusoidal downtrack spacecraft accelerations (to help account for unmodeled dynamics), and scale factors on ΔVs associated with AMD events.[12]

The noise levels for two-way and three-way Doppler data were typically better than 0.08 mm/s. One-way Doppler data were not usable for orbit determination, since the spacecraft lacks an ultra-stable oscillator.[12]

A 75th degree and order spherical harmonic gravity field model (MGS75E),[13] truncated to 65x65, was used for orbit determination. The spacecraft was modeled as a collection of five flat plates, to account for solar radiation pressure. For trajectory reconstruction analyses, an exponential atmospheric drag model was used. Drag was ignored for trajectory prediction analyses, due to its uncertainty and its magnitude relative to other perturbing forces acting on the spacecraft.[12]

The Mars Odyssey spacecraft was used to provide UHF communication relay-link support to the Spirit and Opportunity rovers in the days after their January 2004 landings on Mars. To achieve the proper geometry, an OTM of 0.543 m/s was executed on 22 November 2003 to change the Mars Odyssey orbit period by 3.25 s.[5]

Mars Exploration Rovers

Interplanetary Phase. The MER-A and MER-B spacecraft were launched toward Mars on June 10 and July 8 of 2003, respectively. The spacecraft were subsequently targeted to land on opposite sides of Mars: inside Gusev Crater (at 14.6° south latitude and 175.3° east longitude) in the case of MER-A and in the Meridiani Planum region (at 2.0° south latitude and 354.1° east longitude) in the case of MER-B.[14]

The primary navigation challenge during the interplanetary flight was to deliver each spacecraft to the desired atmospheric interface point (defined to be at a Martian radius of 3522.2 km) with sufficient accuracy that the landed vehicle would touch down (at the 99% probability level) within a specified landing ellipse (about 70 km x 5 km in size), determined to be safe for landing and also judged to be scientifically interesting. To land within the targeted landing ellipse, precise control of the inertial entry flight path angle (relative to a nominal value of -11.5°) was required: ±0.12° (3σ) for MER-A and ±0.14° (3σ) for MER-B. These error levels were significantly more stringent than those associated with the prior Mars Pathfinder and Mars Polar Lander missions discussed in Ref. 2 and references therein. The achievement of the required accuracies was aided by the development of new processes and software for orbit determination, propulsive maneuver design, and EDL trajectory simulation.[14,15]

The design of the MER flight system was an adaptation of the Mars Pathfinder spacecraft design. During flight, the spacecraft was spin stabilized, with a nominal spin rate of 2 rpm. The MER flight system consisted of four major components: a cruise stage, an EDL system or aeroshell (consisting of a heatshield and a backshell), a lander, and a rover (Spirit in the case of MER-A and Opportunity in the case of MER-B). Direct-to-Earth communications throughout the mission were carried out at X-band, using electronics located on the rover. During the interplanetary flight, low-gain and medium-gain antennas located on the cruise stage were used for transmitting and receiving signals.[14]

Radio metric data accuracies observed in flight were typically 0.015-0.035 mm/s (1σ) for Doppler and 1 m (1σ) for range (with a 10-minute integration time). ΔDOR data accuracies were typically about 0.04 ns, or 1.5 nanoradians. By the final approach phase, the processing time for ΔDOR measurements (the total time from recording of raw data to delivery of a tracking data file to the radio metric data conditioning team) had been reduced to eight hours or less.[14]

The quantities estimated in generating orbit determination solutions included spacecraft position and velocity at an epoch time, solar radiation pressure parameters, velocity changes from TCMs and attitude control events, space plasma effects on the radio signal, and range biases for each DSN site. In addition, a number of stochastic parameters were estimated. These stochastic parameters included tropospheric and ionospheric transmission effects on the radio signal and Earth rotation and polar motion parameters. Mars and Earth ephemerides, tracking station locations, and quasar locations were “considered” as error sources.[14,15,16,17]

Six TCMs were planned during cruise and approach for each spacecraft. TCM-1 was executed 10 days after launch in each case. Its purpose was to correct injection errors and to remove the deliberate bias in launch vehicle targeting included to reduce to below 10^{-4} the probability that the launch vehicle upper stage would impact Mars. Certain telecommunication, power, and thermal constraints limited the spacecraft pointing direction during TCM-1. TCM-1 for MER-A was executed in “vector” mode, with no turning of the spacecraft needed. It involved a continuous axial burn (along the spin axis), followed by a sequence of lateral burn pulses during successive spin cycles of the spacecraft. The burn was targeted centrally among the landing sites then under consideration. TCM-1 for MER-B was targeted to the preferred Meridiani Planum landing site. This maneuver was executed by turning the spacecraft to the appropriate spin axis orientation and then performing an axial burn – a more propellant-efficient mode than the vector mode.[14,18,19]

Subsequent to TCM-1, an ACS/NAV characterization activity was performed with each spacecraft. This allowed the determination of how much translational ΔV would result from firing thrusters to turn the spacecraft. The translational ΔV would be nominally zero, since the thrusters were fired as couples in executing turns; but thruster misalignments and thrust imbalances would result in a nonzero ΔV. Understanding this behavior was important because of its effect on spacecraft navigation accuracy. Fortunately, the translational ΔVs observed were significantly smaller than what had been assumed prior to launch.[14,16,17]

TCM-2 was executed at launch plus 52 days for MER-A and at launch plus 62 days for MER-B. Its purpose was to correct execution errors associated with TCM-1 and to target to the proper entry aimpoint, which had now been updated for MER-A to be consistent with landing at Gusev Crater. These (and all subsequent) TCMs were executed in vector mode.[14,19]

TCM-3 was executed at entry (E) minus 50 days for MER-A, for the purpose of correcting flight path errors remaining after TCM-2. After TCM-3, flight path errors had been reduced sufficiently that MER-A was, for the first time, on a Mars impact trajectory. TCM-3 for MER-B, scheduled for E - 65 days, was cancelled. MER-B was already on an impact trajectory after TCM-2, and TCM-3 would have been quite small. While there were certain navigational advantages to executing a small TCM-3, a heavy flight operations workload motivated a project decision to cancel the maneuver.[14,19]

TCM-4 was executed at E - 8 days for each spacecraft, to clean up orbit determination and maneuver execution errors propagated forward from the previous TCM and to target for precise entry conditions. Entry flight path angle (nominally to be held fixed at -11.5°) and latitude and longitude of the landing point were the quantities to be controlled in the execution of TCM-4. In the case of MER-A, it turned out that the execution of a vector mode maneuver involving both axial and lateral components would result in an unsatisfactory landing site if the first component (either axial or lateral) were successfully performed, while the second component, for some reason, failed to execute. Instead, the safer approach of performing a vector-mode lateral-only maneuver was adopted. This would allow full correction of the latitude and longitude of the landing site, while incurring a small, but acceptable, error in entry flight path angle. For MER-B, TCM-4 was somewhat larger than for MER-A, due to the earlier cancellation of TCM-3. TCM-4 consisted of both a continuous axial burn and a sequence of lateral burn pulses, executed in vector mode. This maneuver mode allowed entry flight path angle and latitude and longitude of the landing point all to be controlled.[14,19]

TCM-5 was scheduled for E - 2 days for each spacecraft. In each case, this maneuver was cancelled. For MER-A, the predicted landing point was only 2.3 km from the target. The entry flight path angle was estimated to be -11.486°±0.028° (3σ). The landing ellipse (at the 99% probability level) was 63 km x 3 km, with the size of the ellipse now dominated by atmospheric and spacecraft aerodynamic modeling uncertainties, rather than navigation errors at atmospheric entry. This error ellipse was slightly smaller than that used for pre-flight design purposes. For MER-B, the predicted landing point was 9.6 km from the target. The entry flight path angle was estimated to be -11.465°±0.034° (3σ). The landing ellipse (at the 99% probability level) was 61 km x 4 km, with the size of the ellipse again dominated by atmospheric and spacecraft aerodynamic modeling uncertainties. This error ellipse was also slightly smaller than that used for pre-flight design purposes. The execution of TCM-5 offered a potential science benefit for MER-B by shifting the landing site to a more interesting location. However, MER-A was experiencing a flash memory problem at this time; and the project preferred to devote flight team resources to addressing this issue, rather than performing a TCM for MER-B.[14,15,19]

TCM-6 was scheduled for E - 4 hours for each spacecraft and was for the purpose of correcting any gross modeling errors that might have become evident as the flight path became significantly influenced by Mars' gravity. For both spacecraft, this maneuver was cancelled as unnecessary, with little change observed in the predicted entry flight path angle and landing point since the decisions to cancel TCM-5.[14,15,19]

The orbit determination data cut-off times varied for the six scheduled TCMs. For the first three, it was five days before scheduled execution. For the fourth and fifth, it was 13 hours before scheduled execution. For TCM-6, it was 4.6 hours before scheduled execution.[14,19]

Chances to provide ground-based updates to EDL parameters stored on board were provided at E - 6 days, E - 28 hours, and E - 2.7 hours. On some of these occasions, updates were made, based on changes in the spacecraft's estimated flight path or refinements to atmospheric and aerodynamic modeling. On other occasions, updates were deemed unnecessary.[14]

A final orbit determination update prior to atmospheric entry for each spacecraft included tracking data through the termination of two-way tracking at E - 105 minutes. Again, there were only small changes in the estimated spacecraft trajectory and landing site. At this time, the entry flight path angle error was estimated to be $-0.007°\pm0.010°$ (3σ) for MER-A and $+0.030°\pm0.021°$ (3σ) for MER-B. These flight path angle errors correspond to errors in the magnitude of the impact parameter (B)[1] of only 180 m for MER-A and 750 m for MER-B.[14,15]

In addition to the standard dynamical and measurement models used in prior planetary missions, a high-fidelity solar radiation pressure model was created, the spin signature was removed from the tracking data, the DSN station locations were resurveyed, and a model of interplanetary charged particles was developed for the MER mission. These efforts, in combination with the dynamically quiet nature of the spacecraft, allowed the achievement of unprecedented delivery accuracies at Mars.[17]

During the last two months before entry, 12185 orbit solutions were produced, using 108 filter cases and seven data arcs for each vehicle. The interpretation of the information generated was aided by the automation of statistical tests, display generation, and report generation.[16]

Entry, Descent, and Landing. Tracking of each spacecraft subsequent to E – 105 minutes was in a one-way mode. While not highly accurate, this one-way tracking provided qualitative verification (through real-time Doppler residual displays) that a number of critical events had taken place, including transition from medium-gain to low-gain antenna (at E - 105 minutes), turn to entry attitude (at E – 85 minutes), heat rejection system venting (at E – 40 minutes), cruise stage separation (at E – 15 minutes), onset of atmospheric deceleration (at E), and parachute deployment (at roughly E + 4 minutes).[14]

MER-A and MER-B landed successfully on Mars on 4 January 2004 and 25 January 2004, respectively. Length limitations associated with the present paper make it necessary to defer a full discussion of EDL and post-landing navigation activities to a follow-on paper.

EXPLORATION OF THE OUTER PLANETS

Galileo

The Galileo mission has been described in Ref. 2 and references listed therein, through the prime science mission and the Galileo Europa Mission. Subsequent to the latter mission extension, there was a Galileo Millennium Mission, a preliminary design for which is presented in Ref.

20. Of particular interest in this mission extension was the use of solar gravitational perturbations to decrease the periapsis distance and the 163-km altitude flyby of the moon Amalthea that was achievable on 5 November 2002 as a result.

In total the Galileo spacecraft made 34 orbits of Jupiter, usually flying close to one of its four major moons during each loop around the planet. The mission ended on 21 September 2003, when the spacecraft plunged into Jupiter's atmosphere. This planned maneuver eliminated the risk of an eventual, unwanted impact with the moon Europa.

Cassini

The travels of the Cassini spacecraft through the inner solar system have been described in Ref. 2 and references listed therein.

Jupiter Flyby. Four TCMs were planned for the flight from Earth (18 August 1999) to Jupiter (30 December 2000). Of these four planned TCMs, only the first two were executed. The first was executed on 31 August 1999, to correct for orbit errors subsequent to (and magnified by) the Earth flyby. The next TCM was executed on 14 June 2000, in part just to meet the requirement that a main-engine burn of at least 5 s be executed at least every 400 days. This TCM placed the spacecraft on a sufficiently accurate trajectory relative to Jupiter that the last two pre-encounter TCMs could be cancelled.[21,22]

During the first 150 days of the flight from Earth to Jupiter, the spacecraft declination was within 5° of zero, limiting the effectiveness of Doppler tracking data. By mid-May 2000, however, the declination had increased to 20°.

The availability of range data and the integration times used for their collection (5-30 minutes) were dependent on which spacecraft antenna was being used to communicate with the Earth and the geocentric distance. Between 21 November 1999 and 1 February 2000, no range data were collected, because the low-gain antenna could no longer produce an adequate signal and the high-gain antenna was not used, other than briefly, until the latter date.[21]

The heliocentric range of the spacecraft increased from 1 AU to 5 AU during the flight to Jupiter. This made it easier to distinguish between spacecraft nongravitational accelerations due to solar radiation pressure, falling off with the inverse square of heliocentric distance, and the nearly constant accelerations due to radioisotope thermoelectric generator (RTG) thermal radiation, than had been the case earlier in the mission.[21]

Estimated parameters for orbit determination (specifically, for orbit reconstruction) included the spacecraft position and velocity, nongravitational accelerations, 82 ΔVs (some one-dimensional and some three-dimensional), and the ephemeris of Jupiter. Considered parameters included the ephemeris of the Earth-moon barycenter, tracking station locations, and transmission media calibration errors. Solar radiation pressure affected the spacecraft's motion directly, as a force, and indirectly, through the torque produced by a center-of-pressure/center-of-mass offset. This torque had to be compensated for, sooner or later, by firing RCS thrusters, which produced a net translational force on the spacecraft. When spacecraft attitude was maintained by firing RCS thrusters, an average nongravitational acceleration was estimated. When spacecraft attitude was maintained by reaction wheels, discrete, instantaneous ΔVs were estimated instead. Stochastic accelerations were estimated, in addition, to compensate for inaccuracies in solar radiation pressure and RTG thermal radiation models.[21]

The Cassini spacecraft flew past Jupiter at a periapsis altitude of 9,722,965 km, which imparted a heliocentric ΔV of 2.2 km/s and bent the spacecraft's trajectory by 12.2°. The actual and targeted spacecraft positions in the encounter B-plane[1] differed by 114 km ±1.5 km (1σ) and 160 km ±115 km (1σ) in two orthogonal directions.[21]

Flight from Jupiter to Saturn. A TCM to clean up orbit errors subsequent to the Jupiter flyby and satisfy the "flushing maneuver" requirement was executed on 28 February 2001. Later flushing TCMs were executed on 3 April 2002 and 1 May 2003. TCMs were executed on 10 September 2003 and 2 October 2003 to test a new RCS maneuver block, an energy-cutoff burn algorithm, and yaw steering.[22,23]

In preparation for the Cassini tour of the Saturnian system, the Voyager 1 and 2 trajectories near Saturn (discussed in Ref. 1 and references therein) were recomputed in the International Celestial Reference Frame, rather than the original B1950 reference frame, to facilitate the combined analysis of Voyager and Cassini data. This revised computation allowed a more accurate determination of Saturnian system gravity field parameters (such as the gravitational parameters of Saturn and eight satellites and several of Saturn's gravitational harmonic coefficients), due to improvements in modeling and data processing since the original computations. It also allowed a more complete utilization of Voyager imaging data in the development of Saturnian satellite ephemerides, since not all such data were used in the original analysis. In addition, the satellite ephemerides were obtained from a dynamically complete numerical integration of the satellite orbits, rather than analytic theories.[24]

In addition to a standard X-band downlink (at 8.4 GHz) that is coherent with an X-band uplink (at 7.2 GHz), the Cassini spacecraft is capable of transmitting two Ka-band downlinks (at 32.0 GHz), one coherent with the X-band uplink and the other coherent with a Ka-band uplink (at 34.3 GHz). (The downlink-to-uplink frequency ratios for these three links are 880/749, 3344/749, and 14/15, respectively.) These Ka-band links were included to allow radio science experiments measuring the gravitational deflection of radio signals passing near the sun and searching for low-frequency gravitational waves to be performed. However, the processing of the three independent downlinks also allowed the demonstration of dramatic reductions in the solar plasma-induced noise in Doppler tracking data during solar conjunctions. This multifrequency plasma calibration scheme could be used only for Doppler data collected at the Goldstone, California, Deep Space Station 25, since this is the only DSN station that can generate a Ka-band uplink. During the June-July 2002 conjunction, the standard deviation of Doppler frequency residuals was reduced up to a factor of 200 over uncalibrated two-way X-band data. An additional factor of three improvement in Doppler residuals was obtained using a system of water vapor radiometers, digital pressure sensors, and microwave temperature profilers to calibrate path delays due to dry and wet tropospheric effects.[25]

In addition to the Doppler and range data collected throughout the Jupiter-to-Saturn cruise, optical navigation images of Titan and Saturn's icy satellites were acquired beginning 6 February 2004. On 27 May 2004, a 34.7 m/s deterministic maneuver was executed to target a 2000-km flyby of Saturn's satellite Phoebe, while allowing for an efficient subsequent targeting of a safe crossing of Saturn's ring plane. This flyby took place on 11 June, at a flyby distance of 2071 km. Orbit determination data taken up to five days before closest approach were used to update instrument pointing during the flyby.[22,23]

The next TCM, of 3.7 m/s and executed on 16 June, targeted the spacecraft to pass 158,500 km from the center of Saturn at the ascending ring plane crossing, between the F and G rings, while avoiding possible debris near the orbits of the satellites Janus, Epimetheus, and Mimas. The spacecraft was captured into a 116-day orbit about Saturn by means of a 627-m/s Saturn Orbit Insertion (SOI) maneuver, executed on 1 July 2004 with orbital energy, rather than time or ΔV, used as a termination criterion. One-way noncoherent Doppler data were acquired during the SOI burn. The burn execution was accurate enough that a clean-up maneuver, scheduled for two to three days later, was cancelled. The descending ring plane crossing took place safely at 158,800 km from the center of Saturn, with no explicit targeting required.[22,23] The subsequent Cassini satellite tour and Huygens probe delivery lie beyond the time period covered by this paper.

EXPLORATION OF COMETS AND ASTEROIDS

Near Earth Asteroid Rendezvous (NEAR) Shoemaker

The first several years of the NEAR mission, including flybys of the main-belt asteroid 253 Mathilde and the Earth and an initial attempt to rendezvous with the near-Earth asteroid 433 Eros, which resulted in a flyby instead, have been described in Ref. 2 and references listed therein.

Rendezvous with Eros. After two TCMs in October and December 1999 to target the redesigned Eros rendezvous point, an 11.3-m/s rendezvous maneuver was executed on 3 February 2000. After a TCM on 8 February, to adjust the approach speed to accommodate certain planned science observations, insertion into orbit around Eros occurred on 14 February 2000. By means of this 10.0-m/s insertion maneuver, the spacecraft entered a 321 x 366 km orbit around Eros and became the first spacecraft to orbit a minor body of the solar system. Initial estimates of some of the physical parameters of Eros derived from the December 1998 flyby were helpful in planning the orbit insertion maneuver.[26,27,28,29,30]

Orbiting of Eros. It was known through simulations and covariance analyses carried out in advance that the orbit determination process for the orbiting of Eros would have to be different from that used for previous planetary orbiters about much larger and more symmetric natural bodies. Changes in strategy included the need to simultaneously estimate the spacecraft orbit, spacecraft nongravitational accelerations, and pertinent physical parameters of Eros, as well as the need to augment DSN Doppler and range tracking with optical landmark tracking.[30]

The radius of the orbit was brought down in stages, using a sequence of orbit correction maneuvers to first reduce periapsis distance and then circularize the orbit, to a roughly 200 x 200 km orbit on 3 March, a roughly 100 x 100 km orbit on 11 April, and a roughly 50 x 50 km orbit on 30 April. From 1 May to 26 August, the spacecraft remained in a roughly 50 x 50 km orbit, except for most of July, when the orbit was modified to be roughly 35 x 50 km or 35 x 35 km in size.[27,30]

Physical parameters of Eros, such as its mass, gravity field, shape, spin state, and rotation pole orientation, were determined with increasing accuracy during this orbital progression. The information gleaned from each larger orbit allowed for subsequent use of a smaller orbit, with the resulting orbital behavior adequately understood in advance. The orbit phase of the mission was redesigned multiple times as the physical properties of Eros became better known. The low mass of Eros resulted in low speeds for an orbiting spacecraft, so that a small but poorly planned velocity change could result in either impact or escape. In addition, large values of oblateness and ellipticity coefficients for the irregularly shaped asteroid tended to produce large transient and secular changes in spacecraft orbital elements relative to Eros, again resulting in possible impact or escape.[30,31,32,33]

The inclination of the spacecraft's orbit plane relative to the equator of Eros began at 35° on 14 February, gradually increased to 90° by 30 April, and remained there through 8 August. Over the remainder of the orbital operations, the inclination gradually increased to 179°. Orbits retrograde to the asteroid's equator provided improved orbital stability. (The direct orbits employed prior to 30 April were high enough to avoid excessive instability.) From 26 August 2000 to 12 February 2001, the spacecraft orbits varied from roughly 200 x 200 km to 35 km x 35 km, with intermediate elliptical orbits used to transfer among the various near-circular orbits. In addition, certain roughly elliptical orbits were used to produce low-altitude flybys of Eros (as close as 2.74 km) on 26 October 2000 and in late January 2001.[27,30,32]

Two-way Doppler and range data were weighted at 0.1 mm/s (for a 60-s count time) and 200 m, respectively, during the orbit phase. The deweighted range data were used largely for adjusting the ephemeris of Eros; the Doppler data were much more important for spacecraft orbit determination.[30]

The optical landmark tracking process for the NEAR Shoemaker mission consisted of first identifying and characterizing a set of landmark craters on Eros, so as to construct a landmark database, and then finding and using those same landmarks in subsequent pictures as tracking data. Many images were collected, with landmarks viewed from various directions, distances, and lighting conditions as the spacecraft orbited Eros and Eros rotated about its spin axis. The processing of a large number of such images in the orbit determination filter allowed both the spacecraft orbit and the landmark locations to be estimated. The building and maintenance of the landmark database was an ongoing process throughout the orbit phase, as the portion of Eros illuminated by the sun gradually changed. The database ultimately contained 131,337 useful observations of 1,590 valid landmarks, in this first ever operational use of optical landmark tracking in a deep space mission. Body-fixed coordinates of the landmarks were ultimately obtained to a typical accuracy of 1 m (1σ).[30,34,35]

The NEAR Laser Rangefinder instrument, with a range of more than 50 km, could be used to provide data for spacecraft orbit determination, alone or in combination with other tracking data, as well as to solve for an accurate shape model of Eros. The laser range data were never used operationally in the former sense, but still provided helpful consistency checks on orbits determined using DSN radio metric and optical landmark tracking data. The use of laser range data to construct a shape model allowed a preliminary gravity harmonic model for Eros to be developed (by assuming a uniform density). In addition, an accurate shape model allowed the more accurate cataloguing of landmark locations.[30,36]

As time went by in orbit about Eros and the estimates of its physical parameters improved, the accuracy with which spacecraft position could be predicted two to five days into the future improved from over 1 km to 10-20 m or better.[30]

25 orbit correction maneuvers (OCMs) were carried out during almost a year of orbital operations near Eros, at a ΔV cost of 17.2 m/s. These OCMs were designed to control the angular momenta of the four spacecraft reaction wheels, in addition to their primary function of controlling the spacecraft's orbit. OCMs were usually constrained to be at least one week apart, to simplify flight operations.[26,27,30]

Landing on Eros. On 12 February 2001 the spacecraft was deorbited from a 36 x 36 km orbit and allowed to descend toward the surface of 33 x 17 x 12 km Eros. The deorbit maneuver made a substantial change in orbit inclination to target the desired landing area, in addition to depleting orbital energy to allow impact. Four braking maneuvers were used during the descent to control the vertical and horizontal speeds at touchdown. The maneuvers were executed at altitudes ranging from about 5 km to a few hundred meters. The timing of the second braking maneuver needed to be adjusted after the first braking maneuver was executed, based on an orbit estimate derived from radio metric and optical tracking data collected between the maneuvers. The deorbit maneuver and the four braking maneuvers involved a total ΔV of 23.3 m/s.[27,37]

The descent took place over 4.5 hours. Because there was insufficient time to program and completely check on-board algorithms for landing and because of flight computer memory limitations, the landing process was executed with preprogrammed open-loop burns. The fact that the fixed-mounted telecommunication antennas, solar arrays, and scientific instruments all needed to point in particular directions (or a limited range of directions) in order to operate complicated the planning of the descent path. Because a spherical harmonic representation of the gravity field of an irregular body such as Eros diverges inside of the sphere circumscribing the body, a polyhedral gravity field, based on the shape model for Eros (derived from a spherical harmonic shape representation to degree and order 34), was used for integrating the spacecraft trajectory within this sphere. Descent images were returned down to an altitude of 125 m. At this first spacecraft landing on one of the solar system's small bodies, the vertical speed was 1.5-1.8 m/s; and the transverse speed was 0.2-0.3 m/s.[27,37]

Deep Space 1

The primary Deep Space 1 (DS1) mission (1998-1999) and pertinent technology demonstrations carried out have been described in Ref. 2 and references listed therein.

In November 1999, early in an extended mission following the July 1999 flyby of the asteroid 9969 Braille, the DS1 stellar reference unit failed. The spacecraft was subsequently placed in a safe-hold configuration for seven months. During this time, new software and techniques were developed to allow the science camera to serve as a replacement star tracker. In June 2000, software modifications were loaded on board; and thrusting with the ion engines was resumed to achieve a flyby of comet 19P/Borrelly.[38,39]

Cruise and Approach Navigation. The autonomous navigation (autonav) system that had been demonstrated during the cruise to Braille could not be used for autonomous orbit determination during the cruise to Borrelly, because the science camera was no longer available for this purpose. Thus, ground-based radio metric navigation was performed instead. The autonav system continued in use, however, for the new purpose of processing science camera images to extract star locations for attitude determination.[38,39]

The ground-based orbit determination relied on the two-way Doppler and range data that were collected during the one or two tracking passes that occurred each week. To avoid interrupting the scheduled ion propulsion system (IPS) thrusting in order to point the high-gain antenna at the Earth and the expenditure of hydrazine needed to rotate the spacecraft back and forth, many of these tracking passes made use of one of the spacecraft's low-gain antennas, which allowed the collection of Doppler, but generally not range, data.[39]

The use of the science camera as a star tracker also imposed constraints on thrusting directions for the IPS. Only a limited number of relatively bright stars could now be used as attitude references, so that the actual thrusting directions could be chosen to approximate only roughly the most favorable directions. In addition, there were several occasions on which the new star tracking software lost its inertial reference star for several days, during which time the IPS thrust vector gradually shifted away from its intended direction. (This loss of lock was sometimes caused by noise in science camera images due to the impacts of high-energy particles from coronal mass ejections, for example.) It was important after such misdirected thrusting to be able to determine the new spacecraft trajectory quickly, so as to modify subsequent thrusting to lead once again to a Borrelly flyby.[38,39]

Comets are subject to relatively large nongravitational forces due to outgassing. Thus, their ephemerides can be relatively difficult to predict even for short periods into the future. An intensive ground-based observational campaign was undertaken to improve the ephemeris of Borrelly, beginning with its first encounter-apparition detection during May 2001. Over 200 observations were obtained from various observatories in the United States and Australia. These observations were used, either alone or in combination with later on-board observations, to improve the cometary ephemeris.[38,40]

Once Borrelly had first been sighted with the on-board camera, optical data became a dominant data type for orbit determination and were relied upon heavily to target the spacecraft to its flyby aimpoint. A month prior to encounter, Borrelly was not bright enough to show up in any single imaging frame. Thus, several imaging frames were added together to produce a composite image. The individual frames were aligned by locating a minimum of two stars in each, to allow a determination of image boresight direction. The location of the comet was then isolated to within a region 20-40 picture elements in size in each frame, and these regions were added together to produce a cometary image of detectable signal-to-noise ratio. The first usable set, consisting of 12 co-added frames, was obtained on 29 August 2001. The resulting cometary ephemeris derived from on-board images disagreed with that based on ground-based observations by the larger than expected amount of 1500 km. The same apparent mismatch was evident in four-frame composite images obtained on 7 September.[38]

In order to help resolve this discrepancy, ΔDOR data were collected and found to confirm the accuracy of the determination of the spacecraft's position relative to the Earth based on Doppler and range data. Ultimately, it was found that associating the comet's position in a ground-based image with the brightest pixel, rather than using a Gaussian fit to the brightness profile, resulted in better agreement with the spacecraft observations. Improvements were also made to the ground software model of cometary outgassing and the resulting nongravitational accelerations experienced by the comet.[38,39]

Once the comet became bright enough to be visible in individual frames (15 September), the multi-frame addition technique was abandoned. The spacecraft was guided by referencing its position and target aimpoint to Borrelly, rather than a location in inertial space. In each orbit determination solution, a best-fit trajectory was first obtained using radio data alone. Then, optical data were used to shift the spacecraft position estimate, with the velocity estimate unmodified.[38]

TCMs on approach to Borrelly were carried out using the IPS, without use of the hydrazine thrusters. The calculation of the appropriate TCMs was complicated by the fact that the IPS was being used to maintain spacecraft attitude (to conserve hydrazine propellant, at least until 14 September), which resulted in net translational forces also. These translational forces were difficult to predict because the attitude corrections that would be needed in the future were themselves difficult to predict. In addition, just as the thrust vector directions for achieving major trajectory modifications were now constrained by the need to have a star of sufficient brightness within the field of view of the science camera for attitude maintenance purposes, the thrust vector directions for TCMs were similarly constrained. The desire to keep the high-gain antenna pointed at the Earth as encounter became closer imposed additional restrictions on targeting conditions. Of the several planned TCMs on approach to Borrelly, only four were actually executed – on 11 September, 17 September, 21 September, and 22 September 2001 (12 hours before encounter).[38,39]

Encounter Autonomous Navigation. The one aspect of the autonomous navigation system that remained usable after the diversion of the science camera to primary use as a star tracker was the closed-loop on-board tracking system. This on-board system was initialized with a final ground-based orbit determination solution about eight hours before encounter (containing data through 12 hours before encounter). This ground-based orbit solution was accurate to about 15 km perpendicular to the relative flight path and to better than 200 km along the flight path. The first images of the comet were recorded 1.5 hours before encounter and processed on board. The resulting information was not actually used on board, but was merely sent to the ground to provide evidence that the on-board system was working properly.[38,41]

Beginning at 32 minutes before encounter, images of Borrelly were recorded roughly every 30 s, with the comet center location computed and stored. These images were recorded using the science camera's charge-coupled device photodetector, rather than the active pixel sensor, which had been used during the Braille encounter. Ten minutes before encounter, the accumulated center locations were used to estimate the spacecraft's position relative to the comet, along with gyro biases and drift rates perpendicular to the boresight direction. Subsequently, the trajectory and attitude estimates were updated with the acquisition of each image. This information was passed along to the attitude control subsystem, to allow it to point the camera in the proper direction to capture additional images of Borrelly. The final image was shuttered 166 s before encounter. The cometary nucleus drifted off center in some of the later images, most likely due to latency in the response of the attitude control subsystem; however, all planned images were obtained. The spacecraft passed the cometary nucleus at a distance of 2171 km (as planned), at a speed of 16.6 km/s.[38,41]

Stardust

Early Cruise and Earth Flyby. The Stardust spacecraft was launched on 7 February 1999. Early in the mission, the Navigation Team observed stochastic spacecraft accelerations of 10^{-11} to 10^{-9} km/s^2, along with a constant acceleration bias of 1.0-3.0 x 10^{-11} km/s^2. These accelerations were attributed to some combination of solar radiation pressure and small ΔVs arising from unbalanced thruster firings used to control spacecraft attitude. Considerable difficulty was experienced in determining and predicting consistent estimates of the spacecraft's orbit, due to the difficulty of properly modeling these nongravitational accelerations.[42]

While it was found important to develop a more sophisticated model for solar radiation pressure than was initially used, the major source of nongravitational acceleration mismodeling turned out to be associated with the ΔVs produced during a typical attitude control cycle of pointing at the sun, followed by a deadband walk to Earth pointing, followed by pointing at the Earth, followed by a deadband walk back to pointing at the sun.[42]

These difficulties were addressed by treating deadband walk, Earth pointing, and sun-pointing spacecraft modes separately and constructing characteristic stochastic batch sizes and levels of process noise for each mode. Obtaining telemetry information on each ΔV event, rather than summed over several events; doubling the time to execute a deadband walk to Earth pointing; and developing a variety of special-purpose utility programs all resulted in force model accuracy improvements. Constant solar radiation pressure parameters and stochastic "small force" scale factors were estimated as part of the orbit determination process and subsequently used for orbit propagation.[42]

The accurate prediction of small force activity was critical in preparing for an Earth gravity assist flyby on 15 January 2001, as well as for events later in the mission. The Earth flyby was used to boost the spacecraft aphelion and change the orbit inclination.[42]

Annefrank Flyby. On 2 November 2002, Stardust flew within 3076 km of asteroid 5535 Annefrank at a relative speed of 7.2 km/s. A principal purpose of this flyby was to perform an engineering test of flyby events to be carried out in a subsequent cometary encounter, including the use of an autonomous navigation system derived from that of the DS1 mission. As in the DS1 mission, simplified orbital dynamics were used in the on-board estimator, with rectilinear motion of the spacecraft relative to the asteroid assumed and no numerical integration needed. Three components of spacecraft position and three attitude corrections were estimated. The on-board estimator was initialized 20 minutes before encounter using a radio-based spacecraft orbit determination solution and a ground-based asteroid ephemeris, the latter being accurate to better than

80 km (1σ). Optical data taken between 38 and 12 hours before encounter did not show the asteroid, due to its high phase angle (150°) and small size (3 km) and thus could not be used to initialize the on-board estimator. Optical observations of the asteroid were made roughly every 30 s during the 20 minutes before encounter. A number of observations were collected before the spacecraft position and attitude estimates were first updated, 10 minutes before encounter. Subsequently, updates were made after the processing of each new measurement. The asteroid was successfully tracked through closest approach, with the closest image taken less than a second before.[41]

Wild 2 Flyby. A sample-collection encounter with comet 81P/Wild 2 took place on 2 January 2004 at a distance of 236 km and a relative speed of 6.1 km/s. The spacecraft's 16 thrusters were all mounted on one side of the spacecraft, to avoid contaminating the aerogel collector grid. However, this arrangement resulted in unbalanced forces acting on the spacecraft whenever the thrusters were fired for attitude maintenance or modification. As noted above, the accurate modeling of these effects was a challenge throughout the mission. Scale factors applied to thruster firing events in the orbit determination process were assumed uncertain to 15% (1σ) a priori. Predictions of future forces were frequently updated based on the most recent trends and anticipated changes in spacecraft attitude.[43]

The navigation camera, essential for generating optical navigation data during the month and a half leading up to the Wild 2 encounter, suffered from contamination of its optical surfaces at times during the mission, causing the image of a point source to spread over an extended area, with its peak intensity substantially diminished. This effect could be mitigated to some degree by turning on heaters inside of the navigation camera or rolling the spacecraft to place the camera radiator in sunlight. In addition, the scan mirror, needed to view the comet when the spacecraft was oriented so as to shield sensitive components from dust impacts near encounter, produced images with a pronounced stray light background. This problem was addressed by imaging Wild 2 without using the scan mirror/periscope until 14 hours before encounter and first relying on the periscope only 30 minutes before encounter, at which time the cometary nucleus was bright enough to stand out relative to the stray-light background.[43]

There were two solar conjunction periods between the Annefrank and Wild 2 encounters, with the sun-Earth-probe (SEP) angle less than 10 degrees during more than 56% of the time between encounters, resulting in radio metric data of diminished quality. Doppler and range data were weighted at 2-15 mm/s and 3-14 m when the SEP angle was small and 0.1-0.8 mm/s and less than 3 m when this angle was larger, as was the case around encounter. The spacecraft's transmission of thruster events data to the DSN was delayed by the second solar conjunction period.[43]

Over the first several years of the mission, deep space maneuvers were executed in January 2000, January 2002, and June 2003. The third of these targeted the spacecraft for a Wild-2 flyby distance of 150±50 km, based on ground-based observations of the comet thru June 6. The 71-m/s maneuver was split into two parts and executed on June 17 and 18. The mission requirement on flyby distance was later changed to 300±50 km based on early optical navigation images and finally to 250±50 km based on better optical navigation images a few days before encounter.[44]

Four TCMs were scheduled during the last 30 days before encounter, to correct the spacecraft's trajectory as better comet-relative orbit solutions were obtained. The first three of these TCMs (at encounter minus 30 days, ten days, and two days) were executed; but the final one was not needed. To ensure spacecraft safety and avoid contamination of the aerogel collector grid, the third TCM was chosen to be a roll-only maneuver, to adjust flyby distance but not time of flight. The latter would be compensated for by adjusting the encounter sequence start time. A contingency maneuver six hours before encounter was not needed.[44]

The autonomous navigation system was initialized 30 minutes before encounter with the best available ground-based orbit determination solution, based on spacecraft radio metric data, a ground-based cometary ephemeris, and optical navigation data. With the aid of an intensive ground-based observing campaign, the cometary ephemeris error had been reduced from 1100 km to 300 km (1σ) in the spacecraft's downtrack direction. With the final ground-processed optical navigation images taken 14 hours before encounter, the spacecraft position uncertainty relative to the comet was about 5 km in the crosstrack directions. Images were shuttered every 30 s until six minutes before encounter and then more frequently through encounter. The first filter update occurred 10 minutes before encounter. A spacecraft roll maneuver was executed six minutes before encounter, to allow the scan mirror to rotate about the proper axis near closest approach. The comet stayed within the camera's field of view throughout the encounter, with the closest image shuttered three seconds before encounter. The final spacecraft position updates calculated on board were 530, 20, and 11 km in the downtrack and two crosstrack directions.[41]

Missions flying past small natural bodies prior to Deep Space 1 and Stardust had captured images of the small bodies by collecting image mosaics of sufficient extent that the body was almost certain to appear somewhere within the mosaic. This was a relatively inefficient approach, however, since a large fraction of the images returned would contain no information of interest. The use of autonomous navigation in the Deep Space 1 and Stardust missions eliminated this inefficiency by allowing the spacecraft's camera to be accurately pointed at the small body in each image.[41]

Successful delivery of the sample return capsule at Earth atmospheric entry required controlling the entry flight path angle to $\pm 0.08°$ (3σ). The small ΔVs arising from unbalanced thruster firings used to maintain spacecraft attitude control made it challenging to meet this requirement. Tests of deadband walks, limit cycling, and simulated entry maneuvers were carried out in June and July of 2003, at a heliocentric distance of roughly 1 Astronomical Unit, to assess the likelihood of meeting the entry angle requirement. These tests resulted in changes in the way maneuvers on approach to entry would be carried out.[45] The return of the sample capsule to Earth on 15 January 2006 lies beyond the time period covered by this paper.

CONCLUSION

Deep space navigation capabilities, which had evolved enormously from the 1960s through the 1990s, continued to evolve thereafter, benefiting the eight planetary missions that have been described. Increases in computing power allowed more accurate orbit determination by permitting more detailed dynamical and measurement modeling and allowing large numbers of scenarios to be investigated. An improved delta-differential one-way ranging system was developed, and optical landmark tracking was put into operational use.

Unprecedented accuracies were achieved in delivering spacecraft to the top of the Martian atmosphere and subsequently to the planet's surface. Further experience was gained in using aerobraking to reduce long-period elliptical orbits about Mars to low-altitude, near-circular orbits, allowing a large saving in propellant usage. The orbiting of and landing on a small, irregular, natural body were achieved for the first time. Autonomous navigation was used to obtain high-resolution images of small natural bodies during flybys.

ACKNOWLEDGMENTS

The research described in this paper was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

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