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THE EVOLUTION OF DEEP SPACE NAVIGATION: 2006-2009*

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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 fifth in a chronological sequence dealing with the evolution of deep space navigation. The time interval covered extends from 2006 to 2009. 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 14 planetary missions.

INTRODUCTION

Four previous papers[1,2,3,4] have described the evolution of deep space navigation over the time interval 1962 to 2006. 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 third and fourth papers covered the portions of the Galileo, Near Earth Asteroid Rendezvous, Mars Global Surveyor, Cassini, Deep Space 1, Stardust, 2001 Mars Odyssey, Hayabusa, Mars Express, Mars Exploration Rover, Rosetta, MESSENGER, Deep Impact, Mars Reconnaissance Orbiter, and Venus Express missions that took place between 1999 and 2006. In addition, Butrica[5] has recently described the history of deep space navigation from the perspective of a professional historian.

The current paper extends Ref. 4 by three years. It covers the portions of the Cassini, Stardust-NExT, 2001 Mars Odyssey, Hayabusa, Mars Express, Mars Exploration Rover, Rosetta, MESSENGER, Deep Impact/EPOXI, Mars Reconnaissance Orbiter, Venus Express, New Horizons, Mars Phoenix Lander, and Dawn missions that took place between 2006 and 2009. 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.

EXPLORATION OF THE TERRESTRIAL PLANETS

2001 Mars Odyssey

The interplanetary, aerobraking, and primary science phases of the 2001 Mars Odyssey mission have been described in Reference 3 and references listed therein, as well as Reference 6. The primary Mars Od-

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yssey mission was completed in August 2004, and an extended mission began thereafter. The Mars Odyssey spacecraft, in a 400-km, near-circular, sun-synchronous orbit, provides telecommunication relay services to other spacecraft on or near Mars through a UHF transceiver operating at 437 MHz on the forward link and 402 MHz on the return link, as in the case of the Mars Phoenix Lander mission described below.[7]

Mars Express

The interplanetary and early operational orbit phases of the European Space Agency's Mars Express mission have been described in Reference 4 and references listed therein. The spacecraft remained in an eccentric, near-polar orbit, in an 11:3 resonance with the rotation period of Mars, through the majority of its nominal orbiting mission (completed in November 2005) and the first mission extension, culminating in November 2007.[8]

Five periapsis maneuvers in late 2007 increased the orbit period from 6.72 to 6.84 h, an 18:5 resonance with the rotation period of Mars. In early 2009, two additional maneuvers increased the period to 6.895 h (a 25:7 resonance). Precession of the periapsis over time allowed all sub-satellite latitudes to be viewed from relatively low altitudes.[8]

The Mars Express spacecraft was tracked by both ESA sites (New Norcia in Australia and Cebreros in Spain, primarily the former) and DSN sites (Goldstone and Madrid, primarily the former). Primary science data collection near periapsis precluded pointing the high-gain antenna toward Earth, so that tracking data were not available around that time. Orbit determination was typically performed weekly, based upon 10 days of tracking data, allowing three days of overlap between solutions. Thruster firings were used to off-load reaction wheel angular momentum near every fourth apoapsis. Mars Express ranging data have been used to improve the ephemeris of Mars.[8]

Mars Exploration Rovers

The interplanetary and entry, descent, and landing phases of the Mars Exploration Rover mission have been described in References 3 and 4 and references listed therein, as well as References 9, 10, and 11. The Spirit and Opportunity rovers have derived navigational information (involving both attitude and position) from accelerometers, gyros, wheel encoders, and cameras while traversing the surface of Mars. A detailed description of the techniques used is beyond the scope of this paper. Further information about surface navigation early in the rover missions may be found in References 12, 13, 14, and 15.

MESSENGER

The Earth flyby and Venus flyby 1 portions of the MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) mission have been described in Reference 4 and references listed therein, as well as Reference 16.

Venus Flyby 2. A 25-m/s trajectory-correction maneuver (TCM) was executed on 2 December 2006 to correct targeting errors from Venus flyby 1, 39 days earlier. A cleanup TCM scheduled for 24 January 2007, after solar conjunction, was not needed. TCMs correcting targeting for Venus flyby 2 were executed on 25 April (0.6 m/s) and 25 May (0.2 m/s). Unexpected, autonomous angular momentum dumps and resulting spacecraft velocity changes were avoided by designing momentum dumps into TCMs. The desire to keep the spacecraft's sunshade pointed within a specified angle relative to the sun placed constraints on the directions in which the latter two TCMs (and TCMs in general) could be performed and precluded control of the time of closest approach. The first of these burns ended prematurely due to anomalous behavior by the spacecraft's guidance and control system, necessitating that the second burn be replanned.[17,18]

Spacecraft orbit determination was carried out by processing two- and three-way Doppler, two-way range, and delta-differential one-way range (ΔDOR) data generated by the Deep Space Network (DSN). The ΔDOR data increased the accuracy and robustness of the orbit determination process. It also improved the estimation of parameters other than spacecraft position and velocity, such as solar radiation pressure model parameters.[17]

The second Venus flyby, on 5 June 2007, took place at a considerably lower altitude than the first (338 versus 2987 km). The spacecraft was delivered to within 6 km of its targeted aim point in the hyperbolic impact plane (or B-plane) and 2 s in time of closest approach. Optical navigation using the narrow- and wide-angle cameras was tested on departure from Venus, a noncritical time period that resembled, in time-reversed fashion, the upcoming first Mercury flyby, where it would be more important.[17,18,19]

Mercury Flyby 1. A deep space maneuver (DSM-2) of 227 m/s was executed on 17 October 2007, before entry into a seven-week solar conjunction period. (The DSM was executed about two weeks before its optimal execution time to avoid the conjunction.) The scheduling of ΔDOR tracks near DSM-2 and conjunction entry and exit allowed quick and accurate reconstruction of DSM-2 and accurate trajectory propagation toward encounter for planning a clean-up TCM (1.1 m/s), which was executed on 19 December after conjunction exit. In designing this TCM, the time of closest approach for the upcoming encounter was allowed to shift in order to allow a lateral orientation for the TCM. Careful analysis of the planned spacecraft attitudes and solar array orientations leading up to the Mercury encounter and the rescheduling of a solar array reorientation to take advantage of solar radiation pressure forces allowed approach TCMs scheduled for 10 and 13 January 2008 to be cancelled.[18,19,20,21,22]

The Mercury flyby involved the mission's first operational use of optical navigation. Individual images of the planet and stars had to be combined due to the large differences in brightness of these objects. Optical navigation images were used to confirm the accuracy of Mercury's ephemeris and rule out any need for a late contingency maneuver.[20]

Radio metric observation geometries at the first Mercury flyby were less favorable than for the second Venus flyby, in that the geocentric distance was greater, resulting in a lower signal-to-noise ratio and decreased range data precision; the encounter took place a month after superior conjunction; and the encounter was at a sufficiently southerly declination that Goldstone-Madrid station overlap opportunities for generating ΔDOR data were limited. The final targeting maneuver before the encounter was designed just after exit from a long solar conjunction and was based on data largely from within that conjunction. The smaller heliocentric range and numerous attitude changes and solar panel offsets relative to the sun increased the importance of solar radiation pressure modeling. Because the timely execution of TCMs was needed to maintain an adequate propellant margin and avoid violating certain spacecraft pointing constraints relative to the sun, a number of contingency maneuvers were planned in case the primary maneuvers did not execute as planned. The contingency maneuvers were ultimately not executed.[20,22]

The Mercury flyby took place on 14 January at an altitude of 201 km. The spacecraft was delivered to within about 10 km of its targeted aim point in the B-plane (with periapsis altitude accurate to 1.4 km) and 3 s in time of closest approach. A post-encounter TCM scheduled for 5 February was cancelled as unnecessary. After this and other planetary flybys, as well as DSMs, the remaining trajectory was reoptimized to accommodate trajectory uncertainties and maneuver execution errors.[18,19,20,22]

Mercury Flyby 2. A deep space maneuver (DSM-3) of 72 m/s was executed on 19 March 2008, with no subsequent clean-up maneuver needed. This DSM allowed the testing of the turn-while-burn maneuver execution mode that would be needed later for propellant-efficient insertion into orbit about Mercury.[18,19,20,22]

With the successful use of solar radiation pressure forces for trajectory control in Mercury Flyby 1, this procedure was used more extensively to control the encounter conditions for Mercury Flyby 2, by developing and adjusting a multi-month plan for articulating solar arrays and modifying spacecraft attitude, consistent with various power, thermal, and communication constraints. (The modeling of spacecraft body and solar array attitudes was extended in sophistication and accuracy several times during the interplanetary flight.) Similarly, solar radiation pressure torques were manipulated so as to minimize the need for thruster firings to unload accumulated reaction-wheel angular momentum. Consequently, three approach TCMs were cancelled (saving propellant and reducing operational risk); and the 6 October 2008 Mercury flyby (at 199-km altitude) was accurate to 2.6 km in the B-plane and 0.8 km in periapsis altitude. A departure TCM was cancelled also.[18,21,22]

Mercury Flyby 3. A deep space maneuver (DSM-4) was executed in two parts totaling 247 m/s on 4 and 8 December 2008. The second part (about 10% of the total) was carried out with a timed thrust cut-off, as a test of a backup mode for Mercury orbit insertion in the event that accelerometer data were unavailable. Continued use of the "solar sailing" technique successfully demonstrated on approach to the prior Mercury flybys allowed the cancellation of four scheduled TCMs before Mercury Flyby 3, which occurred on 29

September 2009 at 228-km altitude. This flyby was controlled to an accuracy of 3.5 km in the B-plane and 0.5 km in altitude.[18,22]

After the flyby, a corrective TCM was cancelled; and a final deep space maneuver (DSM-5) of 178 m/s was performed on 24 November 2009. The three Mercury flybys and DSMs that followed had the effect of achieving spacecraft heliocentric orbits that were resonant with Mercury's orbital motion in the approximate ratios of 2:3, 3:4, and 5:6 (with approach speeds to Mercury gradually decreasing). The Earth flyby (at 2348-km altitude), the two Venus flybys, and the three Mercury flybys produced heliocentric velocity changes of 6.00, 5.52, 6.94, 2.30, 2.45, and 2.84 km/s, all much larger than the ΔVs (velocity changes) for the deep space maneuvers that were needed to set up the flybys.[18]

Mars Reconnaissance Orbiter

Primary Science Orbit. The interplanetary and aerobraking phases of the Mars Reconnaissance Orbiter (MRO) mission and the establishment of an initial Primary Science Orbit (PSO) have been described in Reference 4 and references listed therein, as well as References 23, 24, 25, and 26. The PSO was initially established (in an approximate sense) and then refined between September and December of 2006, with 8 November considered to be the start of the mission's Primary Science Phase. The PSO was roughly a 255 x 320 km altitude, sun-synchronous (inclination of 92.65 deg) orbit with periapsis frozen over the south Martian pole and ascending node at 3 PM local mean solar time. The desired ground track pattern was to be generated by a 211-orbit cycle that walked 0.5 deg in longitude (32.5 km) west of the previous cycle every 16 sols (Martian solar days).[27,28]

Atmospheric drag uncertainties were the dominant error source for orbit prediction, with the MRO spacecraft orbiting Mars at lower altitudes than the Mars Global Surveyor and 2001 Mars Odyssey spacecraft (by 120-140 km), thereby incurring increased drag effects due to the tenuous upper atmosphere of Mars. Atmospheric variations were thus important to model. Several different versions of the Mars Global Reference Atmospheric Model were used at various times in flight operations and preparatory analyses. With the solar arrays and high-gain antenna free to articulate relative to the spacecraft bus and the spacecraft sometimes pointed at off-nadir targets, the area presented by the spacecraft to the upper atmospheric flow varied significantly over time.[23,27]

Through the first few months of the PSO, the navigation team used a 95th-degree and -order gravity field based on Mars Global Surveyor and Odyssey tracking data through 6 December 2004. Periodically throughout the subsequent science phase, the navigation team received preliminary gravity-field solutions that included MRO tracking from the project's Gravity Investigation Team.[27]

The spacecraft was equipped with 100 N-m-s reaction wheels and balanced, monopropellant thrusters for attitude control. Momentum desaturation events were generally spaced two days apart. However, even with nominally balanced thrusters and relatively infrequent desaturations, the residual ΔVs imparted due to misalignments and inconsistent thruster performance comprised a significant error source for trajectory prediction. For trajectory reconstruction, spacecraft telemetry contained thruster pulse information that could be used in trajectory integration. For trajectory prediction, the Attitude Control System (ACS) Team modeled the momentum and provided a file of predicted desaturation thruster pulses for the upcoming two to three weeks.[27]

DSN tracking (yielding one-, two-, and three-way Doppler data) was carried out for 12-16 h per day during the Primary Science Phase. Quantities estimated in the orbit determination process included spacecraft position and velocity, near-resonant gravity field harmonic coefficients ($J12$ and $J13$), orbit trim maneuver ΔV components (when applicable), solar pressure scale factor, drag scale factor (assumed to vary from one orbital revolution to the next), one-way Doppler bias and drift rate, and stochastic angular momentum desaturation scale factors. The one-way Doppler data were helpful in estimating drag scale factors for those orbits without two-way Doppler tracking. Tracking data residuals were minimized in a batch least-squares filter. Modeling of the displacement of the phase center of the high-gain antenna from the spacecraft's center of mass was necessary to minimize the Doppler residual noise.[23,27]

Accurate instrument pointing during the PSO required that long-term (28-day) orbit prediction errors be less than 195 km in the downtrack direction (so that planned off-nadir instrument pointing would be accurate to within 3 deg). Results of more than 200 orbit predictions derived from reconstructing the first 8000 science orbits showed that this prediction accuracy requirement was consistently being met.[23,27]

Orbit prediction errors were required to be less than 1.5 km in the downtrack direction for short-term spacecraft ephemerides, nominally delivered by the navigation team twice per week and used on board the spacecraft. This requirement was met most of the time over 8000 science orbits, with the option of three ephemeris deliveries per week available, if needed, during high-drag seasons. In addition, the navigation team implemented a daily, automated, quick-look orbit solution procedure to allow a comparison to the on-board ephemeris. This allowed rapid identification of and response to occasional, unanticipated orbit perturbations (due to the onset of a dust storm, for example).[23,27]

The spacecraft's Mars-relative position was to be reconstructed to within 100 m, 40 m, and 1.5 m (3σ) in downtrack, crosstrack, and radial directions. With only Doppler tracking data available, the downtrack position had diminished observability for face-on orbit geometries as seen from Earth (which occurred in a rough sense during the first half of 2008). Similarly, the crosstrack position had diminished observability for edge-on orbit geometries (as in July and November of 2007). The MRO trajectory was reconstructed in 20-24 orbit batches, with four or five such batches subsequently merged together to form a weekly reconstruction delivery. One orbit of overlap was typically common to successive batches, to allow checks for solution consistency. Occasional spacecraft safe mode entries (as in November 2007 and February 2008) caused degradations in orbit reconstruction accuracy due to transitions to thruster-based attitude control and the ΔVs that resulted from attitude changes.[27,29]

Orbit Trim Maneuvers (OTMs) were nominally scheduled for every 28 days to maintain the orbit semimajor axis close to its desired value and keep the spacecraft ground track within 10 km of the desired track. (Atmospheric drag would cause the semimajor axis to decrease, reducing the orbital period and causing the ground track to drift eastward.) OTM opportunities were sometimes bypassed during seasons of low drag or as a matter of convenience. For example, by deliberately initiating the PSO with the ground track near the western edge of the allowable window, it was possible to omit the first scheduled OTM on 10 January 2007, with the first actual OTM delayed to 7 February. In addition, the OTM scheduled for March was not performed due to spacecraft entry into a safe mode. Subsequent OTMs were performed on 18 April, 23 May, 27 June, 25 July, 22 August, 19 September, 31 October, and 12 December 2007. All were 7-23 cm/s in size. OTMs typically alternated between apoapsis and periapsis locations in order to minimize departures from the frozen-orbit conditions. It was desired to maintain the mean argument of periapsis within ±3 deg, for example.[23,27,30]

Use as a Relay Satellite. The MRO spacecraft can provide telecommunication relay services to other spacecraft on or near Mars through the Electra Proximity Link Payload. The Electra UHF Transceiver makes use of a flexible, software-defined radio architecture, which allows post-launch software and firmware upgrades to support new functional capabilities, respond to unforeseen mission scenarios, etc. Forward-link frequencies can be tuned between 435 and 450 MHz and return-link frequencies between 390 and 405 MHz for full-duplex operations.[7]

In order to provide UHF communications to the Phoenix spacecraft during its entry, descent, and landing (EDL), it was necessary to shift the phasing of the MRO orbit by about 45 min. In terms of accuracy, it was desired to cross a given latitude on an ascending track within 30 s of the specified time. Orbit synchronization maneuvers were executed on 6 February (15 cm/s) and 30 April 2008 (12 cm/s) in order to achieve the desired phasing, with the usual ground track control for MRO science purposes suspended for several months. The desired 30-s timing accuracy was met by a substantial margin. Indeed, the trajectories of both the MRO and Phoenix spacecraft were controlled accurately enough that MRO's high-resolution camera was able to capture an image of the Phoenix lander descending toward the Martian surface on its parachute. The standard ground track walk pattern was reestablished after Phoenix EDL with an OTM on 25 June.[27,31]

While in orbit about Mars, MRO ΔDOR data have been acquired on a monthly basis to improve the ephemeris of Mars in the International Celestial Reference Frame for the benefit of future missions.[32]

Venus Express

The interplanetary phase of the European Space Agency's Venus Express mission and the establishment of an initial operational orbit have been described in Reference 4 and references listed therein. The operational orbit was polar and highly elliptical, with a period of 24 h plus or minus no more than 6 min. The apoapsis altitude was about 66500 km, and the periapsis altitude was controlled within the range of 185 to 390 km. The periapsis remained close to the north pole of Venus, with little precession of the line of apsides.[8]

Between late April 2006 and early May 2009, 97% of tracking passes involved ESA's Cebreros station, with the remainder involving either New Norcia or DSN stations. Tracking data were obtained over the majority of the ascending leg of each orbit. Orbit determination was typically performed weekly, based upon 10 days of tracking data, allowing three days of overlap between solutions. Thruster firings were used to off-load reaction wheel angular momentum near each apoapsis, with the thrust direction and spacecraft attitude chosen to control orbit phasing. Various factors caused the Venus Express orbit determination to be less accurate than for Mars Express. Venus Express ranging and (limited-bandwidth) ΔDOR data were used to improve the ephemeris of Venus.[8]

Mars Phoenix Lander

Interplanetary Phase – General Characteristics and Early Cruise Flight. The Phoenix spacecraft, consisting of a cruise stage, an entry system, and a lander, was launched toward Mars on 4 August 2007, on a 10-month, type-II trajectory (heliocentric transfer angle between 180 and 360 deg). DSN tracking of the Phoenix spacecraft was scheduled as continuous during the first 14 days after launch, the last 60 days before entry (E), and several eight-day periods centered about TCMs. Three eight-hour tracks per week were scheduled at other times. Two-way X-band Doppler and range data were collected during these tracking passes, through the use of a Small Deep Space Transponder on board the spacecraft. Weekly ΔDOR measurements were made during much of the interplanetary flight, with three measurements per week scheduled between E-60 days and E-18 days and two to three measurements scheduled per day thereafter.[33,34]

To perform a TCM in a particular direction using the spacecraft's monopropellant hydrazine propulsion subsystem, the spacecraft would turn to the desired attitude and turn on the TCM thrusters until the desired ΔV had been obtained, as measured by on-board accelerometers. The first TCM, of 18.5 m/s, was performed six days after launch to correct for the majority of the injection errors and the launch vehicle targeting bias (included to reduce the probability of a Mars impact by the spent third stage below 10^{-4}).[33,34,35]

One navigational challenge in this mission was to understand the effect of many small ΔVs imparted to the trajectory by the unbalanced thrusters of the ACS. The spacecraft used thrusters to maintain spacecraft pointing in order to communicate with Earth and allow power to be generated by the solar arrays. Each time a thruster was pulsed to maintain attitude, a ΔV of as much as 0.07 mm/s was imparted to the trajectory. These thruster firings, occurring as frequently as 160 times per day, needed to be understood in order to properly predict and reconstruct the spacecraft trajectory. Every time a thruster pulse was fired, a telemetry packet with thruster information was recorded and later transmitted to the ground for use in orbit determination and trajectory propagation.[33,34,35]

A second navigational challenge was that in order to perform TCMs, the spacecraft used its thrusters to slew the spacecraft to the proper direction for the ΔV correction. The turn itself caused a ΔV because of the unbalanced thrusters. In order to meet the stringent atmospheric entry and landing site conditions, the ΔVs associated with spacecraft slews needed to be understood and the TCM design strategy adjusted to minimize the overall effect of this error source (as demonstrated by the more than 19-deg pointing error associated with TCM-1).[33,34,35]

The four Reaction Control System (RCS) thrusters, used for attitude control and smaller than the TCM thrusters, were oriented such that the firing of any number of them would always produce a net force along one vehicle axis. The net forces along the other two vehicle axes would tend to cancel over time, given a balanced distribution of thruster firings. However, solar torque imbalances caused certain thrusters to fire more than others. In addition, any thrust magnitude or direction asymmetries caused additional unbalanced forces and net ΔV accumulation.[33,34,35]

RCS Thruster Calibrations and Mid-Cruise Flight. On 14 September, an active RCS thruster calibration was carried out, where the thrusters were pulsed in a designed sequence at two specific spacecraft attitudes while high-rate two-way Doppler tracking data and gyroscope data from the spacecraft's inertial measurement unit (IMU) were collected. The Doppler and gyroscope data were combined in a sigma-point consider filter (SPCF) to estimate the average RCS thrust vectors, with 10% being the desired accuracy for each thruster. The previous Mars Odyssey and MRO active thruster calibrations had allowed for complete observability of the RCS thrust vectors by using reaction wheels to maintain specific spacecraft attitudes. For the Phoenix spacecraft, without reaction wheels, this was not possible. In addition, thermal and telecommunication attitude constraints limited the observability of the thrust vectors. The inclusion of the gyroscope data allowed better observability of the thrusting behavior than would have been the case with Doppler tracking data alone. After every 18 thruster pair firings, it was necessary to fire the thrusters in longer pulses to restore the calibration attitude. These interruptions of the calibration process made it necessary to estimate a new Doppler data bias for each set of firings, further complicating the calibration analysis relative to those for Mars Odyssey and MRO. After full data analysis, each thruster was thought to have been calibrated to an accuracy of 7% (1σ), consistent with the active calibration goal.[33,34,36]

A 3.6-m/s TCM-2 was performed on 24 October to remove the remaining launch vehicle targeting bias and TCM-1 execution errors. Ground and on-board models of RCS thrusting behavior were updated before the TCM, based on the active calibration data. The design and execution of the TCM were complicated by a spacecraft safing event (due to a computer reset) that had occurred a couple of weeks earlier. TCMs 1 and 2 were jointly optimized to minimize their combined ΔV cost. Both required relatively small slews to reach the TCM attitude.[33,34,35]

In early November, the spacecraft was reoriented so as to point the solar panel at the sun (thereby making solar radiation pressure effects more symmetrical); and the spacecraft's attitude control deadbands were tightened. On 15 November, a passive RCS thruster calibration was begun. The objectives were to characterize the thruster activity in the late cruise attitude with its associated tight deadbands and to assess the accuracy of the active calibration results by comparing the accumulated ΔV of all thruster firings from the Doppler tracking data to the SPCF model. After an initial period of outgassing due to the attitude change, the RCS thruster firings remained stable until atmospheric entry, with the passive calibration activity extended to cover this entire time interval. The validity of the active calibration results was confirmed.[33,34]

The thruster calibrations performed throughout cruise were designed to calibrate minimum impulse (15-ms) isolated pulses. Slews, however, were mostly made up of pulse trains designed to establish and maintain a specific turn rate. Therefore, the calibrations discussed above were not adequate to characterize slews. TCM slews consisted of turns both to and from the TCM ΔV attitude. The slews were completed using the RCS thrusters, which imparted ΔV to the trajectory. While this ΔV was predicted and accounted for as a part of the TCM implementation process, there were significant errors associated with the predictions. One such error source was the randomness of the attitude within the deadbands just prior to the start of the slew. Therefore, there was an uncertainty in the number of pulses needed to impart the desired slew rate, as well as in the inertial direction of the pulsing. Other error sources included thruster misalignments and pulse-to-pulse variations. In addition, rate damping and inertial hold modes within TCM execution sequences introduced ΔV uncertainties.[33]

With much of the slew ΔV error coming from the thruster pulsing used to start and stop the slews, using relatively low slew rates was beneficial. However, spacecraft thermal constraints limited how long the spacecraft could depart from its nominal attitude. Thus, rotations through larger angles required higher slew rates than rotations through smaller angles. By allowing the entry flight path angle to float to some degree (still consistent with the constraints on entry flight path angle and time), while explicitly targeting the latitude and longitude of the landing site, the slew angle for a given TCM could be minimized, allowing a reduced slew rate.[33,35]

TCM execution errors posed a significant challenge in navigating the Phoenix spacecraft. These errors can be partitioned into two major parts, errors associated with the main ΔV execution and errors associated with the spacecraft attitude control actions before and after the main burn. The main burn errors were predominantly in pointing, due to effects like thruster imbalances, misalignments, and mass property uncertainties. The main burn errors dominated the execution error for the larger early burns of approximately 2 m/s or more; however, they were not of concern for the smaller, fine-tuning TCMs that were scheduled for later. The dominant error source for the smaller turns was the attitude control activities surrounding the main burn. In particular, the slews to and from the main burn attitude caused the most ΔV error because of the unbalanced RCS thrusters.[33,35]

Orbit Determination Modeling. Estimated parameters in the orbit determination solutions included spacecraft position and velocity components at a reference time (or epoch); TCM execution errors and associated slew ΔVs; and stochastic spacecraft accelerations due to RCS thrusting, RCS thrusting scale factors, and range biases per tracking pass. Earth orientation parameters, tropospheric and ionospheric signal delays, tracking station locations, quasar locations, and Earth and Mars ephemerides were "considered" as error sources, rather than estimated.[2,34]

At the time of the Phoenix mission, typical accuracies for DSN metric observables were about 0.06 mm/sec for line-of-sight velocity, 75 cm for line-of-sight distance, and 2.5 nrad for angular position. Various calibrations needed for processing radio metric data were discussed in Reference 1 and references listed therein, with the means of calibration and the accuracies stated there as of about 1980, in general. By the time of the Phoenix mission, calibration accuracies had improved considerably, in part from the use of several new techniques.[37]

With sufficient very long baseline interferometry (VLBI) measurements involving a large number of extragalactic natural radio sources and multiple ground antennas, it became possible to determine the positions of the various sources on the celestial sphere, as well as the positions of the various antennas on the Earth's surface. Error standard deviations in the locations of DSN tracking stations relative to the Earth's crust had been reduced to about 2 cm by the time of the Phoenix mission. Earth orientation parameters, determined from both VLBI and Earth-orbiting Global Positioning System (GPS) satellite observations, had become accurate to about 3 cm (1σ). Quasar locations were accurate to about 0.8 nrad (1σ).[32,37]

GPS satellite observations are very useful in calibrating both tropospheric and ionospheric media delays. The error standard deviations for zenith troposphere delay and line-of-sight ionosphere delay had been reduced to about 1 and 2.5 cm (at X-band frequencies), respectively, by the time of the Phoenix mission. Calibration of signal path delay through tracking station electronics was accurate to about 50 cm.[37]

Navigation Software. NASA's planetary missions were navigated for many years using the Double-Precision Trajectory Software (DPTRAJ)/Orbit Determination Program (ODP) software set, FORTRAN-based software that had originated in the late 1960s and expanded over the next three decades. In 1998 the strategic decision was made (and funding secured) to develop a new set of software tools that would exploit the advances in computing capabilities and coding techniques that had taken place during the interim.[1,38]

The new software was given the name Mission Analysis, Operations and Navigation Toolkit Environment (MONTE). With various computing constraints imposed by late 1960s/early 1970s computing technology no longer of consequence, MONTE was designed to reproduce the capabilities of the legacy software, while being based on an entirely new design consistent with the best software practices of 1998 and thereafter. With open-source software having become a foundation for most computing, the open-source Linux operating system was chosen for MONTE's development and operational environment. From the start it was decided that MONTE would be developed in an object-oriented language, with C++ ultimately adopted as the backbone of the implementation. It was also felt that scripting languages provide users with a flexible, rapid-analysis system more readily than compiled languages. Thus, the Python language with its object-oriented interface, wide cross-platform support, extensive suite of built-in and third-party languages, and ability to interact with compiled libraries was chosen as a scripting interface.[38]

At the time of the Phoenix mission, MONTE was able to perform all of the functions in the legacy software, including force modeling, trajectory propagation, residual and partial derivative generation, filtering, and mapping, as well as maneuver design and optimization. Graphical user interface-based tools were provided for residual display, data editing, multi-scenario filter/editing runs, case management, and solution display and comparison. The software could be used for both flight operations and covariance analysis. Its throughput performance was comparable to that of DPTRAJ/ODP. Extensive on-line documentation was made available, including user reference, formulation, training, and search features. The entire software suite was put under configuration management to assure its integrity.[34]

Phoenix was the first mission to use MONTE in mission operations. As such, the Phoenix navigation team developed an operations environment from scratch. In addition, matching DPTRAJ/ODP cases had to be maintained to validate the MONTE solutions and protect against an unexpected MONTE failure during critical events. The MONTE cases were considered the baseline; and tools were developed to generate ODP inputs, such as stochastic filter update controls, data edits, and data weights, from MONTE inputs.[34]

Mars Approach. At E-45 days, a 1.4-m/s TCM-3 was executed, correcting TCM-2 delivery errors as well as trajectory errors that had accumulated over more than five months since TCM-2. With the ACS deadbands constant and in their final state and the active and passive thruster calibration results extensively analyzed, the orbit determination solutions were quite stable leading up to the TCM. The time from navigation data cutoff to TCM execution was five days for TCMs 1-3. TCM-3 was executed accurately despite a considerably larger slew angle than for the first two TCMs. In this case, the round-trip slew was executed by traveling in a full circle, which resulted in partial ΔV error cancellation.[33,34,35]

Beginning with TCM-3, the targeting of appropriate atmospheric entry interface conditions became an iterative process. Targeting for a previously determined B-plane position and entry time would result in a shift in landing location if the incoming trajectory asymptote had meanwhile shifted. Retaining the desired landing location would then require an adjustment in B-plane angle and entry time. An automated script was developed to perform in an expeditious fashion the roughly three iterations of the maneuver targeting and the EDL atmospheric flight trajectory propagation and targeting software that were typically required.[35]

As arrival at Mars drew closer, certain software tools that had been under development were put into use. A multi-function data pre-processing tool was used to deliver orbit determination products within one hour of the tracking data cutoff. This automated tool removed blunder points from the tracking data and generated per pass data weight commands for each pass based on its data noise characteristics. The tool would determine the time of the last delivered data calibrations and deweight any uncalibrated data. The tool would also generate commands in MONTE and ODP input formats to assure that the two software sets were using identical data sets and weightings. A case management tool was developed to help in the analysis of the many orbit determination solutions that were generated each day (25 filter/data scenarios for each of four data arc lengths).[34]

The purpose of a planned TCM-4 at E-15 days was to limit the ΔV size and error contributions of later TCMs. However, the ΔV needed to correct for TCM-3 targeting errors was below the minimum of 0.05 m/s that the spacecraft could execute. (The ΔVs unavoidably accompanying spacecraft rotations tended to limit how small a TCM could be executed accurately.) Moreover, omitting TCM-4 increased the likelihood that the next TCM would be large enough to perform with reasonable accuracy. Thus, TCM-4 was cancelled. (However, an entry parameter update was performed.)[33,34,35,39]

The landing site selection process identified a suitable landing area that could accommodate a 99%-probability landing ellipse that was 100 by 20 km in size. One main contributor to the size of the landing ellipse was the uncertainty in the trajectory at the Mars atmospheric interface point (defined to be at a Mars radius of 3522.2 km), and most specifically, the uncertainty in the entry flight path angle. A second major contributor to the landing error ellipse size was the trajectory dispersion resulting from flight through the Martian atmosphere, determined by factors such as center-of-mass knowledge, atmospheric density knowledge, and aerodynamic properties of the entry vehicle. These EDL dispersions combined with the selected landing area required that the navigation entry flight path angle be accurate to ± 0.2 deg (3σ). The absolute entry flight path angle target was a critical parameter of the EDL process, in that too shallow an angle would risk the spacecraft skipping out of the atmosphere and too steep an angle would risk overheating of the entry vehicle. The entry flight path angle also impacted EDL timing and details of the terminal descent/touchdown phase. An entry flight path angle target of -13.0 deg was deemed optimal for the EDL system. An additional requirement on the Mars entry conditions involved the time of entry. The Phoenix EDL system was designed to communicate with the Mars Odyssey and MRO spacecraft using a UHF signal, since no reliable direct-to-Earth link would be available. The orbiters would store the data sent by Phoenix and subsequently send that data to the Earth using their X-band telecommunication systems. The orbiters needed to be in the correct locations, orientations, and configurations at the time of EDL. Consequently, control of the Phoenix entry time to within 30 s was required.[33,39,40]

TCM-5 was placed at E-8 days to meet the EDL system limits, thereby mitigating the risk of failure of the next TCM. This strategy allowed a reasonable chance for the EDL system to execute properly even if the landing site constraints were not met. The time from navigation data cutoff to TCM execution was 26 h for TCM-5. The TCM was executed as a minimum-size maneuver of 0.05 m/s, with the entry flight path angle and time allowed to shift so as to reduce the slew angle required for the TCM and increase the maneuver size slightly.[33,34,35]

TCM-6 was planned for E-21 h to guarantee that the landing site constraints were fulfilled and the target reached. The final orbit determination solution before TCM execution indicated that all of the landing site and EDL system constraints would be met without executing the TCM. Consequently, it was cancelled. An option existed to perform a later contingency TCM (to be selected from a set of 25 pre-designed, pretested maneuvers), but no reason arose for doing so. The final orbit determination solution (with tracking data through 15 min before entry) showed the B-plane position to be 2 km from the location targeted by TCM-5 and the entry flight path angle and time within 0.007 deg and 14.9 s of their nominal values. The final pre-entry 99%-probability landing error ellipse was 55 by 19 km in size and was centered 17 km from the desired target point. This offset could have been reduced by executing TCM-6. However, the risks associated with performing a TCM so close to entry more than outweighed the potential modest gain from landing closer to the target, given that all EDL and landing site requirements were already satisfied. Martian atmospheric models were updated twice in May based on weather observations.[33,34,35,39,40,41]

A technology demonstration collecting and processing spacecraft-spacecraft, phase-referenced, interferometric data generated using the National Radio Astronomy Observatory's (NRAO's) Very Long Baseline Array was carried out during the Phoenix spacecraft's approach to Mars. The Mars Odyssey and MRO spacecraft, with their accurately known orbits about Mars, were the reference signal sources. This approach produced significant improvements in Phoenix orbit determination accuracy; however, the results were not available until after arrival at Mars due to the time required for data processing and error debugging.[42]

Entry, Descent, and Landing. Upon Mars arrival on 25 May 2008, the capsule containing the lander was separated from the cruise stage at E-7 min, initiating the EDL phase of the mission. With this event, communication at X-band ceased; and transmission of a UHF carrier signal through the antenna on the backshell of the entry capsule (to orbiting relay spacecraft) began. At E-6.5 min, the capsule began a rotational maneuver to achieve zero angle of attack at entry interface, completing the rotation at E-5 min. Starting at E-2 min, telemetry information was modulated on the UHF carrier. The capsule entered the Martian atmosphere at an inertial speed of 5.6 km/s (at 125-km altitude, referenced to Mars' equatorial radius).[41,43]

Early in the entry, the ACS deadbands were widened sufficiently to make thruster firings unlikely, due to concerns about a control system instability arising from an interaction between the aerodynamic flowfield and plumes from RCS thruster firings. Instead, the capsule relied upon its inherent aerodynamic stability to traverse the various flight regimes prior to parachute deployment. Many thruster firings would have taken place without the deadband widening, given the angular rates that Phoenix experienced during entry; but no firings actually occurred. (Thus, Phoenix became the first ballistic, non-spinning spacecraft to land on Mars without hypersonic thruster control.) Static instabilities were, in fact, experienced during hypersonic flight; and a dynamic instability was experienced during supersonic flight. Trim angles of attack were higher than expected both during and away from these instabilities, but were within allowable ranges. Phoenix experienced a peak deceleration of 8.5 *g*s at E+123 s, a bit lower and later than expected.[41,44]

Parachute deployment was triggered by IMU accelerometer measurements and took place at E+228 s (6 s later than predicted), a deceleration of 7.39 m/s^2, a dynamic pressure of 489 N/m^2, a Mach number of 1.7 (388 m/s), and an altitude (above ground level) of 13.3 km. Heatshield separation was timed to occur 15 s after parachute deployment. Lander leg deployment was timed to occur 10 s after heatshield separation. The descent radar was timed to be activated 15 s after lander leg deployment, but did not begin searching for the ground until 35 s later at an altitude of 7.2 km. The Martian surface was actually detected at an altitude of 2.4 km, consistent with expectations.[41,44]

Lander separation from the backshell/parachute system was based on radar measurements and occurred at E+405 s, a wind-relative speed of 57 m/s, and an altitude of 927 m above ground level. The start of a gravity turn was initiated at an altitude of 720 m, with a transition to a constant speed descent at 52 m. The radar data were not considered reliable at altitudes less than 30 m and were disregarded thereafter. Landing took place at E+446 s, with a vertical speed of 2.4 m/s and a horizontal speed of 0.06 m/s. The landing location was 21 km downtrack from that expected, which can be attributed to the shallower-than-expected atmospheric flight, due to the aerodynamic lift force generated by the larger-than-expected angle of attack. In addition, there was a 5-km crosstrack landing error, since the lift vector gradually drifted to one side from its initial vertical orientation.[41,44]

The MRO spacecraft acquired a high-fidelity, open-loop recording of the Phoenix UHF signal throughout the EDL process. However, the nature of MRO's on-board data architecture is such that this data set was not available on Earth until about two hours after the landing. The Mars Odyssey spacecraft, on the other hand, was able to relay a real-time carrier signal prior to parachute deployment and carrier plus near real-time telemetry data thereafter. As a back-up, the Mars Express spacecraft acquired an open-loop recording of the Phoenix UHF carrier signal, which was available to the Phoenix project about 1.5 h after touchdown. In addition, NRAO's 100-m Green Bank Telescope was able to detect the Phoenix UHF carrier signal in real time.[43]

Surface Positioning. The Phoenix lander, by design, had no ability to communicate directly with the Earth. Instead all communications involved relay links through the Mars Odyssey or MRO spacecraft, with the Mars Express spacecraft available as a back-up. Navigational information could be obtained from these relay links in the form of two-way, coherent Doppler data. UHF Doppler data were stored on board the orbiters, sent down to the DSN in spacecraft telemetry, and subsequently processed to determine the position of the lander (with the trajectories of the orbiting spacecraft already well known). The resulting solution for the lander position, based on eight Odyssey and three MRO UHF passes during overflights of the Phoenix lander, was 68.219 deg north latitude and 234.248 deg east longitude, with a 3σ accuracy of 12 m. This position determination was consistent with the integration of IMU accelerometer data during EDL (accurate to about 10 km, including uncertainties in entry conditions) and imaging by the MRO high-resolution camera (allowing for the inherent uncertainty in the definition of the prime meridian of Mars). The landing site was 7 km from the originally targeted landing site (which differs from the pre-entry predicted site mentioned above).[33,34,35,43,44]

The Phoenix lander was not designed to survive a Martian winter at its high northerly latitude. As days became shorter and less solar energy became available, several low-energy faults were triggered; and the lander made its last successful data transmission on 2 November 2008, having survived for 157 sols, versus the planned minimum lifetime of 90 sols.[43]

EXPLORATION OF THE OUTER PLANETS

Cassini

The interplanetary flight of the Cassini spacecraft and the first two years in orbit about Saturn, including the delivery of the Huygens probe to Titan, have been described in References 2, 3, and 4 and references listed therein, as well as Reference 45. General descriptions of Cassini orbit determination and control operations processes and interfaces are presented in References 46 and 47.

Navigational Computations. Over an extended period of time and with multiple upgrades, a fault-tolerant, high-reliability/high-availability computational environment was assembled to support Cassini navigation data processing. The initial computational system, used during the mission design stages prior to the 1997 launch, consisted of a dozen Hewlett-Packard and two Sun Microsystems workstations running on an Ethernet local area network. After the launch operations, the Sun workstations were retired; and the HP workstations were upgraded to the Hewlett-Packard Unix-based operating system HP-UX 10.20. In 2002, in preparation for the orbital tour about Saturn, the navigation computational environment was upgraded with the purchase of high-end Dell Computer Corporation Intel (32-bit x86) workstations and servers, with Red Hat Linux operating systems. Three servers, more than two dozen workstations (as of 2008), and an eight-terabyte NetApp, Inc., Network-Attached Storage (NAS) network disk array were interconnected by means of a gigabit Ethernet backbone. The disk array served as the NFS (Network File System) protocol primary file server. Several Sun workstations were also included in this computational environment to provide support for flight operations ground software components used in other parts of the Cassini project that would not be ported to the Red Hat Linux operating system.[48]

Built on top of the navigation software was the JPL *navshell* system, which helped automate and tie together essential pieces of the Orbit Determination Program and other software tools into an efficient, robust, and flexible system. The orbit determination operations of the Cassini mission at Saturn could not easily be automated due to the complex dynamical environment in which the spacecraft flew; however, several sub-processes were automated for quick turnaround of products. In addition, Maneuver Automation Software was developed for generating maneuver designs and presentation materials and producing information used to facilitate operational decisions on such issues as maneuver cancellation and alternative maneuver strategies.[46,49,50]

Spacecraft telemetry data contained information about ΔV events, which was potentially useful for improving the accuracies and timeliness of predicted and reconstructed trajectories. Prior to 2005, this information was of limited value because of its poor resolution, necessitated by the need to avoid saturation during three large maneuvers (hundreds of m/s in size), and the lack of data sources for accurate characterization and calibration. In the latter regard, the small sample size of TCMs; the deadband control of spacecraft attitude with frequent (roughly every two hours), small firings of the RCS; the slow variations in orbital dynamics; and the influence of solar radiation pressure all limited the ability to calibrate the ΔV-related telemetry data during interplanetary cruise. Entry into orbit about Saturn resulted in a rapidly growing accumulation of TCM samples; Reaction Wheel Assembly control of spacecraft attitude, with less frequent (every few days) but larger (and more resolvable) RCS firings to maintain wheel speeds within specifications; more rapid variations in orbital dynamics (observable in radio metric tracking data); and diminished force due to solar radiation pressure. With ΔV resolution substantially improved by means of a flight software update in May 2005, the processing of telemetry data became useful for modeling many ΔV events, as an augmentation to standard navigational computations.[51,52]

Titan-16 Through Titan-33 Encounters. The 18 encounters numbered Titan-16 through Titan-33 had the effect of rotating the long axis of the spacecraft's orbit about Saturn such that the encounter locations in Titan's orbit about Saturn shifted by about 180 deg (a so-called pi transfer). The first nine of these encounters took place on 22 July, 7 and 23 September, 9 and 25 October, and 12 and 28 December 2006, and 13 and 29 January 2007. These encounters all took place inbound toward Saturn periapsis and gradually increased the inclination of the spacecraft's orbit about Saturn. The transfer in encounter orbital location occurred between the Titan-24 and Titan-25 encounters, with the spacecraft's orbital inclination having reached a maximum of 59 deg. The second nine encounters, on 22 February, 10 and 26 March, 10 and 26 April, 12 and 28 May, and 13 and 29 June 2007, were all outbound from Saturn periapsis and gradually decreased the orbital inclination to 4 deg. The Titan flyby altitudes ranged from 950 to 2631 km for these 18 encounters.[53,54]

Using the maneuver strategy described in Reference 4, a total of 54 maneuvers were initially planned to achieve these 18 encounters. OTMs after encounters and around apoapsis were typically designed by means of a chained two-impulse optimization strategy, in which 10 such OTMs, over the next five encounters, were optimized. Only the first two such OTMs would actually be implemented before the optimization process was repeated. Fifteen of the 54 initially planned OTMs wound up being cancelled, to simplify mission operations, improve reconstruction of previous maneuvers, improve orbit determination convergence for upcoming maneuvers, or avoid performing a maneuver of less than the minimum allowable size of 10 mm/s, assuming in each case that the downstream trajectory deviations and ΔV cost of cancellation were sufficiently small. Of the 39 OTMs actually executed, 24 were performed using the bipropellant main engine assembly and 15 using the monopropellant RCS thrusters, based on maneuver size. On three occasions, maneuver design targets were biased in either time of flight or B-plane position, in order to avoid performing too small a maneuver or reduce downstream ΔV costs.[53]

Most encounters took place within 1 km of their locations along the reference trajectory in the B-plane coordinates *B•R* and *B•T*. The exceptions resulted from cancelled maneuvers or target biasing. Many of the encounters were at low flyby altitudes, with 950 km being the lowest. 14 of 18 pairs of consecutive encounters were 16 days apart, 16 days being Titan's orbital period about Saturn. Planning and executing an OTM every five days necessitated expeditious orbit determination and maneuver design and assessment.[53]

Although radio metric tracking data were not available during most satellite encounters due to the incompatibility of collecting the majority of the desired scientific data while simultaneously communicating with the Earth, the Titan-11, -22, and -33 encounters were designed so that radio metric data were indeed available. In addition, the spacecraft's attitude in these encounters was controlled using reaction wheels, so as to avoid thrusting perturbations from the RCS. From the resulting near-encounter data, it was possible to improve the estimates of Titan's mass and quadrupole gravity field.[54,55]

Titan-34, Titan-35, and Iapetus-1 Encounters. The Titan-34 and -35 encounters took place on 19 July and 31 August 2007 at altitudes of 1332 and 3324 km, with a solar conjunction period in between. The Titan-34 encounter was inbound, while the Titan-35 encounter was outbound. The outbound Iapetus-1 encounter took place on 10 September at an altitude of 1651 km. The Iapetus ephemeris turned out to be in error by 19 km, with significant improvement derived from this flyby. The accuracy of the flyby trajectory reconstruction was enhanced by estimating the oblateness of Iapetus.[54,56]

Titan-36 Through Titan-41 and Enceladus-3 Encounters. After closest approach to Iapetus, system fault protection events caused spacecraft attitude control to switch to the RCS mode. This induced spacecraft rotations and dead-band attitude maintenance, with significant, difficult-to-measure ΔV activity resulting. Consequently, the next OTM was designed based on an inaccurate orbit determination solution, with a better solution obtained through modeling improvements just in time for the following scheduled OTM.[54]

The Titan-36, -37, -38, -39, -40, and -41 encounters took place on 2 October, 19 November, and 5 and 20 December 2007 and 5 January and 22 February 2008 at altitudes of 973, 999, 1298, 970, 1014, and 1000 km. All occurred outbound from periapsis. Over this time period the orbit inclination increased from 5 to 57 deg. The inbound Enceladus-3 encounter took place on 12 March at an altitude of 54 km, with the spacecraft passing through the icy plume emanating from the satellite's south polar region. Pre-encounter analyses led to the conclusion that scaling the Enceladus ephemeris covariance upward by a factor of three would lead to a more accurate orbit solution for the spacecraft than using an unscaled covariance. (An accumulation of 28 optical navigation images, taken over a number of months, indicated a consistent shift in the downtrack position of Enceladus. Optical navigation data residuals could be reduced by this scaling.) The desirability of the scaling was confirmed by post-encounter reconstruction analyses.[54,56,57]

Accumulated experience with the Cassini spacecraft through September 2007 allowed the development of an updated maneuver execution error model with reduced uncertainties for use beginning in February 2008.[58]

Titan-42, Titan-43, and Titan-44 Encounters. The Titan-42, -43, and -44 encounters took place on 25 March and 12 and 28 May 2008 at altitudes of 999, 1001, and 1400 km. All occurred outbound from periapsis. Over this time the orbit inclination continued to increase to 76 deg.[54,56]

Of the 39 OTMs planned to achieve the 13 encounters Titan-34 through Titan-44, 27 were performed in their prime locations, one was executed in the back-up location, 9 were cancelled (mostly approach OTMs), and 2 were deleted (i.e., cancelled well in advance). One OTM was executed with a biased time-of-flight target. 19 OTMs were performed with the main engine assembly, while 9 used the RCS.[56]

Transition from Prime Mission to Equinox Mission. Cassini's prime mission concluded, and the extended Cassini Equinox Mission began, on 1 July 2008, after 45 Titan and 7 icy satellite targeted encounters over the course of 75 revolutions about Saturn. 400 m/s of propellant ΔV were used to derive 33 km/s of gravity-assist ΔV from the various satellite encounters. 112 OTMs were performed, out of a planned total of 161. The satellite gravity-assist tour reference trajectory was updated eight times after the spacecraft's arrival at Saturn, either to improve science return or reduce mission risk. At the end of the prime mission, the gravitational parameters of Saturn's nine largest satellites had been determined to 0.2% or better. The accurate analysis of the numerous flybys of Titan at relatively low altitudes required careful efforts to separate the dynamical effects of atmospheric drag, gravity field irregularities, and spacecraft thrusting for attitude control.[49,51,54,59,60]

Over the course of the prime mission, the ΔV cost to correct statistical errors was found to average 0.3 m/s per encounter, considerably less than had been assumed before the satellite tour began (2.35 m/s). This was due in large part to orbit determination accuracies at satellite encounters turning out to be about 3 km

(1σ), whereas 10 km had been assumed. Trajectory deviations relative to the reference trajectory were often considerably larger away from satellite encounters, since it was not necessary or practical to remain on the reference trajectory continually. Maximum trajectory variations were less than 1000 km on 75% of all satellite-to-satellite orbit transfers.[61]

Titan-45, Enceladus-4, Enceladus-5, Enceladus-6, and Titan-46 Encounters. The outbound Titan-45 encounter took place on 31 July 2008 at an altitude of 1613 km. It had been set up primarily by a 12.2-m/s trajectory-shaping maneuver more than a month earlier. The outbound Enceladus-4 encounter took place on 11 August at an altitude of 54 km. Only a single OTM was used to set up this encounter, with the approach maneuver cancelled to eliminate the small possibility of an impact with Enceladus in the event that a thruster were to develop a leak.[57,62]

Three deterministic OTMs and a statistical approach maneuver, totaling 18.3 m/s, were used over multiple orbital revolutions to set up the outbound Enceladus-5 encounter on 9 October at an altitude of 28 km. Consideration was given to cancelling the approach maneuver, as for the Enceladus-4 encounter; however, the complexities of and accuracies required for subsequent encounters dictated against this. Instead, the encounter time was adjusted slightly, so as to produce an OTM of at least the minimum allowable size, which was at this point 9 mm/s. The flyby altitude in this very low encounter was within 0.4 km of that predicted.[57,62]

The outbound (non-targeted) Enceladus-6 encounter on 31 October at 176-km altitude was quickly followed by the outbound Titan-46 encounter on 3 November at 1100-km altitude. Targeting for Titan, rather than Enceladus, in this double encounter resulted in lower ΔV cost. As in the case of the Enceladus-4 and -5 encounters, the spacecraft passed through the icy plume emanating from the south polar region.[62]

Titan-47 Through Titan-51 Encounters. The Titan-47, -48, -49, -50, and -51 encounters, all outbound, took place on 19 November, 5 and 21 December 2008, and 7 February and 27 March 2009 at altitudes of 1023, 960, 970, 960, and 960 km. Various OTMs setting up the first three of these encounters were impacted by the poor thruster performance in the approach maneuver before the Enceladus-6/Titan-46 encounters. To address this performance issue, thruster branches were swapped before the Titan-51 encounter, which led to an OTM rescheduling to first use the new thruster branch comfortably before the encounter.[62]

Titan-52 Through Titan-58 Encounters. The Titan-51 to -52 transfer was an eight-day pi transfer, with the spacecraft encounter longitude shifting by 180 deg. The Titan-52, -53, -54, -55, -56, -57, and -58 encounters, all inbound and separated by 16 days, took place on 4 and 20 April, 5 and 21 May, 6 and 22 June, and 8 July 2009 at altitudes of 4150, 3600, 3245, 965, 965, 955, and 965 km. The transfers to Titan-53 through -56 required deterministic maneuvers near apoapsis of several m/s each. The two subsequent transfers required deterministic flyby clean-up maneuvers of comparable size. Many other OTMs were cancelled due to the accuracy of those that were executed, the small downstream ΔV penalty associated with their cancellation, spacecraft risk and thruster life-cycle considerations, possible undesirable features of backup maneuver scenarios, the desire to avoid time biasing associated with very small computed maneuvers, and the desire to simplify spacecraft operations. A new maneuver execution error model was introduced after the Titan-53 encounter, reflecting observed thruster performance.[62]

Over the first year of the Cassini Equinox Mission, 45 maneuvers were planned well in advance, with 29 executed in their prime locations and 16 cancelled. An additional maneuver was executed in a location not originally planned. 70 percent of these OTMs were executed using the main-engine assembly.[62]

New Horizons

The New Horizons spacecraft was launched on 19 January 2006 toward Jupiter, on the way to its primary destinations of Pluto and Charon. The spacecraft uses a non-coherent transceiver instead of the usual deep space transponder, so that radio metric tracking data must be pre-processed with downlink telemetry for calibration before their use in orbit determination. The spacecraft could be operated in either a spin-stabilized mode (at about 5 rpm) or a three-axis stabilized mode. The former mode requires occasional thruster firings to reorient the spin axis. The latter mode requires much more frequent thruster firings to maintain the spacecraft attitude within a certain deadband, there being no reaction wheels on the spacecraft to provide attitude stabilization. The thrusters are normally fired in pairs to adjust spacecraft attitude, but are not perfectly coupled and thus impart translational forces also.[63]

Trajectory errors after launch were quite small, requiring a first TCM of only 18 m/s, well below the amount budgeted. This TCM was executed in two parts on 28 and 30 January, with the first limited to 5 m/s, as a precaution during this first use of the spacecraft's thrusters. These burns were constrained to be in the same direction and were performed in an open-loop mode with a timed cutoff, while the spacecraft was spinning. The next TCM of 1 m/s was performed on 9 March in a three-axis stabilized, closed-loop mode with an accelerometer-determined cutoff. This TCM corrected for an underburn in the first, two-part TCM and targeted the spacecraft accurately for a Jupiter gravity assist flyby on 28 February 2007. Three later opportunities for performing TCMs prior to the Jupiter flyby were judged to be unnecessary and were not used. Orbit solutions were based on X-band Doppler, range, and (after 15 September 2006) ΔDOR data. The larger thruster firings were estimated as discrete ΔV events, while the deadband thruster firings were estimated as constant accelerations.[63]

The passage by Jupiter offered the opportunity to test the Long-Range Reconnaissance Imager on the spacecraft. Prior to arrival at Jupiter, the camera was geometrically calibrated by analyzing 58 images of an open cluster of stars. Closer to encounter, optical navigation images of eight Jovian satellites were obtained (which were useful for improving the satellite ephemerides).[64]

EXPLORATION OF COMETS AND ASTEROIDS

Stardust-NExT

The primary Stardust mission has been described in References 3 and 4 and references listed therein. In July 2007, the extended Stardust-NExT (New Exploration of Tempel 1) mission was approved. In August the spacecraft was awakened from a state of hibernation. A 3.6-m/s deep space maneuver was executed on 10 October. With the spacecraft in a heliocentric orbit with 1.5-year period, an Earth gravity assist flyby took place on 14 January 2009 at an altitude of about 9200 km, three years after the prior flyby and cometary sample return. This Earth gravity assist included a change in orbit plane, to set up a flyby of the comet 9P/Tempel 1 in 2011, beyond the time period covered by this paper. The passage through the Earth-moon system also provided an opportunity for calibrating the navigation camera.[65]

Hayabusa

The interplanetary flight and near-asteroid operations of the Japan Aerospace Exploration Agency's Hayabusa mission have been described in Reference 4 and references listed therein, as well as References 66, 67, 68, 69, 70, 71, and 72.

On departure from the asteroid 25143 Itokawa in November 2005, the telecommunication link was lost for 46 days before restoration. In March 2007, the spacecraft embarked on a return to Earth using its ion engines. In November 2009, ion source and neutralizer failures left none of four ion engines fully operational. However, it was found possible to combine an ion source from one engine with a neutralizer from another to enable a resumption of thrusting. The subsequent return of the sample capsule to Earth lies beyond the time period covered by this paper.[73,74]

Rosetta

The early interplanetary flight of the European Space Agency's Rosetta spacecraft, including the first Earth flyby, has been described in Reference 4 and references listed therein.

Mars Flyby and Second Earth Flyby. Trajectory errors after the March 2005 Earth flyby were too small to warrant the execution of a TCM. Any needed corrections were deferred until the execution of a 31.8-m/s deterministic TCM on 29 September 2006, near perihelion. Reaction wheel off-loadings were performed about once per week over an extended period of time. Velocity changes of less than 1 mm/s were observed during the off-loadings, with the thrusters nominally balanced when firing for this purpose.[75]

A 9.9-cm/s TCM was executed on 13 November to improve the targeting for the upcoming Mars flyby. A smaller TCM was executed on 9 February 2007 to further improve the targeting. No spacecraft slews or reaction wheel off-loadings were allowed between this TCM and the activation of science payload instruments two days before the Mars encounter. Four potential TCM opportunities subsequent to this were not used, since the remaining trajectory errors were calculated to be relatively small.[75,76,77]

During the several-month approach to Mars, X-band two-way Doppler and range data were collected from ESA's New Norcia 35-m antenna and DSN stations at Goldstone and Madrid. In addition, ΔDOR data were collected along ESA's New Norcia-Cebreros baseline and the DSN's Goldstone-Canberra and Goldstone-Madrid baselines (with little data available along the latter due to the spacecraft's southerly declination). The ΔDOR data were found to be a useful complement to the Doppler and range data.[75,77]

Orbit determination solutions were obtained by estimating spacecraft position and velocity at an epoch time, radial solar radiation pressure acceleration, TCM and wheel off-loading ΔV components, and range biases per station pass. In addition, the effects of transverse solar radiation pressure accelerations, tracking station locations, Earth orientation, quasar locations, tropospheric and ionospheric signal propagation, transponder group delay, and Mars ephemeris were treated as "considered" parameters.[75]

The Rosetta spacecraft flew past Mars on 25 February 2007 at an altitude of 251 km. The flyby location in the B-plane was 8.3 km from the point targeted. No TCM was needed immediately after the Mars flyby. Any small trajectory corrections associated with the flyby were instead incorporated into a primarily deterministic 6.5-m/s TCM made on 26 April to set up a subsequent Earth flyby.[75,77]

The second Earth flyby, targeted for an altitude of 5301 km, took place on 13 November 2007, raising the spacecraft's aphelion distance into the main asteroid belt. A 1.53 m/s TCM was executed 10 days later to correct errors in flyby conditions.[77,78]

Steins Flyby. A TCM of 0.25 m/s was executed on 21 February 2008 to target the spacecraft for a flyby of the 5-km diameter, main-belt asteroid 2867 Steins. For several months thereafter, the spacecraft was tracked only once per week while in near-sun hibernation mode. More normal operations resumed in July, with the spacecraft tracked daily by the New Norcia station beginning in August, as well as DSN stations at Madrid or Goldstone, all acquiring two-way Doppler and range data. Reaction wheel off-loadings took place once per week.[78]

An approach TCM of 12.8 cm/s was executed on 14 August, based on determination of the spacecraft's heliocentric position and velocity from radio metric data, determination of the asteroid's ephemeris from ground-based astrometric observations, and (beginning 4 August) optical navigation images derived from several on-board cameras. On 4 September, a final approach TCM of 11.8 cm/s was performed. Twenty minutes before the 5 September closest approach, the spacecraft switched into an asteroid flyby mode, in which the asteroid was tracked by a navigation camera and the spacecraft attitude adjusted to maintain science instrument pointing.[78,79,80]

The flyby took place at a distance of 803 km and was accurate to 6.6 km in the B-plane and 4 s in time of flight. The use of optical navigation data was essential to the achievement of these accuracies. The ground-based astrometric observations appeared to contain systematic errors due to biases in the star catalogs used for data reduction. The pointing of the instrument payload in the asteroid flyby mode reached a peak error of 0.4 deg, which exceeded the system requirement of 0.3 deg.[78,79,80]

During the interplanetary cruise portions of 2004 to 2009, it was found that the spacecraft's nongravitational accelerations could be modeled more accurately by including the effects of thermal radiation from the solar panels and the spacecraft body than by treating only solar radiation pressure effects, with the former about 5-10% as large as the latter.[81,82,83,84]

Deep Impact/EPOXI

The primary Deep Impact mission has been described in Reference 4 and references listed therein. Although the primary mission was only six months long, the Deep Impact spacecraft was traveling in an orbit with a 1.5-year period, which would result in a passage somewhere near the Earth three years after launch and the possibility of an extended mission. Thus, a TCM of 97 m/s was executed on 20 July 2005, a few weeks after the primary encounter with comet 9P/Tempel 1, to target more accurately for an Earth flyby and a subsequent encounter with the periodic comet 85P/Boethin. Afterward, the spacecraft was put into a spinning, sun-coning hibernation mode.[85,86]

Earth Gravity Assist-1. The EPOXI mission formally began in July 2007 as a merger of the separate proposed investigations, Extrasolar Planet Observation and Characterization (EPOCh) and Deep Impact eXtended Investigation (DIXI). On 25 September 2007, the spacecraft was awakened from hibernation and returned to three-axis stabilized attitude control. Preparations began for a TCM to target for an Earth gravity assist flyby on 31 December and the Boethin encounter to follow. However, questions arose as to the viability of the infrequently observed Boethin as a flyby target (it could not be found in telescopic observations); and in October the decision was made to switch to periodic comet 103P/Hartley 2, with a much better known ephemeris, as the target. On 1 November, a 14.6-m/s TCM was executed to modify the targeting accordingly for the 31 December Earth flyby. A TCM scheduled for 11 December to correct any remaining trajectory errors was found to be unnecessary. The Earth flyby took place at a radial distance of 21,944 km. A statistical TCM scheduled for 16 January 2008 was found to be unnecessary and was cancelled.[85,86]

Earth Gravity Assist-2 and Thereafter. The switch to Hartley 2 as the cometary destination required additional time and Earth gravity assists relative to the original mission plan involving Boethin, as well as a search among trajectory options to avoid dangerously long Earth eclipse durations, solar ranges below 1 Astronomical Unit, and a low solar phase angle on approach to the comet. On 19 June and 11 December 2008, TCMs of 31.6 and 0.6 m/s were executed to set up a second Earth gravity assist flyby on 29 December at a radial distance of 49,835 km. On 19 February and 8 December 2009, TCMs of 0.8 and 0.5 m/s were executed to set up a third Earth gravity assist flyby in 2010, beyond the time period covered by this paper. Distant flybys of Earth, which were of little navigational consequence, occurred in June and December 2009.[85,86]

Dawn

The Dawn spacecraft was launched on 27 September 2007, on its way to eventually orbit the large main-belt asteroids 4 Vesta and 1 Ceres. The first 80 days of flight were an initial checkout phase, with particular attention devoted to the ion propulsion system (IPS). The IPS operated for 278 h during this time, producing a velocity change of 65 m/s.[87]

The interplanetary cruise phase began on 17 December, with the IPS thrusting for 6486 h, consuming 71.7 kg of xenon propellant, and providing a velocity change of more than 1.8 km/s, before the termination of deterministic thrusting on 31 October 2008. With a gravity assist flyby of Mars occurring on 18 February 2009, thrusting was non-optimal after 31 October, other than to execute a single statistical TCM on 20 November. This TCM, executed with the IPS, provided a ΔV of 0.63 m/s over 121 min of thrusting. A second planned TCM opportunity in January turned out not to be needed. The Mars gravity assist flyby at 542 km altitude resulted in a ΔV of 2.6 km/s, increasing spacecraft perihelion and aphelion distances and orbit inclination.[87,88]

Deterministic thrusting resumed on 8 June. By the end of 2009, the IPS had produced thrust for 11,365 h since launch (57% of the time since then), imparting a ΔV of 3.2 km/s and expending 126 kg of xenon. During 2008, coast periods of three to five days had been included in each five-week command sequence, to allow time to perform activities incompatible with optimal IPS thrusting. Subsequently, the amount of forced coasting was substantially reduced. Every five weeks, the thrust vectors from the start of the command sequence to arrival in a 3000-km orbit about Vesta (in 2011) were re-optimized to minimize time to arrival (subject to the constraint that there be no thrusting during certain time intervals).[88,89]

CONCLUSION

Deep space navigation capabilities, which had evolved enormously from the 1960s through the early 2000s, continued to evolve thereafter, benefiting the 14 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. Further experience was gained in delivering a spacecraft very accurately to the top of the Martian atmosphere and subsequently to the planet's surface. Accurately controlled gravity assist flybys of planets and planetary satellites allowed the execution of missions that would have been infeasible using chemical propulsion alone. The challenge of accurately modeling spacecraft nongravitational accelerations, particularly due to attitude control subsystem activity, was a recurring theme.

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