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THE EVOLUTION OF DEEP SPACE NAVIGATION: 2004-2006*

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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 fourth in a chronological sequence dealing with the evolution of deep space navigation. The time interval covered extends from roughly 2004 to 2006. 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 eleven planetary missions.

INTRODUCTION

Three previous papers[1,2,3] have described the evolution of deep space navigation over the time interval 1962 to 2004. 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 paper covered 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 2004. In addition, Butrica has recently described the history of deep space navigation from the perspective of a professional historian.[4]

The current paper extends Reference 3 by about two years. It covers the portions of the Mars Global Surveyor, Cassini, Stardust, Hayabusa, Mars Express, Mars Exploration Rover, Rosetta, MESSENGER, Deep Impact, Mars Reconnaissance Orbiter, and Venus Express missions that took place between 2004 and 2006. 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

Mars Global Surveyor

The interplanetary, aerobraking, and primary and extended mapping orbit phases of the Mars Global Surveyor (MGS) mission have been described in Refs. 2 and 3 and references listed therein. The MGS spacecraft last communicated with Earth on 2 November 2006. After studying Mars four times as long as

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originally planned, the spacecraft appears to have succumbed to battery failure caused by a complex sequence of events involving the onboard computer memory and ground commands.

Mars Express

Interplanetary Phase. The European Space Agency's Mars Express spacecraft was launched toward Mars on 2 June 2003. NASA was a supporting partner in this ESA mission, providing Deep Space Network (DSN) tracking and certain navigation services. Cross-verification of ESA and NASA navigation software and validation of radio metric tracking data types were carried out both before and after launch, with Ulysses spacecraft tracking data used before launch. During the interplanetary flight of the Mars Express spacecraft, extensive interactions between the European Space Operations Centre (ESOC) and the Jet Propulsion Laboratory took place to refine the dynamic modeling of the spacecraft. Solar radiation pressure modeling received particular attention.[5,6,7]

Reaction wheel off-loadings were performed using unbalanced thrusters, thereby producing changes in spacecraft velocity averaging 21 mm/s every 4.3 days during cruise. Only a small component of the resulting velocity change could be directly observed in Doppler tracking data when the high-gain antenna was pointed at the Earth, because of the way that the antenna and thrusters were mounted on the spacecraft. In addition, several entries into safe mode caused unpredictable velocity changes of about 15 cm/s each, due to thruster-controlled attitude slews. Moreover, outgassing effects showed up in Doppler data residuals when certain spacecraft surfaces, normally in darkness, were exposed to the sun and increased in temperature.[8]

X-band radio metric tracking data for the mission were provided roughly equally by ESA's new 35-m antenna at New Norcia in southwestern Australia and antennas of the DSN (particularly at the Madrid complex). Format conversions were needed to process the New Norcia radio metric data with JPL's navigation software. Arrangements were made for generating ionospheric and tropospheric calibration data associated with the New Norcia tracking site.[5,8]

In addition to conventional radio metric data, the DSN supplied delta-differential one-way range (ΔDOR) data for the Mars Express spacecraft, generated along the Goldstone-Madrid and Goldstone-Canberra baselines. The relatively narrow bandwidth of the Mars Express signal spectrum limited the nominal accuracy of the ΔDOR delay observable to 0.25 ns. During the cruise phase of the mission, ΔDOR measurements were made every three or four days. Prior to critical events, such as Beagle 2 separation and Mars orbit insertion, these measurements were made as frequently as twice per day. Nearly all of the 55 scheduled ΔDOR measurements were obtained successfully, and the scatter in the data residuals turned out to be less than expected.[5,8]

Two months after launch, JPL and ESOC performed a set of software and auxiliary file cross-verification tests during a two-week tracking campaign, to verify the equivalence of JPL and ESOC navigation solutions, test the file exchange interfaces, and ensure that the output products were compatible and complete. During the last several weeks prior to Mars orbit insertion, JPL and ESOC exchanged orbit determination solutions daily.[5,6,7]

After correcting for launch injection errors early in the mission, trajectory-correction maneuvers (TCMs) executed on 10 November and 16 December 2003 served largely to remove trajectory errors that had been introduced by spacecraft safing events. Leading up to the release of the Beagle 2 lander, there were some inconsistencies between JPL and ESOC orbit solutions, which were resolved by ensuring that Earth orientation parameters were updated daily.[8,9]

No reaction wheel off-loadings were allowed to take place between the 16 December TCM and separation of the Beagle 2 lander from the Mars Express spacecraft on 19 December. The lander, spinning at about 14 rpm after its release, could not be tracked after its separation; but its trajectory could be inferred from the pre-separation Mars Express flight path and the observed Mars Express velocity change experienced during separation (an event tracked using the spacecraft's S-band low-gain antenna). The JPL and ESOC Beagle 2 trajectory solutions agreed to within 3 km at atmospheric entry (on 25 December), were about 6 km from the targeted conditions, and were well within the flight path angle requirement (at a height of 120 km above the reference ellipsoid) of -16.5 ± 1.0 deg (3σ). Errors in delivery of the lander to the top of the Martian atmosphere were dominated by position and velocity errors prior to separation, rather than separation velocity change (ΔV) errors or solar radiation pressure uncertainties.[5,10,11]

The entry, descent, and landing scenario for Beagle 2 involved the use of an aeroshell, drogue and main parachutes, and airbags. Based on the available data and various modeling assumptions, the 3σ landing error ellipse was calculated to have semi-axes of 28.5 and 3.8 km, with its center displaced 14.3 km from the nominal landing point. Contact with the lander after its atmospheric entry was never established.[11]

A final TCM was executed by the Mars Express spacecraft on 20 December to shift the Mars arrival conditions by a planned 2311 km in the B-plane,[1] to place the spacecraft on a different trajectory from that of the Beagle 2 lander and set up proper arrival conditions at Mars. Orbit determination solutions using Doppler data only, Doppler and range data, and Doppler, range, and ΔDOR data were found to be statistically consistent, with the addition of data types resulting in significant improvements in accuracy.[5,8]

The B-plane location achieved by the final TCM was off by 11.3 km, but was known to an accuracy of 0.7 km at the time of the final design of the Mars Orbit Insertion maneuver. (Around this time JPL and ESOC orbit solutions differed by just 0.5 km.) An 807-m/s Mars Orbit Insertion maneuver was executed on 25 December, accurately placing the spacecraft into a low-inclination, highly eccentric, 10-day orbit about Mars.[8]

Operational Orbit. A maneuver to change orbit inclination was performed on 30 December. Apoapsis reduction maneuvers were performed on 3, 6, and 11 January 2004. These four maneuvers were all performed with the 400-N main engine and varied from 55 to 160 m/s in size. Seven further maneuvers were executed with the 10-N attitude control thrusters to complete the apoapsis reduction, with an operational orbit achieved on 28 January.[9,12]

The operational orbit was chosen to maximize the scientific return of the mission and satisfy various instrument requirements pertaining to surface coverage from low altitude, ground track spacing, illumination conditions, and lander relay support, while limiting propellant usage. Initial orbit characteristics included a periapsis radius of 3670 km, an apoapsis radius of 15039 km, an inclination of 86.6 deg, and an argument of periapsis of -2.0 deg. In May 2004 the orbit period was reduced so as to change the 13:4 resonance with Mars' rotation period to an 11:3 resonance. Throughout the operational orbit, maneuvers were executed around apoapsis (except at very high periapsis latitudes), either parallel or antiparallel to the velocity vector, to control orbit period in addition to off-loading reaction wheel angular momentum. An upcoming series of such maneuvers was calculated twice per week.[12,13]

Mars Exploration Rovers

The interplanetary phase of the Mars Exploration Rover (MER) mission has been described in Reference 3 and references listed therein. Reference 3 left off with a brief mention of various events during final approach to Mars that could be monitored with one-way Doppler data, including transition from medium-gain to low-gain cruise-stage antenna, turn to entry attitude, heat rejection system venting, cruise-stage separation, atmospheric deceleration, and parachute deployment, all occurring at times ranging from atmospheric entry minus 105 minutes to entry plus four minutes. Once the switch to the cruise-stage low-gain antenna had been made, only one-way Doppler data were available, first from this antenna and later from the backshell low-gain antenna, with the data accuracy limited by on-board oscillator instability, temperature variations, and other effects. These data, when viewed with real-time Doppler residual displays, were adequate for qualitative verification that the various critical events had taken place.[14]

The atmospheric entry, descent, and landing (EDL) scenario for the MER spacecraft was generally similar to that used for the Mars Pathfinder mission (described in Reference 2 and references therein). MER-A and MER-B entered the Martian atmosphere on 4 January 2004 and 25 January 2004, respectively, at inertial speeds of 5.63 and 5.70 km/s. The EDL process was executed autonomously, but was subject to various parameter values that were updated on board prior to entry. The cruise stage was jettisoned 15 minutes before entry interface (defined to occur at a radius of 3522.2 km). The MER-A and MER-B aeroshells experienced maximum decelerations of 5.6 and 6.3 *g*. The MER-A and MER-B parachutes were deployed 251 and 250 s after entry interface, at altitudes of 7.5 km, wind-relative speeds of 411 and 434 m/s, and dynamic pressures of 730 and 764 N/m^2, with the deployment times selected by flight software processing of vehicle deceleration measurements obtained from two inertial measurement units.[15]

The heatshields were jettisoned 20 s after parachute deployment, followed by the initiation of the descent of each lander along its bridle 10 s later. The radar altimeters acquired the ground at altitudes of a few km. Their distance measurements to the local surface allowed the flight software to calculate when to fire the Rocket Assisted Deceleration (RAD) systems, in an effort to reduce the vertical velocity of each lander to zero about 12 m above the ground. These firings began 339 and 336 s after entry, at altitudes of 99 and 127 m and wind-relative speeds of 69 and 71 m/s. Airbag inflation began 0.5 s before RAD firing. The lander bridles were cut at heights of 8.5 and 6.9 m and wind-relative speeds of 12 and 9 m/s, with the inflated airbag/lander configurations falling freely to the ground and bouncing and rolling many times thereafter. The backshell and parachute were meanwhile transported a safe distance from each landing site using the remaining RAD system propellant.[15]

Extensive numerical simulations of the EDL process were carried out in advance, using three- and six-degree of freedom dynamical modeling software and taking into account a broad variety of error sources. The in-flight performance of the EDL system was fully consistent with the simulation results.[15]

The relative contributions to the landing error ellipses of interplanetary navigation uncertainties, atmospheric density variations, spacecraft aerodynamic modeling uncertainties, wind variations, and margins included to account for unmodeled or unknown effects are discussed in Reference 16. (The error ellipse sizes stated in Reference 3 did not include the latter margins, which were superimposed after the other effects had been combined in a standard statistical manner.)

EDL for both MER spacecraft occurred within sight of two DSN complexes, which permitted the generation of differenced one-way Doppler data, allowing the removal of oscillator drift and other error sources common to the two received signals. The resulting data were not accurate enough to decrease the orbit determination uncertainty in the spacecraft position and velocity at entry, but were adequate to constrain error growth during the atmospheric flight. Because the low-gain antenna boresights were pointed 22 and 39 degrees away from Earth at entry attitude for MER-A and MER-B, resulting in relatively weak signals, only the 70-m antennas at each complex had any chance of maintaining carrier lock through most of EDL. These large antennas were, in fact, able to track both spacecraft down to parachute deployment, at which point carrier lock was lost. These differenced Doppler data were processed within a few hours of each landing and resulted in quick solutions for landing locations accurate to within 10 km.[14,17]

Later radio metric solutions (as well as landmark triangulation techniques) indicated that MER-A and MER-B had landed 10.1 km and 24.6 km, respectively, from their targeted landing points on the Martian surface. These results were consistent with the statistical properties of the respective landing error ellipses.[16]

MESSENGER

Earth Flyby. The MErcury Surface, Space ENvironment, GEochemistry, and Ranging (MESSENGER) spacecraft was launched on 3 August 2004. The eventual insertion into orbit around Mercury was preceded by six planetary gravity-assist flybys (most of which occurred outside of the time interval covered by this paper). The first of these flybys was with Earth on 2 August 2005. This gravity assist was preceded by TCMs on 24 August, 24 September, and 18 November 2004, and 23 May and 21 July 2005. The first two of these TCMs were for the principal purpose of controlling orbit energy in the presence of launch dispersions (which were larger than expected), with the full maneuver executed in 80% and 20% portions due to its overall size of about 23 m/s. The third TCM was a small deterministic maneuver to correct the spacecraft's orbit plane. The last two of these TCMs refined the targeting for the Earth flyby. Images of the Earth and moon against star backgrounds were acquired with the Mercury Dual Imaging System, to test the optical navigation techniques to be used later on approach to Mercury.[18,19,20,21]

Venus Flyby 1. Two TCMs scheduled for shortly after the Earth flyby were cancelled as unnecessary, given the accuracy of the flyby. A Deep Space Maneuver (DSM-1) of 316 m/s was executed on 12 December 2005, the first use of the large velocity adjust, bi-propellant, 667-N thruster. The purpose of this DSM was to target for a Venus gravity-assist flyby (at a nominal altitude of 3040 km) on 24 October 2006, which would increase orbit inclination and reduce orbit period. A subsequent trajectory reoptimization, combined with modeling and estimation errors, resulted in the execution of a 1.3-m/s TCM on 22 February 2006. A subsequent TCM of 1.7 m/s was executed on 12 September. Maneuver execution and attitude and solar radiation pressure modeling errors necessitated an additional 0.5-m/s TCM on 5 October, resulting in a B-plane delivery error of about 60 km. This Venus flyby was complicated by a prolonged solar conjunction (increasing the noise in radio metric tracking data), a geocentric declination near zero, difficulty in accurately modeling solar radiation pressure forces because of attitude modeling errors, and larger than expected and unpredictable spacecraft velocity changes resulting from angular momentum dumps. Optical navigation using the narrow- and wide-angle cameras was tested on approach to Venus. Venus was not a good optical navigation target because of the indistinct edge of its atmosphere and the large difference in brightness between Venus and background stars; however, the tests were useful in preparation for later operational use on approach to Mercury. Delta-DOR data were not used operationally during this encounter but were fully validated for later operational use.[20,21,22]

Mars Reconnaissance Orbiter

Interplanetary Phase. The Mars Reconnaissance Orbiter (MRO) spacecraft was launched toward Mars on 12 August 2005. A 7.8-m/s TCM was executed on 27 August to correct launch injection errors and remove injection target biases included for planetary protection and to permit an early test of the 170-N main engines. Execution of the TCM in this fashion provided in-flight verification of the Mars Orbit Insertion modes of the propulsion and attitude control subsystems. Spacecraft velocity changes associated with nominally coupled angular momentum desaturations were calibrated subsequently. Only one additional TCM was needed on the way to Mars, with two later planned TCMs ultimately cancelled as unnecessary.[23,24]

X-band Doppler, range, and ΔDOR data were used for orbit determination during the interplanetary flight. ΔDOR data, continually improving over the years due to advances in digital signal processing techniques, higher volumes of sampled data, improved calibrations of Earth orientation and transmission media parameters, and more accurate and comprehensive radio source catalogs, were at this point accurate to better than 2 nrad.[25]

On approach to Mars, Phobos and Deimos were imaged relative to background stars using an experimental optical navigation camera. Ground-based computations then compared spacecraft orbit solutions derived from this information with solutions derived from radio metric data.[24,26]

Aerobraking Phases. The spacecraft was inserted into orbit around Mars on 10 March 2006 by means of a 1015-m/s Mars Orbit Insertion maneuver, resulting in an initial orbit with a period of about 35 h and a periapsis altitude of 428 km. The ensuing walk-in phase of aerobraking, beginning 25 March, included six propulsive maneuvers, which gradually decreased the periapsis altitude until the desired peak heating rate had been achieved at about 108 km.[24,27,28]

The main aerobraking phase began on 13 April. Solar array temperature, as influenced by the aerodynamic heating rate, was a limiting factor throughout this phase. The spacecraft was designed to have a high effective drag surface area, moderating the dynamic pressure (and hence the peak heating rate) needed for effective aerobraking. Moreover, the need to achieve a specific local mean solar time (LMST) for the eventual science orbit was a principal determining factor for the length of the aerobraking process. (The orbit plane remained nearly fixed in inertial space until the spacecraft began to spend an appreciable fraction of each orbit near the planet.) This resulting length (and the number of atmospheric passes implied) was sufficient to allow a substantial margin between the maximum allowable heating rate and that actually experienced on each pass. Aerodynamic heating rates for MRO wound up being comparable to those for MGS (after the in-flight reduction necessitated by the solar array latching problem mentioned in Reference 2) and about half of those for Mars Odyssey. The MRO spacecraft was designed to be aerodynamically stable during a drag pass, had balanced thrusters, and had better accelerometers than the two prior Mars aerobraking missions.[27,28,29]

Precession of the line of apsides due to planetary oblateness moved the periapsis poleward (from 70 deg S latitude) for the first 143 orbits and then toward the equator. With the periapsis initially well within the polar vortex, orbit-to-orbit variations in atmospheric density were mostly small, but later (after orbit 250) increased as the periapsis moved out of the vortex and toward the equator. The $J3$ long-period variation in eccentricity caused the periapsis altitude to increase (decrease) as the periapsis latitude moved southward (northward). Similarly, the oblateness of the planet caused the periapsis altitude to increase (decrease) as the periapsis latitude moved southward (northward). These combined effects served to drive the periapsis altitude upward initially and then downward, necessitating more frequent maneuvers to keep the periapsis altitude within the aerobraking corridor after orbit 180.[30]

Due to the time constraints associated with aerobraking operations, the orbit determination process involved the weighted least-squares estimation of a minimal set of parameters, including spacecraft position and velocity, a density mismodeling factor, a solar pressure mismodeling factor, periodic (per orbit) accelerations, and a constant acceleration around periapsis, based on two-way X-band Doppler data. The tracking data were available continuously, except around periapsis and DSN station handovers. Although various factors made atmospheric variations less dramatic for the MRO mission than for MGS and Mars Odyssey (solar activity, Martian season, and latitude range of interest), atmospheric modeling was nevertheless a challenging process, with a density multiplicative factor modeled as the sum of a (locally) constant term and terms periodic in longitude.[28,29]

The aerobraking operations process was simplified by limiting maneuver magnitudes to a menu of 20 possible values, ranging from 0.03 to 4.2 m/s and roughly evenly spaced in a logarithmic sense, rather than a continuum. Atmospheric uncertainties made precise maneuver calculations impractical, in any case. 14 maneuvers were executed for the purpose of adjusting the atmospheric corridor through which the spacecraft would pass at periapsis between 27 April and 29 August, a relatively modest number for an extended aerobraking process. The fact that the peak heating rates fell well within the safety limits helped to allow these relatively infrequent maneuvers and also allowed maneuvers to be delayed for a few orbits when operationally advantageous. Several other maneuver types existed, at least in concept, to respond to some sort of perceived threat to the spacecraft (due to an expected increase in atmospheric density, a spacecraft anomaly, or an extended entry into a safe mode), based upon either on-board or ground-based calculations.[27]

An on-board periapsis time estimator was used to update estimates of upcoming periapsis times, reducing the risk and effort of flying through an uncertain atmosphere. This algorithm estimated the velocity change experienced during the most recent drag pass from accelerometer data, updated the predicted time of the next drag pass, and adjusted the associated on-board sequence times, keeping them within the 225-s periapsis timing requirement and reducing the frequency with which new sequences needed to be generated on the ground. The predicted periapsis time was used to determine when the spacecraft needed to slew into a safe drag-pass configuration and when it was safe to slew back to the default orientation for communication with Earth. Errors larger than 225 s could result in the spacecraft not being in a safe orientation over some portion of the drag pass.[28]

Variations in orbit inclination occurred as the orbit period passed through resonances with the Mars rotation period. An effort was made to actively control inclination when performing aerobraking maneuvers, so as to avoid significant propellant expenditure for inclination control at the end of the aerobraking process – the first such instance of active inclination control.[27]

Orbital lifetime became the limiting factor during the endgame phase, which began when the orbit period had been reduced to 4 h. Here, the heating rate was limited such that the orbit apoapsis would remain above 300 km for at least 48 h in the absence of a maneuver. During this time there were up to 13 drag passes per day, and atmospheric predictions were refined daily. A number of close approaches to the MGS, Mars Odyssey, and Mars Express spacecraft took place, necessitating maneuvers to minimize the risk of collision. Six such maneuvers were executed between 22 and 27 August. Aerobraking was terminated with a 25 m/s maneuver on 30 August when the apoapsis altitude had reached 486 km (with an orbit period of 1.9 h). (Termination had been planned at an apoapsis altitude of 450 km, but collision avoidance analyses made it seem undesirable to seek this lower altitude.) Overall, aerobraking was accomplished over 445 orbits (including 23 before the main phase began), saving 1200 m/s in ΔV compared to the use of chemical propulsion.[27,28,31]

The 486 by 215 km orbit resulting from the aerobraking termination maneuver departed from the required Primary Science Orbit (PSO) in a number of respects. Thus, five additional maneuvers totaling 75 m/s were performed during a transition phase to establish the desired near-polar 250x315 km orbit with an ascending node at roughly 3:00 PM LMST, a sun-synchronous orbit inclination, the proper line of apsides for a frozen orbit condition, and the desired ground-track walk pattern. The first two maneuvers were executed on 5 and 11 September, before a one-month solar conjunction period began, producing an orbit approximating the PSO, along with a temporary, non-zero LMST drift rate. A final adjustment of orbit inclination was made on 15 November, to achieve the sun-synchronous value and cancel the LMST drift rate. Two final orbit adjustments were made on 13 December to modify eccentricity and argument of periapsis to achieve frozen-orbit conditions and adjust semimajor axis to establish the desired ground-track walk rate. LMST needed to be controlled to ±15 min in this mission, a tighter tolerance than for MGS (±35 min), but the same as for Mars Odyssey. A late, in-flight change in mission design had resulted in a revised target of 3:10 PM for LMST at aerobraking termination, a time that was achieved to within 20 s. The transition phase maneuvers allowed LMST to be shifted closer to the more desirable 3 PM. The PSO was to have a near repetition of ground tracks every 211 orbits (16 sols), with the ground track shifting 32.5 km west over this time. There was also to be a theoretical exact repetition of ground tracks every 4602 orbits (349 sols).[27,29]

Estimates of the variability of the Martian atmosphere during the science phase of the MRO mission were derived from orbit reconstructions of the Mars Express spacecraft during maneuver-free periods of February 2004, based on radio metric data from the New Norcia and northerly DSN tracking sites. From these reconstructions, the atmospheric density near periapsis could be evaluated, along with its variability and predictability.[32]

Venus Express

Interplanetary Phase. The European Space Agency's Venus Express spacecraft was launched on 9 November 2005. A TCM was performed two days after launch to correct launch injection errors and target the spacecraft relative to Venus. This and other smaller TCMs were executed using two sets of four 10-N thrusters. The 400-N main engine was test fired on 16 February 2006, to achieve confidence in its eventual use for orbit insertion. A TCM was executed on 24 February to retarget the spacecraft for Venus. Thereafter, only essential activities were executed on the spacecraft, to minimize the likelihood of entry into a safe mode. The spacecraft was oriented in early March to illuminate the face that nominally remains cold, with significant outgassing resulting, in order that little outgassing would result on approach to orbit insertion, with the spacecraft in a similar orientation. On 29 March a 0.13-m/s TCM was performed, largely to compensate for solar radiation pressure acceleration uncertainties and implementation errors in reaction wheel offloadings. Later TCM opportunities at 6 days, 2 days, and 6 h before Venus orbit insertion were not used, given the achieved flight path accuracy. Due to the in-flight calibration of solar radiation pressure disturbance torques, no reaction wheel offloadings were required during last two weeks before orbit insertion. On 11 April, the spacecraft was captured into orbit about Venus with a 1251-m/s maneuver using the main engine. Pointing a high-gain antenna at Earth during orbit insertion was not feasible. However, it was possible to monitor the burn through a two-way, carrier-only, link between a low-gain spacecraft antenna and a 70-m DSN antenna.[33]

Operational Orbit. The spacecraft was captured into a highly eccentric orbit about Venus with a period of about 9 days. On 15 April, at the first apoapsis, a 5.8 m/s maneuver was performed to lower the periapsis altitude to 250 km, the low end of the desired operating range. On 20 and 23 April, 200- and 105-m/s burns were executed with the main engine at periapsis to reduce the orbit period. Additional, much smaller, periapsis maneuvers were performed on 26 April, 30 April, and 3 May to further reduce the orbit period to about 24 h. Another apoapsis maneuver was performed on 6 May to reset the periapsis altitude to 250 km, since solar third-body gravitational perturbations were causing it to increase over time, thereby putting the spacecraft into its operational orbit about Venus.[33]

EXPLORATION OF THE OUTER PLANETS

Galileo

The Galileo mission did not extend into the time frame covered by this paper. However, References 2 and 3 included only very brief discussions of navigation during the Galileo Europa and Millennium Missions, due to the lack of pertinent references authored by members of the Galileo Navigation Team. In the appendix to this paper, additional material is presented regarding these extended mission phases, drawing from a sequence of papers written by the Galileo project management.

Cassini

The travels of the Cassini spacecraft in transit to Saturn, up through Saturn Orbit Insertion (SOI), have been described in References 2 and 3 and references listed therein. While the Cassini spacecraft remains in orbit about Saturn as of the date of this paper, the discussion below covers only the first few years in orbit.

General Characteristics of the Satellite Tour. While numerous natural satellites of Saturn were of scientific interest during the tour, only Titan was massive enough to provide significant gravity assists for trajectory shaping, with a spacecraft velocity change of 800 m/s achievable from a relatively low flyby altitude of 950 km. Throughout the tour, the larger orbit trim maneuvers (OTMs) were carried out using the 445-N bi-propellant main engine assembly, while the smaller ones ($\leq$ 0.3 m/s, generally) were carried out using the 0.9-N monopropellant reaction control system thrusters. Spacecraft attitude was normally maintained using the reaction wheel assembly. Periodic reaction wheel momentum dumps were performed one or more times per orbit.[34,35]

While in orbit about Saturn, control of the spacecraft trajectory was typically accomplished with three OTMs on each orbit. A cleanup maneuver was performed about three days after the previous satellite flyby to correct errors associated with that flyby. The first targeting maneuver for the upcoming flyby was performed near apoapsis and often included a significant deterministic component to shape the orbit. The second targeting maneuver was executed about three days before encounter to correct errors in the prior maneuver and to achieve accurate flyby conditions. Individual maneuvers were sometimes cancelled when the flight path was already accurate enough that performing the maneuver would add little value.[34,35]

During the cruise trajectory, the spacing of maneuvers was such that the development of a maneuver design could span a month, with the final orbit determination information made available five days before the maneuver. During the satellite tour, the orbit determination data cut-off needed to be shifted to about a day before the maneuver; and the whole maneuver development process needed to take place over less than a week. Consequently, maneuver automation software was developed to propagate and search for trajectories, generate and transfer pertinent files, generate and transmit email announcements, and produce presentation material and other supporting documentation. This resulted in the completion of what had previously been hours to days of work in less than an hour. The new software was, in essence, a master script to produce runstreams for the Double-Precision Trajectory Software (DPTRAJ) and Maneuver Operations Software (MOPS) (preexisting software sets with mission-related extensions) and deliver output products.[36]

Consistent, accurate, and efficient orbit determination was essential for meeting design requirements for targeting satellite encounters and the Huygens probe descent to Titan’s surface and for predicting science instrument pointing for targeted satellite encounters. Orbit determination solutions throughout the tour routinely estimated 90-150 constant bias parameters and 10-20 stochastic terms. New data arcs were typically started at each apoapsis, beginning with the best currently available information about spacecraft and Saturnian system parameters. Orbit determination solutions that included two satellite flybys and Saturn periapsis passages were typically difficult to converge fully due to various nonlinearities.[37]

After a targeted flyby, the first two tracking passes of radio metric data helped to determine the flyby conditions accurately and improve the targeted satellite’s ephemeris state at the orbital longitude of encounter. Improvements in estimates of satellite ephemerides, Saturn and satellite masses, Saturn zonal harmonics, and the Saturn pole vector, relative to those available before arrival at Saturn and which were achieved through the orbit determination process early in the satellite tour, were important for navigating the entire tour. In addition, the collection and processing of spacecraft-quasar, phase-referenced, interferometric data generated using the National Radio Astronomy Observatory’s Very Long Baseline Array from June to October of 2004 allowed improvements in the ephemeris and gravitational parameter of Saturn.[37,38,39]

Optical navigation center-finding accuracies were limited by the accuracies of satellite shape models and variabilities in surface topographies and albedos. Satellite-specific challenges included the fuzzy nature and variability of Titan's atmosphere, the unpredictable nature of Hyperion's spin and its non-spherical shape, and large albedo variations over Iapetus' surface.[26,37,40]

Titan-A Encounter. The first satellite encounter after the 1 July 2004 SOI was an 1174-km altitude flyby of Titan on 26 October 2004 (referred to as Titan-A). An OTM scheduled for three days after SOI was cancelled as unnecessary, given the small size of the calculated maneuver. The first OTM targeting to the desired encounter conditions was executed on 23 August. This Periapsis Raise Maneuver was quite large (393 m/s) and raised the ascending node crossing from between Saturn's F and G rings out to near Titan's orbit, while also establishing the orbital eccentricity to target Titan in subsequent flybys. A second OTM was executed near apoapsis on 7 September to clean up errors in the prior maneuver. A third OTM was performed on 23 October.[34,37]

Orbit determination solutions were derived from two-way coherent X-band Doppler and range data (collected at the Goldstone and Madrid DSN complexes, due to the northerly geocentric declination of Saturn around this time), as well as optical navigation data obtained using the 2000-mm focal length Cassini narrow angle camera. Doppler and range data were typically not available within 12 h of satellite encounters, when the spacecraft was pointed toward the satellite for science data collection, rather than toward Earth for communication. Several dozen optical navigation images were processed for each of Saturn's eight largest natural satellites.[26,37,40,41,42]

Estimated parameters in the orbit determination solutions included spacecraft position and velocity components at an epoch time, ΔV components and either start time or thrust level for each of the three OTMs, stochastic spacecraft nongravitational accelerations along body-fixed axes, radioisotope thermoelectric generator acceleration biases, RCS thrust event characteristics, position and velocity components for Saturn and its eight largest satellites, gravitational parameters for Saturn and its seven largest satellites, several gravity harmonic coefficients and the pole orientation for Saturn, range biases per tracking station and per tracking pass, stochastic camera pointing errors, and a Titan phase bias. Considered parameters[2] included station location errors, tropospheric and ionospheric signal delays, Earth orientation parameters, and the gravitational parameter of Hyperion. (Titan's 3:4 mean longitude resonance with Hyperion caused Titan's ephemeris to be affected by the gravitational parameter of Hyperion. Thus, the latter, uncertain quantity was given special treatment during this first Titan flyby.)[41,42]

The reconstructed flyby conditions for the Titan-A encounter differed from those targeted by 32 km in $B \bullet T$, 5.6 km in $B \bullet R$, and 4.2 s in time of closest approach. Uncertainties in the Titan ephemeris were major contributors to these errors. Optical navigation images of Titan had limited impact on ephemeris improvement because of difficulties in locating the image center of a satellite with a diffuse atmosphere, viewed at a considerable distance and various solar phase angles. The processing of radio metric data collected before and after the encounter improved estimates of Titan's ephemeris and mass, for subsequent use.[26,40,41,42]

Titan-B Encounter. The first use of the standard three-OTM-per-targeted-satellite-encounter strategy was for the orbit connecting the Titan-A and Titan-B encounters. Accordingly, OTMs were scheduled for 29 October, 21 November, and 10 December 2004 for the 1192-km altitude Titan-B encounter of 13 December. The first two of these OTMs were designed by a two-impulse optimization. The third OTM was cancelled, due to the small calculated size resulting from the accurate execution of the first two OTMs. Spacecraft pointing errors, which are due in part to orbit determination errors, were consistent with requirements during this encounter, which was not the case during the prior encounter – outcomes that were consistent with expectations in each case.[34,37,42,43]

Huygens Probe Delivery and Titan-C Encounter. A Probe Targeting Maneuver of 12 m/s was executed on 17 December for the purpose of targeting the combined Cassini spacecraft-Huygens probe to the surface of Titan. On 23 December, a small clean-up OTM was executed to further refine the trajectory. Requirements to be met by the Probe Targeting Maneuver and the subsequent cleanup OTM were the achievement of a probe entry angle corridor of -65.0±3.0 deg (99% probability) and an angle-of-attack corridor of 0±5 deg (3σ) at the time of atmospheric entry interface (1270 km above the surface of Titan). Execution of the cleanup OTM was necessary in order to meet these requirements, execution of the Probe Targeting Maneuver alone being not quite sufficient. Complicating the targeting process a bit was the presence of a nontargeted, 123,000-km flyby of mass-uncertain Iapetus on 31 December.[34,43]

The Huygens probe was separated from the Cassini spacecraft on 25 December and was thereafter on a ballistic flight path until entering Titan's atmosphere and landing on its surface on 14 January 2005. The actual entry flight path angle at entry interface was estimated to be -65.40 deg, with a standard deviation of 0.27 deg; and the actual angle of attack was estimated to be 1.445 deg, with a standard deviation of 0.005 deg.[34,43]

After passing through the atmospheric entry interface, the probe decelerated over several minutes to about Mach 1.5, at which time the pilot parachute was deployed. 2.5 s later, the back cover was released and the main parachute deployed. The main parachute was used for 15 min, then jettisoned, after which a drogue parachute was deployed for use during the remaining descent to the surface, over more than 2 h. The atmospheric flight path of the probe was reconstructed based upon estimated entry conditions and subsequent accelerometer, temperature and pressure sensor, and altimeter measurements.[44]

The Cassini orbiter flight path was modified by a 24-m/s Orbiter Deflection Maneuver (ODM) on 28 December, to target for a 60,000-km flyby during the 14 January Titan-C encounter and to set up the proper geometry for the spacecraft to serve as a telecommunication relay for three hours as the probe descended through Titan's atmosphere and landed on its surface. Several optical navigation images of the probe were obtained prior to ODM execution, which were helpful in separating probe release and orbiter detumble maneuver ΔV effects. On 3 January, a small cleanup OTM was executed. The orbiter was required to be pointed to an accuracy of 6 mrad (99%) while serving as a relay for probe data, with the pointing error arising due to several sources, including spacecraft orbit determination errors (with 3 mrad allocated thereto). Execution of the cleanup OTM was necessary to meet the pointing accuracy requirement over the full three-hour relay period.[26,34,40,43]

Titan-3 Encounter. After flying past Titan at 60,003 km, OTMs with significant deterministic components were needed to bring the flyby altitude down to 1579 km for the Titan-3 encounter of 15 February 2005. Thus, jointly designed OTMs of 22 and 19 m/s were executed on 16 January and 28 January, with the first of these closer than usual to the prior encounter due to a substantial penalty in ΔV associated with further delay. A third, much smaller, OTM was executed on 12 February.[34,37]

Enceladus-1, Titan-4, Titan-5, and Enceladus-2 Encounters. The next several encounters made use of a chained optimization technique, in which six OTMs (the first two on three successive orbits, at encounter plus three days and near apoapsis) were optimized (in terms of total ΔV magnitude) subject to the constraints that $B \bullet R$, $B \bullet T$, and time of flight be maintained at fixed values over each of the next three encounters. The third OTMs on each orbit, on approach to encounter, had no deterministic components and were small. In fact, two of these four OTMs were cancelled as unnecessary, given the orbit accuracies already achieved. In addition, the first cleanup OTM after the Titan-5 encounter was cancelled as unnecessary, due to the high accuracy of that encounter. The Enceladus-1, Titan-4, Titan-5, and Enceladus-2 encounters took place on 9 March, 31 March, 16 April, and 14 July 2005 at altitudes of 502, 2404, 1027, and 175 km. Flyby errors were all less than 10 km in $B \bullet R$ and $B \bullet T$ and 2 s in time of flight. Post-flyby trajectory reconstruction accuracies were typically accurate to a few hundred meters for these and prior satellite encounters. The Titan-4 encounter was the first flyby to occur after a Saturn periapsis passage. The Titan-5 encounter occurred at a low enough altitude that it was found necessary to incorporate an atmospheric drag model for Titan. Separating dynamical effects due to atmospheric drag, Titan's oblateness, and velocity changes due to spacecraft attitude maneuvers was made challenging during early, close Titan encounters due to a lack of radio metric tracking data from 12 h before to 12 h after closest approach. Multiple spacecraft revolutions about Saturn separated the Titan-5 and Enceladus-2 encounters.[34,37,45]

Titan-6, Titan-7, Hyperion-1, and Dione-1 Encounters. The spacecraft was entering a solar conjunction period at the time of the Enceladus-2 encounter, with the result that the cleanup OTM was delayed until 20 days afterward, near the periapsis of an "empty" revolution about Saturn. The Titan-6 and Titan-7 encounters brought the spacecraft's orbit into Saturn's equatorial plane in preparation for flybys of several icy satellites. These Titan encounters took place on 22 August and 7 September 2005 at altitudes of 3660 and 1075 km. Chained optimizations, with eight OTMs calculated over the next four orbits, were used to prepare for both of these encounters.[35,46]

As the result of an in-flight trajectory redesign, a non-targeted, 1497-km altitude flyby of Tethys was accomplished on 24 September. The Hyperion-1 and Dione-1 encounters took place on 26 September and 11 October at altitudes of 488 and 499 km. Of the 12 OTMs nominally scheduled to set up the four targeted encounters discussed in this subsection, five were cancelled because of the accuracy of the previous flyby, a negligible ΔV penalty downstream (<1 m/s) associated with cancellation, easy substitution of a pointing update, or the desire to reduce mission operations effort. All but the last of these encounters took place as the spacecraft was outbound from a periapsis passage. Most encounters during the second year of Cassini's orbital tour about Saturn took place within 10 km and a few seconds of the planned conditions, with the Titan encounters being more accurately controlled than those with the icy satellites.[35,46,47]

Titan-8, Rhea-1, and Titan-9 Encounters. The Titan-8, Rhea-1, and Titan-9 encounters took place on 28 October, 26 November, and 26 December 2005 at altitudes of 1353, 502, and 10411 km. In the orbit leading to the first of these, an OTM was performed just one day after the Dione-1 encounter, to avoid an OTM of excessive size. An OTM was also performed just one day after the Rhea-1 encounter. Over this three-orbit time span, three OTMs out of nine were cancelled due to their small sizes, the low ΔV penalties downstream associated with their cancellation, and the minimal impact on flyby conditions. The flyby of Rhea allowed the estimation of several low-degree and -order gravity field harmonics because continuous radio metric tracking was available throughout this particular encounter. Optical navigation images had been scheduled with a frequency of up to 3-6 per day from several months before SOI through November 2005. Subsequently, the frequency was reduced by 2/3 or more, reflecting the improvement in satellite ephemerides that had by then taken place.[35,40,46,48]

Titan-10, Titan-11, Titan-12, Titan-13, Titan-14, and Titan-15 Encounters. The Titan-10, -11, -12, -13, -14, and -15 encounters took place on 15 January, 27 February, 19 March, 30 April, 20 May, and 2 July 2006 at altitudes of 2043, 1813, 1951, 1855, 1879, and 1906 km. Odd-numbered Titan encounters among these (as well as Titan-9) were outbound, while even-numbered encounters were inbound. (This alternating sequence of flybys had the effect of rotating the spacecraft's orbit around Saturn toward the magnetotail, without changing the orbit period significantly, on the average.) With outbound-to-inbound orbit transfers only 20 days in length, it was hypothesized and then confirmed by analysis that fewer than three OTMs would be adequate on such transfers. Thus, one OTM was deleted from the nominal schedule for each of these transfers. Various other OTMs were ultimately cancelled due to their small sizes, the low ΔV penalties downstream associated with their cancellation, and the minimal impact on flyby conditions. Accumulated experience with the Cassini spacecraft allowed the development of an updated maneuver execution error model with reduced uncertainties in early 2006. Optical navigation images of Titan were not collected after 16 April because the errors in locating the centroid of Titan were too large for the images to be useful in further improving the Titan ephemeris, given the accurate radio metric data collected before and after the multiple flybys to date. By the time of completion of the Titan-10 encounter, the rss satellite ephemeris accuracies, projected 180 days into the future, had been reduced to 7, 2, 3, 3, 3, 0.3, 6, 9, and 14 km for Mimas, Enceladus, Tethys, Dione, Rhea, Titan, Hyperion, Iapetus, and Phoebe, respectively. By mid-2006, the gravitational parameter of Saturn had been determined to 3 parts in 10^8 and the corresponding values for the major satellites determined to better than 1%.[26,35,40,46,49]

EXPLORATION OF COMETS AND ASTEROIDS

Stardust

The first several years of the Stardust mission, through the January 2004 sample-collection flyby of Comet 81P/Wild 2, have been described in Reference 3 and references listed therein, as well as Reference 50. A fourth (and final) deep space maneuver was performed in February of 2004 to set up a return of the spacecraft to Earth.

The three-axis stabilized Stardust spacecraft was not equipped with reaction wheels. Thus, attitude control was performed entirely with the use of the hydrazine thrusters of the reaction control system, operating so as to maintain the spacecraft attitude within a deadband. These thrusters were all located on the same side of the spacecraft and produced unbalanced torques, so that attitude control maneuvers also produced translational velocity changes. It was important to take these unsought velocity changes into account in both orbit determination and propulsive maneuver design. The execution of TCMs was less accurate than anticipated pre-flight, due to uncertainties in what ΔV would result from rotating the spacecraft from its nominal attitude to the propulsive burn attitude and then back, as well as non-rigid body settling effects. A number of calibration activities were carried out over the several years prior to the sample return to Earth (the earliest of which were mentioned in Reference 3) to improve the modeling of these phenomena. Nevertheless, separating the effects of TCM execution errors, solar radiation pressure, and reaction control system (RCS) thrusting activity was a challenge when accurate knowledge and control of the spacecraft flight path were required.[51,52,53,54]

A TCM of 4.2 m/s was executed 60 days before arrival at Earth to clean up the effects of the calibration burns performed during the previous weeks and ensure that the spacecraft was on a near-Earth, but non-impacting trajectory (to avoid an uncontrolled atmospheric entry in the event of a subsequent spacecraft failure). The next TCM of 2.4 m/s was executed 10 days before atmospheric entry and was biased to ensure that no large (and unpredictable) spacecraft slews were needed to reach the burn attitude, as well as to ensure that the solar panels were adequately illuminated during the transitions to and from the burn attitude. This TCM placed the spacecraft on a trajectory to skip through Earth's upper atmosphere. Both this and the prior TCM were designed taking into account the ΔVs associated with expected attitude control subsystem events, the attributes of upcoming TCMs, and the sample return capsule (SRC) ΔV. Slow deadband walks, rather than abrupt attitude changes, were used as the Earth was approached, because they introduced smaller ΔV uncertainties. The baseline orbit determination filter strategy was modified as the spacecraft dynamics became better understood. Variations in data type, data arc length, and estimated parameters around the baseline case were studied.[51,52,53,54]

A final TCM of 1.3 m/s was executed 29 h before entry into the Earth's atmosphere. This TCM was performed in a fixed direction (that needed for SRC release), so as to minimize the introduction of trajectory errors due to spacecraft rotations, with the consequence that entry flight path angle (and thus downtrack landing position) was being controlled, but crosstrack position was not. (Each mm/s of velocity error at this maneuver time would result in about 1 km of landing error.) In the TCM design process, this TCM and the previous one were constrained to be larger than 1.0 and 1.4 m/s, respectively, to increase the likelihood of accurate maneuver execution.[51,52,53,54,55]

The SRC was spun up to 13.5 rpm and released from the main spacecraft 4 h before atmospheric entry, with the main spacecraft's flight path deflected by a 19.7-m/s burn 3.7 h before SRC entry to fly past Earth with a 250-km perigee. The SRC entered the Earth's atmosphere at 12.9 km/s, the highest entry speed of any Earth-returning mission to date. The inertial flight path angle at a reference entry altitude of 125 km was -8.21 deg, with a 3σ error of ±0.0017 deg, well within the requirement of ±0.08 deg. A supersonic drogue parachute was deployed (at approximately Mach 1.37), 15.0 s after a drag deceleration of 3 *g*s was sensed. The main parachute was deployed 350.6 s later, about 1.8 km above ground. The actual landing location within the Utah Test and Training Range (UTTR) on 15 January 2006 was about 8.1 km from the targeted landing point. (The SRC passed within 1 km of its B-plane target. The bulk of the landing error was due to atmospheric dispersions.) The expected landing dispersion ellipse, as determined from extensive numerical simulations, was 45 km x 19 km (at a 99% confidence level). The selection of the targeted landing point as well as the ultimate decisions to proceed with SRC entry, descent, and landing were made based on extensive probabilistic calculations regarding SRC safety and the safety of people and physical assets on or near the ground, taking into account various possible failure scenarios.[54,55,56]

The SRC lacked the electronics needed to obtain coherently transponded radio metric tracking data. Consequently, the DSN was not able to track the SRC after it separated from the main spacecraft. Instead, the U.S. Strategic Command tracked the SRC between separation and atmospheric entry, using its Space Surveillance Network's optical tracking facility on Maui, HI, and its long-range tracking radar on Kwajalein Atoll in the western Pacific Ocean. Once the SRC had entered the atmosphere, the UTTR acquired the SRC and tracked it during its descent to determine the landing location, using two radar sites and four tracking telescopes.[57]

The Stardust experience underscored the importance of balanced thrusters in sample return missions. While the sample return was executed very successfully, continuous small-force monitoring and trade studies absorbed a large amount of time and resources.[51,52]

Hayabusa

The Hayabusa spacecraft (called MUSES-C before launch) was launched on 9 May 2003 by Japan's Institute of Space and Astronautical Science. It was intended to demonstrate five technologies: the use of ion engines for primary propulsion in interplanetary cruise, autonomous guidance and navigation based on optical measurements, sample collection in an ultra-low gravity environment, direct entry into Earth's atmosphere from an interplanetary trajectory for sample recovery, and the combination of low-thrust propulsion and gravity-assist trajectory-modification techniques.[58,59,60]

An Earth gravity assist flyby, at an altitude of about 3700 km, took place on 19 May 2004. This was achieved by thrusting with the ion engines until 1 April and then executing small maneuvers with chemical thrusters on 20 April and 12 May. The Earth gravity assist flyby was executed quite accurately, with ISAS- and JPL-generated orbit determination solutions in close agreement. Ion engine thrusting resumed on 25 May. Throughout the interplanetary flight, the thrusting with the ion engines was paused every three weeks to allow three radio metric tracking passes to occur during a dynamically quiet period. The spacecraft arrived 20 km from and roughly stationary relative to the near-Earth asteroid 25143 Itokawa, 0.3 by 0.6 km in size, on 12 September 2005. Thrusting with the ion engines had been terminated on 28 August, at a distance of 4800 km and a relative speed of 9 m/s, with the bipropellant chemical thrusters used for subsequent maneuvering relative to the asteroid. After a solar conjunction period in July, orbit determination solutions were obtained by processing both X-band radio metric data (derived from Usuda Deep Space Center and DSN tracking facilities) and optical navigation data (acquired first using the star tracker and then the telescopic optical navigation camera).[58,59,60,61,62,63,64,65]

During September and October, the spacecraft performed remote sensing of the asteroid from distances typically ranging from 20 km to 7 km (and once as close as 3 km for an asteroidal mass determination). The force on the spacecraft due to solar radiation pressure was larger than the gravitational force exerted by the asteroid at ranges greater than 5 km. A lidar device and the wide-angle optical navigation camera were available for on-board autonomous navigation. Such an approach was found to be less accurate than ground-based orbit determination, however, since the lidar data underwent large excursions due to the irregular shape of the asteroid and were not always available due to attitude variations resulting from the failures of two out of three reaction wheels (which necessitated the expansion of pointing control deadbands). Under favorable conditions, ground-based navigational computations were found to be accurate to a few 10s of meters and a few mm/s for spacecraft position and velocity relative to the asteroid.[58,59,60,62,66,67]

Test or rehearsal descents were conducted on 4, 9, and 12 November, dropping to within 55 m of the surface. (The spacecraft returned to its "home position" at a range of about 7 km between descents.) Optical navigation camera images were available almost every 15 min. In addition, the surface was detected using the lidar device (designed to operate at ranges of 50 m to 50 km) and a laser range finder (designed to operate within 100 m). A highly reflective target marker was deployed toward the surface (to serve as an artificial landmark for optical navigation within 35 m) and successfully tracked, various facets of the navigation and guidance process were tested, and possible landing sites were investigated. Beginning 12 November, heavy reliance was placed on expedited ground processing of wide-angle optical navigation camera images of ground control points (or surface landmarks) at ranges greater than 500 m. An attempt to deploy the Minerva hopping lander to the surface was unsuccessful. It was inadvertently released when the Hayabusa spacecraft was slowly rising from its minimum altitude, and it subsequently escaped from the asteroid's weak gravity field.[58,60,67,68,69,70,71,72,73,74,75,76,77,78,79]

The Hayabusa spacecraft first landed on the asteroid on 20 November, deploying a target marker on descent, hovering at 17 m and then falling freely from that height, touching down at 10 cm/s twice and bouncing upward, and then settling on the surface for 34 min before being commanded to ascend. The landing took place within 30 m of the desired location. No surface sampling was attempted because of the detection of an obstacle by the fan beam sensors. (This detection should have terminated the touch-down attempt, but did not due to an attitude error large enough to prevent an automatic, emergency ascent thruster firing. The target marker was tracked as a navigational reference until the interruptions caused by the obstacle detection.) Much of the descent navigation and guidance process was ground commanded, based on the experience gained in the rehearsal descents, given that Itokawa turned out to be a challenging target on which to land. Round-trip light-time delays necessitated that the terminal portions of the landing process (within 500 m of the surface) be carried out autonomously, however. The resulting landing was more tightly controlled than that of the NEAR spacecraft on Eros.[3] The ascent was the first from any natural body outside of the Earth-moon system.[60,66,68,69,70,71,72,73,74,75,79,80]

A second landing took place on 25 November, with the spacecraft again touching down at about 10 cm/s. The firing of two sampling projectiles into the asteroid's surface was commanded (but later found not to have occurred). The subsequent lift-off from the surface took place at 50 cm/s. A reaction control system failure and propellant leak after ascent from the surface precluded an immediate departure of the spacecraft toward Earth.[60,68,69,70,71,72,73,74,79]

Rosetta

The European Space Agency's Rosetta spacecraft was launched on 2 March 2004, with its ultimate destination being Comet 67P/Churyumov-Gerasimenko. X-band Doppler and range data were collected at ESA's New Norcia station and at the DSN's Goldstone and Madrid facilities. Reaction wheel off-loadings took place every 5 days on the average during the first year of flight. These off-loadings were executed by firing thrusters in a coupled fashion and resulted in velocity changes of only a small fraction of a mm/s.[81]

The spacecraft made a gravity-assist flyby of the Earth on 4 March 2005, after performing perihelion maneuvers totaling 158 m/s on 10 and 16 May 2004. Three statistical TCMs were executed between 25 November 2004 and 17 February 2005 to set up the Earth flyby, with the second TCM needed to correct for inaccurate pulse counting control in the first. The flyby took place within 3.7 km of the desired B-plane location, and no statistical TCM was needed thereafter. An anomalous increase in hyperbolic excess speed of 1.8 mm/s was observed in connection with this flyby. Similar small, unexplained increases in hyperbolic excess speed had been seen in connection with Earth flybys in the Galileo and Near Earth Asteroid Rendezvous missions. The passage through the Earth-moon system provided the opportunity to test the spacecraft's Asteroid Fly-by Mode by tracking the moon.[81]

Deep Impact

The Deep Impact flight system was launched on 12 January 2005 toward Comet 9P/Tempel 1. The flight system consisted of a flyby spacecraft and an impactor.[82]

Cruise and Approach Phases. A 28.6-m/s TCM (TCM-1) was executed on 11 February to correct launch vehicle injection errors of a random nature, as well as a targeting bias error (related to second-stage modeling) that was discovered too close to launch to be fully corrected. For several reasons, it was found desirable to omit a planned TCM-2 in favor of a later 5.0-m/s TCM-3A, which was executed on 5 May. This TCM included a bias component that was omitted from TCM-1 to avoid violating a geometric pointing constraint, corrected known trajectory errors relative to the most current ephemeris for Tempel 1, and adjusted the time of cometary impact to enhance viewing conditions from the Hubble Space Telescope, the Keck Observatory in Hawaii, and the DSN's Goldstone site.[82,83,84]

X-band Doppler, range, and ΔDOR data were collected and processed throughout the mission. Thruster firings to desaturate momentum wheels were executed in an uncoupled mode throughout most of the mission and occurred much more frequently than had been expected prelaunch, resulting in larger than expected trajectory perturbations. The availability of ΔDOR data helped prevent a significant degradation in orbit determination accuracy due to this effect. Early in the mission, it was difficult to distinguish between vehicle accelerations due to outgassing and solar radiation pressure effects.[82,83]

At the time of launch, it was expected that optical navigation data would become available about 60 days before the cometary encounter, at a distance of 50,000,000 km. Initial calibrations of the High Resolution Imager after launch revealed a focus problem, which turned out to be uncorrectable. Thus, the optical navigation process had to be revised to make use of the Medium Resolution Imager (MRI), with picture elements five times as large in angular terms (10 versus 2 μrad), resulting in initial detection of the comet later and measurement accuracy less than planned. In-flight calibrations were needed to align the camera boresight with respect to the flight system coordinate frame. The optical navigation images transmitted to the ground consisted of a number of "snip boxes" including the comet and the brightest stars, rather than full imaging frames containing much blank space. Picture registration (the alignment of image features with their expected locations) for initializing precision centerfinding routines was performed in an automated fashion, rather than by a more traditional manual approach. While image centroiding for stars was relatively straightforward, this was not the case for the comet because of the obscuration of the nucleus by the coma. Six comet centerfinding algorithms (eventually increased to eight) were used in a comparative fashion to see which produced the best data fits.[26,82,85]

A third TCM (TCM-3B) of 6.0 m/s was executed on 23 June to correct B-plane errors and shift the time of impact by 10 minutes. Deferring this final shift in timing until this TCM avoided constraint violations in either this or the prior TCM. It became possible to distinguish between the cometary coma and nucleus in optical navigation images about a day later. A planned TCM-4 was deleted since TCM-3B was executed later than originally planned.[26,82,84]

Orbit solutions (a suite of 10 to 15) were calculated every two hours during the week prior to encounter, with the solutions found to exhibit the desired degree of consistency. Pointing at the comet and collection of optical navigation data were carried out continuously during this time. Over the course of the mission, more than 2500 usable optical navigation images were produced, 1800 of which were taken during the final week.[82,83,85]

A fourth TCM (TCM-5) of 0.3 m/s was executed on 3 July (at E-30h, where E denotes encounter, the time of cometary impact), to reduce the B-plane error to within the required 5 km prior to impactor release. This was a “critical plane” maneuver, with adjustments made in B-plane coordinates but not in time of flight.[82,83,84]

Efforts were made before encounter to improve the knowledge of the physical properties of Tempel 1, such as its size, shape, topography, pole location, rotation period, and relative brightness of nucleus and coma. The cometary ephemeris was updated before and during flight by ground-based and on-board astrometric measurements. This ephemeris refinement was made challenging by the nongravitational accelerations experienced by the cometary nucleus due to stochastic outgassing as well as obscuration of the nucleus by the coma. Ground-based observations of the comet (in right ascension and declination) provided the most accurate ephemeris information until the last week before encounter. Even then, with optical navigation data providing the most accurate ephemeris information in two dimensions, the comet’s position along the line of sight from the spacecraft was determined principally from ground-based observations. Roughly 6000 ground-based astrometric observations of Tempel 1 were collected, spanning the time interval 1967-2005, with almost 80% collected during the first half of 2005. Distinguishing between poorly characterized observational biases and errors in modeling cometary nongravitational accelerations was a challenge. As the encounter drew closer and the prediction interval of interest grew shorter, the use of more recent observations was increasingly emphasized, particularly those acquired using the more accurate small pixel-scale instruments.[82,86]

The impactor was released at E-24h. About 16 minutes later, the impactor performed a detumbling maneuver to null the angular rates induced by separation. The translational ΔV resulting from this attitude maneuver shifted the impactor’s B-plane position by 1 km.[82,84]

Twelve minutes after impactor separation, the flyby spacecraft executed a divert maneuver of 102.5 m/s to target for a cometary flyby altitude of 500 km and delay closest approach to 850 s after impact. This maneuver was executed quite accurately (as was the case with the earlier TCMs), resulting in predicted errors in flyby altitude and time of closest approach within about 3 km and 1 s and making a contingency divert trim maneuver unnecessary.[82,84]

Radio metric data involving the impactor were not available after separation; however, optical navigation data from the Impactor Targeting Sensor (ITS) were transmitted to the ground by means of an S-band impactor-to-flyby spacecraft communication link. The ITS camera and photodetector were substantially identical to those of the flyby spacecraft’s MRI. The final ground-based orbit determination solution for the impactor (with data up to E-3h) showed a shift in impactor position in the B-plane of about 2 km relative to the final pre-release orbit determination solution.[82,83,84,87]

Autonomous Navigation. At E-2h autonomous navigation (Autonav) systems took control aboard both spacecraft. Autonomous navigation techniques and algorithms used in this mission were derived from prior experience with the Deep Space 1 mission,[2,3] with some simplification and a focus specifically on encounter. In addition, the timing of many events was determined by scripted sequences, rather than in a fully autonomous fashion. Optical navigation was carried out without searching for stars in imaging frames; instead, ITS pointing information was derived from star tracker data, due to the large difference in brightness between the comet and background stars. The key modules of the Autonav systems were image processing, orbit determination, maneuver computation (for the impactor only), and an executive to handle interfaces among these modules and with the attitude determination and control system and other spacecraft functions.[26,87]

Several image processing algorithms were used to locate the center of brightness of the cometary nucleus and select a suitable impact site. Orbit determination was performed using a batch-sequential least-squares filter of fixed data arc length, sliding over the roughly 2-hour encounter interval. The spacecraft's trajectory was integrated using the previously determined spacecraft state plus accelerometer data, predicted pixel and line locations were calculated for optical navigation images, and residual differences between observed and predicted values were evaluated. The residuals were minimized in a least-squares sense to solve for the three position and velocity components and (optionally) two cross-line-of-sight attitude drift bias parameters. Aside from the first 10 min of the encounter, where images were accumulated before producing an orbit determination solution, orbits were calculated every minute, based on the most recent 20 min (if available) of optical data, with new images available every 15 s on the average.[87]

Impactor targeting maneuvers (ITMs) were executed autonomously at E-90 min, E-35 min, and E-12.5 min. Only lateral trajectory errors were corrected by these maneuvers, which were 1.3, 2.2, and 2.3 m/s in size. The maneuvers were terminated when accelerometer data indicated that the commanded velocity changes had been achieved. The first of these ITMs actually shifted the trajectory farther from the cometary nucleus, rather than closer, perhaps due to an attitude bias error. The second ITM, again using a center-of-brightness targeting technique, restored an impact trajectory; and the third ITM, based on a scene-analysis algorithm, shifted the trajectory to the preferred sunlit impact location. The 10.3-km/s impact occurred on 4 July, just 2 s after the targeted impact time and within 220-270 m of the targeted impact site.[82,87]

Over the final two hours, the flyby spacecraft independently calculated its own trajectory and that of the impactor, in order to execute a series of slews to aim instruments at the expected impact point so as to record the collision and subsequent plume formation, turn to a shielded attitude during the flyby, and again point instruments at the comet after the flyby.[26,82,87]

On 20 July the flyby spacecraft performed a retargeting maneuver of 97 m/s to establish an Earth-return trajectory, allowing for the possibility of an extended mission to another natural body of the solar system.[84]

CONCLUSION

Deep space navigation capabilities, which had evolved enormously from the 1960s through the early 2000s, continued to evolve thereafter, benefiting the 11 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. 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. Autonomous navigation was used to impact a small natural body.

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APPENDIX: GALILEO EXTENDED MISSIONS

Galileo Europa Mission

The navigation of the Galileo spacecraft during the prime mission while in orbit about Jupiter was sufficiently accurate that less propellant than expected was needed for trajectory correction. Thus, a substantial amount of propellant remained for use during extended mission phases.[88]

On 7 December 1997, the Galileo prime mission came to an end; and the Galileo Europa Mission began. This extended mission began with flybys of Europa on 16 December, 10 February, 28 March, 31 May, and 20 July. The first four of these took place at altitudes of 200, 3562, 1645, 2515 km, respectively. During the fifth encounter, three transient bus resets caused the spacecraft to enter the safe mode, which halted the science data gathering sequence.[89]

Additional flybys of Europa took place on 26 September 1998 at 3582-km altitude, 22 November 1998 at 2273-km altitude, and 1 February 1999 at 1439-km altitude. A despun bus reset event occurred during the second of these encounters, resulting again in spacecraft safing and termination of the science sequence.[90]

Flybys of Callisto took place on 5 May 1999 at 1315-km altitude, 30 June at 1047-km altitude, 14 August at 2296-km altitude, and 16 September at 1057-km altitude, as essential parts of a Perijove Reduction Campaign to enable subsequent flybys of the innermost Galilean satellite Io.[90]

The first Io encounter took place on 11 October 1999 at an altitude of 612 km. A second Io encounter took place on 26 November at an altitude of 300 km.[91]

Galileo Millennium Mission

The Galileo Europa Mission ended and the Millennium Mission began with a 351-km altitude flyby of Europa on 3 January 2000. This was followed by a 198-km flyby of Io on 22 February. An 809-km altitude flyby of Ganymede followed on 20 May, which began the Cassini phase of the Galileo Millennium Mission.[91]

To maximize the likelihood that the Galileo spacecraft would be fully functional when the Cassini spacecraft passed through the Jovian system in December, the Galileo spacecraft spent much of 2000 in a large orbit outside of the intense radiation environment close to Jupiter. On 28 December, the Galileo spacecraft flew past Ganymede at an altitude of 2337 km. The Cassini flyby of Jupiter on 30 December has been described in Reference 3 and references therein. Many coordinated scientific observations were made possible by the simultaneous presence of the Galileo and Cassini spacecraft in the vicinity of Jupiter.[92]

Still in a large orbit after the 28 December flyby of Ganymede, the Galileo spacecraft flew past Callisto on 25 May 2001 at an altitude of 138 km, using this outbound gravity assist to lower perijove and begin the Io phase of the Galileo Millennium Mission. An Io flyby took place on 6 August at an altitude of 194 km.[92]

The next flyby of Io took place on 16 October 2001 at an altitude of 184 km. The flyby conditions were in error by less than 3.5 km in the B-plane and 0.7 s in time of flight. The final flyby of Io took place on 17 January 2002 at an altitude of 102 km. Both pre- and post-encounter TCMs were cancelled as unnecessary for this encounter, for the first time during Galileo orbital operations. The flyby conditions were in error by 2.4 km in the B-plane and 5 s in time of flight – an accuracy sufficient to place the spacecraft on a ballistic trajectory that would impact Jupiter two orbits later. As had happened earlier during the Galileo extended missions, transient bus resets during this encounter resulted in spacecraft entry into safe mode and cancellation of the science sequence, until corrective measures could be taken. Imaging of Amalthea was carried out to update the ephemeris of this small inner satellite in preparation for the next encounter.[93]

The final satellite encounter, with Amalthea, took place on 5 November 2002 at an altitude of 163 km, deep within the high-radiation environment in the vicinity of Jupiter. A new closest approach to Jupiter for the spacecraft (at 71500 km altitude) took place shortly thereafter. Two-way Doppler data were not obtained during the Amalthea encounter; nevertheless, a mass determination was made using one-way data. One orbit later, on 21 September 2003, the Galileo spacecraft entered the Jovian atmosphere and was destroyed, precluding the possibility of impacting (and contaminating) one of Jupiter's moons in the future.[94]

REFERENCES

[1] L.J. Wood, "The Evolution of Deep Space Navigation: 1962–1989," *Advances in the Astronautical Sciences: Guidance and Control 2008*, Vol. 131, ed. M.E. Drews and R.D. Culp, Univelt, San Diego, 2008, pp. 285-308.

[2] L.J. Wood, "The Evolution of Deep Space Navigation: 1989–1999," *Advances in the Astronautical Sciences: The F. Landis Markley Astronautics Symposium*, Vol. 132, ed. J.L. Crassidis, et al., Univelt, San Diego, 2008, pp. 877-898.

[3] L.J. Wood, "The Evolution of Deep Space Navigation: 1999–2004," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2014*, Vol. 152, ed. R.S. Wilson, et al., Univelt, San Diego, 2014, pp. 827-847.

[4] A.J. Butrica, *The Navigators: A History of NASA's Deep-Space Navigation*, Andrew J. Butrica, 2014.

[5] D. Han, et al., "Mars Express Interplanetary Navigation from Launch to Orbit Insertion: The JPL Experience," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[6] F. Budnik, T.A.Morley, and R.A. Mackenzie, "ESOC's System for Interplanetary Orbit Determination," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[7] T. Martin-Mur, et al., "Exchange of Standardized Flight Dynamics Data," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[8] T. Morley, et al., "Mars Express Navigation for Beagle 2 Separation and Mars Orbit Insertion," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[9] V. Companys, et al., "Operational Implementation of Mars Express Orbit and Attitude Control," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[10] R. Cramm, J.M.deJ. Gamo, and R. Bauske, "Reconstructed Beagle 2 Trajectory," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[11] R. Bauske, "Dependence of the Beagle2 Trajectory on the Mars Atmosphere," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[12] M. Lauer, S. Kielbassa, and U. Herfort, "Assessment of AOCS In-Orbit Performance for Mars Express and Rosetta," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[13] J.A. Pulido-Cobo and J. Schoenmaekers, "Combined Control of the Operational Mars Express Orbit and the Spacecraft Angular Momentum," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[14] D.T. Baird, et al., "Mars Exploration Rovers Entry, Descent, and Landing Navigation," AIAA Paper 2004-5091, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Providence, RI, Aug. 2004.

[15] P.N. Desai and P.C. Knocke, "Mars Exploration Rovers Entry, Descent, and Landing Trajectory Analysis," AIAA Paper 2004-5092, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Providence, RI, Aug. 2004.

[16] P.C. Knocke, et al., "Mars Exploration Rovers Landing Dispersion Analysis," AIAA Paper 2004-5093, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Providence, RI, Aug. 2004.

[17] T.P. McElrath, et al., "Differenced Doppler Applications for Mars Exploration Rover Navigation," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[18] J.V. McAdams, et al., "Trajectory Design and Maneuver Strategy for the MESSENGER Mission to Mercury," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2005*, Vol. 120, Pt. II, ed. D.A. Vallado, M.J. Gabor, and P.N. Desai, Univelt, San Diego, 2005, pp. 1185-1204.

[19] B. Williams, et al., "Early Navigation Results for NASA's MESSENGER Mission to Mercury," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2005*, Vol. 120, Pt. II, ed. D.A. Vallado, M.J. Gabor, and P.N. Desai, Univelt, San Diego, 2005, pp. 1233-1250.

[20] R.M. Vaughan, et al., "Return to Mercury: The MESSENGER Spacecraft and Mission," Paper 1562, IEEE Aerospace Conference, Big Sky, MT, Mar. 2006.

[21] J.C. Leary, et al., "MESSENGER Operations and Critical Events," Paper ISTS 2006-o-2-09v, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 1582-1587.

[22] A. Taylor, et al., "Earth to Venus-1 Navigation Results for NASA's MESSENGER Mission to Mercury," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. I, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 1081-1100.

[23] J.E. Graf, et al., "The Mars Reconnaissance Orbiter Mission," Paper IAC-05-A3.3, 56th International Astronautical Congress, Fukuoka, Japan, Oct. 2005.

[24] J.E. Graf, et al., "Status of Mars Reconnaissance Orbiter Mission," Paper IAC-06-A3.3.01, 57th International Astronautical Congress, Valencia, Spain, Oct. 2006.

[25] J.S. Border, "A Global Approach to Delta Differential One-Way Range," Paper ISTS 2006-d-49, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 581-586.

[26] W.M. Owen, Jr., et al., "A Brief History of Optical Navigation at JPL," *Advances in the Astronautical Sciences: Guidance and Control 2008*, Vol. 131, ed. M.E. Drews and R.D. Culp, Univelt, San Diego, 2008, pp. 329–348.

[27] C.A. Halsell, et al., "Mars Reconnaissance Orbiter Aerobraking Control," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. II, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 2071-2088.

[28] S.W. Demcak, et al., "Mars Reconnaissance Orbiter Orbit Determination During Aerobraking," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. II, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 2103-2118.

[29] J.L. Prince and S.A. Striepe, "Mars Reconnaissance Orbiter Operational Aerobraking Phase Assessment," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. II, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 2089-2102.

[30] R. Tolson, et al., "Atmospheric Modeling Using Accelerometer Data During Mars Reconnaissance Orbiter Aerobraking Operations," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. II, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 1245-1255.

[31] R.S. Bhat, et al., "Mars Reconnaissance Orbiter Primary Science Orbit Acquisition," *Advances in the Astronautical Sciences: Astrodynamics 2007*, Vol. 129, Pt. II, ed. R.J. Proulx, et al., Univelt, San Diego, 2008, pp. 1051-1070.

[32] D.E. Highsmith, et al., "Mars Atmospheric Variability Above 250 km Altitude," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[33] V. Companys and M. Lauer, "Operational Experience During Orbit Insertion of Venus Express – ESA's Second Planetary Mission," Paper ISTS 2006-d-07, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 428-433.

[34] S.V. Wagner, et al., "Cassini-Huygens Maneuver Experience: First Year of Saturn Tour," *Advances in the Astronautical Sciences: Astrodynamics 2005*, Vol. 123, Pt. I, ed. B.G. Williams, et al., Univelt, San Diego, 2006, pp. 609-628.

[35] S.V. Wagner, et al., "Cassini-Huygens Maneuver Experience: Second Year of Saturn Tour," AIAA Paper 2006-6663, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[36] T.D. Goodson, et al., "Cassini-Huygens Maneuver Automation for Navigation," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1867-1885.

[37] P.G. Antreasian, et al., "Cassini Orbit Determination Performance During the First Eight Orbits of the Saturn Satellite Tour," *Advances in the Astronautical Sciences: Astrodynamics 2005*, Vol. 123, Pt. II, ed. B.G. Williams, et al., Univelt, San Diego, 2006, pp. 933-962.

[38] R.A. Jacobson, "The Orbits of the Major Saturnian Satellites and the Gravity Field of Saturn from Spacecraft and Earth-Based Observations," *The Astronomical Journal*, Vol. 128, July 2004, pp. 492-501.

[39] T. Martin-Mur, et al., "Use of Very Long Baseline Array Interferometric Data for Spacecraft Navigation," Paper ISTS 2006-d-50, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 587-592.

[40] S.D. Gillam, et al., "Optical Navigation for the Cassini/Huygens Mission," *Advances in the Astronautical Sciences: Astrodynamics 2007*, Vol. 129, Pt. I, ed. R.J. Proulx, et al., Univelt, San Diego, 2008, pp. 3-19.

[41] I.M. Roundhill, et al., "Orbit Determination Results for the Cassini Titan-A Flyby," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2005*, Vol. 120, Pt. I, ed. D.A. Vallado, M.J. Gabor, and P.N. Desai, Univelt, San Diego, 2005, pp. 185-202.

[42] J. Stauch, et al., "Preparing for the Huygens Probe Mission, Cassini Orbit Determination Results for the First and Second Targeted Titan Encounters," 6th International ESA Conference on Guidance, Navigation and Control Systems, Loutraki, Greece, Oct. 2005.

[43] J. Bordi, et al., "Orbit Determination Results and Trajectory Reconstruction for the Cassini/Huygens Mission," Third International Planetary Probe Workshop, Anavyssos, Greece, June-July 2005.

[44] S.A. Striepe, et al., "Huygens Titan Probe Trajectory Reconstruction Using Traditional Methods and the Program to Optimize Simulated Trajectories II," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. II, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 1853-1880.

[45] F.J. Pelletier, et al., "Atmospheric Drag Model for Cassini Orbit Determination During Low Altitude Titan Flybys," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. I, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 633-651.

[46] P. G. Antreasian, et al., "Cassini Orbit Determination Performance During Saturn Satellite Tour: August 2005 – January 2006," *Advances in the Astronautical Sciences: Astrodynamics 2007*, Vol. 129, Pt. I, ed. R.J. Proulx, et al., Univelt, San Diego, 2008, pp. 21-40.

[47] K.E. Criddle, et al., "Cassini Orbit Determination: The Only Targeted Hyperion Flyby of the Prime Mission," *Advances in the Astronautical Sciences: Guidance and Control 2006*, Vol. 125, ed. S.D. Jolly and R.D. Culp, Univelt, San Diego, 2006, pp. 407-424.

[48] R.A. Mackenzie, et al., "A Determination of Rhea's Gravity Field from Cassini Navigation Analysis," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2007*, Vol. 127, Pt. II, ed. M.R. Akella, et al., Univelt, San Diego, 2007, pp. 2149-2168.

[49] R.A. Jacobson, et al., "The Gravity Field of the Saturnian System from Satellite Observations and Spacecraft Tracking Data," *Astronomical Journal*, Vol. 132, Dec. 2006, pp. 2520-2526.

[50] S. Bhaskaran, et al., "Optical Navigation for the Stardust Wild 2 Encounter," *Proceedings of the 18th International Symposium on Space Flight Dynamics*, ESA SP-548, Munich, Germany, Oct. 2004.

[51] B. Kennedy, T. McElrath, and S. Nandi, "Modeling of Deadbanding Delta-V for the Stardust Earth Return: Calibration, Analysis, Prediction, and Performance," AIAA Paper 2006-6408, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[52] C. Helfrich, et al., "Maneuver Analysis and Targeting Strategy for the Stardust Re-Entry Capsule," AIAA Paper 2006-6406, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[53] S. Nandi, et al., "On-Orbit Maneuver Calibrations for the Stardust Spacecraft," AIAA Paper 2006-6409, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[54] D.T. Baird, et al., "Stardust Earth Return Orbit Determination," *Advances in the Astronautical Sciences: Astrodynamics 2007*, Vol. 129, Pt. III, ed. R.J. Proulx, et al., Univelt, San Diego, 2008, pp. 2785-2797.

[55] P.N. Desai, et al., "Entry, Descent, and Landing Operations Analysis for the Stardust Re-Entry Capsule," AIAA Paper 2006-6410, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[56] J. Tooley, et al., "Stardust Entry: Landing and Population Hazards in Mission Planning and Operations," AIAA Paper 2006-6412, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[57] D.C. Jefferson, et al., "Interfacing with USSTRATCOM and UTTR During Stardust Earth Return," AIAA Paper 2006-6411, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[58] T. Kominato, et al., "Optical Hybrid Navigation in Hayabusa – Approach, Station Keeping & Hovering," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1753-1772.

[59] T. Kominato, et al., “Optical Hybrid Navigation and Station Keeping around Itokawa,” AIAA Paper 2006-6535, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[60] J. Kawaguchi, “HAYABUSA – A New Solar System Cruiser,” Paper ISTS 2006-o-2-07v, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 1576-1581.

[61] J. Kawaguchi, A. Fujiwara, and T.K. Uesugi, “The Ion Engines Cruise Operation and the Earth Swingby of ‘Hayabusa’ (MUSES-C),” Paper IAC-04-Q.5.02, 55th International Astronautical Congress, Vancouver, Canada, Oct. 2004.

[62] J. Kawaguchi, A. Fujiwara, and T. Uesugi, “Hayabusa (MUSES-C) - Rendezvous and Proximity Operation,” Paper IAC-05-A3.5.A.01, 56th International Astronautical Congress, Fukuoka, Japan, Oct. 2005.

[63] T. Ohnishi, et al., “Hayabusa Orbit Determination Just Before the Rendezvous with Itokawa,” Paper ISTS 2006-d-12, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 446-450.

[64] T. Ohnishi, et al., “HAYABUSA Orbit Determination under Ion Engine Propulsion,” Paper ISTS 2004-d-48, *Proceedings of the 24th International Symposium on Space Technology and Science*, Miyazaki, Japan, May-June 2004, pp. 493-498.

[65] M. Yoshikawa, et al., “Orbit Determination of Hayabusa from the Launch to the Asteroid Arrival,” Paper IAC-05-C1.7.03, 56th International Astronautical Congress, Fukuoka, Japan, Oct. 2005.

[66] K. Shirakawa, et al., “Accurate Landmark Tracking for Navigating Hayabusa Prior to Final Descent,” *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1817-1825.

[67] T. Mizuno, et al., “LIDAR for Asteroid Explore Hayabusa: Development and on Orbit Evaluation,” Paper ISTS 2006-k-24, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 1058-1061.

[68] J. Kawaguchi, S. Hagino, and T. Ohshima, “Guidance and Navigation Operation for Rehearsals and Touch-Downs in Hayabusa,” *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1359-1378.

[69] M. Uo, et al., “Hayabusa’s Touching-Down to Itokawa – Autonomous Guidance and Navigation,” *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1805-1816.

[70] T. Kubota, et al., “Use of Laser and Optical Sensors in Terrain Alignment and Touchdowns,” *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1827-1841.

[71] J. Kawaguchi, “Overview of Hayabusa Flight – Landing at Asteroid Itokawa,” Paper ISTS 2006-d-10, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 440-445.

[72] J. Kawaguchi, “Hayabusa, Summary of Guidance, Navigation and Control Achievement in its Proximity Phase,” AIAA Paper 2006-6533, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[73] H. Morita, et al., “Hayabusa’s Real-time Landmark Tracking Navigation for Descents and Touching-Downs,” AIAA Paper 2006-6537, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[74] T. Hashimoto, et al., “Final Autonomous Descent Based on Target Marker Tracking,” AIAA Paper 2006-6538, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[75] T. Kubota, et al., “Touchdown Dynamics for Sampling in Hayabusa Mission,” AIAA Paper 2006-6539, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[76] M. Maruya, et al., “Navigation Shape and Surface Topography Model of Itokawa,” AIAA Paper 2006-6659, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[77] R. Gaskell, et al., “Landmark Navigation Studies and Target Characterization in the Hayabusa Encounter with Itokawa,” AIAA Paper 2006-6660, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[78] D.J. Scheeres, et al., "The Actual Dynamical Environment About Itokawa," AIAA Paper 2006-6661, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[79] M. Uo, et al., "Hayabusa Guidance and Navigation for Autonomous Final Descent and Touchdown," Paper ISTS 2006-d-15, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 451-456.

[80] J. Kawaguchi, S. Aida, and H. Morita, "Hayabusa, Detailed Guidance and Navigation Operations during Descents and Touchdowns," AIAA Paper 2006-6536, AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Keystone, CO, Aug. 2006.

[81] T. Morley and F. Budnik, "Rosetta Navigation at Its First Earth Swing-by," Paper ISTS 2006-d-52, *Proceedings of the 25th International Symposium on Space Technology and Science/19th International Symposium on Space Flight Dynamics*, Kanazawa, Japan, June 2006, pp. 593-598.

[82] R.B. Frauenholz, "An Overview of Deep Impact Navigation with Lessons Learned," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1151-1178.

[83] M. Ryne, et al., "Ground-Based Orbit Determination for Deep Impact," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1179-1202.

[84] R.S. Bhat, P.W. Stumpf, and R.B. Frauenholz, "Deep Impact Ground Navigation Maneuver Design and Performance," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1203-1229.

[85] W.M. Owen, Jr., et al., "Optical Navigation for Deep Impact," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1231-1250.

[86] S.R. Chesley and D.K. Yeomans, "Comet 9P/Tempel 1Ephemeris Development for the Deep Impact Mission," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1271-1282.

[87] N. Mastrodemos, et al., "Autonomous Navigation for Deep Impact," *Advances in the Astronautical Sciences: Spaceflight Mechanics 2006*, Vol. 124, Pt. II, ed. S.R. Vadali, et al., Univelt, San Diego, 2006, pp. 1251-1270.

[88] W.J. O'Neil, et al., "Project Galileo Completing Its Primary Mission," Paper IAF-97-Q.2.02, 48th International Astronautical Congress, Turin, Italy, Oct. 1997.

[89] R.T. Mitchell, et al., "Project Galileo: The Europa Mission," Paper 98-Q.2.01, 49th Congress of the International Astronautical Federation, Melbourne, Australia, Sep.-Oct. 1998.

[90] J.K. Erickson, et al., "Project Galileo: Completing Europa, Preparing for Io," *Acta Astronautica*, Vol. 47, Nos. 2-9, 2000, pp. 419-441.

[91] J.K. Erickson, et al., "Project Galileo: Surviving Io, Meeting Cassini," *Acta Astronautica*, Vol. 53, No. 2, 2003, pp. 85-122.

[92] E.E. Theilig, D.L. Bindschadler, and N. Vandermey, "Project Galileo: From Ganymede Back to Io," Paper IAF-01-Q.2.08, 52nd International Astronautical Congress, Toulouse, France, Oct. 2001.

[93] E.E. Theilig, D.L. Bindschadler, K.A. Schimmels, and N. Vandermey, "Project Galileo: Farewell to the Major Moons of Jupiter," *Acta Astronautica*, Vol. 53, Nos. 4-10, 2003, pp. 329-352.

[94] D.L. Bindschadler, E.E. Theilig, K.A. Schimmels, and N. Vandermey, "Project Galileo: Final Mission Status," Paper IAC-03-Q.2.01, 54th International Astronautical Congress, Bremen, Germany, Sep.-Oct. 2003.