

## THE EVOLUTION OF DEEP SPACE NAVIGATION: 1989-1999

Lincoln J. Wood\*

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 second in a chronological sequence dealing with the evolution of deep space navigation. The time interval covered extends from the 1989 launch of the Magellan spacecraft to Venus through a multiplicity of planetary exploration activities in 1999. 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 a dozen planetary missions.

### INTRODUCTION

A previous paper has described the evolution of deep space navigation over the time interval 1962 to 1989.<sup>1</sup> The missions covered ranged from the early Mariner missions to the inner planets to the Voyager mission to the outer planets. The current paper extends the previous paper by one decade. It covers 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. As in the previous paper, attention is limited to those missions that involved travel well in excess of 1,500,000 km from the Earth and that were targeted to fly close to one or more distant natural bodies. Attention is focused primarily on planetary missions that were managed by the National Aeronautics and Space Administration (NASA).

### EXPLORATION OF THE TERRESTRIAL PLANETS

#### Pioneer Venus Orbiter

The majority of the Pioneer Venus Orbiter mission took place before the time period covered by this paper.<sup>1</sup> In 1992, the periapsis altitude of the spacecraft's highly elliptical orbit about Venus, on a multi-year downward trend due to solar gravitational effects, was approaching the upper atmosphere of Venus. Enough on-board propellant remained to sustain the periapsis altitude high in the atmosphere (130-150 km) for a limited amount of time, enabling the collection of atmospheric data.

Around this time, power limitations on the aging spacecraft limited tracking to a few short intervals spread around each 24-hour orbit, rather than nearly continuous coverage, as had been available early in the mission. A tracking strategy of four short passes per orbit was tested starting in July 1992. Three orbits of

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\* Program Manager, Guidance, Navigation, and Control Section, Jet Propulsion Laboratory, California Institute of Technology, Mail Stop 301-150, 4800 Oak Grove Drive, Pasadena, California 91109. Email: Lincoln.J.Wood@jpl.nasa.gov

data were fit at a time to produce orbit solutions. A sequence of periapsis-raising maneuvers was executed from 8 September into early October 1992, at which point the propellant had been depleted. After a lowest-yet 128.5-km periapsis passage, with a resulting speed reduction due to atmospheric drag of 2 m/s, no radio signal was detected from the spacecraft.<sup>2</sup> The spacecraft entered the atmosphere, never to emerge, some unknown number of orbits thereafter.

## **Magellan**

*X-Band Uplink.* An X-band uplink capability (at 7.2 GHz) was added to one 34-m antenna at each Deep Space Network (DSN) complex in 1990.<sup>3</sup> Although originally motivated principally by the possibility of conducting gravitational wave detection experiments, the increase in the uplink frequency from the previously standard S-band offered numerous other benefits, some of which are related to navigation. For example, at the time of the Magellan mission, S-band and X-band Doppler data had expected measurement accuracies of 1.0 and 0.1 mm/s, respectively, for 60-s averaging times.<sup>4</sup> In addition, X-band radio metric data are much less affected by plasma interactions than S-band data because of the inverse-square frequency dependence of these interactions.<sup>1</sup> An X-band uplink capability was put into operational use at a second subnet of DSN 34-m stations in 1997 (well after the end of the Magellan mission).<sup>3</sup>

*Interplanetary Phase.* The Magellan spacecraft was launched on 4 May 1989 by the Space Shuttle Atlantis, using an Inertial Upper Stage. Coherent uplinks and downlinks were available at both S- and X-band frequencies during the interplanetary cruise toward Venus (with the X-band uplink first available for operational use at some point during cruise). The cruise and approach navigation were carried out using S-band and X-band two-way coherent Doppler data and X-band delta-differential one-way range ( $\Delta$ DOR) data.<sup>1</sup> The X-up/X-down Doppler data were of high quality after some initial problems were resolved.<sup>5</sup> The Magellan X-band high-rate telemetry subcarrier had harmonics that spanned a 30-MHz bandwidth – the spanned bandwidth being a major determinant of  $\Delta$ DOR data accuracy. The overall  $\Delta$ DOR measurement accuracy was expected to be 0.8 ns (24 cm), resulting in an angular accuracy of about 40 nrad using the DSN's intercontinental baselines.<sup>4</sup> The spacecraft transponder did not include a ranging channel, so that ranging data were not available.

The dynamical modeling of the Magellan cruise trajectory included the treatment of solar radiation pressure and reaction wheel momentum dump effects. Solar radiation pressure effects were modeled by representing the spacecraft as a self-shadowing parabolic antenna and several flat plates (with shadowing effects not included). Diffuse and specular reflectivity coefficients associated with each of these components were estimated as part of the orbit determination process.<sup>4</sup> The Magellan spacecraft was the first NASA planetary spacecraft to use reaction wheels for attitude control. The angular momentum accumulated in the reaction wheels was removed by executing thruster firings twice per day. Each week a file describing the size and duration of each burn was supplied to the project's navigation team by the spacecraft team.<sup>4</sup> The most accurate orbit determination solutions were obtained by processing all available Doppler and  $\Delta$ DOR data, with the X-band Doppler data deweighted relative to their inherent accuracy, to minimize the impact of modeling errors and allow the limited amount of  $\Delta$ DOR data available to exert a greater influence on the orbit solution.<sup>4</sup> Three trajectory-correction maneuvers (TCMs) were executed en route to Venus.<sup>6</sup>

*Elliptical Orbit Mapping Phase.* The Magellan spacecraft reached Venus and was inserted into an elliptical orbit on 10 August 1990. Magellan's principal scientific payload was a synthetic aperture radar (SAR), to map the surface of Venus. The SAR imaging of the planetary surface required an orbit inclination between 84 and 86 deg relative to the equator of Venus, an orbit period between 3.1 and 3.3 hours, a periapsis altitude between 275 and 325 km, and a latitude of periapsis between 0 and 10 deg north. With no orbit trim maneuvers executed after Venus orbit insertion, the first mapping cycle of roughly 243 days (one Venus rotation period), began on 15 September with an orbit inclination of 85.5 deg, an orbit period of 3.26 hours, a periapsis altitude of 295 km, and a latitude of periapsis of 9.7 deg north.<sup>4,7</sup> The achievement of these orbital characteristics subsequent to orbit insertion was enabled by the navigational accuracy attained during cruise and on approach to Venus.

The orbital period for the Magellan spacecraft was substantially shorter than for NASA's previous planetary orbiter missions. This made it impractical to conduct orbit determination by processing single orbits of data, with periapsis data deleted, as had typically been done before.<sup>1</sup> For a substantial fraction of an hour during each orbit, around periapsis, Doppler data were not available because radar data were being collected during that time using the high-gain antenna. In addition, Doppler data could not be collected for 15 minutes just after apoapsis, while reference stars were being scanned to calibrate the attitude control gyros.

Both two-way and three-way Doppler data were potentially available for use. Whereas the transmitting and receiving stations are the same with two-way data, the stations are different (and typically lie on different continents) in the case of "three-way" data. The differencing of two-way and three-way Doppler data (referred to as differenced Doppler data) provides information about the spacecraft velocity component perpendicular to the line of sight to the spacecraft and lying in the plane determined by the two tracking stations and the spacecraft.

Plans had existed to use delta-differential one-way Doppler ( $\Delta$ DOD) data during the orbit phase of the mission.  $\Delta$ DOD data is analogous to  $\Delta$ DOR data, involving the tracking of a spacecraft relative to a distant quasar from tracking stations on different continents,<sup>1</sup> but without the need to resolve ambiguities in spacecraft delay. Like differenced Doppler data, it provides information about the spacecraft velocity component perpendicular to the line of sight to the spacecraft and lying in the plane determined by the two tracking stations and the spacecraft. It is theoretically more accurate than differenced Doppler data, due to a greater cancellation of common error sources. However, it was found that interstation clock and frequency offsets could be calibrated to sufficient accuracy for the Magellan mission using Global Positioning System satellite signals. Thus, mission operations could be simplified by eliminating the quasar observations inherent in  $\Delta$ DOD data and relying on differenced Doppler data instead. The recent improvements in station time and frequency standards and the migration from S-band to X-band had made the differenced Doppler data type, which had been understood conceptually for some time, of practical use in flight operations. Tests of both differenced Doppler and  $\Delta$ DOD were carried out in transit to Venus.<sup>5</sup>

Orbit determination accuracy requirements existed for radial, downtrack, and crosstrack position components, both in absolute terms and in orbit-to-orbit relative terms, with the latter requirements (0.15-1.0 km) being considerably more stringent than the former (0.3-14.0 km). In addition there were downtrack and crosstrack orbit prediction accuracy requirements.<sup>7</sup> All of these requirements were motivated by various details of the SAR mapping process.

Uncertainties in the gravity field of Venus were an important error source for orbit determination while in orbit about Venus. Other dynamical effects that were modeled include atmospheric drag near periapsis, solar radiation pressure, and velocity changes due to reaction wheel desaturation thrustings. The modeling of solar radiation pressure was improved substantially during cruise. The angular momentum desaturations occurred every second orbit near apoapsis. Spacecraft position and velocity at an initial (or epoch) time and base density were the standard quantities estimated early in the mapping phase, with a local fifth degree and order (5x5) gravity field added later.<sup>8</sup>

Orbit determination solutions were carried out daily, using 12 orbits (39 hours) of data. Thus, orbit solutions on successive days contained 15 hours of common data, to reduce the relative error between the daily orbit calculations. The two-way Doppler data were deweighted relative to their inherent accuracy, to allow the differenced Doppler data to play a more significant role in the orbit solution. Either a face-on or edge-on geometry occurred every several months, and adversely affected orbit determination accuracy for about three weeks.<sup>8</sup> In face-on, edge-on, and normal, intermediate geometries, the inclusion of differenced Doppler data produced more accurate orbits than the processing of two-way Doppler data alone.

Orbit determination software, which in the past had resided on mainframe computers and minicomputers,<sup>1</sup> was transported to high-performance work stations in the early 1990s. Orbit determination for the Magellan mission was performed on a dedicated local network consisting of two 28.5 MIPS Sun Microsystems SPARCstations, a Sun 3/260 and three Sun 3/60s running UNIX, with 4GB of on-line hard-disk storage.<sup>8</sup>

The spacecraft's orbit plane relative to Venus was rotated by 0.106 deg by a propulsive maneuver after the first mapping cycle, to allow the cycle-2 altimetry data to be interleaved with the cycle 1 altimetry swaths.<sup>9</sup> Over the first three mapping cycles, 99% of the surface of Venus was mapped at least once by the SAR, with some regions mapped three times at different incidence angles. A second antenna collected altimetry data over 98% of the surface, in parallel with the SAR imaging. It was experimentally determined that orbit determination accuracy could be improved significantly, after the fact, by the use of SAR landmark measurements.<sup>10</sup>

Periapsis was lowered to 181 km before the fourth data collection cycle, for the purpose of improving the gravity-field model. The periapsis altitude remained in a  $175\pm 6$  km band throughout this cycle. The collection of high-resolution gravity data was limited to a  $\pm 30$ -deg latitude band near periapsis, due to the elliptical orbit.

*Aerobraking Phase.* After four data-collection cycles in a 3.3-hour orbit, an aerobraking maneuver, in which the spacecraft made repeated passes through the uppermost Venusian atmosphere to reduce the orbital energy and bring the spacecraft into a more circular orbit, was performed from 25 May until 3 August 1993, over the course of 730 orbits.<sup>11</sup> This aerobraking process reduced the apoapsis altitude from 8467 km to 541 km. To accomplish this orbit change, involving a total velocity change ( $\Delta V$ ) of 1219 m/s, in the standard propulsive manner would have required an order of magnitude more propellant than was available.<sup>12</sup>

The aerobraking process consisted of three phases: walk-in, main, and endgame. In the first of these, four propulsive maneuvers were executed over three days to gradually lower the periapsis altitude from 172 km to the aerobraking altitude of 141 km. During the main aerobraking phase, nine small propulsive maneuvers were executed to maintain the orbital periapsis in a time-varying, 2-km portion of a 134 to 141 km range, such that excessive aerodynamic heating would be avoided, with the aerodynamic drag still sufficient to gradually lower the periapsis speed. (It was really dynamic pressure at periapsis that was carefully tracked, rather than altitude, since the density at a given altitude can vary due to solar energy inputs, turbulence, and gravity waves.<sup>13</sup>) Six of these trim maneuvers served to raise periapsis, and three to lower it. The velocity change imparted by each atmospheric pass was about 1.8 m/s. In the endgame phase, five propulsive maneuvers were executed to raise the periapsis altitude to 197 km after the desired apoapsis altitude had been achieved.

Various antenna pointing and geometric constraints limited the theoretical maximum amount of tracking data per orbit to 133 minutes during the main phase and 20 minutes during the endgame phase. Orbit solutions were computed twice per day, with the estimated quantities including spacecraft position and velocity at an epoch time, the harmonic coefficients for a local  $8\times 8$  gravity field, the base atmospheric density (at a reference altitude of 131 km) for each drag pass, and constant accelerations of 600-s duration over each orbit to account for thruster firing activity for attitude maintenance while in the atmosphere.<sup>11</sup> It was particularly important to be able to predict accurately the dynamic pressure and periapsis time for upcoming drag passes. A corridor orbit trim maneuver was planned when an upcoming drag pass was expected to incur a dynamic pressure in excess of some limit, typically  $0.35 \text{ N/m}^2$ . Accurate predictions of periapsis times were needed to put the spacecraft in the correct attitude during the drag pass to reduce attitude control thruster firing. The orbit period at the end of aerobraking was 1.58 hours.

*Near-Circular Orbit Phase.* The next data collection cycle was used to acquire gravity information from the initially 541-km (apoapsis altitude) by 197-km (periapsis altitude) orbit from 3 August 1993 until 29 August 1994, producing high-resolution gravity data for about 95% of the planet. Doppler data could not be collected continuously, because of the need to do star scans to update attitude knowledge, the need to point the high-gain antenna toward the sun some of the time to put the rest of the spacecraft in shadow for thermal control, the need to desaturate the reaction wheels, the time required to establish a two-way link, occultations of the spacecraft by Venus, occasional superior conjunction conditions, and the demand for the use of DSN tracking stations by other projects.<sup>14</sup> Periapsis altitude fluctuations over a data-collection cycle were now considerably larger than in the earlier elliptical orbit, due to gravity field irregularities, necessitating occasional maneuvers to keep the peak dynamic pressure from becoming too large.

In September 1994 a “windmill” experiment took place, in which the solar panels were tilted at an angle so that atmospheric drag put a torque on the spacecraft, which was measured to give information about the atmospheric density at different altitudes. The Magellan spacecraft began its final descent into the Venusian atmosphere on 11 October 1994. On 12 October radio contact was lost, and the spacecraft presumably burned up in the atmosphere shortly thereafter.

### **Mars Observer**

The Mars Observer spacecraft was launched on 25 September 1992. It was the first spacecraft to communicate exclusively at X-band. Three TCMs were executed on the way to Mars. A planned fourth TCM was not executed due to the orbit accuracy achieved with the third TCM months earlier.<sup>15</sup> Orbit determination solutions constructed using various combinations of Doppler, ranging, and  $\Delta$ DOR data over various periods of time were found to be consistent throughout the mission.<sup>16</sup>

Communication with the spacecraft was lost on 21 August 1993, three days before the planned insertion into orbit around Mars. The most probable cause of the spacecraft failure was thought to be a rupture of the fuel pressurization tubing.

### **Mars Global Surveyor**

*Interplanetary Phase.* The Mars Global Surveyor (MGS) spacecraft was launched on 7 November 1996. Three TCMs were executed on the way to Mars. An additional TCM (the third in a planned sequence of four) was not executed. The MGS spacecraft was inserted into an elliptical orbit about Mars on 12 September 1997 by means of a 973-m/s maneuver. The actual periapsis altitude of 262.9 km was within 12.9 km of the targeted altitude of 250 km.<sup>17</sup>

*Aerobraking Phases.* Whereas aerobraking was employed in the Magellan mission only after many of the scientific objectives had already been accomplished, due to concerns about risk, this orbit modification approach was considered mature enough to be used upon arrival in the case of MGS, due to the successful Magellan demonstration. However, structural problems with a solar panel wing<sup>17,18</sup> required a more gradual aerobraking schedule than had been planned, so as to put less pressure on the solar panels.

Aerobraking was initiated with a sequence of six propulsive maneuvers at apoapsis, gradually lowering the orbital periapsis into the Martian atmosphere.<sup>19</sup> Aerobraking operations were suspended on 12 October, after an aerobraking pass with a dynamic pressure of  $0.93 \text{ N/m}^2$ , due to concerns about the damaged solar panel wing. The aerobraking process was replanned, extended in time, divided into two phases, and resumed on 7 November, with average dynamic pressure levels reduced to  $0.2 \text{ N/m}^2$  (and later revised upward somewhat). During the first aerobraking phase, the orbital period was reduced from 45 hours to 11.6 hours.<sup>19,20</sup> (It had already been reduced to slightly less than 35 hours at the time of the suspension and replanning of aerobraking.)

After the termination of this first aerobraking phase on 27 March 1998, the spacecraft entered a science phasing orbit. This phasing orbit avoided the need to carry out aerobraking operations during a solar conjunction period and allowed the ascending node of the orbit to regress toward the local mean solar time desired for the eventual science mapping orbit. A second phase of aerobraking was initiated on 23 September.<sup>21,22</sup> Three propulsive maneuvers were executed at apoapsis to gradually lower the orbital periapsis into the Martian atmosphere.

Throughout the aerobraking process, the orbit was maintained such that the dynamic pressure at periapsis fell into a corridor bounded above to limit the force acting on the spacecraft’s damaged solar array yoke assembly (as well as thermal effects on the spacecraft) and bounded below to ensure that the desired reduction in orbital period could be achieved by the scheduled end date for aerobraking (determined by the desired lighting conditions for the mapping phase). Remaining within this dynamic pressure corridor was complicated by the fact that atmospheric variations of 70% ( $2\sigma$ ) occurred from orbit to orbit and the fact that uncertainties in the Martian gravity field could cause the periapsis altitude to shift unexpectedly. Parameters associated with an atmospheric density model were estimated at almost every periapsis passage, based on the analysis of two-way, and occasionally one-way, Doppler data. For approximately one hour centered about periapsis passage, no Doppler data could be acquired, because the spacecraft orientation employed while passing through the upper Martian atmosphere was not consistent with pointing the high-

gain antenna toward Earth. Thus, the effects of the Martian atmosphere were deduced by comparing Doppler-determined orbit solutions before and after periapsis passage. Spacecraft accelerometer data recorded during the atmospheric passes were helpful in measuring atmospheric properties, particularly the density scale height.<sup>23</sup> It was necessary to be able to predict future periapsis passage times to 225-280 s, since the timing of the spacecraft's sequence of events during an aerobraking pass was relative to the assumed periapsis passage time. This prediction accuracy was achieved a sufficient number of orbits in advance to support the sequence generation process for periapsis passages.

The second phase of aerobraking was terminated by a propulsive maneuver on 4 February 1999. Over the course of the two aerobraking phases, the MGS spacecraft performed 891 drag passes, executed 92 trajectory control maneuvers, survived the onset of a Martian dust storm that tripled dynamic pressure at aerobraking altitudes, and underwent a reduction in spacecraft orbital period of more than 43 hours, while avoiding the propellant expenditure associated with a velocity change of nearly 1250 m/s that would have been needed for orbit modification without aerobraking.<sup>21</sup>

*Mapping Orbit Phase.* From 4 February to 26 February, a gravity calibration period, with nearly continuous Doppler and ranging tracking coverage, was carried out. On 19 February, the spacecraft was transferred into an initial mapping orbit by means of a propulsive maneuver. Science mapping, with the Mars Orbiter Camera and other instrumentation, began on 9 March 1999, with the MGS spacecraft in a 117-minute, nearly circular, nearly polar, frozen orbit with a mean altitude of 402 km.<sup>24</sup> (In a frozen orbit, the eccentricity and argument of periapsis are constrained by the gravitational field structure to vary over narrow ranges with time.<sup>25</sup>) The orbit was sun synchronous and crossed the Martian equator from south to north at 2 PM local mean solar time. The orbit had a 7-day near-repeat cycle, allowing for the mapping of Mars in 26-day cycles.

Major orbit determination challenges during the orbiting phase of this mission involved the modeling of the Martian gravity field and the autonomously commanded spacecraft angular momentum desaturations (AMDs).<sup>24</sup> The Martian gravity field model was significantly improved in accuracy over the course of the mission through the analysis of MGS Doppler data. AMD thruster firings affected the translational motion of the spacecraft, as well as unloading the angular momentum accumulated by the spacecraft's three reaction wheels; occurred about three times per day; and were difficult to model.

X-band Doppler data at 10-s intervals were collected over three to four orbits for trajectory prediction (to avoid the occurrence of more than one AMD event in the data span) and 17 to 22 orbits for trajectory reconstruction.<sup>24</sup> An ultra-stable oscillator was available on board for tracking and gravity field determination. Consequently, one-way Doppler data were found to be usable for orbit determination, provided that two-way data were available to resolve a 4.9-kHz bias in the one-way data. The quantities estimated include the spacecraft position and velocity at an epoch time, a subset of the gravitational field harmonic coefficients, velocity changes associated with AMD events during the data span, and the ultra-stable oscillator frequency bias and drift rate.

Orbit reconstruction errors were typically less than 10 m, 200 m, and 3 m in radial, downtrack, and crosstrack directions.<sup>24</sup> The results improved by a factor of three to ten during tracking data passes. The accuracy was somewhat degraded when the spacecraft orbit about Mars was close to edge-on, as viewed from the Earth.

Orbit prediction errors twenty days into the future were typically less than 0.2 km, 70 km, and 0.04 km in radial, downtrack, and crosstrack directions.<sup>24</sup> The effects of future AMDs were modeled by constant acceleration components, derived from recent AMD behavior. An accurate gravity field, MGS75C,<sup>26</sup> developed during the mission, was needed to allow proper treatment of AMD effects.

## **Mars Pathfinder**

*Interplanetary Phase.* The Mars Pathfinder (MPF) spacecraft was launched on 4 December 1996. Communications were at X-band. The circular polarization of the radio signals and an offset of the spacecraft's antenna boresight from the spin axis of the rotating spacecraft caused both a bias and a sinusoidal modulation to appear in the Doppler data.

The noise level of ranging data was required to be no more than 1m ( $1\sigma$ ). This requirement was not met during much of the interplanetary flight; but the data that were most important, taken during the last 35 days, had a noise level of 0.57 m ( $1\sigma$ ).<sup>27</sup> In previous planetary missions, range data had typically been weighted more loosely than the noise level would imply, due to various measurement and dynamical error sources that had been difficult to model accurately. (To a lesser degree, the same approach had been taken with Doppler data at times.) In the Mars Pathfinder mission, the approach was taken to weight range data near their inherent accuracy level, while using more sophisticated models to represent measurement and dynamical errors that might be important.<sup>28,29</sup>

The quantities estimated in generating orbit determination solutions included spacecraft position and velocity at an epoch time, solar radiation pressure parameters, velocity changes from TCMs and attitude control events, DSN station locations, Earth and Mars ephemerides (near encounter), a range bias due to spacecraft hardware delay, range biases for each DSN site, and a Doppler bias. In addition, a number of stochastic parameters were estimated. These stochastic parameters included pass-dependent biases for range and Doppler data, tropospheric and ionospheric transmission effects on the radio signal, Earth rotation and polar motion parameters, and miscellaneous nongravitational accelerations experienced by the spacecraft.<sup>27</sup> In previous planetary missions, effects that had been thought to be difficult to model had been treated as “considered” parameters – parameters that are not estimated, but with uncertainties that affect the error covariance associated with the estimated parameters. Considered parameters were not included in the Mars Pathfinder orbit determination solutions, due to the effort made to model and estimate all significant effects.

The flight path angle at entry to the Martian atmosphere was required to be  $-14.2\pm 1.0$  deg, to ensure a safe descent to the surface.<sup>27</sup> Four TCMs were executed en route to Mars to achieve the desired delivery accuracy. The first two had biased aimpoints due to planetary contamination issues. The third TCM was not quite sufficient to achieve the desired entry conditions. The fourth TCM was executed 9 days before arrival at Mars. Orbit determination accuracies did not improve significantly after the fourth TCM until the last 48 hours before entry, as the gravitational attraction of Mars began to appear in the Doppler data at levels greater than noise. These improved flight path estimates were used to update key parameters in the flight software; to update trajectory predictions used by the DSN to maintain contact with the spacecraft for as long as possible during the descent; and to allow for the possibility of a fifth TCM either 10.5 or 5.5 hr before entry, should it be necessary. This fifth TCM was not executed. The achieved entry flight path angle was  $-14.06$  deg.<sup>27,30</sup>

*Entry, Descent, and Landing.* On 4 July 1997, MPF entered the Martian atmosphere at an air-relative speed of 7.48 km/s. Whereas the Viking landers had entered the Martian atmosphere from orbit, the MPF spacecraft entered the atmosphere directly, at a considerably higher speed. In addition, whereas the Viking landers were designed to achieve lift as well as drag, the MPF spacecraft maintained an angle of attack near zero by virtue of its aerodynamically stable design and by spinning at 2 rpm. The entry, descent, and landing (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 30 min before entry interface (defined to occur at a radius of 3522.2 km), marking the formal beginning of the EDL process.

The aeroshell protecting the lander experienced a maximum deceleration of  $16 g$  78 s after entry interface (EI). The parachute began to unfurl 171 s after EI, at an altitude of 9 km and a speed of 375 m/s (Mach 1.8). The on-board decision to deploy the parachute was triggered by starting a timer when the accelerometer-sensed aerodynamic deceleration reached 5 g. The sensed deceleration 12 s after the start of the timer was used as an indexing parameter to select the proper time for parachute deployment from a stored functional relationship, which had been determined in advance from extensive simulations. There was a desire to deploy the parachute at a dynamic pressure of about  $600 \text{ N/m}^2$  (and a Mach number between 1.7 and 2.3). The triggering scheme was intended to accomplish this, using the available measurement information. In the event that this scheme appeared to be malfunctioning, a backup scheme was available, which would have initiated parachute deployment at a fixed time stored on board. The detailed parameter values associated with the timing of parachute deployment were updated 35 hours before entry, based on the latest available trajectory and atmospheric information. Opportunities for later updates existed, but changes in parameter values were not needed.<sup>31</sup>

Release of the forebody heatshield and extension of a bridle occurred 21 and 40 s after the initiation of parachute deployment. After the spacecraft had traveled down the bridle, the radar altimeter initiated the search for the Martian surface, at an altitude of about 1.6 km and a speed of 60-75 m/s. These altimetry data were used to infer the descent rate and determine the appropriate time of airbag inflation (295 s past EI) and solid-rocket ignition (299 s past EI, at an altitude of 88 m). After the solid-rocket motors had removed the remaining vertical velocity, the bridle connecting the lander and the backshell was cut, 21 m above the surface, and the spacecraft fell to a first surface impact at 305 s past EI.<sup>30,31</sup> The spacecraft bounced more than 15 times before coming to a stop. The Sojourner microrover, stowed within the lander, rolled onto the surface on 5 July.

Extensive numerical simulations of the EDL process were carried out in advance, using three- and six-degree of freedom dynamical modeling software. Efforts were made to improve the modeling of the Martian atmosphere using Hubble Space Telescope and Earth-based microwave measurements. An aerodynamic model of the vehicle was developed from computational fluid dynamic calculations and existing wind-tunnel and ballistic-range data.<sup>31</sup>

Landing was required to occur within a 100 km x 200 km elliptical footprint on the surface. The landing took place 27 km from the intended target landing site.<sup>31</sup> The knowledge of the spacecraft's trajectory upon entry was sufficiently accurate that atmospheric flight dispersions, due to aerodynamic and atmospheric uncertainties, were significant contributors to the landing error. Based on the successful approach navigation of the Mars Pathfinder spacecraft using the orbit determination filter configuration and the tight weighting of Doppler and range data described above, it was argued that future Mars missions should be navigated using only Doppler and range data, without the addition of optical or  $\Delta$ DOR data.<sup>27</sup>

### **Mars Climate Orbiter**

The Mars Climate Orbiter (MCO) spacecraft was launched on 11 December 1998. During the nine-month journey from Earth to Mars, thrusters were periodically fired to remove angular momentum buildup in the on-board reaction wheels. Preflight analysis had indicated that momentum unloading events would occur approximately once every seven days during cruise.<sup>32</sup> These angular momentum desaturation (AMD) events occurred much more frequently than this (10-20 times per week) and more frequently than in the case of the MGS spacecraft, because the MCO solar array was asymmetrical with respect to the spacecraft body, unlike the MGS spacecraft. This center of pressure/center of mass offset produced torques that resulted in significant accumulations of angular momentum in the MCO spacecraft's reaction wheels. A daily 180-deg reorientation of the spacecraft, which would have caused the solar pressure torque and the resulting angular momentum buildup and AMD thrusting effects to largely cancel over time, had been planned. These daily rotations were ultimately deleted from the spacecraft operations plan, however.<sup>33</sup>

Information on AMD events that was supplied to the navigation team by means of computer files represented these events in lbf-s, whereas the software interface specification called for the use of N-s. Thus, the effect of the AMD events was understated by a factor of 4.45. These files were not available for use by the navigation team until well into the flight. Since the AMD effects were mostly perpendicular to the Earth-spacecraft line of sight, the full extent of AMD mismodeling was not readily apparent in the Doppler data. Some degree of inaccuracy was apparent, however; and, with the problem reported but never fully resolved, the AMD data were assigned relatively large error standard deviations in the orbit determination process. These error standard deviations were small compared to the actual, yet unknown, errors, though. The variation in the capture orbit periapsis altitude was required to be no more than  $\pm 20$  km, with 99% probability. Preflight analysis had indicated that accurate modeling of the angular momentum desaturation maneuvers would be important in meeting this requirement, even at the originally expected rate of once per week.<sup>32</sup>

As the spacecraft approached Mars, orbit determination was carried out by processing Doppler data only, range data only, and both Doppler and range data, with the results of the three strategies compared for consistency. The Doppler-only solutions consistently indicated a closer approach to the planet than the other two solutions, with range-only solutions indicating a more distant approach. Four TCMs were executed on the way to Mars, with the fourth of these, executed on 15 September (and based on an 8 September orbit solution), targeting the periapsis altitude shortly after the Mars Orbit Insertion (MOI) burn

to be 226 km. Over the week after TCM-4, the predicted periapsis altitude decreased to the range of 150-170 km. A contingency plan for a fifth TCM was in place during the approach to Mars. Such a TCM could have been used as to raise the initial periapsis altitude. There were concerns regarding the risk and potential compromises to future mission performance associated with executing such a maneuver, however; and a fifth TCM was not attempted. With Mars exerting a significant gravitational attraction on the spacecraft over the last 24 hours before arrival, increasingly accurate orbit solutions were obtained, with the predicted periapsis altitude dropping to about 110 km about an hour before the MOI burn.<sup>33</sup>

The spacecraft executed the MOI burn on 23 September 1999. The spacecraft was to have entered a 13-14 hour orbit as a result of the MOI burn and then used aerobraking for several weeks to achieve a low-altitude, near-circular, near-polar orbit about Mars, for a two-year study of the Martian atmosphere and surface. The burn was observed in telemetry data for four minutes, until the spacecraft was occulted by Mars, 49 seconds earlier than predicted. A signal from the spacecraft was not reacquired after the 21-minute expected occultation interval. Once the error in the AMD data had been fully diagnosed and estimates of the spacecraft's trajectory recalculated using all available tracking data and corrected AMD data, it was determined that the initial periapsis altitude (with atmospheric effects ignored, for simplicity) would have been 57 km. A "vacuum" periapsis altitude of at least 80 km would have been needed to allow the spacecraft to survive a passage through the upper atmosphere.<sup>33</sup>

This incident was an unfortunate demonstration of the fact that orbit determination solutions based solely on line-of-sight radio metric data can be inaccurate if the forces acting on a spacecraft are modeled incorrectly. Some spacecraft position and velocity components are only weakly observable with line-of-sight data. Highly accurate dynamical and observable modeling allows these components to be inferred after the processing of sufficient data. The interplanetary portion of the Mars Pathfinder mission had been navigated very well using just line-of-sight radio metric data, in conjunction with good modeling. However, the reliance on single-vehicle DSN tracking to support planetary orbit insertion, in the context of MCO, was judged, after the fact, to have involved considerable systems risk, due to the possible accumulation of unobserved perturbations to the long interplanetary trajectory. The development and deployment of alternative navigational schemes were recommended.<sup>34</sup>

### **Mars Polar Lander**

*Interplanetary Phase.* The Mars Polar Lander spacecraft was launched on 3 January 1999. The spacecraft was three-axis stabilized, with attitude control performed using thrusters. Spacecraft attitude control was made complex because of the need to point the medium-gain antenna toward the Earth, the solar array toward the sun, and the star cameras in a direction such that they could update the on-board attitude estimate. In general, it was not possible to perform these three functions from the same spacecraft orientation. Thus, many spacecraft reorientations were needed during the mission to accomplish these three essential tasks. Information about all thruster firings for attitude control was included in spacecraft telemetry data, either individually or in terms of short-term average or accumulated effects. The attitude control activity was a major determining factor for the delivery accuracy at Mars. The solar pressure model consisted of four flat plates and a cylinder, with specular and diffuse reflection coefficients for each.<sup>35</sup>

The main tracking data types used during cruise were two-way Doppler and range data. Constraints on solid-state power amplifier temperature and power availability limited tracking intervals to four hours, followed by five hours of no tracking. Special efforts were made with both ground and on-board telecommunications equipment to obtain ranging data with a 1-m noise level. Late in the cruise, some three-way Doppler data were collected, so as to generate differenced Doppler data. During the last 30 days of cruise, near simultaneous tracking (NST) data, using the MGS spacecraft as a reference, were collected. By combining tracking data and estimating the trajectories of both spacecraft simultaneously, the effects of error sources common to both spacecraft were reduced, allowing a more accurate estimation of the lander's trajectory relative to Mars than could be achieved with lander tracking data alone. The selection of estimated parameters for orbit determination solutions was roughly the same as for the Mars Pathfinder mission. No Doppler bias was needed since the spacecraft was not spinning. A number of standard filter cases were routinely run, with differenced Doppler and NST data included in some cases, but not others, for example.<sup>35</sup>

Four TCMs had been executed on approach to Mars, along with a landing site adjustment maneuver, as of two days before arrival. The flight path angle at entry to the Martian atmosphere (again defined to occur at a radius of 3522.2 km) was required to be  $-13.25 \pm 0.54$  deg with 95% probability, to ensure a safe landing in the designated target area. Landing was to occur at a latitude of roughly  $-75$  deg. The question remained as to whether to execute a fifth TCM 6.5 hr before encounter. A suite of 10 possible TCMs was designed to vary the entry flight path angle in discrete steps, while keeping other entry parameters constant, should a TCM turn out to be necessary, based on future orbit determination solutions. It was ultimately decided to execute the TCM that would decrease the flight path angle by the minimum designed amount. The spacecraft reached Mars on 3 December 1999, entering the atmosphere at a speed of 6.9 km/s. The actual flight path angle was in error by 0.15 deg. The late orbit determination solutions drifted somewhat in the B-plane direction that was not being controlled by the fifth TCM; however, the expected landing site was within the target zone on the Martian surface. (The B-plane is centered on the planet and perpendicular to the spacecraft-planet relative velocity vector on approach, before the flight path has been affected by the planet's gravity.<sup>1)</sup>

*Entry, Descent, and Landing.* Upon arrival at Mars the spacecraft was to orient itself in the entry attitude and then jettison the cruise stage. Deceleration to an acceptable touchdown speed was to be accomplished by a combination of an aeroshell, a parachute, and thrusters. The parachute was to be deployed as the vehicle approached Mach 2, as calculated by the on-board inertial measurement unit, with the heatshield jettisoned after the parachute deployment transients had dissipated. As the lander neared the surface, a short-range Doppler radar was to acquire the surface. After the parachute had been jettisoned, a set of 12 pulse-mode hydrazine engines was to slow the vehicle for final touchdown on the Martian surface.<sup>36</sup>

After the planned communication outage during atmospheric entry and descent to the surface of Mars, contact with the spacecraft was never reestablished. The most probable cause of failure was the generation of spurious signals when the lander legs were deployed during descent. These signals would have falsely indicated that the lander had landed, resulting in a premature shutdown of the engines and a subsequent crash landing. An after-the-fact analysis determined that the Mars '98 project, consisting of the MCO and MPL missions,<sup>37</sup> was significantly underfunded from the start, for the established performance requirements.<sup>37</sup>

## EXPLORATION OF THE OUTER PLANETS

### Galileo

The Galileo spacecraft consisted of two vehicles: a Jupiter orbiter and an atmospheric probe, the first instance of each type of vehicle to explore an outer planet. The Galileo spacecraft was launched on 18 October 1989, carried into Earth orbit in the cargo bay of Space Shuttle Atlantis. It was then propelled onto its interplanetary flight path by an Inertial Upper Stage. Although earlier plans had called for Galileo to use a more powerful upper stage so that it could fly directly to Jupiter, the final flight path took it by other planets first so that it could gain energy from their gravity fields.<sup>38</sup>

*Venus and Earth-1 Flybys.* The Galileo spacecraft flew past Venus on 10 February 1990 at an altitude of 16,000 km and then past Earth on 8 December 1990 at 960-km altitude. Orbit determination solutions were obtained by processing S-band Doppler, range, and  $\Delta$ DOR data. Because the S-band transmissions were circularly polarized and the S-band antennas were mounted on the spinning portion of the spacecraft, a bias proportional to the spin rate (nominally 3.15 rpm) was present in the Doppler data and required modeling. In addition, one of the S-band antennas was offset from the spacecraft's spin axis, producing a sinusoidal fluctuation in the Doppler signal, which had to be modeled.<sup>39</sup>

A flat-plate model, with specular and diffuse reflectivity coefficients, was used to model solar radiation pressure effects on the spacecraft's motion. Trajectory-correction maneuvers could be performed most efficiently by rotating the spacecraft to the desired maneuver orientation, executing the maneuver, and then rotating the spacecraft back to its original orientation (turn-and-burn mode). They could also be performed less efficiently, but more simply, by firing some combination of thrusters without rotating the spacecraft (vector mode). To avoid potential thruster overheating, the thrusters were fired in pulses, rather than

continuously.<sup>40</sup> To prevent the clogging of retro-propulsion module passages, the thrusters were briefly fired every few weeks to flush old propellant from the system. This resulted in small velocity changes periodically, which were estimated as part of the orbit determination process. It was necessary to reorient the spacecraft from time to time (every two or three days to once per month, depending on heliocentric distance) to keep the spin axis pointed toward the sun. It was intended that these attitude changes would be performed by balanced thruster firings that would produce a torque, but no net translational force. However, there were net velocity changes resulting from these attitude updates, which needed to be modeled.<sup>39</sup>

Estimated parameters included spacecraft position and velocity at an epoch time, solar pressure reflectivity coefficients, and  $\Delta V$  events occurring during the data span.<sup>39</sup> Two TCMs were executed on the way to Venus, with a third planned TCM cancelled as unnecessary. The spacecraft passed about 9 km from the B-plane target point, even with no TCMs executed during the last month and a half before the Venus flyby.<sup>39,41</sup>

Between Venus and the Earth, six TCMs were executed.<sup>39</sup> The first two were deterministic maneuvers needed to reach the vicinity of the Earth. The first four were biased away from the desired Earth-relative aimpoint to maintain the probability of the spacecraft's entering the Earth's atmosphere below  $10^{-6}$ . The first usage of S-band  $\Delta DOR$  data in this mission took place during the three months before the Earth flyby, with a data accuracy of 1.67 ns (50 cm) of delay, equivalent to an angular accuracy of about 80 nrad using the DSN's intercontinental baselines. The most accurate orbit determination solutions made use of all three available radio metric data types. The spacecraft passed the Earth about 9 km from the B-plane aimpoint for the final TCM,<sup>39,41</sup> adding 5.2 km/s to the spacecraft's heliocentric speed in the process.

*Gaspra Flyby and Optical Navigation.* Between the first and second Earth flybys, the Galileo spacecraft flew close to the main-belt asteroid 951 Gaspra on 29 October 1991, at a distance of about 1,600 km, the first such visit by any spacecraft. Achieving a successful flyby of Gaspra required a campaign of ground-based astronomical observations to improve its ephemeris and the development of high-accuracy star catalogs for the region of space behind Gaspra. The ground-based observational campaign, which included many observations during Gaspra's oppositions of 1989-90 and 1991, improved the ephemeris errors by roughly a factor of three. High-accuracy star catalogs were constructed for those portions of the celestial sphere near the track of Gaspra as seen from Earth during these two oppositions.<sup>42,43</sup> The same was done for the track as seen from the spacecraft during the final two months before encounter, to allow accurate optical navigation<sup>1,44</sup> to be performed over at least a portion of this time.

The optical navigation software evolved through a number of languages over the years and was hosted on a variety of computers. Before operational use in the Galileo mission, the Optical Navigation Image Processing System (ONIPS), for picture display and image centerfinding, was rewritten in SFTRAN. The Optical Navigation Program set (ONP), for scene prediction, generation of residuals and partial derivatives, and filtering, was rewritten in FORTRAN 77. The software was hosted on Digital Equipment Corporation VAX minicomputers.<sup>44</sup>

The Galileo imaging system had a focal length of 1500 mm, as in the case of the Voyager narrow-angle camera. Unlike the Voyager camera, however, the Galileo imaging system had a charge-coupled device (CCD) light-sensing element, rather than a vidicon sensor. CCDs, with greater geometric fidelity than vidicons, produce much better astrometry. Well exposed images of stars or unresolved solar system objects (in Galileo and subsequent missions) often have rms residuals of less than 0.05 picture elements (pixels) with a CCD photodetector. The prior vidicon systems rarely performed better than 0.25 pixels.<sup>44</sup> Each picture element in the Galileo 800x800 CCD array subtended an angle of 10  $\mu$ rad.<sup>45</sup>

It was on the way to Gaspra that the spacecraft's high-gain antenna was to be deployed, allowing subsequent communication at X-band, with much higher data rates and improved radio metric accuracy. The failure of the high-gain antenna to deploy in April 1991 had many implications – among them a severe limitation on the number of optical navigation images that could be transmitted to Earth prior to the Gaspra flyby, because of the drastically reduced data rates that were available using the low-gain antennas.

Four imaging frames were returned, in single-frame mosaic mode, with the images of Gaspra and background stars appearing as irregular streaks, due to the combination of deliberate and unintended

motions of the scan platform while the shutter was held open for an extended time.<sup>42,44,45</sup> Each such imaging frame contained the information equivalent of multiple conventional optical navigation images.

On-board optical data were essential for determining and controlling two spacecraft position coordinates perpendicular to the line of sight to the asteroid, but could provide little information in the direction along the spacecraft's flight path relative to Gaspra. The combination of radio metric data (Doppler and, when available, range) and the improved ground-based ephemeris for Gaspra provided information in this third, "time-of-flight" direction. The quantities estimated in the orbit determination process include those mentioned previously, as well as the Gaspra ephemeris, camera pointing parameters, and stochastic accelerations included to model miscellaneous spacecraft attitude updating events.

Five TCMs were employed during the flight between the Earth and Gaspra – three to correct flight path errors resulting from the Earth flyby and create a flight path that would pass by both Gaspra and the Earth, while satisfying certain mission constraints, and two to home in on the desired Gaspra aimpoint.<sup>40</sup> It is estimated that the spacecraft passed 6 km from its B-plane aimpoint and arrived 1.5 s early.<sup>41</sup> A planned science pointing update was not needed because of the proximity of the final orbit solution to the Gaspra aimpoint.<sup>45</sup>

*Earth-2 Flyby.* The Galileo spacecraft flew past Earth for a second time on 8 December 1992, at 303-km altitude. Between the Gaspra flyby and the second Earth flyby, four trajectory correction maneuvers were executed. The first was deterministic, and the first two had biased aimpoints to minimize the likelihood of entering the Earth's atmosphere. As on the approach to the first Earth flyby, S-band Doppler, range, and  $\Delta$ DOR data were processed. The most accurate orbit determination solutions made use of all three available radio metric data types. During this mission phase and throughout the remainder of the mission, the various navigational advantages associated with X-band communication links could not be realized, because of the high-gain antenna deployment problem.

The spacecraft passed the Earth about 1.4 km from the B-plane aimpoint for the final TCM, adding 3.7 km/s to the spacecraft's heliocentric speed.<sup>41,46</sup> This accuracy was sufficient that a TCM scheduled for two weeks after the flyby, to correct any errors in the outbound trajectory, was cancelled. Both the first and second Earth flybys provided opportunities for the experimental processing of range data weighted near its actual precision,<sup>47</sup> as would be done later in an operational fashion in the Mars Pathfinder mission.

*Ida Flyby.* The spacecraft flew past the main-belt asteroid 243 Ida on 28 August 1993, at distance of 2,391 km. As on approach to Gaspra, numerous ground-based observations of Ida were acquired during the several years before the flyby. This observational campaign improved the Ida ephemeris by roughly a factor of four.<sup>48</sup>

Again, the single-frame mosaic technique was used to generate optical navigation data. Five optical navigation images were planned. Only two and a fraction were returned, however, due to spacecraft safing events and the preemption of the DSN's 70-m antenna subnet by the Mars Observer project when communication with the latter spacecraft was lost. Each image contained at least five usable data points.

The spacecraft was too far from the Earth for the last several months before encounter to allow S-band ranging with a low-gain antenna. Only a few  $\Delta$ DOR points were available, and they were somewhat inconsistent with other tracking data. Nevertheless, good orbit determination solutions were obtained; and the last of the three Ida targeting TCMs was cancelled as unnecessary. The flyby took place less than 6 km from the desired B-plane aimpoint. The time of closest approach was off by about 5s.<sup>41</sup>

*Atmospheric Probe Delivery.* A deterministic TCM was needed after the Ida flyby to return the spacecraft to a flight path toward Jupiter. A TCM performed on 12 April 1995 was used to place the combined orbiter/probe spacecraft on the proper trajectory for atmospheric probe entry. A subsequent scheduled TCM for further refinement of the flight path proved to be unnecessary. To prepare for release, the combined vehicle was oriented in the desired probe release attitude and spun up to 10.5 rpm. The probe was ejected using pre-loaded springs on 13 July 1995. The probe had no means of propulsion, so that its flight path was established by the pre-release and release events. The  $\Delta$ V uncertainties in the pre-release spinup and the release event, as well as Jupiter ephemeris uncertainties, were the dominant error sources affecting the probe's flight path.<sup>41</sup>

On 7 December the probe entered the top of Jupiter's atmosphere at a speed of 47.6 km/s. It was required that the probe's atmosphere-relative flight path angle at entry (defined to occur at 450 km above the 1-bar pressure level) be  $-8.6 \pm 1.4$  deg, with 99% probability.<sup>41,49</sup> The actual flight path angle error at entry was 0.15 deg. Reconstruction of the entry flight path angle was also important. The achieved accuracy of 0.10 deg compares with the requirement of 0.15 deg.<sup>41</sup> The probe slowed by aerodynamic braking for two minutes, then deployed its parachute and released its heat shield. The probe transmitted data for nearly an hour as it parachuted down through Jupiter's atmosphere. The probe could not communicate with (or be tracked in a conventional fashion from) the Earth directly. Data were transmitted to Earth using the Galileo orbiter spacecraft as a relay.

*Io Flyby and Jupiter Orbit Insertion.* The Galileo orbiter, deflected away from the probe's trajectory by a 61-m/s maneuver on 27 July, also arrived at Jupiter on 7 December 1995. On approach to Jupiter, the Galileo orbiter flew close to the satellite Io and used Io's gravitational field to reduce the propulsive  $\Delta V$  needed to insert into orbit about Jupiter. Two months before arrival at Jupiter, the tape recorder on board the spacecraft had begun to behave in an anomalous manner. This resulted in the cancellation of optical navigation images of Io and the removal of science requirements related to navigational accuracy at Io. A TCM had been performed on 29 August to clean up errors in the orbiter deflection maneuver. Two later TCMs were cancelled as a consequence of the tape recorder anomaly and the cancellation of Io remote sensing observations.<sup>50</sup> The achievement of the desired gravity assist from Io depended principally on the flyby altitude, and the B-plane errors in this direction were too small to warrant executing these TCMs. Later orbit determination solutions predicted a lower than planned flyby altitude at Io, but this allowed the orbital tour to begin with a Ganymede encounter one Ganymede revolution earlier than planned. Since the latter outcome was quite satisfactory, the final pre-Io TCM and an update to the Jupiter Orbit Insertion (JOI) maneuver were cancelled.<sup>50</sup> The spacecraft passed about 67 km from the expected Io B-plane location.<sup>41</sup>

The JOI burn was executed after Jupiter closest approach under accelerometer control, such that the burn was lengthened slightly to compensate for a slight underperformance in thrust. This 644-m/s burn was accurate enough that two planned subsequent orbit trim maneuvers (OTMs) were cancelled.<sup>50</sup>

*Galilean Satellite Tour.* A 378-m/s peri-raise maneuver was executed on 14 March 1996 to target for the first encounter of the Galilean satellite tour, with Ganymede on 27 June 1996. Subsequent encounters with Ganymede during the prime mission took place on 6 September 1996, 5 April 1997, and 7 May 1997. Prime mission Callisto encounters occurred on 4 November 1996, 25 June 1997, and 17 September 1997. Prime mission Europa encounters occurred on 19 December 1996, 20 February 1997, and 6 November 1997. Satellite gravity assists were used throughout the tour to enable subsequent satellite encounters.

There were typically three orbit-trim maneuvers (OTMs) scheduled per orbit about Jupiter: one three days before encounter to fine tune the satellite-relative aimpoint, one three days after encounter to correct for satellite flyby dispersions, and a third near apoapsis. The satellite tour was frequently reoptimized, taking into account any accumulated flight path dispersions arising from orbit determination uncertainties, uncertainties in the ephemerides and masses of the Galilean satellites, and maneuver execution errors, in order to select the aimpoint for the next encounter and minimize propellant expenditure. Five of the first eight pre-encounter OTMs were cancelled as unnecessary.<sup>51</sup>

During the satellite tour, one-way, rather than two-way, Doppler data were frequently collected to boost the power of the spacecraft's telemetry signal for science data return. One-way Doppler biases, drifts, and drift rates were estimated, based on two-way coverage before and after each one-way data span.<sup>51</sup> Loose pointing requirements for the low-gain antenna and the spin-stabilized design of the spacecraft led to relatively few nongravitational forces due to attitude control activities to be dealt with in orbit determination.

Optical navigation images were generated using a new flight software algorithm that detected and extracted from an imaging frame only the limb and terminator regions of the observed satellite and a small box of data around each star.<sup>44</sup> The amount of extracted data returned to Earth was about 1/300 of the data in a full imaging frame, a degree of data compression that was very helpful for a spacecraft with limited

data storage and transmission capabilities. More than 100 optical navigation images were returned through the sixth satellite encounter on 5 April 1997. Optical navigation images were not utilized thereafter because of the improvement in satellite ephemerides that had already been achieved, allowing for a simplification of the sequence design process.

The ephemeris accuracies of the Galilean satellites improved by an order of magnitude with the orbit determination data collected during the prime tour encounters. The satellite encounter delivery errors in the B-plane were typically 10-20 km for these encounters.<sup>41,52</sup>

Subsequent to the prime satellite tour, a two-year Galileo Europa Mission (GEM) was initiated in December 1997, including fourteen more orbits about Jupiter. The spacecraft investigations focused particularly on Europa and Io.<sup>53</sup>

Operational orbit determination solutions for the Galileo mission were generated using the Earth Mean Equator and Equinox of 1950 coordinate system, which had been used in many previous planetary missions. During the GEM, a reconstruction of the spacecraft trajectory throughout the prime tour was undertaken in the newer J2000 International Celestial Reference Frame,<sup>54</sup> together with the development of a consistent set of ephemerides for the Galilean satellites. By necessity, the work also included the improvement of the Jupiter ephemeris, the values of the Jovian system gravitational parameters, and the Jupiter pole orientation angles.<sup>55</sup>

### **Ulysses**

The European Space Agency/NASA Ulysses spacecraft, launched on 6 October 1990 by the Space Shuttle Discovery plus a Centaur upper stage, flew past Jupiter on 8 February 1992 and used Jupiter's gravity to place it in a high-inclination orbit about the sun, allowing observation of the sun's polar regions in 1994 and 1995 and several times thereafter.<sup>56</sup> The requirements on navigational accuracy were less demanding (1000-km circular error in the Jupiter B-plane and one hour in time of arrival) than in many missions. The spinning motion of the spacecraft (nominally 5 rpm), coupled with a circularly polarized antenna (nominally Earth-pointed), produced a bias in the Doppler data. Communications were normally at S-band for the uplink and X-band for the downlink, although an S-band downlink was available also.

Estimated parameters typically included spacecraft position and velocity at an epoch time, Doppler and range biases, velocity changes associated with attitude change maneuvers, and (stochastically modeled) solar radiation pressure parameters.<sup>57</sup> As the spacecraft came closer to Jupiter, ephemerides for Jupiter and the Earth and Jupiter's mass were added to the list of estimated parameters. Three TCMs were performed en route to Jupiter. The Ulysses flyby of Jupiter improved the accuracy of the Jupiter ephemeris, for subsequent use by the Galileo project.

Tracking data acquired during a solar-conjunction period in 1991 provided a means of testing a model of solar plasma-induced variations in ranging data needed to allow the weighting of these data near their inherent precision,<sup>28,58</sup> as would be done later in an operational fashion in the Mars Pathfinder mission.

### **Cassini**

The Cassini spacecraft was launched on 15 October 1997. Because it could not be launched directly to Saturn with the propulsion systems available at the time, it followed a path described as a VVEJGA (Venus-Venus-Earth-Jupiter Gravity Assist) trajectory. The primary telecommunication frequency was X-band, with the NASA Deep Space Transponder being put to its first operational use. Downlink capabilities were available at S-band and Ka-band also.

*Venus-1 Flyby.* The Cassini spacecraft flew past Venus on 26 April 1998 at an altitude of 284 km. Only the first two of three planned TCMs on approach to Venus were actually needed and executed.<sup>59</sup> Even though only two TCMs were executed on the Earth-Venus trajectory segment, a total of 22 velocity change events, most associated with some facet of attitude control, were modeled and estimated as part of the orbit determination process. In addition to these 22 discrete velocity change events, a number of continuously acting nongravitational acceleration effects were estimated. Solar radiation pressure, asymmetric radioisotope thermoelectric generator (RTG) radiation, and spacecraft outgassing effects were estimated over the entire data span. Several short-duration nongravitational acceleration effects were

estimated also. Since the heliocentric distance was nearly constant along portions of the trajectory, it was difficult to separate accelerations due to solar radiation pressure from those due to RTG thermal radiation.<sup>60</sup>

For some period of time, orbit determination solutions drifted toward the Venus impact radius as more data were collected and processed. The drifts in the orbit determination solutions were large compared to the error statistics associated with those solutions – an indication of mismodeling, coupled with relatively tight weighting of the X-band tracking data and a spacecraft declination close to zero.<sup>1</sup> It was deduced that uncoupled spacecraft thrusters were producing greater than expected velocity changes along the sun line while performing pitch and yaw attitude control activities, and the spacecraft torque model was adjusted to compensate. With evidence in hand of a high sensitivity of orbit solutions to mismodeling effects, the weighting of the Doppler and ranging data was diminished (from 0.2 to 1.0 mm/s and 5 to 50 m) to reduce the sensitivity. The Venus periapsis distance predicted from orbit determination solutions a month before arrival was accurate to within 1 km,<sup>60</sup> although the B-plane close approach distance was 67 km less than originally planned.<sup>59</sup>

*Venus-2 and Earth Flybys.* Five TCMs were scheduled for the time between the spacecraft's first and second Venus flybys. The first of these, scheduled for shortly after the first flyby to clean up orbit errors magnified by the flyby, was cancelled due to a post-flyby trajectory redesign.<sup>59</sup> A large deep space maneuver (DSM) was executed on 3 December 1998, to lower the periapsis distance and increase the flight path angle at the second Venus flyby. A subsequent TCM was executed to correct DSM execution errors, primarily in pointing.<sup>61</sup> A subsequent TCM removed a deliberate bias in the Venus aim plane. A final TCM on approach to Venus was cancelled as unnecessary. The spacecraft passed 11 km from the desired B-plane aimpoint on 24 June 1999, with a time of flight error of 2.5 s.<sup>62</sup>

Four TCMs were performed between the second Venus flyby and the Earth flyby (18 August 1999), to gradually remove aim point biases (incorporated to keep the probability of Earth impact less than  $10^{-6}$ ) and target for the Earth flyby. The spacecraft passed 9 km from the desired B-plane aimpoint at Earth, with a time of flight error of less than 1 s.<sup>62</sup>

For the Venus-1 to Earth flyby trajectory segment, the eight TCMs were modeled as finite burns and estimated as part of the orbit determination process. In addition, 47 other  $\Delta V$  events, associated with some sort of spacecraft activity, typically some facet of attitude control, were modeled as impulsive velocity changes and estimated. As during the launch to Venus-1 mission phase, solar radiation pressure and asymmetric RTG radiation effects were estimated as continuously acting nongravitational accelerations. Several short-duration nongravitational acceleration effects were estimated also. Ephemerides for the Earth and Venus were estimated. Tracking station locations and media calibration effects were considered.

The continuation of the Cassini mission to Jupiter and Saturn lies beyond the time period covered by this paper.

## EXPLORATION OF ASTEROIDS AND COMETS

### Near Earth Asteroid Rendezvous

The Near Earth Asteroid Rendezvous (NEAR) spacecraft was launched on 17 February 1996. The first encounter in this mission was with the main-belt asteroid 253 Mathilde. A series of five TCMs to correct upper-stage injection errors, calibrate the thrusters,<sup>63</sup> and target for Mathilde was used to fly past this asteroid on 27 June 1997.

*Mathilde Flyby.* Communications in the NEAR mission were at X-band. Radio metric orbit determination on the way to Mathilde was complicated by passages through solar conjunction (15 to 23 February 1997) and zero declination (22 April 1997) on approach. Radio metric data were unusable for about a month around conjunction, due to solar corona effects. When processing radio metric data outside of this time interval, it was found that the use of ranging data improved orbit determination accuracy significantly relative to the processing of Doppler data alone. It was also found that the tight weighting of ranging data, in combination with the estimation of additional parameters to allow this tight weighting, did not improve orbit determination accuracy substantially.<sup>64</sup>

The inclusion of ranging data helped to compensate for the long outage of Doppler data around solar conjunction, aided in the estimation of solar radiation pressure coefficients, and aided in the determination of spacecraft declination when it was near zero. Range data were not available at all times because the spacecraft's high-gain antenna was needed for ranging; and the pointing of the high-gain antenna toward Earth for extended periods of time would cause unacceptable angular momentum accumulations, due to the solar radiation pressure torque.<sup>64</sup>

Numerous ground-based observations of Mathilde were made at observatories around the world during the asteroid's 1995 and 1996 oppositions, in order to refine its ephemeris. Some observations made in May and June 1997 were reduced relative to the Hipparcos and Tycho star catalogs and were particularly accurate.<sup>64,65</sup>

Optical data derived from the on-board Multi-Spectral Imager (MSI) were available only during the last two days before arrival at Mathilde. The solar phase angle of 140 deg on approach made the asteroid difficult to detect very far in advance. Optical navigation images were acquired in batches of 16 at six separate times. Pointing the MSI at Mathilde required pointing the solar arrays 50 deg away from the sun line, which was undesirable and could be permitted only infrequently. Pointing the MSI only 40 deg from the sun introduced stray light into the optical navigation images, making dim background stars difficult to detect. In addition, the high phase angle displaced the center of brightness of Mathilde from its center of mass. Mathilde became visible during the second optical navigation opportunity, but only when several imaging frames were added together. It became visible in individual frames only in the fifth opportunity.<sup>64</sup>

The accurate modeling of any periods of off-nominal spacecraft attitude was found to be important in obtaining accurate orbit solutions. Estimated constant parameters in the orbit determination solutions on approach to Mathilde included the epoch state of the spacecraft, a scale factor applied to the solar radiation pressure model, the magnitude and direction of the fifth TCM, the coordinates of the DSN tracking stations, and the ephemeris and mass of Mathilde. Estimated stochastic parameters were associated with the Earth's orientation, tropospheric and ionospheric signal delays, range biases, and camera pointing biases. Doppler and range data were weighted at 0.1 mm/s and 1 m, respectively.<sup>64</sup>

A TCM planned for 13 hours before the flyby was cancelled as unnecessary, based on the latest solution for the spacecraft's orbit (based primarily on radio metric data, along with a limited amount of optical navigation data) and the asteroid's ephemeris determined largely from the ground-based observations. A late update to the spacecraft pointing at closest approach to Mathilde was made based on similar information, with additional optical navigation data included. The spacecraft flew past Mathilde at a distance of 1212 km and a speed of 9.9 km/s. The relative position uncertainties in the B-plane were less than 10 km at the time of the final encounter sequence update, with the time of flight uncertainty less than 5 s. The mass of Mathilde was determined from pre-and-post encounter tracking to an accuracy of several percent.<sup>64,65</sup>

*Earth Flyby.* On 3 and 23 July 1997, the spacecraft executed a 274-m/s deep space maneuver in two parts (98%/2%), lowering the perihelion distance from 0.99 AU to 0.95 AU and setting up an Earth gravity assist swingby on 23 January 1998. Two TCMs after the double maneuver were used to correct maneuver execution errors and target the Earth flyby, which occurred at a closest approach distance of 539 km, altering the orbital inclination from 0.5 to 10.2 deg and the aphelion distance from 2.17 to 1.77 AU, nearly matching the corresponding values (as well as the line of apsides) for the near Earth asteroid 433 Eros.<sup>66</sup>

*Eros Flyby.* Two TCMs preceded an effort to initiate a rendezvous with Eros on 20 December 1998, which was aborted shortly after starting due to an on-board software problem. Instead, the NEAR spacecraft flew by Eros on 23 December 1998 at a speed of 0.965 km/s and a distance of 3828 km.<sup>66</sup>

A 932-m/s anomaly-recovery maneuver was performed on 3 January 1999, to roughly match the spacecraft's orbital speed to that of Eros. Two corrective TCMs were performed over a number of months.<sup>66</sup> A successful rendezvous with Eros was achieved beyond the time period covered by this paper.

## **Deep Space 1**

*Technology Demonstrations.* The Deep Space 1 (DS1) spacecraft was launched on 24 October 1998 to demonstrate a variety of new technologies. Propulsion was provided by a xenon ion engine. Xenon ions

were accelerated and ejected from the spacecraft as an ion beam, producing 0.09 N of thrust at maximum power (2300 W) and 0.02 N at the minimum operational power of 500 W. Whereas deep-space trajectories in missions with chemical propulsion are characterized by long periods of coasting, with only brief periods of thrusting, electric propulsion missions involve thrusting a substantial fraction of the time, with the engines shut off when thrusting is least advantageous.

One of the technologies demonstrated by the DS1 mission was AutoNav, the first autonomous navigation system to be used in deep space. AutoNav consisted of an executive, which interacted with the rest of the flight software and was responsible for scheduling and executing events; a real-time ephemeris server, which provided spacecraft and natural body ephemeris information to other flight software elements; and the computational elements that performed the fundamental navigational calculations related to orbit determination, maneuver planning, and encounter target tracking.<sup>67, 68</sup>

Another experiment on board the DS1 spacecraft was the Miniature Integrated Camera and Spectrometer (MICAS). The two visible light channels were potentially usable for optical navigation – one with a CCD chip and one with an active pixel sensor (APS) array.

*Cruise and Approach Autonomous Navigation.* Orbit determination during cruise was done autonomously by imaging “beacon” asteroids, one at a time, relative to star backgrounds. Each such measurement gave a two-dimensional position fix for the spacecraft, to some level of accuracy. Subsequent measurements in different directions provided different two-dimensional position fixes. Several clusters of such sightings were incorporated into a least-squares filter to obtain an orbit determination solution. The accuracy of such solutions depended on the brightnesses of, distances to, and angular separations among the asteroids; the accuracies of the asteroid ephemerides; the resolution of the camera; the ability to pinpoint asteroid locations in camera frames; and the accuracy of the camera pointing information. A list of beacon asteroids to observe during a given mission segment was stored on board the spacecraft, along with ephemerides for all of the beacons. At predetermined times, the spacecraft trajectory was compared to the ephemeris of a particular asteroid to generate a camera pointing direction.<sup>68</sup>

Coarse registration of the asteroid and background stars in each imaging frame was done by computing a center of brightness in a 50-100 pixel region surrounding each star, based on the knowledge of the inertial pointing direction provided by the attitude control subsystem star tracker. The predicted locations for the stars were shifted by the measured center-of-brightness/predicted location offsets, one star at a time, until the light intensity integrated over the specified regions surrounding the stars was maximized. Fine registration of the asteroid and background stars was done using a variation of the single-frame mosaic technique employed during the Galileo mission on approach to Gaspra and Ida. The spacecraft attitude wandered somewhat during the exposure time, creating complex, unpredictable image tracks. The image of each object in a frame could be used as a template to cross-correlate with other image tracks. This information was combined in a least-squares sense to find an ensemble of image shifts where the signals had peak responses with respect to each other. The accuracy with which the pointing estimates provided by the attitude control subsystem could be updated was dependent on the number of stars present in the imaging frame, with single-star frames providing little information regarding rotations about the boresight direction.<sup>68</sup>

The on-board dynamical model of the spacecraft included gravitational attractions of the major bodies of the solar system, solar radiation pressure, thrust forces due to the ion propulsion system or the hydrazine attitude control thrusters, and a three-dimensional bias acceleration to account for small unmodeled forces. The propagation of the spacecraft motion was done in a heliocentric, Earth Mean Equator of 2000 coordinate system using a 7-8<sup>th</sup> order Runge-Kutta numerical integrator. Orbit determination solutions were carried out by linearizing about a nominal trajectory and estimating corrections to parameters that minimized the data residuals in a weighted least-squares sense. A U-D covariance factorization was used.<sup>68</sup>

Observations of beacon asteroids were typically carried out during 4-6 hr time blocks once per week. After all of the imaging frames had been recorded and processed, a new orbit determination solution was generated, based on data going back about 30 days in the past, with the solution mapped from the epoch time to the current time and propagated into the future.<sup>68</sup>

The testing of the autonomous navigation system was delayed for several months due to the need to compensate for some unanticipated attributes of the experimental MICAS camera. There was significant stray light apparent in the images, which could be compensated for by the differencing of sets of frames. There were geometric distortions in the images, which required modeling. The ability to detect dim objects was less than expected, with the result that the dimmer beacon asteroids and background stars were not usable. Independent of this, the estimates of attitude control thruster firings were off by a factor of two, which corrupted the orbit determination solutions, until corrected; and an erroneous parameter describing the orientation of the camera boresight was sent to the spacecraft.

The on-board orbit determination accuracy varied over a broad range, eventually settling down to 700-1000 km, after the various problems noted above had been addressed. The velocity errors similarly varied over a broad range, settling down to 0.2-0.5 m/s. After the upload of a corrected parameter file in mid-June 1999, orbit determination on board the spacecraft was operating fairly independently of the ground. The accuracy being obtained was not sufficient to fully support autonomous maneuver planning, however, even though the algorithms worked well when supplied with sufficiently accurate information.<sup>68</sup>

Maneuver planning was accomplished by means of a linear control process. The initial design of the nominal trajectory using ion propulsion was done on the ground. Partial derivatives of the target conditions with respect to the available thrust-related control parameters were calculated on board. Modifications to the control parameters were determined by inverting the partial derivative matrix and multiplying by the desired change in target conditions.

The first natural body to be encountered was the near-Earth asteroid 9969 Braille, to be flown past on 29 July 1999, at 15.5 km/s relative velocity. Braille was first detected by the ground processing of multiple images three days before encounter, about 450 km from its expected position. A TCM, based on ground computations, was executed taking this information into account 1.5 days before encounter. After the on-board software had locked onto Braille, a latent defect in the software caused a spacecraft safing event, destroying the latest trajectory result. The processing of more imaging frames on the ground refined the orbit solution relative to Braille and precipitated a TCM six hours before encounter to avoid a possible impact. Later images processed on board confirmed the post-TCM trajectory estimate.<sup>68</sup>

*Encounter Autonomous Navigation.* As the spacecraft drew close to the asteroid encounter, a different approach to generating orbit determination solutions was adopted. At this point, no more thrusting maneuvers were to be performed; and the objective was to maintain visual lock on the asteroid. The rapid processing of images for this purpose could not be accomplished by the orbit determination process that has been described so far. Instead, a reduced-state estimator was used, modeling the trajectory as a target-centered straight line through encounter, with initial conditions 27 minutes prior to encounter provided by the full orbit determination process. In addition, the APS array was used rather than the CCD; the image processing consisted solely of finding a brightness centroid; and, since no stars were visible, pointing information was provided solely by the star tracker.<sup>68</sup>

The asteroid was not tracked successfully during the encounter. It was determined after the fact that a spurious signal had appeared above the detection threshold level before Braille itself exceeded that level, causing the on-board orbit solution to shift by 30 km. The asteroid turned out to be considerably dimmer than expected, and the sensitivity of the APS was less than expected. The asteroid was captured in CCD images recorded 15 minutes after closest approach, after the calculation of pointing commands had reverted to the pre-encounter process.<sup>68</sup>

The autonomous navigation system worked well in cruise after the Braille encounter, until the star tracker failed in November 1999. It also worked well in an encounter with comet 19P/Borrelly, beyond the time frame covered by this paper.

## CONCLUSION

Deep space navigation capabilities, which had evolved enormously from the early Mariner missions of the 1960s and early 1970s to the Voyager mission of the late 1970s and 1980s, continued to evolve through the 1990s, benefiting the dozen planetary missions that have been described. The availability of X-band communications on both the uplink and downlink, coupled with the improved calibrations of various error

sources, improved the accuracy of Doppler and ranging data significantly. Optical navigation capabilities improved with the use of charge-coupled device photodetectors and the development of more sophisticated image processing techniques.

Rapid increases in computing power allowed more accurate orbit determination by permitting the estimation of many additional parameters associated with dynamical and measurement modeling. The combination of increasing computational capabilities and declining costs allowed more comprehensive navigational analyses to be carried out. Orbit determination and maneuver planning functions were performed on board a planetary spacecraft for the first time.

The use of aerobraking to reduce a long-period elliptical orbit about an atmosphere-bearing planet to a low-altitude, near-circular orbit was demonstrated, allowing a large saving in propellant usage. The use of a continuous, low level of thrust, provided by an ion propulsion system, was demonstrated as an efficient means of travel through the inner solar system. Atmospheric entry, descent, and landing techniques were improved in accuracy.

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