ORBIT DETERMINATION SUPPORT FOR THE MICROWAVE ANISOTROPY PROBE (MAP)

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NASA's Microwave Anisotropy Probe (MAP) was launched from the Cape Canaveral Air Force Station Complex 17 aboard a Delta II 7425-10 expendable launch vehicle on June 30, 2001. The spacecraft received a nominal direct insertion by the Delta expendable launch vehicle into a 185-km circular orbit with a 28.7° inclination. MAP was then maneuvered into a sequence of phasing loops designed to set up a lunar swingby (gravity-assisted acceleration) of the spacecraft onto a transfer trajectory to a lissajous orbit about the Earth-Sun L2 Lagrange point, about 1.5 million km from Earth. Because of its complex orbital characteristics, the mission provided a unique challenge for orbit determination (OD) support in many orbital regimes. This paper summarizes the premission trajectory covariance error analysis, as well as actual OD results. The use and impact of the various tracking stations, systems, and measurements will be also discussed. Important lessons learned from the MAP OD support team will be presented. There will be a discussion of the challenges presented to OD support including the effects of delta-Vs at apogee as well as perigee, and the impact of the spacecraft attitude mode on the OD accuracy and covariance analysis.

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INTRODUCTION

The Microwave Anisotropy Probe (MAP) is the second Medium Class Explorer (MIDEX) mission of the National Aeronautics and Space Administration (NASA). The main goal of the MAP observatory is to measure the temperature fluctuations, known as anisotropy, of the cosmic microwave background (CMB) radiation over the entire sky and to produce a map of the CMB anisotropies with an angular resolution of approximately 3 degrees. This map of the anisotropy distribution will help determine how structures formed in the early universe, will determine the ionization history of the universe, and will refine estimates of key cosmological parameters. In particular, these data will be used to shed light on several key questions associated with the Big Bang theory and to expand on the information gathered from the Cosmic Background Explorer (COBE) mission, flown in the early 1990s. The L2 lissajous orbit was selected by the MAP program to minimize environmental disturbances, maximize observing efficiency, and to provide instrument thermal stability. A lissajous trajectory is considered as a three-dimensional quasi-periodic orbit. [2] The science mission minimum lifetime is two years of observations at L2 with a desired lifetime of 5 years.[1]

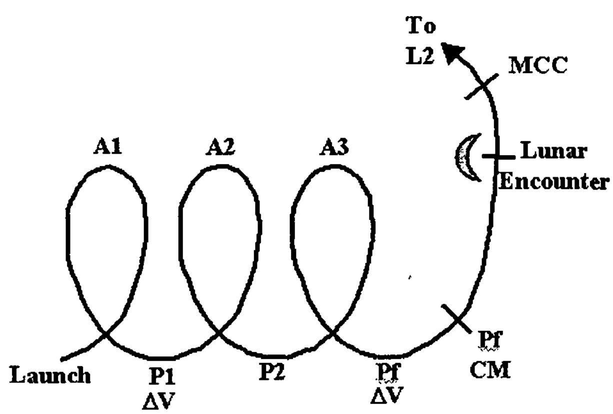

Figure 1 MAP (3.5-loop) Trajectory Schematic

MAP used a lunar gravity assist strategy since it reduced the fuel required to achieve the desired lissajous orbit. The MAP transfer orbit consisted of 3.5 phasing loops. Figure 1 shows the MAP trajectory schematic all the way through L2.[3] The first loop had a period of 7 days, the second and third loops were 10 days long, and the last half loop was 5 days long. A correction maneuver at the third perigee (P_f or Pfinal) was planned for approximately 18 hours after the last perigee maneuver to accurately achieve the targeted lissajous orbit. The lunar encounter took place approximately 30 days after launch. After the lunar encounter, the spacecraft cruised for 60 days before it arrived in the vicinity of the L2 libration point. Two mid-course correction (MCC) maneuvers were performed a week after the periselene (i.e., lunar encounter or swingby) to refine MAP's post-launch trajectory. Now that MAP is at its operational L2 lissajous orbit, the MAP satellite is commanded to perform occasional station-keeping (SK) maneuvers in order to maintain its orbit around L2. At L2, MAP will maintain a lissajous orbit with a MAP-Earth vector between 0.5° and 10.5° off the Sun-Earth vector in order to satisfy its communications requirements while avoiding eclipses.[6,7]

Telemetry, tracking and command is provided by the NASA Deep Space Network (DSN). The Tracking and Data Relay Satellite System (TDRSS) was also used during launch and early orbit operations.

The NASA GSFC Flight Dynamics Analysis Branch (FDAB) performed premission covariance analysis in order to determine MAP orbit determination requirements for maneuver planning and calibration for each phase of the mission. Once the satellite was launched, the Navigation Team from Honeywell and the Computer Sciences Corporation (CSC) performed definitive orbit determination in support of the mission. This paper presents the results of the premission orbit analysis, the technique and results of the post-launch OD process, and evaluates the OD accuracy. Important lessons learned from the MAP Navigation team support are also presented.

PREMISSION ORBIT ERROR ANALYSIS

The purpose of the orbit error analysis was to provide definitive and predicted ephemeris accuracy estimates to help the MAP Project plan for orbit control. Another purpose of this analysis was to determine if the orbit accuracy and tracking requirements as specified in the MAP Detailed Mission Requirements (DMR)[9] document could be met, and to propose other tracking scenarios if necessary. The FDAB's Orbit Determination Error Analysis System (ODEAS) was used to perform the covariance analysis and it was based on a nominal trajectory provided by the trajectory design group. The nominal trajectory did not include the maneuver at perigee 2 (P2). Several tracking scenarios were investigated for each phase of the mission.[5] Estimated ephemeris accuracies were derived for different post-maneuver and pre-maneuver tracking scenarios. Results from the ODEAS runs show that required orbit accuracy can be satisfied if tracking support includes both range and range rate measurements from at least two of three DSN 26-meter stations (Goldstone, California; Canberra, Australia; and Madrid, Spain) under a proposed tracking schedule as follows:

- For maneuver recovery: 6 to 18 hours of continuous tracking support after each planned maneuver (M).
- For maneuver planning:
  From the transfer trajectory injection (TTI) to the first perigee (P1): three 1-hour passes/day (alternating northern (N) and southern (S) hemisphere DSN stations) then continuous tracking from M-16hrs to M-12hr.
  From P1 to Pfinal: three 1-hour passes/day (alternating N&S hemisphere DSN stations) then continuous tracking from M-16hr to M-12hr.
  From Pfinal to periselene (Ps): three 1-hour passes/day (alternating N&S hemisphere DSN stations) then continuous tracking from M-16hr to M-12hr.
  From Ps to L2 insertion: one 37-min pass/day (alternating N&S hemisphere DSN stations)
  L2 Nominal: one 37-min pass/day (alternating N&S hemisphere DSN stations)

Table 1 shows post-maneuver definitive and predicted ephemeris accuracies under different tracking scenarios. For each scenario in Table 1, the maneuver is supported by continuous tracking followed by no tracking support until the next planned maneuver (e.g., from TTI to P1). Three post-maneuver tracking data arcs were evaluated: 6 hours, 12 hours and 18 hours (for Pfinal only). After 12 hours of continuous tracking, RSS position error (3σ) was on the order of 500 m and RSS velocity error (3σ) was on the order of 2 cm/s. With Delta-V magnitudes of 3 km/s at TTI, 22 m/s at P1, and 7 m/s at Pfinal, the estimated velocity error is not a significant fraction of the burn magnitude. For this paper, the phrase "definitive ephemeris" is used for a post-processed trajectory generated by an orbit determination process (i.e., OD with tracking data) and the word "predicted ephemeris" is used for trajectories generated by orbit propagation (i.e., without tracking data). It should be noted that errors on the predicted ephemerides (i.e., no tracking support after 6 or 12 hours) are quite high, which is typical.

Table 1

POST-MANEUVER EPHEMERIS ACCURACY ESTIMATES

Epoch	Tracking Support	Definitive Ephemeris Accuracy (3σ)	Predicted Ephemeris Accuracy (3σ)
TTI	Maneuver+ 6 hours	Pos: 1.039km Vel: 2.72 cm/s	Pos: 63.419 km (at P1) Vel: 9.06 m/s
	Maneuver+ 12 hours	Pos: 559 m Vel: 1.14 cm/s	Pos: 35.062 km (at P1) Vel: 5.01 m/s
P1	Maneuver+ 6 hours	Pos: 300 m Vel: 1.05 cm/s	Pos: 167.19 km (at Pf) Vel: 19.47 m/s
	Maneuver+ 12 hours	Pos: 241 m Vel: 0.35 cm/s	Pos: 146 km (at Pf) Vel: 17 m/s
Pfinal	Maneuver + 6 hours	Pos: 416 m Vel: 3.5 cm/s	Pos: 4.85 km (at Ps) Vel: 33.81 cm/s
	Maneuver + 12 hours	Pos: 886 m Vel: 2.57 cm/s	Pos: 4.340 km (at Ps) Vel: 29.44 cm/s
	Maneuver + 18 hours	Pos: 564 m Vel: 1.14 cm/s	Pos: 2.729 km (at Ps) Vel: 21.88 cm/s

The orbit determination process used for this error analysis is a batch least-squares method and the orbit error depends on many factors including spacecraft position on the orbit. It is expected that more tracking coverage would improve the overall post-processed and predicted orbit accuracy but not necessarily at any specific time (e.g., the estimated definitive orbit error at the end of the 12-hr arc from Pf is 886 m while that of the 6-hr arc is only 416 m).

Table 2

DEFINITIVE EPHEMERIS ACCURACY ESTIMATES FOR MANEUVER PLANNING

Epoch	Definitive Ephemeris Accuracy (3σ) at Maneuver – 24 hours	Definitive Ephemeris Accuracy (3σ) at Maneuver – 12 hours	Definitive Ephemeris Accuracy (3σ) at Maneuver
TTI (To plan P1 maneuver)	Pos: 233 m Vel: 0.18 cm/s	Pos: 149 m Vel: 0.27 cm/s	Pos: 33 m Vel: 0.89 cm/s
P1 (to plan Pfinal maneuver)	Pos: 554 m Vel: 0.24 cm/s	Pos: 243 m Vel: 0.38 cm/s	Pos: 53 m Vel: 0.93 cm/s
Pfinal (to plan periselene maneuver)	Pos: 194 m Vel: 0.15 cm/s	Pos: 161 m Vel: 0.14 cm/s	Pos: 90 m Vel: 1.06 cm/s
L2 – 3 weeks (to plan SK)	Pos: 2.376 km Vel: 0.16 cm/s	Pos: 2.364 km Vel: 0.16 cm/s	Pos: 2.354 km Vel: 0.16 cm/s
Epoch	**Definitive Ephemeris Accuracy (3σ) at Epoch + 1 week**	**Definitive Ephemeris Accuracy (3σ) at Epoch + 2 weeks**	**Definitive Ephemeris Accuracy (3σ) at Epoch + 3 weeks**
Periselene (to plan MCC)	Pos: 1.514 km Vel: 0.11 cm/s	Pos: 1.872 km Vel: 0.07 cm/s	Pos: 1.990 km Vel: 0.06 cm/s
L2 Insertion (to plan MCC)	Pos: 2.634 km Vel: 0.08 cm/s	Pos: 2.644 km Vel: 0.08 cm/s	Pos: 2.593 km Vel: 0.1 cm/s

Table 2 shows the results of the error analysis for maneuver planning for a range of pre-maneuver tracking data cut-off times. For all phases of the mission, the estimated RSS definitive position error is ≤ 2.644 km and the estimated RSS definitive velocity error is ≤ 1.06 cm/s at the start time of the planned maneuver. It should be noted that where there is uncertainty about the exact time of the planned maneuver at the time of the analysis, only estimated accuracy at the end of the tracking arc (e.g., 1 week) was given.

Table 3 describes the proposed orbit accuracy and tracking requirements for the MAP mission. It should be noted that the estimated accuracies are three-sigma (3σ) values. The error covariance analysis showed that these orbit requirements could be met if the existing GSFC operational orbit related systems (e.g., the Goddard Trajectory Determination System (GTDS)) are used to support the mission under the specified tracking support from DSN. Based on the covariance analysis, this tracking support was different from the original DMR document.

Table 3

ORBIT ACCURACY AND TRACKING REQUIREMENTS

Mission Phase	Service	Data Type	Pass Frequency	Definitive Ephem Requirements (3 σ)	Data Sample Rate	Predicted Ephem Requirements (3 σ)
LEO (L+0 to L + 1 day)	26-m or 34-m	Range and Doppler	Continuous 2 stations for first 12 hours, one station second 12 hours	Best Obtainable	Doppler:1/10s Range:1/10s	Position: 25 km
Transfer Trajectory Phase-nominal (2-F+3 days) Nominal Support	26-m or 34-m	Doppler, range, and angles from 26-m	2 – 4 one hour passes/day	Position: 5 km Velocity: 5 cm/s	Doppler:1/10s Range: 1/10s	Position: 50 km Velocity: 10 cm/s (1-week arc)
Transfer Trajectory Phase-maneuvers & lunar gravity Assist (phasing loops)	26-m or 34-m	Doppler, range	Near Continuous M – 16h to M– 12h (4 hour span) and M start to M+8 hours	Position: 5 km Velocity: 5 cm/s	Doppler:1/10s Range: 1/60s	Position: 50 km Velocity: 10 cm/s (1-week arc)
Cruise (Gravity Assist to L2 Insertion) (~70 days)	70-m or 34-m	Doppler, range	Nominal: One 37-min pass/day*	Position: 5 km Velocity: 5 cm/s	Doppler:1/10s Range: 1/60s	Position: 50 km Velocity: 10 cm/s (1-week arc)
Cruise-maneuvers	70-m or 34-m	Doppler, range	Near Continuous M-16h to M-12h (4 h span) and M start through M+8 hr	Position: 5 km Velocity: 5 cm/s	Doppler:1/10s Range: 1/60s	Position: 50 km Velocity: 10 cm/s (1-week arc)
L2-nominal (2 years)	70-m or 34-m	Doppler, range	One 37 min pass/day*	Position: 5 km Velocity: 5 cm/s	Doppler:1/10s Range: 1/60s	Position: 100 km Velocity: 40 cm/s (1-week arc)
L2-maneuvers Delta V/Delta H	70-m or 34-m	Doppler, range	Near Continuous from M-4h to M+8h	Position: 5 km Velocity: 5 cm/s	Doppler:1/10s Range: 1/60s	Position: 100 km Velocity: 40 cm/s (1-week arc)

*alternating N & S hemisphere DSN stations

POSTLAUNCH ORBIT DETERMINATION SUPPORT

MAP was launched on June 30, 2001 at 19:46 UTC from Kennedy Space Center aboard a Delta II rocket. Over the course of the following four weeks, MAP executed 3.5 phasing loops about the Earth prior to periselene on July 30, 2001.

The maneuver team conducted thruster firings at every apogee and perigee of the phasing loop period. A thruster calibration was performed at apogee 1 (A1), and engineering burns were executed at apogee 2 (A2) and apogee 3 (A3). Perigee maneuvers (P1, P2, P3) and a final correction maneuver (P_{fc}) at 18 hours after P3 were executed to give MAP necessary energy to reach periselene, correct errors, and fine tune the trajectory.

As a result of these thruster events conducted during the phasing loops, the maximum tracking data arc available for orbit determination (OD) was one half of an orbit. Other perturbations affecting OD were also present during the phasing loops. Additional thruster testing was performed between spacecraft separation and A1. Furthermore, the solar radiation force on MAP is strongly dependent on the spacecraft attitude mode, which changed frequently during the phasing loop period. A timeline of MAP orbit events is shown in Table 4.

Table 4

MAP ORBIT EVENTS

Event	Start Time	Duration (s)	Delta–V (m/s)
S/C Sep	2001/06/30 21:12		
Apogee 1 Delta-V	2001/07/04 13:22	106.0	1.92
Perigee 1 Delta-V	2001/07/08 04:43	1274.4	20.19
Apogee 2 Delta-V	2001/07/12 16:11	40.6	0.25
Perigee 2 Delta-V	2001/07/17 03:36	1777.2	2.51
Apogee 3 Delta-V	2001/07/21 18:54	43.4	0.29
Perigee 3 Delta-V	2001/07/26 10:29	546.2	7.41
P3 Correction Maneuver	2001/07/27 04:30	23.9	0.31
Lunar Gravity Assist	2001/07/30 16:37		
Mid-course Correction 1	2001/08/06 16:37	18.0	0.10
Mid-course Correction 2	2001/09/14 16:37	6.6	0.04
StationKeeping # 1	2002/01/16 16:51	72.9	0.43
StationKeeping # 2	2002/05/08 16:03	53.8	0.35
StationKeeping # 3	2002/07/30 16:39	71.8	0.47
StationKeeping # 4	2002/11/05 19:21	94.3	0.56

MAP orbit determination was performed using GTDS, running on a Windows-NT platform. GTDS employs a batch least-squares estimator with the option to solve for additional force parameters such as the solar radiation pressure coefficient (C_r).

During the phasing loop period, MAP tracking predominantly consisted of range and Doppler measurements from DSN 34-meter antennas, with DSN 26-meter antennas providing some additional support. Near perigee, TDRS 2-way Doppler tracking was received. For all the solutions presented in this report, Doppler observations were sampled down to roughly equal the number of range observations to balance the effects of both data types in the solutions. Angle data, when available, was only used in short-arc solutions, due to the noise of angle observations and their increased uncertainty at larger radial distances.

Intensive orbit determination analysis was performed to evaluate ephemeris accuracy, orbit perturbations (e.g., due to changes in spacecraft attitude), major systematic errors, and performance of short post-maneuver tracking data arcs and of L2 operations.

For most of the OD solutions presented in this paper, the 1-sigma standard deviation of post-fit range residuals varied between 2 m and 10 m. For those solutions over tracking data arcs without attitude or other disturbances, the 1-sigma standard deviation was usually below 5 m. Large residual divergences, like that illustrated in Figure 2, do not significantly drive up the residual standard deviation because most of the observations in the "residual tail" are edited out of the solution. The 1-sigma standard deviation of range-rate residuals was typically between 0.1 cm/sec and 2 cm/sec, depending on the attitude mode of the spacecraft.

Ephemeris Consistency

The ideal method of determining the accuracy of OD solutions is by comparison to precision definitive ephemeris; however, these methods are not available for most missions. Thus it is common to assess OD accuracy through the examination of overlapping definitive ephemeris spans. When using this method, one takes the ephemeris consistency to be a measure of definitive ephemeris accuracy, under the assumption that no systematic errors bias both solutions.

For an operational mission, it is not difficult over time to collect a statistical set of overlapping well-determined solutions. In the case of MAP, the presence of delta-Vs executed at all apogees and perigees made the task of generating overlapping definitive ephemeris difficult during the phasing-loop period since it reduced the maximum tracking data arc to a half of an orbit. The approach used here to generate a set of overlapping half-orbit definitive ephemerides was to modify the tracking data spans and other input parameters in the orbit determination process. This procedure is additionally complicated by the perturbations induced by attitude dependent changes in the solar radiation force.

A set of orbit solutions was generated using all available data between each apogee and perigee. A second set was also generated by deleting the 24 hours of arc from each half-orbit span. In order to mitigate the potential for solution degradation caused by the loss of advantageous geometry, the data was deleted from the middle of the half-orbit spans. These ephemerides generated from these solutions were then compared over their common definitive span or overlapping timespan. The results are shown in Table 5. In addition to position and velocity, C_r was determined for all of these solutions.

Table 5

DEFINITIVE OVERLAP COMPARES

Definitive Overlap Comparison Span	Maximum Definitive Position Difference (m)	Maximum Definitive Velocity Difference (cm/s)
S/C Sep to A1	530	34.8
A1 to P1	32	0.3
P1 to A2	64	0.4
A2 to P2	389	1.5
P2 to A3	871	11.2
A3 to P3	46	0.1

While Table 5 shows a significant dispersion in ephemeris position and velocity differences, the largest differences are associated with spans where thruster firings, deployments, or attitude changes are present in the arc. In the case of the separation-to-A1 arc, initial deployments and thruster testing which occurred during the first 3 days of the mission make solutions very sensitive to the tracking data arc. In addition, many attitude changes took place during the flight to A1 that impacted the force of solar radiation experienced by the spacecraft. Solutions over the separation-to-A1 arc generally exhibit higher range residual standard deviations than over other, less perturbed arcs. Cleaner solutions were obtained by cutting the tracking data arc right after the last thruster testing activity performed, but this resulted in the loss of useful perigee tracking.

Orbit Perturbations Due to Changes in Spacecraft Attitude Mode

Attitude mode changes are likely responsible for larger differences in the A2 to P2 and P2 to A3 tracking arcs. Those with the lowest differences, A1 to P1, P1 to A2, and A3 to P3, are largely unperturbed by attitude mode changes.

Throughout the phasing loops, for the purposes of OD considerations, MAP's attitude could be characterized generically in two modes. The first is a group of states, identified here as "flat spin" modes, in which the spacecraft rotates about its Z-axis while the Z-axis is on or close to (generally within 2 degrees of) the spacecraft to Sun line. The other was the operational science "observing mode" configuration in which the spacecraft rotates about its Z-axis, while the Z-axis precesses along a 22.5-degree cone about the Sun line. In each of these two modes the spacecraft presents a different area to the Sun. As a result, MAP experienced a different force of solar radiation during each mode. When solving for C_r as part of the OD process, a value of 1.45 was typically found for observing mode arcs, while a value of 1.78 was characteristic of flat spin mode. When tracking data arcs contained data from both modes, the OD solutions determined a value somewhere between these two.

The effect of the difference in solar radiation force in each mode is clearly visible in Figure 2. This plot displays final vector residuals computed in an A2 to P2 solution using all available data. This solution features a prominent divergence when the spacecraft transitioned from a flat spin mode to observing mode. From the range residuals, it can be seen that the solution solved to the data taken when the spacecraft was in observing mode, due to the greater proportion of data in this mode. This effect is characteristic of the least-squares estimation process. In fact, most of the range data prior to Day 194 as seen in Figure 2 was excluded from the solution by the differential corrector sigma edit criteria. The 80-meter residuals at the beginning of the tail are at the 30-sigma level. This resulted in an effective reduction in tracking data due to the loss of nearly all usable data in flat spin mode. In both the A2 to P2 and P2 to A3 arcs, the difference in solar radiation force in both modes is too big to allow the least-squares estimation to average well over the attitude change, resulting in states which solve to one or the other set of data.

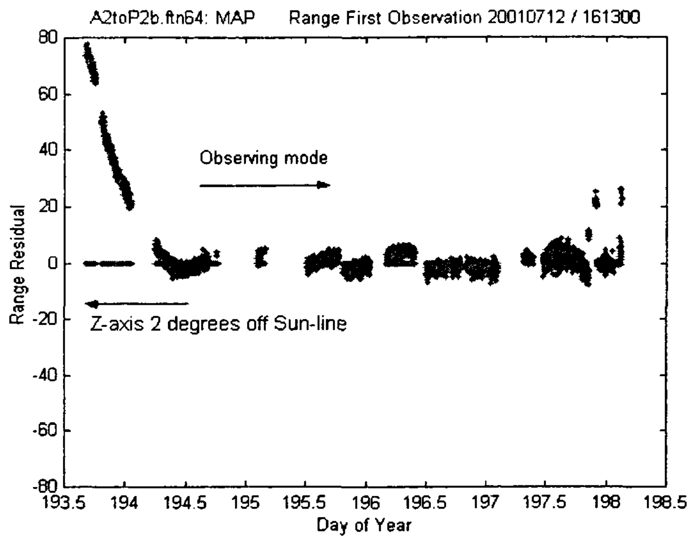

Figure 2 Post-fit Apogee 2 to Perigee 2 Range Residuals
(Range in meters)

Systematic Errors

An effort was made to examine two potential sources of systematic errors that may bias the overlap comparisons; that due to station location uncertainty and that due to uncertainties in a-priori range biases. In operational GTDS runs that apply given station locations and a-priori range biases instead of solving for these parameters, no consideration of their uncertainties is incorporated in the covariance of the estimated state.

Table 6

DEFINITIVE EPHEMERIS DIFFERENCES DUE TO VARIATIONS IN DSN STATION LOCATIONS

Solution Arc and Comparison Span	Maximum Definitive Position Difference (m)	Maximum Definitive Velocity Difference (cm/s)
S/C Sep to A1	185	1.4
A1 to P1	140	0.5
P1 to A2	130	1.6
A2 to P2	205	0.7
P2 to A3	105	0.7
A3 to P3	89	0.1
P3 correction to MCC	35	0.03

To examine OD uncertainty due to station geo-location, a set of runs were made using a set of DSN station locations obtained from Telecommunications and Data Acquisition (TDA) Progress Report by Folkner,[8] which differs slightly from the operational set. The operational geodetics employed in Flight Dynamics Facility (FDF) for these stations are currently being evaluated against those in the TDA report. MAP orbit solutions employing the TDA report locations have lower weighted root-mean-squares and observation residual standard deviations than solutions using the operational locations. Again, two sets of runs were made, the first using FDF operational station locations and the second using the station locations reported in the TDA report. The same tracking data was employed in each series of runs, and each solution solved for position, velocity, and C_r. The results are presented in Table 6.

Finally, a set of solutions was gathered to assess the effects of station range bias uncertainties. Historical analysis employing WIND and SOHO observations indicates that the sequential ranging assembly (SRA) range data received from DSN 34-meter stations and processed in FDF is observed to have a 100 ± 30 meter bias. In operations, this bias is typically solved-for, when possible. However, due to the predominance of 34-meter tracking in the MAP solutions, it was not possible to accurately determine the biases during MAP phasing-loop support, therefore a-priori value of 100 m was applied. Since MAP has been on station at L2, a series of parametric solutions have been performed to establish a new set of a-priori biases for MAP support, 88 meters for DS24, 98 meters for DS34, and 90 meters for DS54.

A series of runs were also made applying a 100-meter bias on all 34-meter DSN ranging, and another set using the new biases as a-priori. The same tracking data were used in each series of runs, and each solution solved for position, velocity, and C_r. The results are presented in Table 7.

As Tables 5-7 show, in all cases the definitive ephemeris uncertainty is less than 900 m in position, and less than 35 cm/s in velocity. In fact, most cases show position differences of less than 400 m and velocity differences of less than 5 cm/s. The largest position and velocity differences are associated with spans where thruster tests, deployments, or attitude mode changes were present. Comparison of Table 5 and Table 7 shows that in the absence of orbit disturbances, the largest consistent component of definitive ephemeris uncertainty appears to be due to uncertainties in range biases.

Table 7

DEFINITIVE EPHEMERIS DIFFERENCES INDUCED BY RANGE BIAS DIFFERENCES

Solution Arc and Comparison Span	Maximum Definitive Position Difference (m)	Maximum Definitive Velocity Difference (cm/s)
S/C Sep to A1	366	4.1
A1 to P1	344	0.9
P1 to A2	550	1.6
A2 to P2	490	0.9
P2 to A3	402	1.1
A3 to P3	247	0.7
P3 correction to MCC	703	0.2

Performance of Short Post-maneuver Tracking Data Arcs

Table 8 displays definitive and predictive differences between solutions that employ just 6 hours of post-maneuver tracking versus baseline definitive ephemeris. All of the baseline solutions used for the short-arc comparisons employed the JPL TDA Progress Report 42-128 station locations for the DSN sites, the parametrically determined 34-meter range biases, and solved for a coefficient of solar radiation. Baseline and short-arc runs apply a Doppler bias to compensate for the spacecraft's rotation about its Z-axis. The 6-hour and 12-hour short-arc solutions did not solve for solar radiation, but applied the value computed in the baseline solution. The 6- and 12-hour solutions included angle data, when available, but angle data was excluded from the baseline solutions. The tracking data arcs used in the 6-hour solutions were cut at six hours after the maneuver end time, regardless of whether six hours of tracking data had been received by that time.

It is apparent from Table 8 that post-perigee solutions performed much better than post-apogee solutions. This better performance is due to advantageous orbit geometry provided by the TDRS tracking at perigee. Excluding TDRS data from the post-P1 solution results in a single-station solution (DS46 only) with definitive error of 95 km in position and 39 m/s in velocity, and a predictive error of 1549 km in position and 8 m/s in velocity. The errors are primarily in the cross-track direction. Without TDRS data, the post-P2 definitive solution accuracy degraded to 13 km in position and 19 m/s in velocity and 77 km in position and 47 cm/s in velocity when predictive.

The separation +6-hour solution is worse than other near-perigee short arc solutions due to thruster testing, deployments, and attitude disturbances, which occurred later in the arc. In the case of the post-A1 six-hour solution, only approximately 4 hours of usable tracking were available during the first 6 hours. To obtain convergence with that amount of tracking, it was necessary to de-weight the Doppler observations from their nominal values. A 7-hour solution with nominal data weights gave significantly better results. No angle data was available in this arc. The post-A3 solution is degraded as a result of being a single-station solution.

Table 8

6-HOUR POST-MANEUVER TRACKING DATA ARC PERFORMANCE

Predictive Span	6-hour Maximum Definitive Position Difference	6-hour Maximum Definitive Velocity Difference	6-hour Solution Maximum Predictive Position Difference	6-hour Solution Maximum Predictive Velocity Difference
S/C Sep to A1	1.7 km	8.9 cm/s	17.8 km	8.9 cm/s
A1 to P1	90.7 km	1.3 m/s	927 km	397 m/s
P1 to A2	332 m	2.5 cm/s	1.9 km	1.3 cm/s
A2 to P2	820 m	6.6 cm/s	49.0 km	24.5 m/s
P2 to A3	624 m	33.4 cm/s	1.9 km	0.9 cm/s
A3 to P3	5.4 km	13.4 cm/s	140 km	53.2 m/s
P3 correction to periselene	17.6 km	28.4 cm/s	62.6 km	3.4 m/s

For orbits lacking good observability of the orbit plane, post-maneuver short-arc OD accuracy has been shown to benefit from constraining the plane of the post-maneuver state to be very close to that of the a-priori estimate. This technique is implemented operationally by performing the solution in Keplerian coordinates and applying tight a-priori covariances for the inclination and right ascension of ascending node. Typically, covariances of 1.0×10^{-12} degrees^2 are applied. Table 9 shows the results of reprocessing the post-apogee 2 and post-apogee 3 solutions under these constraints. For each of these solutions, the a-priori vector employed was taken from a longer, definitive post-maneuver solution.

The technique of tightening a-priori covariances on the right ascension and inclination likely represented a better estimate of the plane than would have been available for the 6-hour solution operationally, especially in the case of maneuvers with out-of-plane components. Both cases show considerable improvement under the constrained plane scheme.

Performance of the 12-hour post-maneuver tracking data arc was also studied. Nearly all 12-hour solutions showed improvement over the 6-hour solutions. Definitive overlap compares were 254 m position and 2 cm/s velocity for the post-P1 arc; 639 m position and 0.2 cm/s velocity for the post-P3 arc. The worst 12-hour definitive overlap compares were 2.7 km position for the post-A1 12-hour arc and 33.3 cm/s velocity for the post-P2 12-hour arc. The worst predictive compares were 48.1 km and 24 m/s, both for the post-A2 12-hour solution. In the case of the post-A2 12-hour solution, definitive overlap compares of 2.4 km and 8.1 cm/s were obtained, both worse than the 6-hour solution. This result may be an indication that the definitive ephemeris uncertainty after 12 hours of post-apogee tracking is truly on the order of 2 km.

Table 9

6-HOUR TRACKING DATA ARC WITH CONSTRAINED PLANE

Predictive Span	6-hour Maximum Definitive Position Difference (km)	6-hour Maximum Definitive Velocity Difference (cm/s)	6-hour Solution Maximum Predictive Position Difference (km)	6-hour Solution Maximum Predictive Velocity Difference (m/s)
A2 to P2	1.034	1.2	8.6	4.3
A3 to P3	2.251	8.9	101	38.6

L2 Operations

During routine operations at L2, MAP typically receives 45 minutes of range and Doppler tracking per day, almost exclusively from the DSN 70-meter antennas. L2 station-keeping requirements are for a minimum of 4 hours pre-burn and 4-hours post-burn near-continuous tracking.[9] MAP will execute approximately 4 station-keeping maneuvers per year.
MAP reached the proximity of L2 in early October 2001. The first station-keeping maneuver occurred on January 16, 2002 and the second on May 8, 2002. A series of solutions were generated using 6-week tracking data arcs between October 3, 2001 until SK1 and then between SK1 and SK2, with 3 weeks of data overlap from span to span.

Table 10 shows the definitive and predictive ephemeris differences between these solutions. The predictive compares are taken from the end of the definitive arc until the next maneuver, compared against the final OD prior to the maneuver. All solutions employed the JPL TDA Progress Report 42-128 geodetics for DSN station locations, solved for C_r, and applied optimal range observation biases.

The high predictive and definitive compares prior to the 5 December solution are due to an extreme solar storm that occurred on November 6, 2001. The effect of this storm is very evident, when compared to the ephemeris differences seen in the unperturbed cases. In all other cases, both predictive and definitive ephemeris differences are less than 5 km in position and 1 cm/s in velocity. In this orbit regime, definitive ephemeris differences induced by differences between JPL TDA Progress Report geodetics and the operational FDF geodetics are about 3 km position and 0.1 cm/s velocity. The definitive ephemeris difference between application of the optimal range biases and nominal (100 meters) range biases is about 2 km and 0.1 cm/s.

Table 10

6-WEEK SOLUTION ARCS WITH 3-WEEK DEFINITIVE OVERLAP (L2 OPERATIONS)

Solution Tracking Data Span	3-Week Maximum Definitive Overlap Position Difference With Prior Ephemeris (km)	3-Week Maximum Definitive Overlap Velocity Difference With Prior Ephemeris (cm/s)	Maximum Predicted Position Difference With Final Pre-manuever Ephemeris (km)	Maximum Predicted Velocity Difference With Final Pre-manuever Ephemeris (cm/s)
Arrived at L2, approximately 1 Oct 2001				
10/03/01 To 10/14/01	-	-	145	9
10/24/01 To 12/05/01	30	3	45	2
10/14/01 To 12/26/01	26	2	2.4	0.1
12/05/01 To 01/16/02	1.7	0.1	-	-
SK1 16 Jan 2002				
01/17/02 To 02/28/02	-	-	3.5	0.2
02/07/02 To 03/21/02	1.5	0.1	1.6	0.1
02/28/02 To 04/11/02	1.2	0.1	3.4	0.1
03/21/02 To 05/08/02	2.7	0.1	-	-
SK2 8 May 2002				

To date MAP has successfully executed a total of 4 L2 station-keeping maneuvers. No significant problems have been observed concerning definitive orbit determination at L2.

PREMISSION AND POST-PROCESSED OD COMPARISON

Comparing the premission covariance analysis results with those from the post-processed OD is a complicated and delicate task. The comparison of predictive ephemeris accuracies is not necessary since the two processes use the same algorithms and force models for orbit propagation. The problem is therefore limited to the comparison of definitive ephemeris accuracies.

As discussed in the previous section, in the absence of an independent precision ephemeris, the common method of accessing the post-processed OD accuracy is through definitive ephemeris overlap compares, under the assumption that no perturbations and systematic errors bias the solutions. The covariance analysis provides the estimated ephemeris accuracy under certain tracking scenarios to plan for maneuver control, whereas post-processed maximum overlaps provide information about the orbit consistency. Results from the covariance analysis and those from post-processed OD are, therefore, not meant to be equivalent. However, by leaving aside the data that are dominated by perturbations and systematic errors, a general correlation between results from the two processes, as displayed in Table 11, can be seen. This correlation suggests that orbit errors estimated by the premission covariance analysis are relatively close to the actual OD errors, enough to assist mission analysts with maneuver planning and recovery. It also shows that even with changes in tracking scenarios due to actual tracking support availability, the requirements for definitive and predicted ephemeris accuracies to be performed by the post-launch OD support (Table 3) can be met.

Table 11

PREMISSION ANALYSIS AND POST-PROCESSED OD COMPARISON

Event	Estimated Definitive Ephemeris Accuracy (3σ) from Premission Analysis (From Tables 1 & 2)	Post-processed Maximum Definitive Overlap Compares (From Tables 5 & 8 and 12-h performance)
At P1	Pos: 33 m Vel: 0.89 cm/s	Pos: 32 m Vel: 0.3 cm/s
At P3 (Pfinal)	Pos: 53 m Vel: 0.93 cm/s	Pos: 46 m Vel: 0.1 cm/s
6 hours from P1	Pos: 300 m Vel: 1.05 cm/s	Pos: 332 m Vel: 2.5 cm/s
12 hours from P1	Pos: 241 m Vel: 0.354 cm/s	Pos: 254 m Vel: 2 cm/s
12 hours from P3	Pos: 886 m Vel: 2.57 cm/s	Pos: 639 m Vel: 0.2 cm/s
3-week arc after L2 insertion	Pos: 2.593 km Vel: 0.1 cm/s	Pos: 2.7 km Vel: 0.1 cm/s

ORBIT DETERMINATION SUPPORT LESSONS LEARNED

A number of lessons for OD may be taken from this analysis. The advantageous orbit geometry provided by TDRS tracking at perigee passage gave 6-hour and 12-hour post-perigee solutions greater predictive accuracy than post-apogee solutions. Application of tight a-priori covariances on right ascension and inclination can improve the predictive accuracy of short-arc solutions by reducing the state-space of the estimation to those solutions in the vicinity of the expected orbital plane. This technique requires that the a-priori state represent a good estimate of the post-maneuver plane. In the case of in-plane maneuvers, the last pre-maneuver state may be used, but for plane-changing maneuvers, the accuracy of the constrained prediction will be dependent upon the accuracy of the predicted post-maneuver state. This technique may be especially helpful when TDRS tracking is not available.

Attitude mode changes impact the definitive ephemeris uncertainty by effectively reducing the usable tracking data span. This is primarily a consequence of least-squares estimation and could be ameliorated by the implementation of an attitude-dependent area model or potentially by filter estimation. This effect did not have a negative impact on predicted ephemeris accuracy as these errors did not have time to grow appreciably before the next delta-V, but did notably degrade definitive accuracy. The magnitude of this effect depends on the spacecraft structure. In the case of MAP, the spacecraft solar arrays and Sun shade system act similarly to a solar sail in the flat-spin modes.

Studies on MAP OD accuracy are impacted adversely by the apogee-perigee maneuver scheme during the phasing loops, which makes obtaining consecutive overlapping full-orbit solutions almost impossible. The magnitude of the ephemeris differences can be driven by the choice of comparison end time, where radial and along-track differences change rapidly. Future analysis of deep-space mission ephemeris accuracy would benefit from the generation of independent precision orbit ephemeris whose inherent accuracy is well known.

Finally, results from the premission covariance analysis are used as a baseline for mission and maneuver planning only. They could be significantly different from position overlap obtained from postlaunch OD data, especially when involving major purturbations, biases, and differences in tracking scenarios.

CONCLUSION

Intensive premission covariance analysis and actual postlaunch orbit determination were performed in support of MAP. The premission analysis provided valuable information on frequency of tracking, definitive arc length, and measurement types in supporting various mission phases. Several challenges were presented to the postlaunch orbit determination accuracy analysis including perturbations due to changes in spacecraft attitude, systematic errors due to uncertainties in station location and a-priori range bias, and the lack of independent precision definitive ephemeris. Even though results from the covariance analysis and the actual OD are not equivalent, they show a general correlation. They suggest that errors estimated from the covariance analysis can be close to actual errors, enough to benefit the mission to plan for maneuver control. Many important lessons learned from the MAP orbit determination support would help improve future mission support. The technique of tightening a-priori covariances on right ascension and inclination can be especially helpful when TDRS tracking is not available.

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