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Mars Odyssey Navigation Experience

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The 2001 Mars Odyssey Mission has returned an orbiter to Mars to map the planet and search for water. The success of this mission has reestablished confidence in Mars exploration that will pave the way for future orbiters, landers, and rovers. The spacecraft has completed its journey and is now in the science gathering phase of the mission. The orbital mission began in February 2002 and will continue through at least 2006. We present an overview of the navigation performance, with a comparison of the prelaunch requirements and expected performance to the in-flight experience.

Introduction

THE Mars Odyssey orbiter (Fig. 1) is the latest in an ongoing program of Mars missions to map the red planet and search for water. Odyssey was designed to launch in the 2001 opportunity and carry the last of the lost Mars Observer instruments, the gamma ray spectrometer (GRS), as well as a thermal and visible imager (THEMIS), and a radiation monitor (MARIE). The goals of the payload suite are to globally image the planet, determine surface mineralogy and morphology, determine the elemental composition of the surface and shallow subsurface, and study the Mars radiation environment from orbit.

The Odyssey spacecraft was launched 7 April 2001 aboard a Boeing Delta II 7925 launch vehicle from Cape Canaveral Air Station in Florida. The relatively short seven-month journey to Mars concluded with a successful orbit insertion burn on 24 October 2001. The spacecraft then employed aerobraking techniques over the next three months to reduce the orbit from the elliptical 18.6-h capture orbit, down to the desired 2-h circular mapping orbit. This was accomplished by flying through the upper atmosphere of the planet and allowing the atmospheric drag to remove energy from the orbit. Odyssey successfully finished aerobraking on 11 January 2002, after 332 drag passes through the Martian atmosphere. Following several weeks of orbit trim maneuvers and spacecraft reconfiguration, the primary science mapping mission began on 19 February 2001. The science mission was planned to extend for 917 days, concluding in August 2004.

The Odyssey mission underwent intense scrutiny following the losses of the Climate Orbiter and Polar Lander missions, and a number of measures were taken to increase the robustness of the mission and the navigation performance. The cruise phase of the mission was the most challenging from a trajectory determination perspective, as the driving navigation requirement was to deliver the spacecraft to an altitude 300 km above the north pole of Mars with an accuracy of ± 25 km. The navigation enhancements and robustness measures paid off, because the achieved altitude at encounter was less than 1 km from the target altitude.

The aerobraking phase demanded 24-h operations and quick reactions to accommodate the ever-changing Mars atmosphere. The primary requirement for the navigation team was to provide a trajectory that would predict the time of the upcoming drag passes to within 225 s. Because the density profile experienced during each drag pass was highly variable, the amount of energy removed with each drag pass, and therefore the change in orbit period, was difficult to predict to high accuracy. Trajectory updates were delivered up to four times per day to accommodate the uncertain atmosphere and meet the strict timing requirement.

Now in the mapping phase, the navigation task is to provide high-accuracy trajectory predicts and reconstructions for science planning and processing. To meet the strict position and pointing requirements of the high-resolution camera, trajectory predicts are delivered once per week.

Flight System

The Odyssey flight system was developed under a Jet Propulsion Laboratory contract with Lockheed Martin Astronautics (LMA) in Denver, Colorado. The spacecraft bus was a build-to-print of the Mars Climate Orbiter, with minor modifications to accommodate the science payload package.

Telecommunications between the Earth and the spacecraft are conducted via an X-Band radio system. The primary communications path is through the high-gain antenna, which is deployed on a boom and gimbaled to track the Earth. The telecom system also includes a receive-only low-gain antenna and a medium-gain antenna for alternate communications scenarios, as well as a uhf system to support data relay from future landed assets.

Spacecraft attitude determination is achieved through the use of star cameras and sun sensors with an intertial measurement unit to propagate the attitude between star camera updates. The spacecraft is three-axis stabilized, with three orthogonally mounted reaction wheels (and a spare skew wheel) that spin to absorb excess angular momentum. When the wheel momentum threshold is reached, generally 2 N-m-s, this excess momentum must be unloaded. This event, known as an angular momentum desaturation (AMD), is accomplished by firing the small attitude control thrusters to counteract and unload the angular momentum. Because the thrusters are not coupled, the thrusting imparts a net translational ΔV to the spacecraft.

The propulsion system is a pressure-regulated dual-mode system utilizing a high-pressure helium tank, two hydrazine fuel tanks, and one nitrogen-tetroxide oxidizer tank. The propulsion system is used to provide low-thrust trajectory correction maneuvers (TCMs), which are performed with the four axially mounted 22 N (5 lb_f)TCM thrusters. Momentum management is maintained via the four 1-N (0.2-lb_f) reaction-control-system (RCS) thrusters. The RCS thrusters can also be used to provide backup attitude control in the event of a failed reaction wheel. Both sets of thrusters are operated in a monopropellant blowdown mode.

The Mars orbit insertion (MOI) burn was performed by the axially mounted LEROS-1B bipropellant main engine, located in the center of the propulsion module. The main engine was used only once, and this was the only time that the propulsion system was operated in bipropellant mode. The oxidizer load was designed to be completely spent at MOI. The main engine performed as planned, delivering a thrust of 695 N (151.9 lb_f) at an I_{sp} of 317 s.

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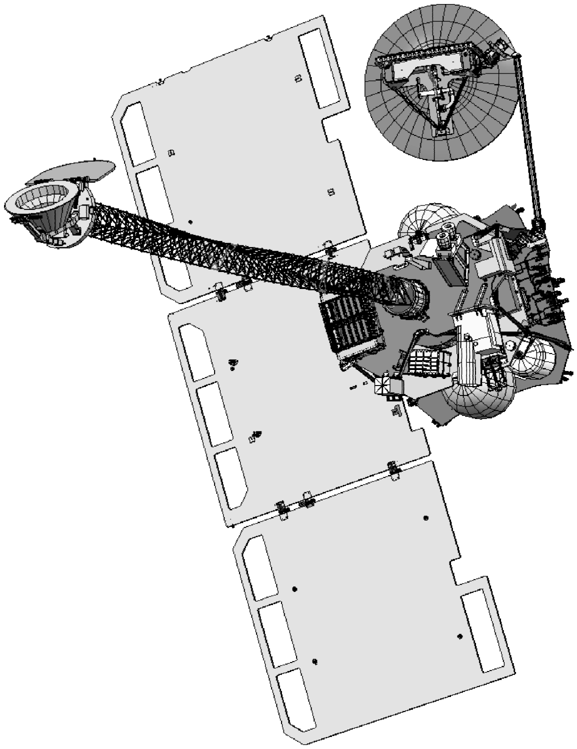

Fig. 1 Mars Odyssey spacecraft.

The spacecraft wet mass at launch was 730 kg. The total fuel mass distributed between the two tanks was 225 kg. The single oxidizer load was 122 kg.

Spacecraft power is generated by a single solar array mounted on a two-axis gimbal. The two gimbals allow the array a hemispheric range of motion to maintain sun-point during the various mission phases. There has typically been enough power margin to leave the arrays fixed in a convenient position for each mission phase, mitigating the need to vector-track the sun.

Science Payload Suite

The Odyssey spacecraft carries three science payload packages. The Thermal Emission Imaging System (THEMIS) is a visible and infrared imager, which can map the planet surface at up to 20-m spatial resolution and determine surface mineralogy by using multispectral thermal-infrared images at 100-m resolution. The GRS experiment suite maps the elemental composition of the planet surface at 300-km spatial resolution with the gamma sensor head (GSH) and the hydrogen and CO_2 abundances with the high-energy neutron detector. The GSH is deployed on the 6-m boom. The Martian Radiation Environment Experiment (MARIE) characterizes the radiation environment in Mars orbit and during the interplanetary cruise to Mars.

Launch Phase

The Orbiter was launched from Space Launch Complex 17A (SLC-17A) at Cape Canaveral Air Force Station in Florida, on a Delta II in the 7925 configuration. The mission had designed a 21-day launch period that opened on 7 April 2001 and closed on 27 April 2001. There were two daily launch opportunities corresponding to a launch azimuth of 65 deg, a long coast trajectory profile, and park orbit inclinations of 52 and 49 deg. The launch window for each opportunity was designed to be instantaneous. The arrival date was dependent on the launch date, and Mars arrival was designed to occur between 17 and 28 October 2001. Launch occurred on the first available opportunity.

A typical Eastern Range flight profile was employed to achieve the 100-n mile circular parking orbit at the first cutoff of the second stage (SECO-1). Two plane change (dog-leg) maneuvers were employed to fly up the coast of North America and achieve the desired parking orbit inclination. A thermal conditioning roll was employed during the coast period as the spacecraft flew over Europe, followed by the second-stage restart. Over the Middle East, the third-stage burn injected the upper-stage/spacecraft stack onto the required escape trajectory, and after spinning up the spacecraft separated nominally from the upper stage.

Launch Targeting and Performance

Planetary protection policies require that, after injection, both the upper stage and the spacecraft must not be on an impacting trajectory with Mars to a probability level of 1 in 10,000. To meet this constraint, it is necessary to bias the injection aimpoint away from Mars to ensure that neither of the terrestrial vehicles will contaminate the Martian environment. The upper stage does not have the ability to perform maneuvers after injection, and so the injection must be biased far enough to accommodate the upper stage. The size of the aimpoint bias is dependent on the expected injection dispersions. The dispersions are provided in the form of an injection covariance matrix, which is a Cartesian covariance about the nominal injection state, at the time of the third-stage engine cutoff. Propellant is required to correct for the aimpoint bias, so it is desirable to keep the aimpoint correction as small as possible.

Injection accuracy is also described via the figure of merit (FOM), which is a statistical measure of the ΔV required at the first maneuver opportunity to correct for the injection errors in the absence of any other errors or target biasing. Because the majority of the interplanetary ΔV is spent at TCM-1, primarily to correct for injection errors, injection accuracy has a significant impact on propellant budget. The statistics associated with the injection covariance matrix are listed in Table 1.

The biased injection targets are expressed in terms of the energy (C3), declination (DLA), and right ascension (RLA) of the outgoing hyperbolic trajectory asymptote. The targets and expected 3σ delivery uncertainty for the Odyssey launch opportunity are given in Table 2.

The actual injection result can be compared against the target and expected dispersion statistics and presented as a σ-level miss from the target as shown in Table 3.

The injection dispersions can be mapped along the nominal trajectory to the Mars target plane to illustrate the expected arrival conditions. The target and expected dispersions are presented in Fig. 2, along with the achieved. The launch dispersion happened to occur in a favorable direction, toward Mars.

Table 1 Injection covariance statistics

Parameter	Value, m/s
FOM	18.7
Aimpoint bias correction	15.4
Mean TCM-1	23.4
99% TCM-1	47.7

Table 2 Injection accuracy

Injection parameter	Target	Accuracy
C3	10.692	± 0.22 km^2/s^2
DLA	−51.727 deg	±0.20 deg
RLA	235.041 deg	±1.14 deg

Table 3 Launch vehicle performance

Injection parameter	Achieved	Delta from target	Miss vs expected
C3	10.767	0.075	1.0σ
DLA	−51.669 deg	0.058 deg	1.0σ
RLA	235.169 deg	0.128 deg	0.3σ

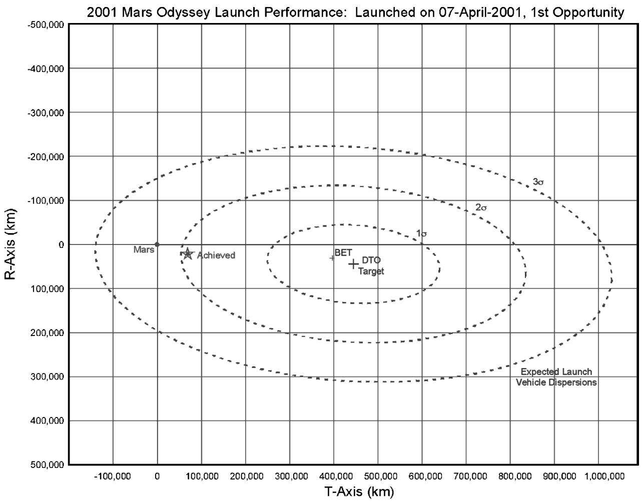

Fig. 2 Odyssey launch performance.

Cruise Phase

The relatively short interplanetary cruise phase of the mission took less than seven months. Activities during this phase included initial deployment and checkout of the spacecraft in its cruise configuration, checkout and calibration of the spacecraft and payload subsystems, and navigation activities necessary to determine and correct the flight path to Mars.

Communication with the spacecraft is accomplished via the deep-space network (DSN) of ground-based radio antennas, distributed around the world. Engineering telemetry, science data, and radiometric tracking data are collected during each tracking pass. One contact or tracking pass per day with a 34-m antenna was standard for the cruise phase, with continuous tracking provided around the critical events such as launch, maneuvers, and final approach.

One peculiarity of the high negative declination trajectory is that for the first two months of cruise only the southern-hemisphere stations were able to view the spacecraft. During that time, the Canberra tracking stations had very long view periods, greater than 16 h. Late in May, Goldstone came into view, and Madrid could not view the spacecraft until early June. A tracking station in Santiago, Chile, was contracted to supplement the DSN for the first month of cruise.

During the majority of cruise, the spacecraft was configured such that the stowed high-gain antenna pointed toward the Earth, and the solar array pointed generally toward the sun with an offset angle profile.

The primary navigation responsibility during this phase was to accurately determine and control the trajectory of the spacecraft to deliver it to the desired aimpoint at Mars encounter. This was accomplished by tracking the spacecraft radio signal to determine the orbit and designing propulsive maneuvers to alter the trajectory. Four maneuvers were scheduled to achieve the necessary delivery accuracy, with a fifth maneuver as a contingency in the final hours prior to encounter.

Although standard radiometric orbit determination techniques were employed to navigate the spacecraft, the traditional Doppler and ranging measurements were supplemented by a series of interferometric measurements known as delta-differenced one-way ranging (ΔDOR). This measurement is independent of the traditional radiometric measurements, and provides crucial out-of-plane trajectory information that is difficult to determine from more traditional data types. This technique has been utilized with success in the past, but the aging hardware and software system was completely rebuilt in preparation for this mission. A total of 45 ΔDOR measurements were planned and obtained over a four-month period. The cruise-phase orbit determination techniques and results are described in detail in Ref. 1 and will not be discussed further here.

Thruster Calibrations

Reaction-wheel assemblies provide primary attitude control and are desaturated via RCS thrusters. This event, known as an AMD, is accomplished by firing the attitude control thrusters to unload the momentum. The thrusters fire in pairs to desaturate each spacecraft axis sequentially, but are not coupled. Desaturation events occurred on a daily basis throughout cruise. Because each thruster firing imparted a net ΔV to the spacecraft, the thruster telemetry was recorded and downlinked for flight team evaluation. Although the net translational ΔV from each event was small (less than 10 mm/s), the cumulative trajectory perturbation was quite large, on the order of 10,000 km. So careful trending and calibration was required to meet the delivery accuracy requirements.

Three inflight thruster calibration activities were scheduled in the baseline reference mission, and only two were actually performed. An active calibration occurred shortly after launch, which involved slewing the spacecraft to view the RCS thrusting from several different angles. Monitoring continued throughout cruise, and one passive calibration, which did not involve attitude changes, was performed.

The goal of the active calibration effort was to completely characterize the magnitude and direction of the thrust vector for each RCS thruster pair. The calibration was designed to fire thruster pairs in sequence to spin up, then spin down each reaction wheel. The translational velocity change was then measured with the Doppler, and the body and wheel rates were captured in telemetry. This sequence was performed in an Earth-pointed attitude, as well as three off-Earth attitudes. The combination provided a viewing profile that enabled the Doppler to sense the vector components of the velocity change from three nearly orthogonal attitudes.

The passive calibration was performed three months before encounter to ensure that the thruster behavior had not changed significantly. It involved all of the data collection, analysis, and interaction between the teams that was required for the active calibration, but did not involve any spacecraft attitude changes.

Trajectory Correction Maneuvers

Although the ideal Orbiter trajectory does not require any deterministic deep-space maneuvers to reach Mars, a schedule of four TCMs was established to provide for sufficient control of the arrival conditions.

TCM-1 was designed as part of a multimaneuver optimization strategy to correct the injection errors, aimpoint bias, and other trajectory errors while maintaining appropriate conditions for planetary protection. Although the TCM-1 aimpoint was selected as part of the maneuver optimization strategy with some consideration for planetary protection requirements, the TCM-2 aimpoint was explicitly biased to satisfy overall planetary protection requirements for the cruise phase. TCM-3 and TCM-4 corrected for the remaining trajectory errors and targeted directly to the desired encounter conditions in preparation for the MOI burn. The relative execution time for each maneuver is given in Table 4.

All maneuvers were executed with the four 22-N TCM thrusters. In all cases, the ΔV direction was constrained to ensure that the medium-gain antenna could maintain communications with Earth at the burn attitude. This constraint was incorporated into the aimpoint biases for the early maneuvers to ensure a telecom link for TCMs 2 and 3. TCM-4 was a statistical clean-up maneuver, and so could not be constrained a priori, but a strategy was developed to maintain communications.

Because of the beneficial injection error, TCM-1 was delayed by several weeks with no propellant penalty. The ΔV required for the cruise phase turned out to be substantially less than planned,

Table 4 Interplanetary trajectory correction maneuvers

Maneuver	Planned, days	Actual, days	Actual date
TCM-1	L+9	L+46	23 May 2001
TCM-2	L+90	L+86	02 July 2001
TCM-3	E−40	E−37	17 Sept. 2001
TCM-4	E−12	E−12	17 Oct. 2001

again because of the positive injection results. Table 5 presents the planned (99%) and actual ΔV and fuel usage associated with each maneuver, as well as the angle from the antenna boresight to the Earth.

The total propellant budget for the cruise phase was dominated by the anticipated maneuvers, but also included fuel allocations for momentum management, as well as for specific events and contingency. Of the total cruise allocation of 21.9 kg, only 3 kg was actually spent. This windfall was recognized shortly after launch, and the unspent fuel was termed strategic propellant, as the project then had the decision of how best to utilize this propellant to mitigate risk or potentially extend the mission. The allocation of strategic propellant was an ongoing analysis that continued throughout the cruise and aerobraking phases.

Delivery Accuracy

The orbit determination and delivery accuracy for all of the cruise TCMs are documented in Ref. 1. Presented next is the final aimpoint and the actual delivery resulting from TCM-4. The final 0.08-m/s maneuver was designed to correct the residual targeting error left after TCM-3.

The 3σ delivery requirements were to meet the targeted altitude to within ± 25 km, the targeted inclination to within 0.2 deg, and the closest approach time to within ± 10 s. Table 6 presents the target and achieved values and demonstrates that the requirements were met. Figure 3 illustrates the achieved and targeted values in the Mars B-plane, along with the constraints.

Table 5 Maneuver performance

Maneuver	Planned (99%), m/s	Actual, m/s	Fuel, kg	Off-Earth angle, deg
TCM-1	48	3.6	1.2	7
TCM-2	38	0.9	0.3	14
TCM-3	3.5	0.45	0.15	2
TCM-4	0.5	0.08	0.04	20
Total	53	5	1.7	2/m

Table 6 Mars delivery accuracy

Parameter	Target	Achieved	Delta
Altitude	300.0 km	300.7 km	0.7 km
Inclination	93.47 deg	93.51 deg	0.04 deg
Periapsis time (ET)	02:29:58	02:29:58	<1 s

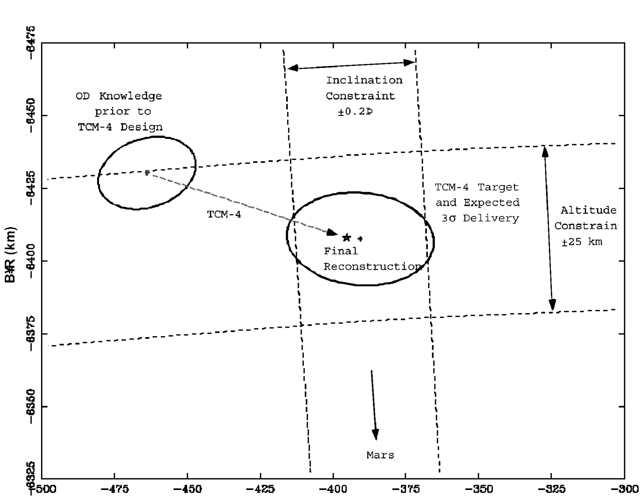

Fig. 3 Final delivery accuracy.

Orbit Insertion Phase

The orbit insertion phase (Fig. 4) began with the initiation of the MOI sequence, 10 days before encounter. Two days prior, the solar array was stowed to configure the spacecraft for the MOI burn. The final TCM-5 maneuver opportunities at 24 and 6 h before MOI were not needed, and system-level fault protection was disabled at 22 h out. Fifteen minutes before the burn, the propellant lines were filled, and shortly thereafter the fuel and oxidizer tanks were pressurized. Seven minutes before, the spacecraft slewed to the preburn attitude, and at this point could only communicate via the medium gain antenna (MGA) in a carrier-only configuration. The main-engine valves were opened simultaneously to initiate MOI, and the engine fired for almost 20 min. Throughout the burn, the spacecraft maintained a constant angular rate, termed a pitchover, to increase the efficiency of the capture maneuver. Ten minutes into the burn, the spacecraft passed behind Mars, as viewed from the Earth, and communications ceased as planned. The burn terminated when the onboard accelerometers detected a 15% thrust decay, indicating that the oxidizer had been depleted. Pyros were then fired to permanently isolate the pressurant from the propellant tanks. Ten minutes after the end of the burn, the spacecraft came back into view, and communications were reestablished.

The nominal main-engine thrust of 703 N and I_{sp} of 322.4 s, along with the 121.3 kg of available oxidizer, defined the expected MOI burn time of 1183 s, expected total propellant expenditure of 266 kg, and ΔV of 1420 m/s. Min and max timers were set at 1115 and 1225 s, respectively, to ensure proper burn termination in the event that the accelerometer did not accurately sense the depletion event.

The postmaneuver analysis indicated that the main engine burned for 1219 s, expended 146.3 kg of fuel and 121.4 kg of oxidizer, and produced a total ΔV of 1433.1 m/s.

In addition to the altitude and inclination constraints placed on the navigation delivery, it was also necessary to achieve a capture orbit period that would permit a nominal aerobraking plan to follow. Based on statistics associated with the main-engine performance, expected RCS performance during the burn, and expected spacecraft mass properties, it was possible to construct a statistical distribution of the capture orbit period by means of Monte Carlo analysis. Figure 5 presents the range of expected capture orbit periods. The mean was 19 h. Based on the capture orbit statistics and associated aerobrake profiles, the project was able to define a maximum capture orbit period of 22 h from which aerobraking could be initiated. If the capture orbit period exceeded this value, then a period reduction maneuver (PRM) would need to be executed before aerobraking to reduce the orbit period to an acceptable level.

If needed, the PRM would, have executed about 48 h after MOI in monopropellant mode on the TCM thrusters. However, the achieved capture orbit period came in below the mean at 18.6 h and eliminated the need for a period reduction maneuver.

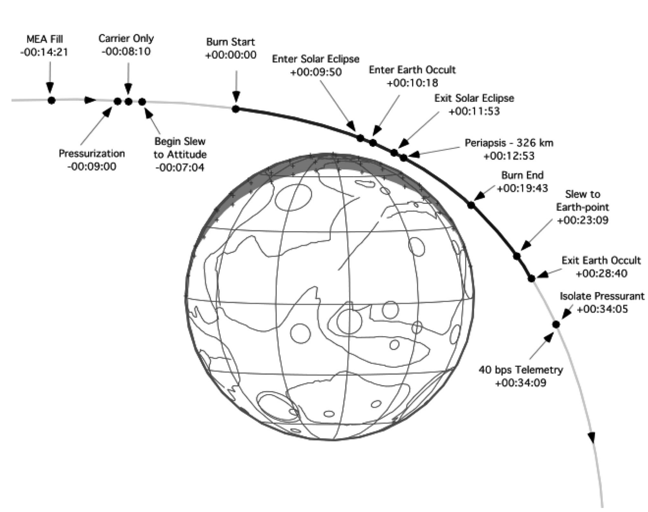

Fig. 4 Mars orbit insertion.

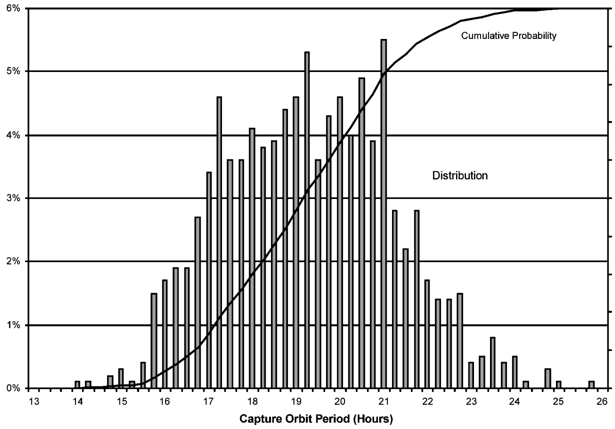

Fig. 5 Capture orbit statistics.

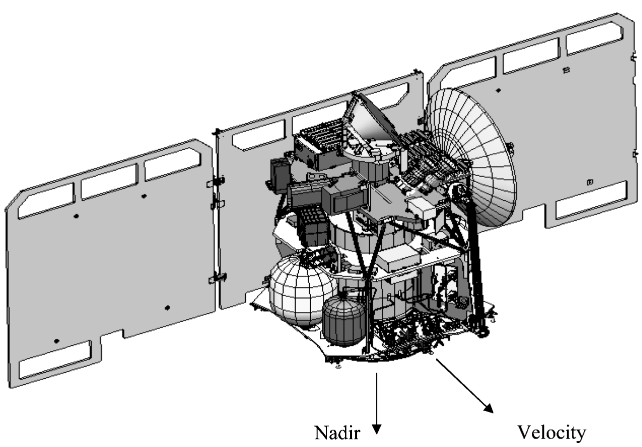

Fig. 6 Odyssey aerobraking configuration.

One other event that was closely monitored in the orbit-insertion phase was potential close approaches with Phobos, the inner moon of Mars. The elliptical, polar capture orbit would permit close encounters with Phobos for particular capture orbit periods. Fortunately, the 18.6-h capture orbit was out of phase with Phobos, and no collision-avoidance maneuvers were necessary.

Aerobraking Phase

On 24 October 2001, the successful MOI maneuver left Odyssey in an 18.6-h orbit. Over the next three months, a total of 332 passes through the Martian atmosphere slowed the vehicle, reducing the orbit period to just under 2 h. The aerobraking phase required 24-h per day operations at both Jet Propulsion Laboratory and LMA and daily support of teams across the country. The aerobraking configuration is shown in Fig. 6.

The aerobraking campaign was subdivided into three separate phases. The walk-in phase gradually lowered the periapsis altitude from the 300-km capture orbit altitude down to the 110-km altitude desired to initiate main-phase aerobraking. Main-phase constituted the bulk of the aerobraking mission both in terms of time and number of drag passes. The aerobraking rate was determined by the heat-rate corridor, and the bulk of the period reduction took place during this phase. As apoapsis decays, the aerobrake rate is eventually constrained by orbit lifetime. Lifetime in this context is defined as the time it takes for apoapsis to decay to 300 km. Below this apoapsis altitude, the orbit will become unstable, and the spacecraft will quickly spiral in. The walk-out phase of this mission was defined to occur when the orbit lifetime reached 24 h.

The dominant mission constraint throughout aerobraking was heating on the solar panels during the drag pass, which limited the total drag that could be achieved with each pass. On the other hand, a desire to limit the total number of drag passes, and a power constraint that limited the aerobraking duration, required that a minimum level of drag be attained in order to finish aerobraking successfully. These constraints were satisfied by instituting a range of acceptable heating rates (function of density and relative velocity) that were used to define the targeted aerobraking trajectory profile. The heat-rate corridor and aerobraking strategy are described in detail in Ref. 2.

Variability of the Martian atmosphere significantly limited the ability to predict the density that would be observed on any given orbit. Overall, the observed variability exceeded 35% (1σ), measured as the ratio of the heating rates derived by the navigation process on successive passes. To accommodate this uncertainty, a 100% margin was generally maintained between the maximum targeted heating rate and the flight allowable limit. Although this margin protected the vehicle from excessive heating, the uncertainty in the periapsis times that resulted from the limited prediction capability necessitated frequent updates to the onboard sequences.

Prior to each drag pass, a series of spacecraft activities was performed to prepare the vehicle for the atmospheric conditions, including slewing the spacecraft to the drag attitude and stowing the solar array. If these activities were not completed before entering the atmosphere, the vehicle's safety could be compromised potentially resulting in loss of the mission. As the timing of these events was tied to the sequenced periapsis time, predicting the periapsis timing was critical to mission safety and was a primary responsibility of the navigation team.

There were two basic requirements on the navigation team during the aerobraking phase. The first was to predict the periapsis altitude to within 1.5 km. This constraint was intended to limit the atmospheric variability caused by altitude uncertainty and in practice was an easy requirement to meet. The second was to predict the time of each periapsis to within 225 s. This constraint was driven by the need to sequence the spacecraft events and was by far the most demanding navigation requirement. Atmospheric variability was the largest source of uncertainty in predicting the orbit timing. If the observed density for a given drag pass did not match the predicted value, the amount of energy removed from the orbit, and therefore the period reduction achieved by the pass, would also not match the prediction. Thus, the time of the next periapsis would be different from the prepass prediction. If that error was determined to be greater than 225 s, the timing for the next sequence would need to be adjusted to reflect the new expected periapsis time.

Given a reference aerobraking profile and an assumption for the maximum level of variability that was expected, a period error profile could be computed. Early in aerobraking, with the assumption of 80% variability and a nominal delta-period per orbit of more than 20 min, the difference between the expected periapsis time and the actual time could easily be more than 225 s after only one pass; therefore, a new trajectory predict was delivered after every drag pass to build the sequence for the upcoming pass. Later in aerobraking, as the nominal delta-period per orbit decreased, the error in the periapsis time prediction also decreased, so that gradually, more than one, and up to six orbits, could be predicted within the 225-s constraint.

Figure 7 indicates the prediction capability as a function of orbit period. The bottom curve is the number of predicted orbits that met the 225-s requirement, the middle curve is the corresponding number of hours the prediction was good for, and the top curve is the number of sequence builds per day that were required. Note that the most intense time in terms of build frequency was not the end when the orbit period was short, but in the 6- to 9-h orbit periods, where the time between orbits was relatively short, but the prediction capability was not yet greater than one orbit.

One key milestone in orbit predict capability was when the timing requirement could be met for more than one orbit. Based on the reference profile and an assumption of 80% variability, the two-orbit predict was not considered to be viable until a 6-h orbit period was reached. In practice this proved to be a reasonable estimate, because the orbit-to-orbit variability was observed to be no better than the 80% assumption. Figure 8 presents the period change per revolution and periapsis timing error that was observed throughout the aerobraking phase. Note that the 225-s requirement was not reliably met until the orbit period was less than 9 h.

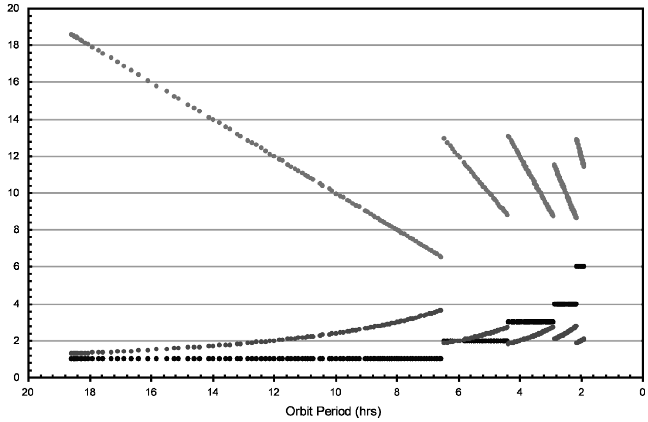

Fig. 7 Aerobraking predict capability.

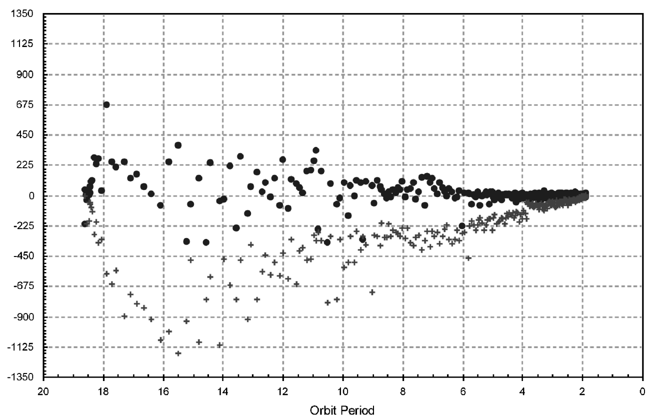

Fig. 8 Two-orbit predict timing accuracy.

At approximately the 9-h orbit period, Odyssey began to experience unusually low atmospheric variability over the north pole. We did not rely on this phenomenon ahead of time, but were able to take advantage of it in flight. The heat-rate upper corridor limit was effectively raised, and the aerobraking rate was increased for a time. Although the two-orbit predicts met the 225-s requirement, we did not build two-orbit sequences until the 6.5-h orbit, after we had demonstrated consistent prediction capability for some time. The increase in aerobraking rate, and corresponding period reduction per revolution, should have degraded the two-orbit prediction, but even this was compensated by the extremely low atmospheric variability.

The navigation timing and sequence build processes operated flawlessly throughout the grueling aerobraking phase, and the orbit timing requirement was never exceeded. The daily strategic planning process also operated as planned; the flight allowable solar-array temperature was never exceeded.

Strategic Propellant

To satisfy the low-risk posture this mission required, an ongoing mission trade was the option to allocate fuel to terminate aerobraking early and mitigate the risk of additional sequence uplinks and drag passes. Prior to orbit insertion, the projected mission propellant budget was reassessed, and depending on the MOI performance and aerobraking usage anywhere from 4 to 35 kg of propellant (20 to 190 m/s) could have been available to use for risk reduction. MOI fuel usage was nominal, and as aerobraking progressed it became clear that the aerobraking fuel usage (9.6 kg) would be significantly less than the 17.8 kg allocated. However, as the transition and science phase planning progressed, conservative analysis indicated that more fuel would be needed in those phases, bringing the transition phase allocation to 14.5 kg and the science phase allocation to 33.7 kg.

The algorithm for allocating strategic propellant was to first allocate 8.2 kg (40 m/s) for contingency use, 1.9 kg (10 m/s) for extending the science mission, and the remainder to early aerobraking termination. For conservatism, the available strategic propellant was chosen to be a 1% low value, which would guarantee a 99% probability of successfully completing the mission. At the final tally, 6.4 kg (35 m/s) of propellant was allocated for strategic risk reduction. This enabled the project to declare that aerobraking would be concluded when the apoapsis altitude reached 520 km. The remainder of the apoapsis altitude reduction would be accomplished propulsively with the strategic propellant allocation.

Transition to Mapping Orbit

The transition phase (Fig. 9) was designed to provide the time required to perform the propulsive burns needed to achieve the final mapping orbit, deploy the high-gain antenna, and configure the spacecraft and the science payloads for mapping operations. The spacecraft was maintained in an inertially fixed, Earth-pointed attitude, with the solar array on the sun for the majority of this phase. Momentum management was accommodated by angular momentum desaturations that occurred up to four times daily.

Because there was no strict requirement for orbit timing accuracy, the primary navigation task during this phase was to design and execute the five transition maneuvers. The first was simply the aerobraking termination maneuver, ABX1.

The ABX1 attitude was selected from a predesigned menu that was used for all aerobraking maneuvers (Ref. 2). The magnitude was also preplanned to be sufficiently large to raise periapsis altitude out of the atmosphere and keep it there for several weeks. The 20-m/s ABX1 maneuver executed on apoapsis 336 on 11 January 2002 raised the periapsis altitude to 201 km, and marked a successful conclusion to the aerobraking phase.

ABX2 was planned to execute several days after ABX1 at a time when the periapsis point had naturally drifted to the equator. At this time, it was optimal to perform a small inclination change and raise the periapsis altitude again. The inclination change was designed to set up the desired local mean solar time (LMST) drift desired for the science orbit. ABX2 was the largest maneuver of the mission (outside of MOI), and the 56-m/s maneuver was executed at apoapsis 393 on 15 January 2002. Following ABX2, the periapsis altitude was 419 km, and the inclination was 93.1 deg.

ABX3 was designed at the same time as ABX2 and was scheduled to execute just two days later to lower apoapsis and freeze the orbit. The frozen orbit condition requires the periapsis point to be at the south pole, but at this time periapsis was still close to the equator. The option to wait until periapsis naturally drifted to the south pole was undesirable, as it would delay the start of the science mission by several weeks, and the natural eccentricity variation would require even more propellant to compensate. The 27-m/s ABX3 maneuver executed successfully on orbit 417 on 17 January 2002. This maneuver established the frozen orbit by rotating the periapsis point to the south pole, and at the same time reducing the apoapsis altitude to 450 km and periapsis to 387 km.

Following these large transition maneuvers, two orbit trim maneuvers were planned to clean up any residual orbit error. ABX2 and 3 had executed just as planned, however, small execution errors, and

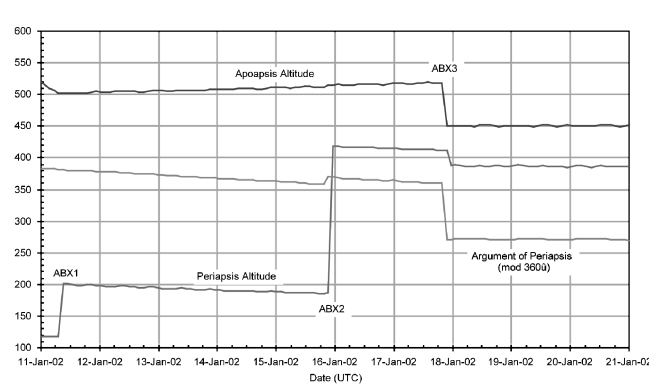

Fig. 9 Transition to science orbit.

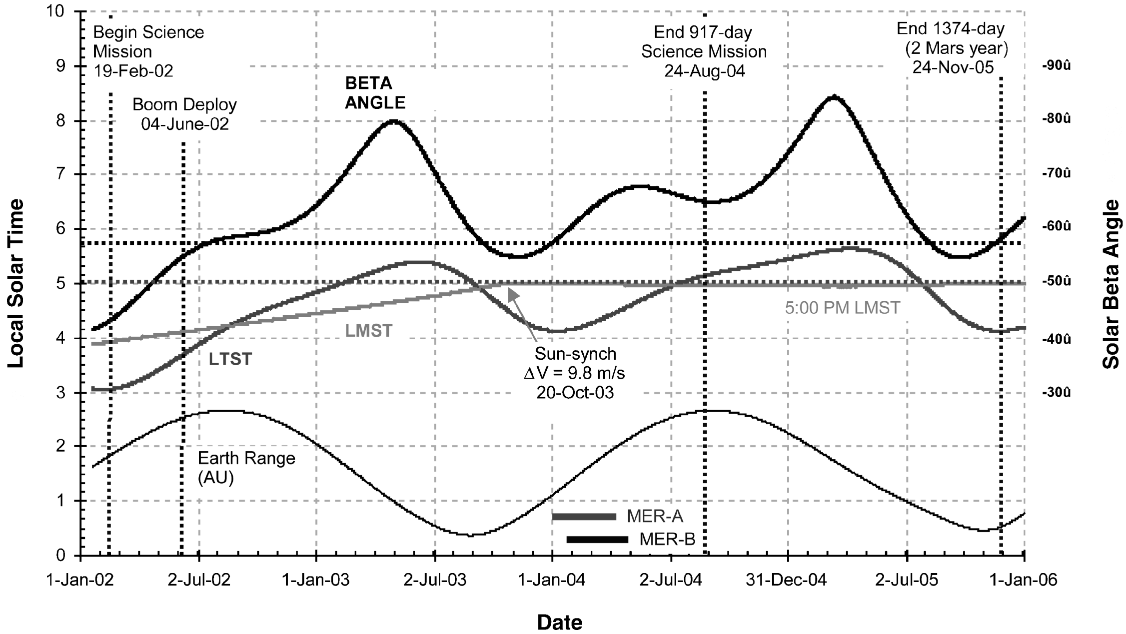

Fig. 10 Mapping orbit evolution.

orbit propagation uncertainty left the need for some small cleanup maneuvers. A week of tracking and maneuver design was allocated, and the two trim burns ABX4 and ABX5 executed on 28 and 30 January 2002. The combined ΔV of the two maneuvers was 3 m/s.

Propellant utilization during transition phase was dominated by the large transition maneuvers. A total of 20.25 kg of fuel was used, of which 14.5 kg was allocated for the nominal transition, and the remaining 5.75 kg came from the strategic propellant allocation.

Once the propulsive maneuvers were completed, the high-gain antenna was successfully deployed on 8 February 2002. Several spacecraft and science payload housekeeping and checkout sequences were accommodated in the following weeks, and the spacecraft turned to nadir-point on 18 February 2002. The THEMIS instrument was the first to be turned on, and it began imaging the planet on 19 February 2002, signaling the start of the 917-day science mission. The GRS door was opened the following day, and in early March the MARIE operations commenced.

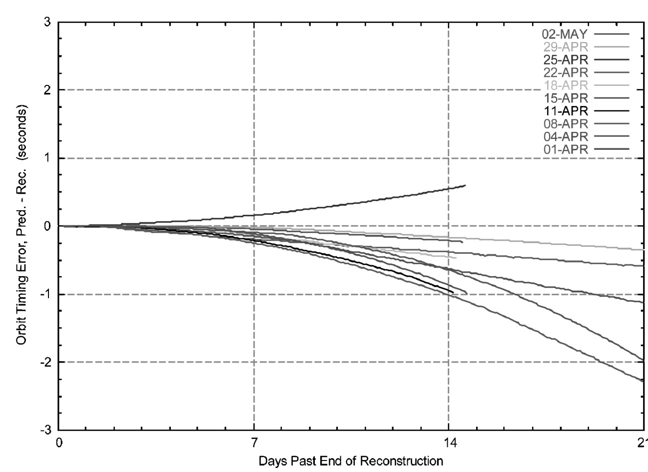

Fig. 11 Orbit predict accuracy.

Science Mapping Phase

The science orbit was established on 30 January 2002 following the final transition orbit trim maneuver. The 400-km, near-circular, frozen orbit provides the observational geometry desired by the science instruments. The orbit period of just under 2 h results in roughly 12.5 revolutions per Martian day, or sol. Successive ground tracks are separated in longitude at the equator by approximately 28.8 deg, and the entire ground track pattern nearly repeats every 2 sols, with a 1-deg shift to the west.

The science orbit design was negotiated to balance the observational desires of THEMIS with those of GRS. The MARIE investigation is insensitive to the orbit design. The somewhat conflicting requirements that drive the orbit design are that high-quality high-THEMIS infrared data can be obtained only at local true solar times (LTST) earlier than 1700 hrs, whereas high-quality GRS data are only obtained for solar beta angles less than −57.5 deg. The LTST and beta-angle profiles are controlled by the orbit inclination, which affects the orbit nodal precession rate (the rate at which the orbit plane rotates in inertial space). Figure 10 displays the time history of the science orbit LTST and beta angle for the planned science mission. The figure also includes LMST and Mars to Earth range.

The frozen orbit condition maintains a relatively fixed eccentricity and argument of periapsis for a given semimajor axis. The periapsis point is "frozen" at the south pole, and the orbit altitude at any given latitude is constant at all longitudes. The real benefit of a frozen orbit for the Odyssey mission is that it keeps the Odyssey orbit away from the orbit of the Mars Global Surveyor, which is also in a similar frozen orbit.

The initial orbit is frozen, but long-term perturbations will cause the orbit to wander from the frozen condition. The primary source of orbit perturbation is the angular momentum desaturations that are performed daily on the spacecraft. Each event imparts several millimeters per second to the trajectory. The events tend to occur at the same relative point in the orbit, so that over time the perturbation can build to the point where the cumulative effect is similar to an orbit trim maneuver.

The navigation requirements during this phase are to provide short-term trajectory predicts and reconstructs to support the science observations and long term predicts for background sequence development. The predictability of the desaturation events dominates the timing predict capability, and the regularity of the execution and propellant expenditure has enabled excellent prediction accuracy.

Figure 11 presents the predict accuracy over time for a typical month of the mapping mission. The plot illustrates how the timing error grows as a function of time past data cutoff, and each curve represents the delivery of a trajectory predict. The driving requirement

Table 7 Propellant allocation[a]

Allocation	Fuel, kg	ΔV, m/s
Contingency	8.2	40
Safe mode	5.0	20
Momentum management	11.5	47
Orbit trim maneuvers	3.7	18
Extended science	1.9	10
PQ orbit raise (EOM)	3.4	18

[a]Total fuel available: 45.1 kg ± 3 kg (1 March 2001). Unallocated fuel remaining: 11.4 kg.

is to provide predicts that are better than 6 s (3σ). The trajectory timing is updated each week, and the performance to date has easily met the requirement.

Only two planned maneuvers remain during the mission. The first is a plane-change maneuver to establish a sun-synchronous orbit that would be desirable for an extended mission. The second is a planetary quarantine (PQ) orbit raise maneuver that would occur at the end of mission. Propellant has also been allocated for any unplanned orbit trim maneuvers that might be required to compensate for the orbit perturbation from the desaturation events. Propellant for momentum management is required to maintain attitude and normal mapping operations. Contingency propellant has been allocated to accommodate off-nominal operations or safe-mode entries. The propellant budget was designed to provide a 99% probability of accommodating a 1374-day (two Mars-year) mission. The propellant windfall continues, as a significant amount of propellant remains unallocated. The detailed propellant allocations are (in kilograms and equivalent ΔV) given in Table 7.

Conclusions

The Odyssey mission has successfully returned an Orbiter to Mars and is well on its way to achieving all of the planned science objectives. The navigation performance has been excellent throughout and has met all of the project requirements at each stage.

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