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Abstract

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 re-established 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 orbital science-gathering phase of the primary mission, which will continue through August 2004. This paper will describe the strategy that was designed to safely and accurately navigate the spacecraft to Mars, and also relate the in-flight experience.

INTRODUCTION

The Mars Odyssey Orbiter 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 map the planet, determine the surface mineralogy and morphology, determine the elemental composition of the surface and shallow subsurface, and study the Mars radiation environment from orbit.

Odyssey launched atop a Boeing Delta II 7925 from Cape Canaveral on April 07, 2001. The relatively short six-month journey to Mars concluded with a successful orbit insertion burn on October 24, 2001. The spacecraft then employed aerobraking techniques over the next three months to reduce the orbit from the elliptical 18.6-hour capture orbit, down to the desired 2-hour circular mapping orbit. This feat 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 January 11, 2002, after 332 drag-passes through the Martian atmosphere. Following several weeks for orbit trim maneuvers and spacecraft reconfiguration the primary science mapping mission began on February 19, 2002. The science mission is planned to extend for 917 days, concluding in August 2004.

A twenty-one day launch period was selected that provided two launch opportunities on each day. The launch targets were designed to inject the spacecraft on the desired interplanetary ballistic trajectory to Mars, and at the same time accommodate planetary quarantine and propellant budget requirements. Launch occurred on the first opportunity and the launch vehicle performance was excellent, placing the spacecraft on the desired trajectory to Mars.

The 200-day journey to Mars was filled with spacecraft checkout and calibration activities, as well as four trajectory correction maneuvers. The cruise phase of the mission was challenging from a trajectory determination perspective, as the driving navigation requirement was to deliver the spacecraft to an altitude 300 $\pm$ 25 km above the North Pole of Mars. The corresponding maneuver strategy had to accommodate the delivery accuracy requirements, and also respect the planetary quarantine and propellant budget constraints.

The schedule of four trajectory correction maneuvers (TCM) was designed to provide sufficient control of the arrival conditions. Due to the favorable injection from the launch vehicle, the first trajectory correction maneuver was delayed from Launch+9 days to Launch+46 days. This delay was quite beneficial to the flight team schedule, and actually resulted in a small propellant savings. TCM-1 was designed as part of a multi-maneuver optimization strategy to correct the injection errors, aimpoint bias and other trajectory errors while maintaining appropriate conditions for planetary protection. 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 Mars Orbit Insertion (MOI) burn.

Two contingency maneuver opportunities were built into the schedule in the final day prior to Mars encounter. It was anticipated that these opportunities would not be needed in a nominal scenario, but all of the planning was in place to execute a maneuver, had the spacecraft been off course. A precise set of criteria were outlined, the plans were thoroughly reviewed, and flight products were built and tested. The navigation strategy and spacecraft operations were designed to provide robust trajectory determination performance, and in the final weeks this proved out as the trajectory estimates held steady. The contingency maneuvers were not executed, and the final trajectory easily met the requirement, as the achieved altitude at encounter was less than one kilometer from the target altitude.

FLIGHT SYSTEM

The Odyssey flight system was developed under a Jet Propulsion Laboratory contract with Lockheed Martin Astronautics in Denver, Colorado. The spacecraft bus was a build-to-print of the Mars Climate Orbiter, shown in Figure 1, 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 (HGA), 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.

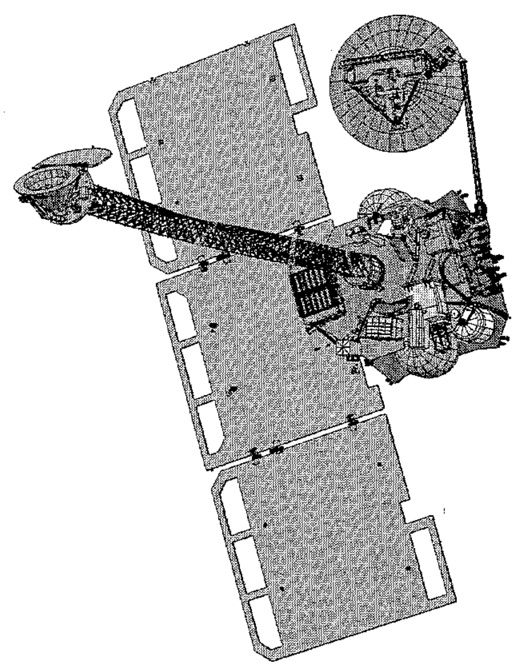

Figure 1: Mars Odyssey Spacecraft Mapping Configuration

Spacecraft attitude determination is achieved through the use of star cameras and sun sensors with an IMU 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 is provided for by the axially-mounted LEROS-1B bi-propellant 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 bi-propellant 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 Isp of 317 sec.

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 2-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-meter spatial resolution, and determine surface mineralogy using multi-spectral thermal-infrared images at 100-meter resolution. The Gamma Ray Spectrometer (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 (HEND). The GSH is deployed on the 6-meter boom. The Martian Radiation Environment Experiment (MARIE) characterizes the radiation environment in Mars orbit and during the interplanetary cruise to Mars.

LAUNCH PHASE

Launch Period

A twenty-one day launch period was designed that would minimize arrival velocity at Mars. Minimal arrival velocity implies less propellant required to capture into orbit. As the design of the Orbiter and the on-board propellant was fixed, it was necessary to find a launch/arrival space that would satisfy the existing Orbiter design. The following figure presents contours of arrival velocity (V-infinity), injection energy (C3), and injection declination. Injection energy is also a prime consideration, as this parameter will ultimately specify the size of the launch vehicle that will be required to place the spacecraft on the correct trajectory to Mars. Declination turned out to be a factor as well, as the optimal combination of injection energy and arrival velocity implied a high injection declination. Range constraints and launch vehicle capability make it difficult to reach high declinations when launching from the Eastern Test Range.

The selected launch period opened on April 7, 2001 and closed on April 27, 2001, with two launch opportunities on each day, distinguished by the launch vehicle parking orbit inclination. As each launch opportunity was instantaneous, two daily opportunities were designed to maximize probability of a successful launch within the launch period. Given the high injection declination that steadily decreased through the launch period, it was possible to design two distinct parking orbits that would satisfy the declination constraint. On each day, the first daily opportunity launched to a 52° inclination parking orbit, the second daily opportunity launched to a 49° inclination. The daily launch opportunities were separated by at least 30 minutes to allow time to recycle, and no more than 90 minutes, to accommodate launch vehicle oxidizer loading constraints.

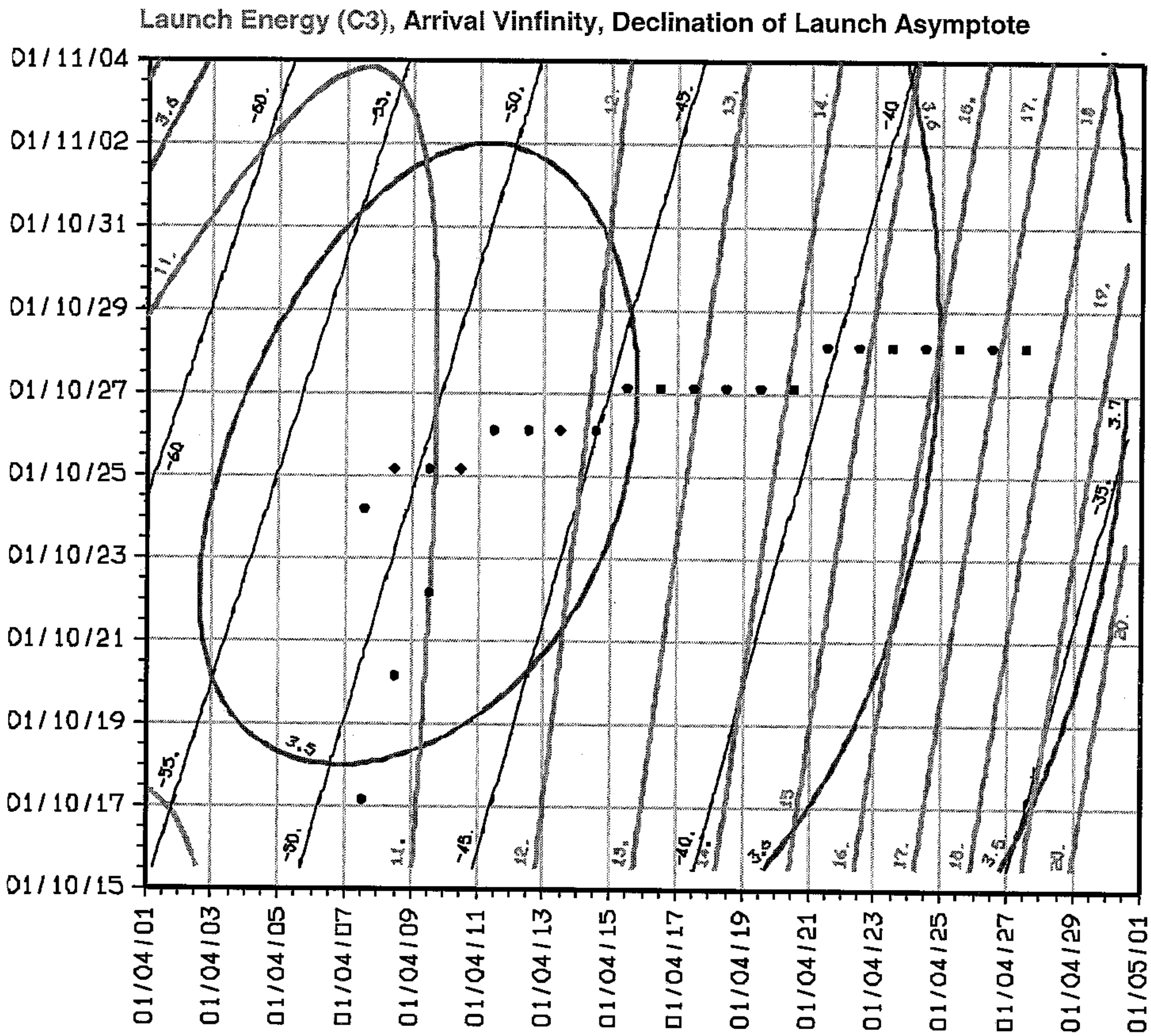

Figure 2: Launch And Arrival Date Space

Planetary Protection

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, 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 (ICM), which is a Cartesian covariance about the nominal injection state, at the time of the third stage engine cutoff (TECO). Propellant is required to correct for the aimpoint bias, so it is desirable to keep the aimpoint correction as small as possible.

Figure 3 presents a set of contours that map out the minimum launch bias that must be accommodated to meet the probability of impact requirements. Contours are presented for several dates in the primary 21-day launch period. Also shown are the selected launch targets, and the corresponding anticipated upper-stage arrival conditions. The data are presented in the Mars targeting plane, or B-plane, which is a plane normal to the incoming V-infinity, with Mars at the origin of the system. These targets, one for each day in the launch period, were selected to minimize the propellant required to correct the injection bias, and to satisfy planetary quarantine requirements. These targets corresponds to the first daily launch opportunity, characterized by the 52° parking orbit inclination. A similar strategy was adopted for the second daily launch opportunity, characterized by the 49° parking orbit inclination. In general, the deterministic ΔV required to correct for the selected injection bias was about 15 m/sec for each opportunity.

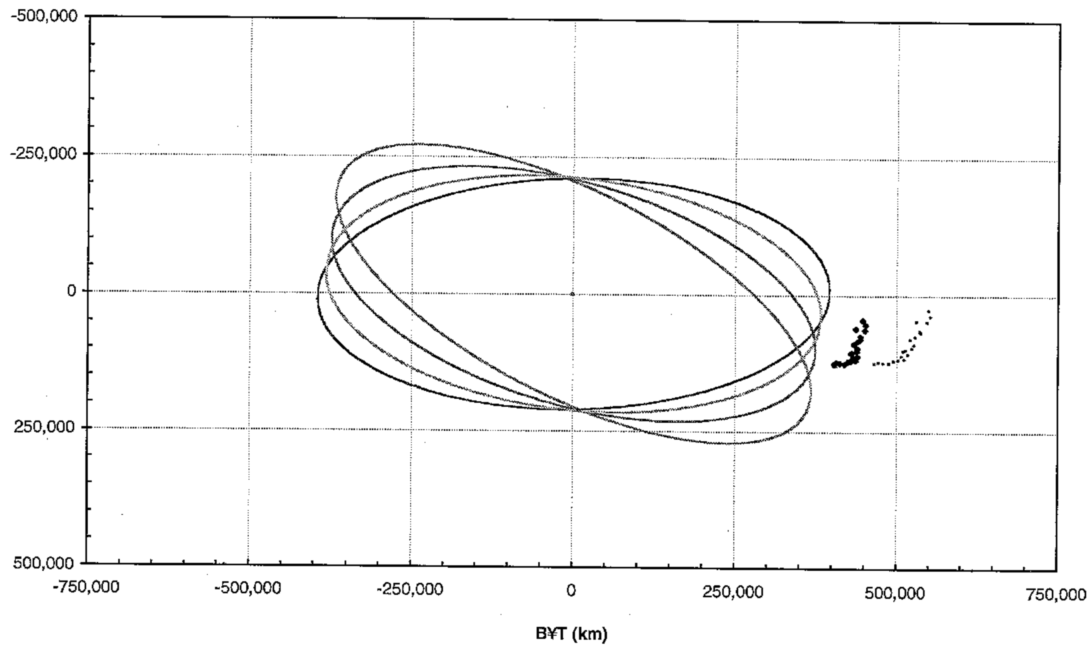

Figure 3. B-plane Arrival Targets and Planetary Quarantine Constraints

Injection Accuracy

Anticipated injection accuracy can be described via the Figure of Merit (FOM), a parameter calculated from the injection covariance matrix, 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. Since 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 range of FOMs for all opportunities was between 17 and 20 m/sec, indicating that the injection dispersions were reasonably similar for each case.

Based on the selected aimpoint bias, and the expected injection dispersions, the statistics for the first trajectory correction maneuver can be calculated. In general, the first maneuver will remove the majority of the aimpoint bias, and will also correct any injection dispersions. The maneuver statistics are generated via a Monte-Carlo analysis that includes launch vehicle dispersions, orbit determination dispersions, and maneuver execution errors. Prior to launch, the statistics for the first launch opportunity were calculated to be:

FOM	18.7 m/s
Aimpoint Bias Correction	15.4 m/s
Mean TCM-1	23.4 m/s
99% TCM-1	47.7 m/s

In general, the aimpoint bias correction was selected to be about 15 m/sec for each launch opportunity, while continuing to meet the planetary protection requirements. On the worst opportunity, the ΔV required at the 99% level to accomplish the cruise phase of the mission was 52 m/sec.

Launch Performance

Launch occurred on the first available opportunity on 7 April 2001, from Space Launch Complex 17A (SLC-17A) at Cape Canaveral Air Force Station in Florida. A typical Eastern Range flight profile was employed to achieve the 100 nmi 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 52° 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.

The biased injection targets were 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 first launch opportunity are presented in Table 1, below. The actual injection result are then be compared against the target and expected dispersion statistics and presented as a sigma-level miss from the target.

Injection Parameter	Target	3σ Expected Dispersions	Achieved	Delta from Target	Miss vs Expected
C3 (km^2/s^2)	10.692	± 0.22	10.767	0.075	1.0 σ
DLA	-51.727°	±0.20°	-51.669°	0.058°	1.0 σ
RLA	235.041°	±1.14°	235.169°	0.128°	0.3 σ

Table 1: Launch Vehicle Injection Accuracy

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 Figure 4, along with the achieved.

The launch dispersion happened to occur in a favorable direction. The biased launch target would have sent the spacecraft to an encounter 450,000 km from Mars at the closest approach. The actual injection left the spacecraft on a trajectory that would take it within just 70,000 km of the planet. Even though the launch dispersion moved the spacecraft closer to the planet, planetary quarantine requirements were never violated. The spacecraft and the upper stage were never on a trajectory that would potentially impact the planet.

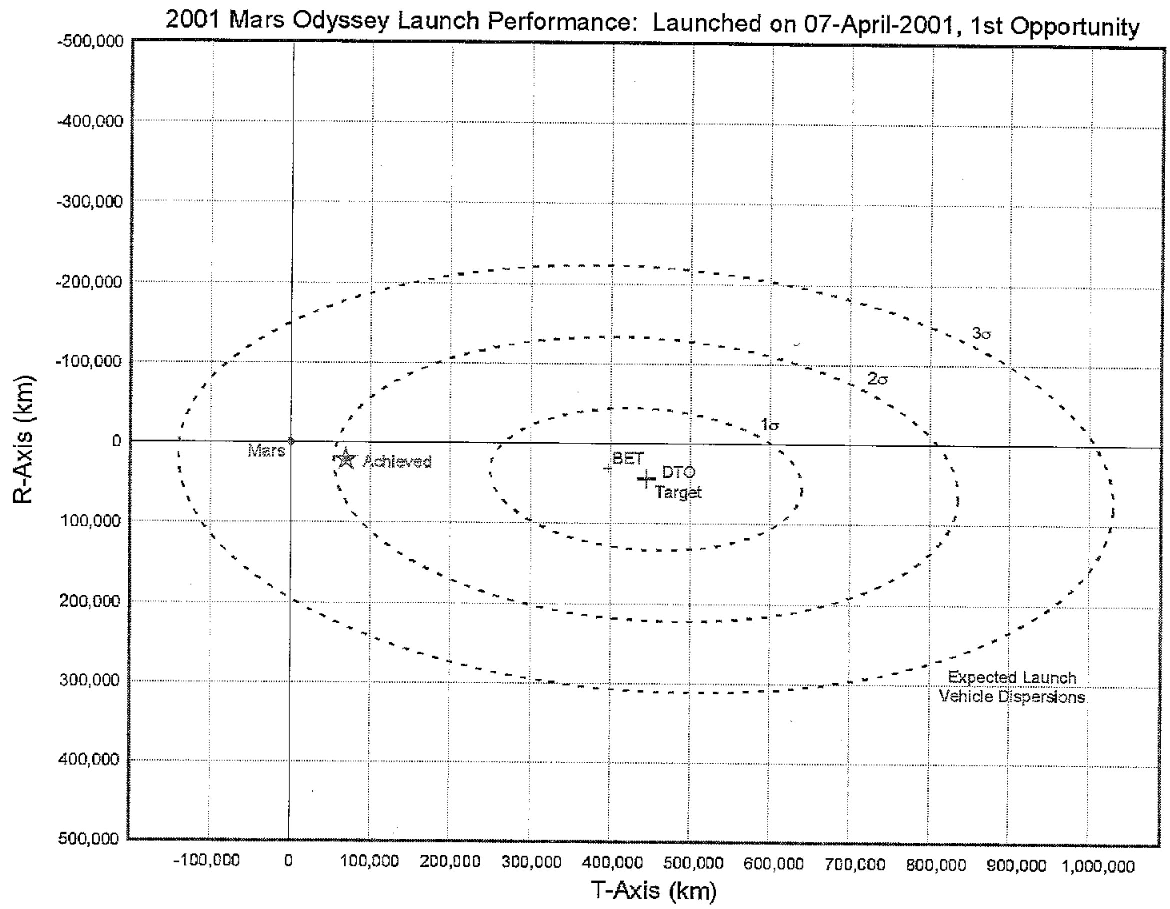

Figure 4: Odyssey Launch Performance

CRUISE PHASE

The relatively short interplanetary cruise phase of the mission lasted 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-meter 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 viewperiods, greater than 16 hours. 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 towards the Earth, and the solar array pointed generally towards 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 re-built 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 Reference 2, and will not be discussed further here.

Thruster Calibrations

Reaction Wheel Assemblies (RWAs) provide primary attitude control and are desaturated via RCS thrusters. This event, known as an angular momentum desaturation (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/sec) 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 in-flight 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 was performed which did not involve attitude changes.

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 prior to 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.

Planetary Protection

In addition to the launch vehicle requirement, there is also a mission requirement on Planetary Quarantine that is portioned out by mission phase. The total allocation for the cruise phase is to not exceed a cumulative probability of impact of 1 in 1000. So with each maneuver design, the probability of impact was calculated. It is always assumed that there is a small chance of 1 in 100 that the spacecraft would not physically be able to execute the next maneuver. So after each maneuver, the probability of impact along with the probability of being able to execute another maneuver must be sufficiently small to meet the cumulative cruise requirement. To accommodate this requirement, the aimpoints for some of the maneuvers were biased away from Mars to reduce the probability of impact statistics to help meet the overall cruise planetary protection requirements.

TRAJECTORY CORRECTION MANEUVERS

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

Maneuver	Planned	Actual	Actual Date
TCM-1	Launch+9 days	Launch+46 days	23-May-2001
TCM-2	Launch+90 days	Launch+ 86 days	02-July-2001
TCM-3	Enc-40 days	Enc-37 days	17-Sep-2001
TCM-4	Enc-12 days	Enc-12 days	17-Oct-2001

Table 2: Maneuver Schedule Planned vs Actual

TCM-1 was designed as part of a multi-maneuver optimization strategy to correct the injection errors, aimpoint bias and other trajectory errors while maintaining appropriate conditions for planetary protection. While the TCM-1 aimpoint is 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 Mars Orbit Insertion (MOI) burn. The relative execution time for each maneuver is given below.

All maneuvers were executed with the four 22N TCM thrusters. In all cases, the ΔV direction was chosen 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, so could not be constrained a-priori, but a strategy was developed to maintain communications.

In the event that the maneuver direction was not favorable for communications, an option was maintained to perform the maneuver in a vectorized mode. In this manner, the maneuver would be broken up into segments that would individually satisfy the telecom constraint, and the vector sum of the ΔV would be preserved. This capability was not needed during operations, as the maneuver design strategy, and a bit of luck, allowed each maneuver to be executed as a single burn with communications at all times.

Maneuver execution error sources are the hardware and control software errors that could alter the designed maneuver ΔV magnitude and direction. For the small maneuvers (less than 5 m/sec) that were performed, the magnitude error (3σ) was required to be less than 2% + 0.020 m/sec, and the pointing error (3σ) was required to be less than 100 mrad or 5.7°. All of the cruise maneuvers met the 3σ execution error requirements.

Based on the pre-launch requirements and orbit determination accuracy analysis, a Monte-Carlo analysis was performed to determine the statistics for each maneuver. Statistics for two cases, a launch on the first opportunity, and the last opportunity are given below in Table 2. The majority of the propellant is needed to accommodate potential launch vehicle injection errors. As TCM-1 and TCM-2 are optimized to produce the most efficient combination, both TCM-1 and TCM-2 have large ΔV estimates at the 99% level. However, the mean for TCM-2 is much smaller than TCM-1.

So there was a good chance that one or the other would be the larger burn, but they would not both be large. TCM-3 and TCM-4 are relatively small burns used mainly to adjust for trajectory perturbations and orbit estimation uncertainties along the way.

		ΔV Statistics in m/sec			
Maneuver	Date	Mean - μ	Std Dev – 1σ	ΔV 95%	ΔV 99%
		Open of Launch Period			
TCM-1	15-Apr-01	20.527	9.626	37.369	47.705
TCM-2	06-Jul-01	3.647	7.428	19.886	37.727
TCM-3	14-Sep-01	0.544	0.637	1.808	3.444
TCM-4	12-Oct-01	0.108	0.089	0.265	0.453
Total		**24.826**	**9.575**	**43.614**	**52.856**
		Close of Launch Period			
TCM-1	05-May-01	22.522	8.555	37.887	47.714
TCM-2	26-Jul-01	1.509	2.851	6.290	17.065
TCM-3	18-Sep-01	0.438	0.291	0.760	1.964
TCM-4	16-Oct-01	0.117	0.070	0.255	0.337
Total		**24.586**	**8.816**	**40.696**	**50.430**

Table 3: Pre-Launch Maneuver Statistics

TCM-1

The pre-launch plan for TCM-1 was to execute it as soon as reasonably possible to minimize the propellant cost to correct injection errors from the launch vehicle. Based on the spacecraft activities and flight team schedule, the maneuver execution time was chosen to be nine days after launch. At this time, given the launch vehicle dispersions, the total cruise propellant cost could be kept to less than 52 m/sec at a 99% confidence level.

TCM-1 was also exempt from the constraint of maintaining a telecom-favorable attitude during the burn. The cost of performing this maneuver in a vectorized mode was potentially large if the launch vehicle dispersion was unfavorable. However, although no formal constraint was placed on TCM-1, the desire to maintain communications during critical events was ever-present.

Post-launch, the orbit determination estimates indicated that the injection dispersion had occurred in a favorable direction, and it was determined that TCM-1 could be delayed by several weeks with no propellant penalty. This delay was of great value to the flight team, as the intense maneuver planning could be postponed to a more quiet time in the mission. It was also of benefit to the science goals, as flight team efforts could be focused on attaining a THEMIS Earth-Moon image. This image was successfully acquired twelve days after launch. It was calculated that the optimal time to execute TCM-1 from a propellant standpoint was May 23, forty-six days after launch.

The design of TCM-1 was performed as part of a two-maneuver optimization with TCM-2. Therefore, TCM-1 did not target to the final aimpoint, it targeted to an intermediate aimpoint that would minimize the total ΔV of both TCM-1 and TCM-2 together. In this way it was also possible to design each maneuver to have a high probability of occurring in a communication-favorable attitude.

TCM-1 executed as planned on May 23, 2001. The figure below presents the change in the trajectory that TCM-1 accomplished. Shown are the Orbit Determination (OD) solution estimate prior to the maneuver. Note that prior to the maneuver, the trajectory would have a closest approach to the planet of about 70,000 km. After the maneuver, the closest approach point has moved in to about 10,000 km, near the equatorial region. Not shown, the time of the closest approach also changed to be 3.5 hours earlier.

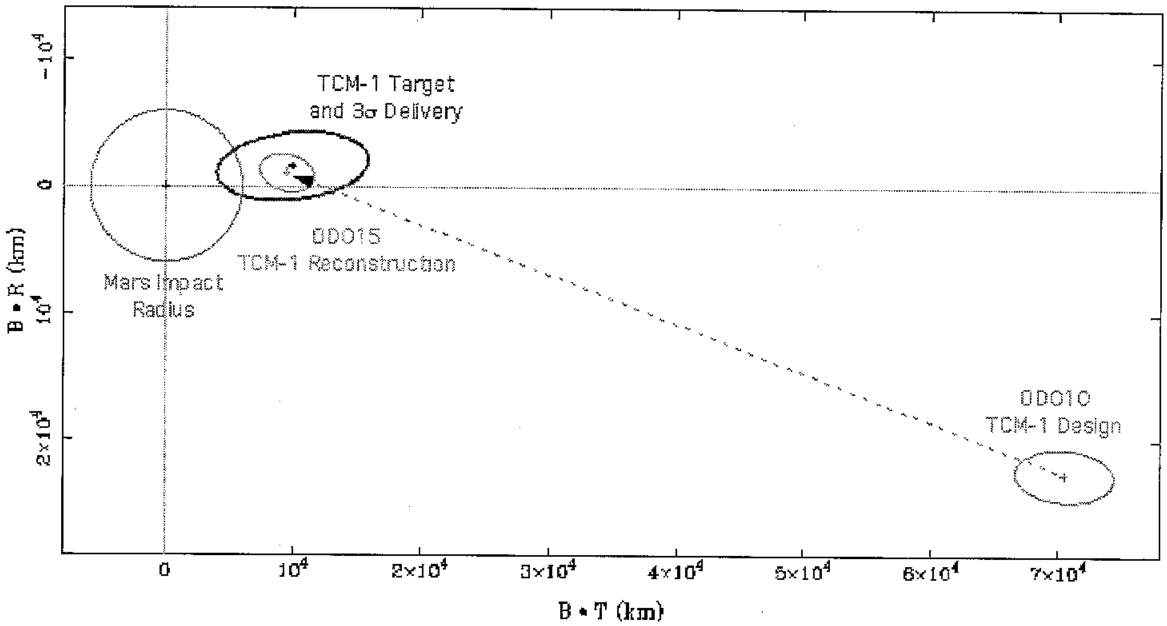

Figure 5: TCM-1

Parameter	Design	Reconstructed	Difference
ΔV Magnitude (m/s)	3.558 m/s	3.563 m/s	+0.14%
ΔV Right Ascension	-28.98°	-29.35°	-0.37°
ΔV Declination	-0.55°	-3.39°	-2.84°
Total Pointing Error		2.87°	

Table 4: TCM-1 Design and Performance

TCM-2

Following a nominal TCM-1, TCM-2 would ideally target to the final aimpoint. However, the cruise-phase planetary protection requirements required that the TCM-2 target be biased away from the planet, such that if no maneuver could be performed after TCM-2, the spacecraft would not be on an impacting trajectory. The final aimpoint was designed to be 300 km above the North pole of Mars, so the TCM-2 aimpoint was biased up to 1300 km above the North pole.

As TCM-2 was part of a dual-maneuver optimization along with TCM-1, the ΔV direction was designed to have a favorable angle for communications. Once the TCM-2 target bias was selected, the deterministic portion of TCM-3 was known to also be favorable for comm. Thus by appropriate selection of targets, it was possible to maximize the probability that each maneuver could be executed while in communications with the Earth, TCM-2 executed as planned on July 2, 2001. Originally scheduled to execute a Launch+90 days, the TCM-2 execution time was moved four days earlier to deconflict with the 4th of July holiday. The figure below presents the change in the trajectory that TCM-2 accomplished. Shown are the Orbit Determination (OD) solution estimate prior to the maneuver (OD005), along with the target, the expected delivery ellipse, and the final reconstruction (OD027). Note that prior to the maneuver, the trajectory would have a closest approach to the planet of about 70,000 km near the equatorial region. After the maneuver, the closest approach point has moved to about 1,300 km above the North pole.

Although the maneuver executed quite accurately, the plot seems to indicate that the maneuver did not achieve the target (OD027). This is misleading as the spacecraft trajectory was altered in a different way between the time of the maneuver design and the final reconstruction. The angular momentum desaturations provide a deterministic ΔV that perturbs the interplanetary trajectory. This effect was predicted and modeled. However, around the time of TCM-2, the AMD strategy was altered to provide less trajectory perturbation for the final sixty days of the mission. As the trajectory modeling to this point had assumed that this trajectory perturbation would occur, once the change in strategy was approved, the predicted effect was removed from the trajectory propagation. This had the effect of moving the predicted encounter point away from the planet, or higher in altitude.

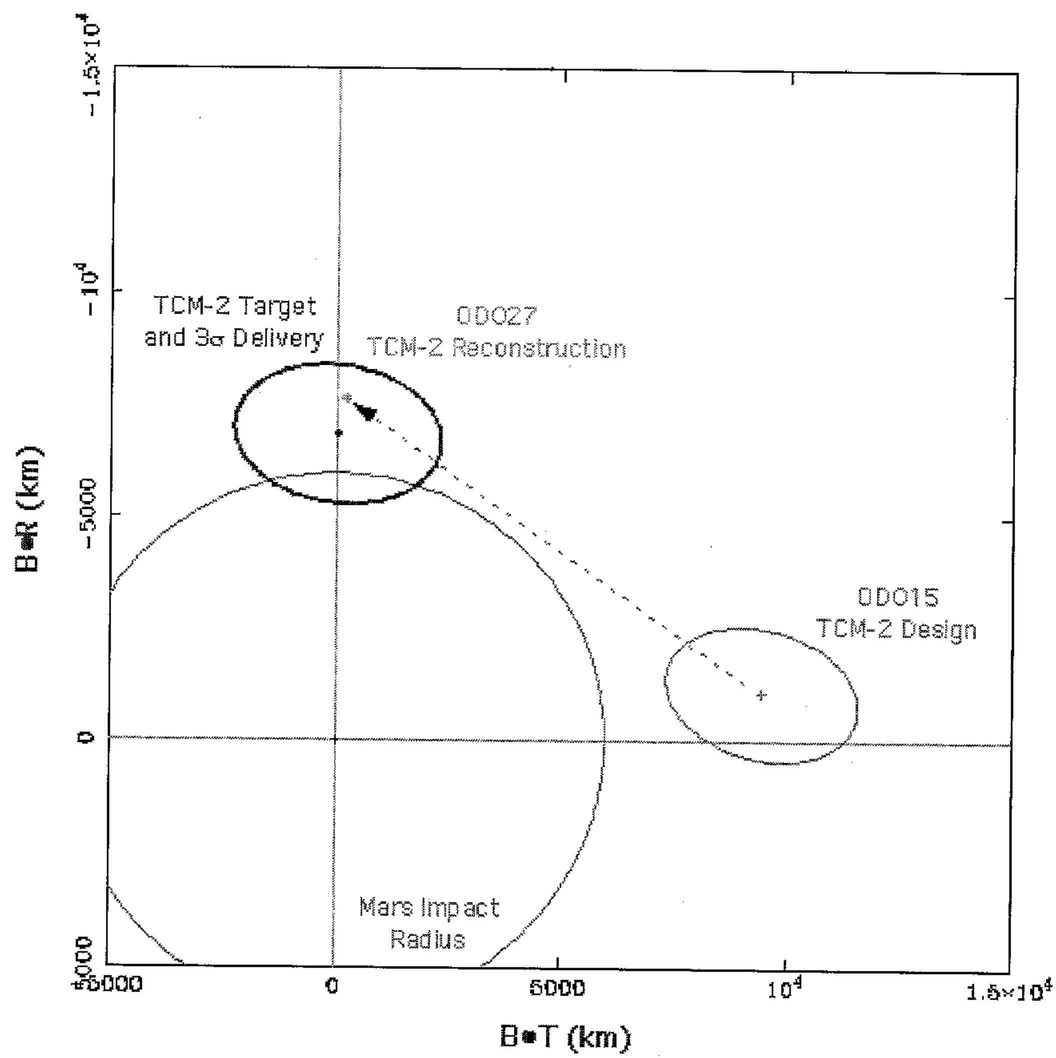

Figure 6: TCM-2

Parameter	Design	Reconstructed	Difference
ΔV Magnitude (m/s)	0.899 m/s	0.909 m/s	+1.12%
ΔV Right Ascension	-21.17°	-22.09°	-0.93°
ΔV Declination	8.34°	8.62°	+0.28°
Total Pointing Error		0.95°	

Table 5: TCM-2 Design and Performance

TCM-3

Because TCM-2 targeted to a biased aimpoint, it was almost assured that TCM-3 would be needed to meet the encounter requirements. The TCM-2 aimpoint bias was chosen in such a way as to ensure a high probability that the maneuver direction would be favorable for communications.

By the time of TCM-3, the final approach strategy for spacecraft operations was finalized. A new solar array position had been designed to minimize the angular momentum desaturations that produced the undesirable orbit perturbations. In this new *low-torque* attitude, it was anticipated that no further autonomous desaturations would occur. This was incorporated into the trajectory propagations and TCM-3 design.

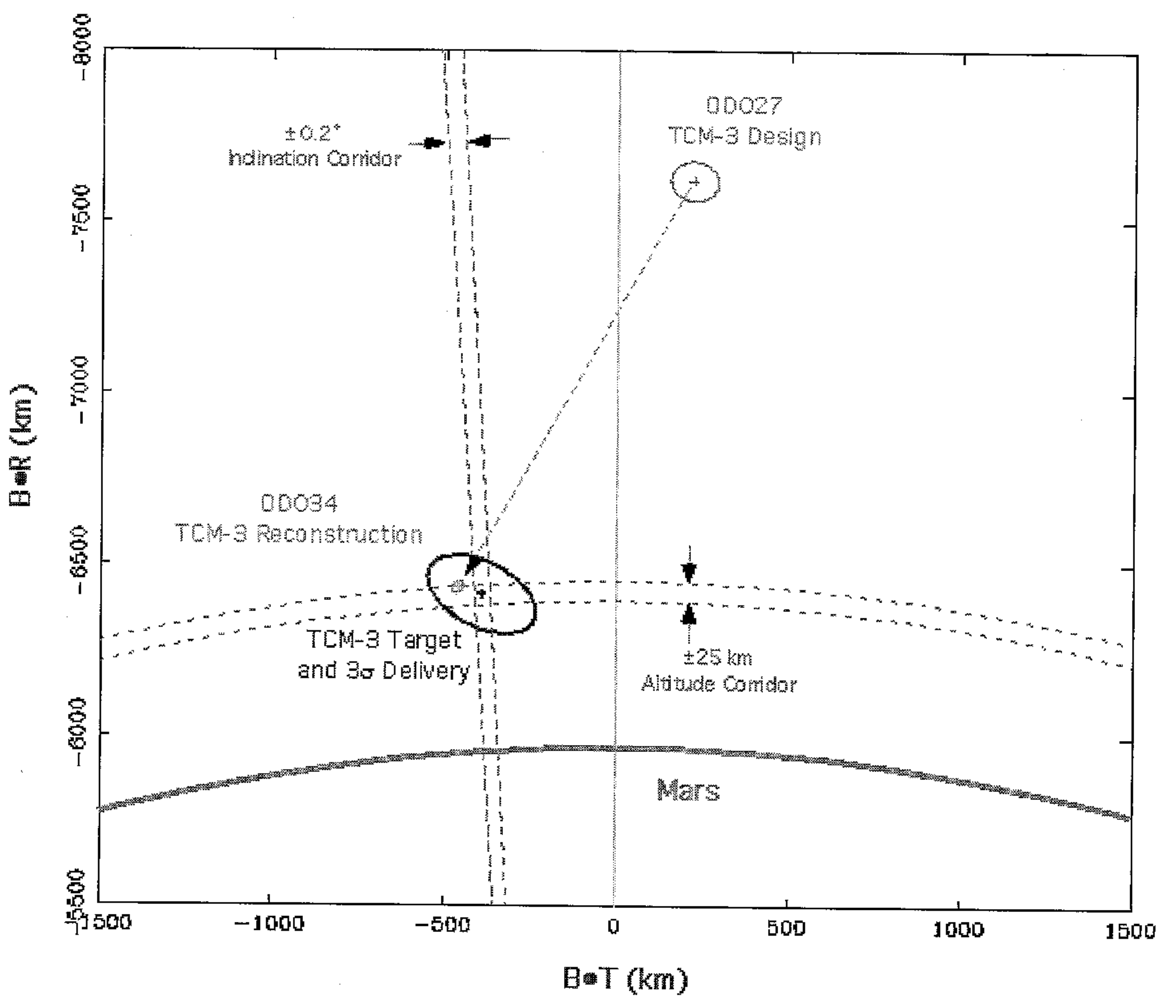

Figure 7: TCM-3

TCM-3 was the first maneuver to target directly to the final aimpoint, 300 km above the North pole of Mars. The maneuver was scheduled to execute 40 days prior to encounter, and was delayed three days to avoid operations over the weekend. Although the trajectory was very well determined, the pre-maneuver delivery statistics indicated that there was only a 10% chance that TCM-3 would meet the all three of the encounter corridor requirements: altitude, inclination and timing. TCM-3 executed as planned on September 17, 2001. The delivery met the execution error requirements and left the trajectory just outside the encounter corridor.

Parameter	Design	Reconstructed	Difference
ΔV Magnitude (m/s)	0.450 m/s	0.463 m/s	+2.98%
ΔV Right Ascension	87.97°	84.13°	-3.84°
ΔV Declination	-63.53°	-63.32°	+0.21°
Total Pointing Error		1.73°	

Table 6: TCM-3 Design and Performance

TCM-4

The trajectory estimates following TCM-3 were very closely monitored, and TCM-4 estimates were generated daily. The maneuver was scheduled to execute 12 days prior to encounter, which fit in well with the load of the MOI sequence 9 days out. In the days prior to TCM-4, it became evident that the trajectory altitude would be very close to the desired 300 km altitude requirement, and was almost within the ±25 km corridor requirement. It was also clear that the trajectory did not meet the inclination requirement. Violation the inclination corridor did not pose a risk to the Orbiter health and safety or to the mission goals, but would require propellant expenditure of approximately 2 kg to correct once in orbit. The propellant cost to correct the inclination at TCM-4 was less than 0.1 kg.

TCM-4 executed as planned on October 12, 2001. Figure 8 presents the trajectory estimate prior to the maneuver, the target and expected delivery dispersions, and the final reconstruction, along with the altitude and inclination corridor constraints. The final altitude was within 1 km of the target altitude and the inclination was within 0.04°, meeting the corridor constraints. Not shown, the time of encounter was less than one second from the target arrival time. As the MOI (Mars Orbit Insertion) burn was scheduled to initiate at a fixed time, an off-nominal arrival time would reduce the effectiveness of the orbit insertion maneuver and increase the capture orbit period.

Parameter	Design	Reconstructed	Difference
ΔV Magnitude (m/s)	0.077 m/s	0.077 m/s	-
ΔV Right Ascension	-174.58°	-174.55°	-0.03°
ΔV Declination	10.89°	10.83°	-0.06°
Total Pointing Error		0.07°	

Table 7: TCM-4 Design and Performance

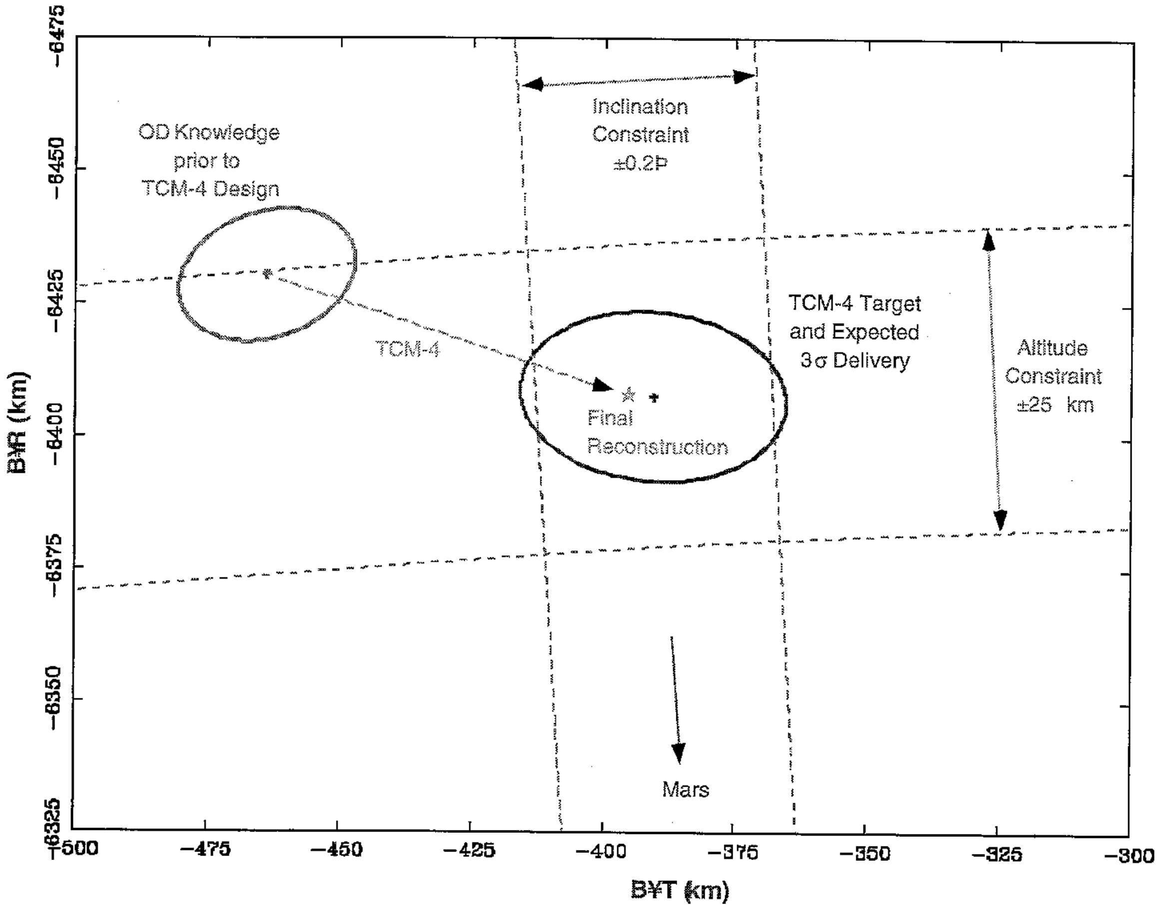

Figure 8: TCM-4

TCM-5

Although TCM-4 was the last scheduled trajectory correction maneuver, two opportunities were held in the final day prior to encounter to perform an altitude raise maneuver, should the trajectory be found to be off-nominal on the final approach. The inclusion of these contingency opportunities followed from the lessons learned on the Mars Climate Orbiter mission. Although there were many steps taken to ensure the navigation accuracy and robustness, this final contingency was prepared to guarantee success. The first opportunity, labeled TCM-5A, occurred at Encounter-24 hours, just prior to the safe-mode inhibit at Encounter-22 hours. If the trajectory was found to be off-nominal in the final few days, this was a relatively safe opportunity to execute a burn, with a fully functioning fault protection system active on the spacecraft. A second opportunity, labeled TCM-5B, was scheduled at Encounter – 6.5 hours. At this time the spacecraft would be within the gravity well of Mars, and the Doppler data at that time would indicate almost conclusively the arrival altitude. At this point the system fault protection was already turned off to guarantee that the Mars Orbit Insertion burn would execute at the proper time. This time was chosen as the latest reasonable time that a maneuver could be executed prior to the initiation of the Mars Orbit Insertion events.

The primary goal of a TCM-5 maneuver was to raise the orbit altitude to ensure orbiter health and safety. An off-nominal inclination or arrival time, while potentially expensive from a propellant standpoint, would not jeopardize the safety of the spacecraft. A high arrival altitude would reduce the efficiency of the MOI burn somewhat, but would pose no risk to the spacecraft. A series of

TCM-5 maneuvers were pre-built to raise the trajectory altitude in the event that it was determined to be too low.

The TCM-5 strategy specifically stated that these maneuvers would only be executed to significantly raise the altitude to ensure Orbiter health and safety. They would not be used to correct inclination errors or fine-tune the trajectory targeting. A single attitude was designed that would accommodate all TCM-5 opportunities and raise the altitude. Seven maneuver magnitudes were chosen for each opportunity to raise the altitude anywhere from 50 km to 350 km in 50 km increments.

In the final weeks prior to encounter, the altitude and the trajectory uncertainty were closely monitored by the project Navigation Team and the Navigation Advisory Group (NAG), a peer review team chartered to independently monitor the Odyssey navigation performance. At the TCM-5 decision point, the Navigation Team in conjunction with the NAG selected the *Approved* solution, which was the best consensus trajectory determination solution. Analysis of the Climate Orbiter history discovered that the formal trajectory uncertainties did not sufficiently represent the true trajectory errors. So although steps were taken to ensure the accuracy of the Odyssey Navigation process, an artificially large uncertainty of 50 km was applied to the trajectory estimate. The NAG held the option to increase this uncertainty, should the trajectory solution history warrant. By this method, a *Minimum* trajectory solution was defined to be 50 km (or more) below the best *Approved* trajectory estimate. This would help to ensure that the actual trajectory would be no lower than this value.

Specific criteria were developed to govern the execution of these contingency maneuvers. TCM-5 would only be executed in the event that the *Minimum* trajectory determination solution indicates a periapsis altitude of less than 200 km. In this event, a TCM-5 maneuver magnitude would be selected to bring the *Minimum* solution altitude above 250 km and bring the *Approved* solution altitude above 300 km.

Figure 9 presents a plot versus time of the altitude estimates (in red), along with the formal 3σ uncertainty associated with the trajectory estimates. Also shown is the *Minimum* solution (in green), 50 km below the best trajectory estimate. The *Minimum* solution never fell below the 200 km criteria, so TCM-5 was not performed at either opportunity.

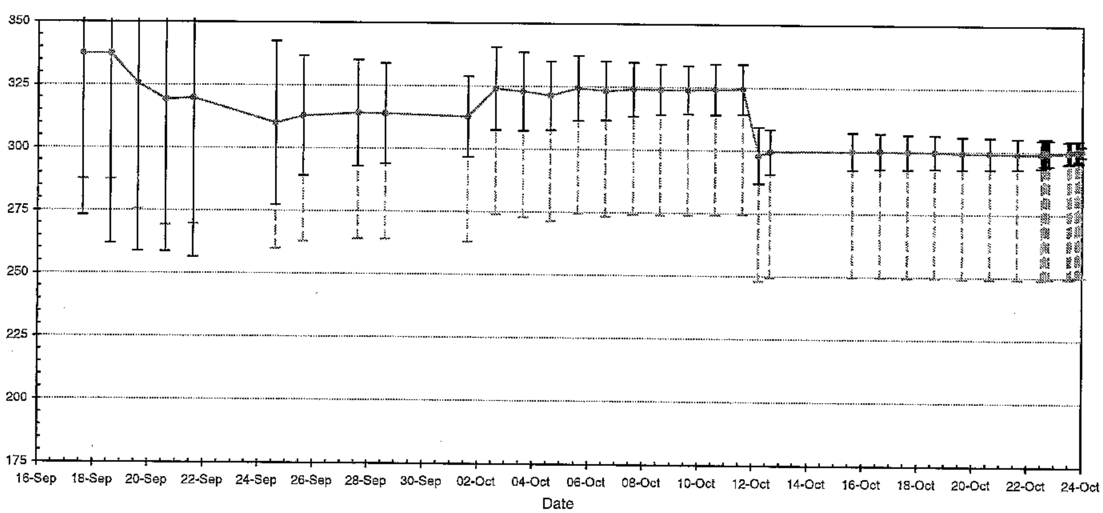

Figure 9: Odyssey Altitude Estimates and TCM-5 criteria

Final Delivery Accuracy

The encounter delivery was ultimately determined by TCM-4, the final maneuver. Figure 8 presented the final reconstructed solution compared to the target and encounter corridor constraints. These data are also presented in Table 8 below.

Parameter	Target	Corridor Requirement	Achieved	Delta
Altitude	300.0 km	± 25 km	300.7 km	0.7 km
Inclination	93.47°	± 0.2°	93.51°	0.04°
Periapsis Time (ET)	02:29:58	± 10 sec	02:29:58	< 1 sec

Table 8: Navigation Delivery Accuracy

Propellant Expenditure

The ΔV required for the cruise phase turned out to be substantially less than planned, again due to the positive launch vehicle injection results. Table 9, below, presents the planned (99%) and actual ΔV and fuel usage associated with each maneuver, as well as the angle from the Medium-Gain Antenna (MGA) boresight to the Earth.

Maneuver	Planned ΔV (99%)	Actual ΔV	Fuel	MGA Off-Earth Angle
TCM-1	48 m/s	3.6 m/s	1.2 kg	7°
TCM-2	38 m/s	0.9 m/s	0.3 kg	14°
TCM-3	3.5 m/s	0.45 m/s	0.15 kg	2°
TCM-4	0.5 m/s	0.08 m/s	0.04 kg	20°
Cruise Total	52.9 m/s	5.0 m/s	1.7 kg	

Table 9: TCM Propellant Expenditure

The total propellant budget for the cruise phase was dominated by the anticipated maneuvers, but it also included fuel allocations for momentum management, as well as for specific spacecraft events and contingency. Of the total cruise allocation of 21.9 kg, only 3 kg were actually spent. This windfall was recognized shortly after launch, and the unspent fuel was termed *strategic propellant*, as the project then had the option to utilize this propellant to mitigate risk and potentially extend the mission.

CONCLUSIONS

The Odyssey project has successfully returned an Orbiter to Mars and is well on its way to achieving all of the planned science objectives. The launch vehicle performance was fortuitous, and with that in conjunction with excellent Navigation and spacecraft performance during the cruise phase, the project realized a propellant savings of almost 19 kg. The Navigation strategy and analysis developed pre-launch held up well during operations. The result was an accurate delivery to Mars, easily meeting the targeting requirements.

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References

1. R. Mase, et al, *The Mars Odyssey Navigation Experience*, AIAA-2002-4530, AIAA/AAS Astrodynamics Specialist Conference, Monterey, California, 5-8 Aug 2002

2. P. Antreasian, et al, *2001 Mars Odyssey Orbit Determination During Interplanetary Cruise*, AIAA-2002-4531, AIAA/AAS Astrodynamics Specialist Conference, Monterey, California, 5-8 Aug 2002