Orbit Determination for the 2007 Mars Phoenix Lander

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The Phoenix mission is designed to study the arctic region of Mars. To achieve this goal, the spacecraft must be delivered to a narrow corridor at the top of the Martian atmosphere, which is approximately 20 km wide. This paper will discuss the details of the Phoenix orbit determination process and the effort to reduce errors below the level necessary to achieve successful atmospheric entry at Mars. Emphasis will be placed on properly modeling forces that perturb the spacecraft trajectory and the errors and uncertainties associated with those forces. Orbit determination covariance analysis strongly influenced mission operations scenarios, which were chosen to minimize errors and associated uncertainties.

I. Introduction

Phoenix was the first mission of the NASA Mars Scout Program. Its purpose was to study the history of water and the habitability potential of the soil in the Martian arctic region. To achieve these goals, the spacecraft needed to enter the atmosphere of Mars in a narrow corridor that allowed the vehicle to survive entry, control its descent, and soft land in an area that was scientifically interesting while minimizing exposure to surface hazards. The target entry corridor was approximately 20 km wide, which corresponds to a three-sigma flight path angle uncertainty of 0.21 degrees.

It was the responsibility of the Phoenix Orbit Determination Team to determine the spacecraft state and predict its future trajectory and characterize the uncertainty associated with those estimates. This paper will discuss the details of the Phoenix orbit determination effort and emphasis will be placed on the modeling of the forces that perturb the spacecraft trajectory and the errors and uncertainty associated with those forces. The dominant error source for this mission arose from the attitude control system, followed by maneuver implementation and execution, and then solar pressure. A number of techniques were employed to mitigate these errors and their associated uncertainties. These include the use of interferometric tracking data types, in-flight calibration of thruster performance, and long term trending of small forces behavior in an environment that matched the terminal approach phase of the mission. The spacecraft orientation and deadbanding profiles, as well as maneuver placement and slew types, were chosen to reduce errors and their associated uncertainties, consistent with the goal of meeting the Mars entry requirements.

A. Overview and Nominal Mission Plan

The Phoenix Project was designed with the objective of delivering a Lander to a high-latitude landing site on Mars. During the baseline surface mission, the Lander's instruments will perform in-situ and remote sensing investigations that will characterize the chemistry of the materials at the local surface, subsurface, and atmosphere, and will identify potential provenance of key indicator elements of significance to the biological potential of Mars, including potential organics within any accessible water ice.

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Phoenix was launched from Kennedy Space Center on August 4, 2007. After traveling along a type II trajectory, Phoenix arrived at Mars on May 25, 2008. During the approximately 10-month interplanetary transfer, which includes the Cruise and Approach mission phases, four of six planned Trajectory Correction Maneuvers (TCMs) were used to deliver the flight system to the specified Mars atmospheric entry aim-point. The Phoenix interplanetary cruise trajectory is shown in Figure 1. During the Cruise phase, navigation, telecommunications, and science instrument checkouts were performed. The Entry, Descent and Landing (EDL) phase began three hours before the atmospheric entry interface point, which is defined to be at a Mars radius of 3522.2 km. Telecommunications during the interplanetary phase was accomplished via a Low Gain Antenna (LGA) and a Medium Gain Antenna (MGA) using an X-band link.

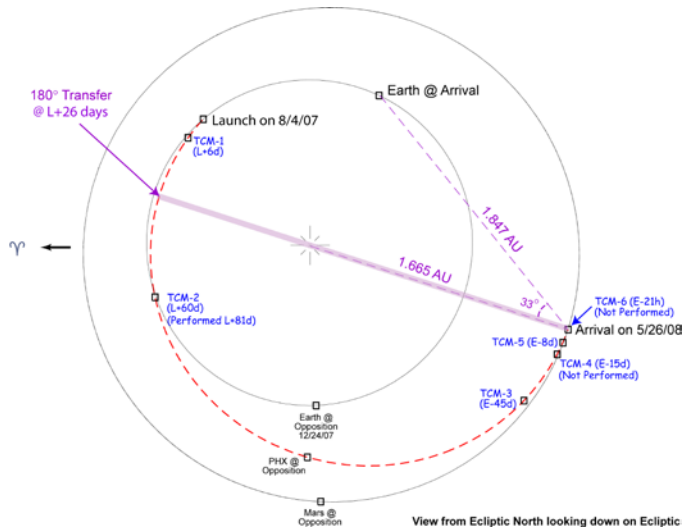

Figure 1. Phoenix Interplanetary Cruise Trajectory

The EDL system design was adapted from the concept developed for Mars '01 Lander baseline: a parachute to slow descent through the Martian atmosphere, and then the terminal descent engines to place the Lander on the Martian surface. Once the Lander sensed the touchdown event, the system began to deploy its solar arrays, robot digging arm, camera and meteorological instruments. The entry vehicle communicated with Mars orbiters during EDL via a UHF antenna mounted on the backshell. After successfully landing at the "Green Valley" landing site, located near 65° deg North and 234° deg East, the Phoenix Lander began its 90-Sol (93.5-Earth-day) primary Surface Operations phase.

B. Spacecraft Description

The Phoenix flight system is almost entirely made up of existing Mars Surveyor Program 2001 (MSP'01) spacecraft hardware. The MSP'01 design is adapted from the Mars Polar Lander (MPL) spacecraft design. The Phoenix flight system consists of a Cruise Stage, an Entry, Descent and Landing (EDL) system (including heat shield, backshell, parachute, and terminal descent engines), and a Lander, which is enclosed inside the entry vehicle. The launch mass allocation for the flight system was 705 kg. The spacecraft in cruise configuration is shown in Figure 2.

The cruise stage includes solar panels, attitude control sensors (sun sensor, star tracker), telecom antennas (LGA and MGA), and two X-band transponders. Both antennas are oriented generally in the spacecraft -X direction. The MGA boresight is in the X-Z plane offset 33 degrees from the -X and - Z axes in order to maintain optimal telecom performance in the nominal cruise attitude. The cruise stage is separated from the entry vehicle assembly approximately 5 min prior to Entry.

During the interplanetary transfer to Mars, the Lander structure (containing the payload) is enclosed by the backshell/heatshield assembly, which is also referred to as the aeroshell. The aeroshell protects the Lander from extreme heat loads experienced during atmospheric entry. The heatshield thermal protection system dissipates energy as the spacecraft enters the Martian atmosphere. The backshell includes the parachute canister used to slow the Lander prior to terminal descent. The terminal descent system, located on the Lander, is used to control the Lander for touchdown onto the surface. The backshell also includes 3 UHF patch antennas for communications during EDL.

The propulsion system, which includes two propellant tanks, four RCS thrusters, four TCM thrusters and plumbing is located on the entry vehicle and is used during cruise and the hypersonic phase of EDL. The Lander also includes the science instruments, solar arrays, batteries, and a UHF antenna for the relay link to the Mars orbiters. All of the electronics that perform spacecraft functions (including during interplanetary cruise) are contained on the Lander.

Phoenix is a three-axis-stabilized spacecraft with four RCS thrusters. The Phoenix propulsion system includes the hardware needed to perform attitude control, TCMs, and terminal descent control during Cruise, Approach and EDL. The propulsion system hardware, inside the entry vehicle, is a monopropellant hydrazine system that operates in a blowdown mode.

Cruise phase attitude control and TCMs are performed using four smaller reaction control system (RCS) thrusters and four larger TCM thrusters. The hydrazine propellant is contained in two propellant tanks that are pressurized with Helium. There are four rocket engine modules (REMs) that are used during the Cruise phase. Each of these REMs consists of an RCS thruster and a TCM thruster. Both the RCS and TCM thrusters are mounted on the Lander and extend through the backshell.

The Phoenix telecommunications subsystem uses X-band for direct-to-Earth (DTE) communications during the cruise phase. A UHF system is used during EDL and surface phases of the mission for relay communications through the Mars orbiters. The X-band telecommunications system design is dual-string coherent X-Band Uplink/X-Band Downlink with electronics located on the cruise stage. The same X-band electronics are used from launch through cruise, but two different X-band antennas, an LGA and MGA, are required. The UHF equipment is also in the Lander and is used with two different antenna sets: a UHF antenna on the backshell for communicating during EDL and a UHF antenna on the Lander for surface operations.

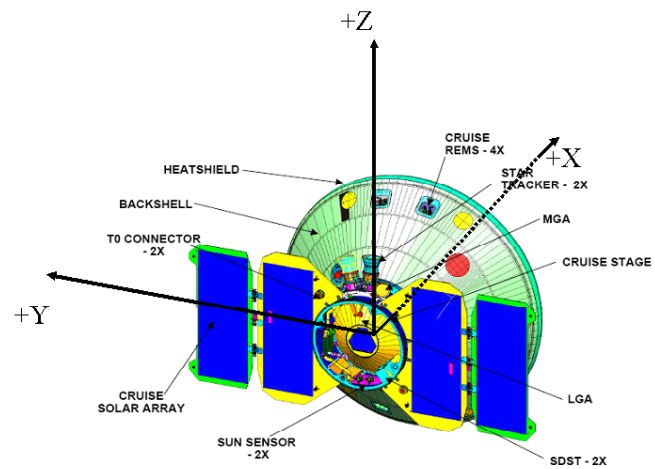

Figure 2. Spacecraft in Cruise Configuration

The X-band telecommunications equipment includes two Small Deep Space Transponders (SDST) and two Solid State Power Amplifiers (SSPAs). The SDST includes a Command Detector Unit (CDU) and a Telemetry Modulation Unit (TMU). The heart of the X-Band telecommunications systems is the SDST, which supports phase coherent turn-around Doppler and ranging, command signal demodulation and detection, telemetry coding and modulation, and DOR tone generation (+/- 19 MHz).

The Lander will employ six science instruments and a robotic arm to record data about the landing sites and selected rock and soil targets. The science payload consists of three imaging instruments, two instruments for in-situ observations, and a meteorological station.

II. Operational Considerations

A. Attitude Control and Small Forces

The cruise attitude strategy is to maintain the -X axis pointed between the direction to the Earth and direction to the Sun through November 5, 2007. Following that, the spacecraft –X axis is pointed at the Sun for the remainder of Cruise and Approach. This strategy allows a telecom link to Earth using the LGA antenna through January 8, 2008, while providing sufficient power for spacecraft operations. After that, telecommunications is through the MGA, which is generally pointed in the Earth direction in this orientation.

Since the Phoenix spacecraft is three-axis stabilized using thrusters but no reaction wheels, its attitude is not fixed. The attitude will vary within a set of deadbanding constraints defined by spacecraft telecom, power and thermal subsystems. The ACS will command the thrusters to fire each time the attitude reaches one side of the deadband. The deadbanding strategy varies during cruise based on the constraints, the Sun-Earth-probe (SEP) angle, and the spacecraft range to the Sun and Earth. The tighter the deadbands, the more thrusting is needed to keep the attitude inside the constraints, which imparts more ΔV and uncertainty into the trajectory. A plot of the pre-launch deadband profile is shown in Figure 3.

A number of factors complicate the total ΔV imparted by the attitude control system while maintaining the spacecraft attitude. Both the RCS and TCM thrusters are mounted on the Lander and extend through the backshell. Each thruster is scarfed to the contour of the backshell. Due to the scarfed thrusters the thrust direction is offset from the nozzle direction. Additionally the thrust can vary due to "dribble volume" effects (non-steady-state propellant flow for short pulses) and other random effects (such as valve variations). Each RCS thruster has a component of thrust in all three spacecraft axes. For pure 2-sided deadbanding, the thrust directions are designed to be balanced in the Y- and Z- axes but not in the X-axis. Therefore every time an RCS thruster is fired there is a ΔV imparted to the spacecraft in the X direction. Although the Y and Z directions are designed to be balanced, the non-determinism of attitude maintenance (deadbanding) may cause one thruster to be fired more than another, which causes an overall

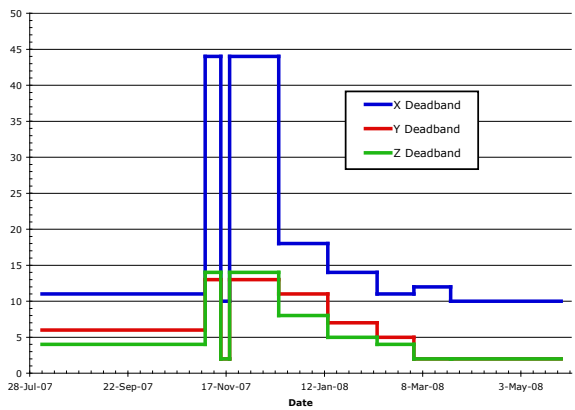

Figure 3. Pre-Launch Deadband Profile

imbalance that imparts ΔV in the Y and Z directions. Additional ΔV is imparted to the spacecraft in those directions due to thruster misalignments and thruster-to-thruster thrust variations. The thrust values for each thruster vary depending on the inlet pressure and duty cycle.

The magnitude of the ΔV from and individual RCS pulse-pair firing is quite small – on the order of 0.05 mm/s. But the cumulative effect of attitude maintenance activity over long time periods can be substantial. It is therefore important to model the ΔV imparted to the system in the orbit determination process in order to meet the delivery accuracy requirements for atmospheric entry. For this reason the flight system records a telemetry packet with thruster information every time a thruster pulse is fired. That telemetry is downlinked and transformed into a text file known as the Small Force File (SFF), which is directly used in the orbit determination and trajectory propagation process. The SFF contains information such as pulse time, pulse length, thruster number, estimated ΔV (based on a fixed model), and attitude at the time of the pulse.

B. Solar Pressure

The solar pressure model, which was based on analysis of the Mars Polar Lander (MPL) and Mars Exploration Rover (MER) missions, consists of five components representing the solar arrays, the launch vehicle adapter on the cruise stage, the cruise stage outer ring, and the backshell (2 components). Each component has a mean specular and diffuse coefficient of reflectivity associated with it. The dimensions and geometry of the solar pressure components have been updated appropriately for the Phoenix spacecraft design. The spacecraft attitude during inner and outer cruise is shown in Figure 4.

C. TCM and Associated Slew Errors

In order to achieve a successful landing on Mars, the Phoenix spacecraft must be delivered to the proper Mars atmospheric entry aimpoint by a series of trajectory correction maneuvers (TCMs). A total of six TCMs are planned during interplanetary cruise. These maneuvers are required to compensate for launch vehicle injection errors and subsequent maneuver execution and orbit determination errors.

For each maneuver, the magnitude and direction of the velocity change required to correct for errors in the desired Mars arrival conditions must be computed. These quantities are determined from an estimate of the actual arrival conditions obtained through the orbit determination process outlined below. In addition, a means of estimating the statistics of the residual guidance errors due to imperfect maneuver execution is needed. These statistics are derived from estimates of the maneuver execution accuracy and the orbit determination error statistics computed as part of the orbit determination process.

The Phoenix spacecraft performs TCMs in a turn and burn mode. The four TCM thrusters are collocated with the RCS thrusters, which are evenly distributed around the spacecraft backshell. Each thruster is aligned to be nearly

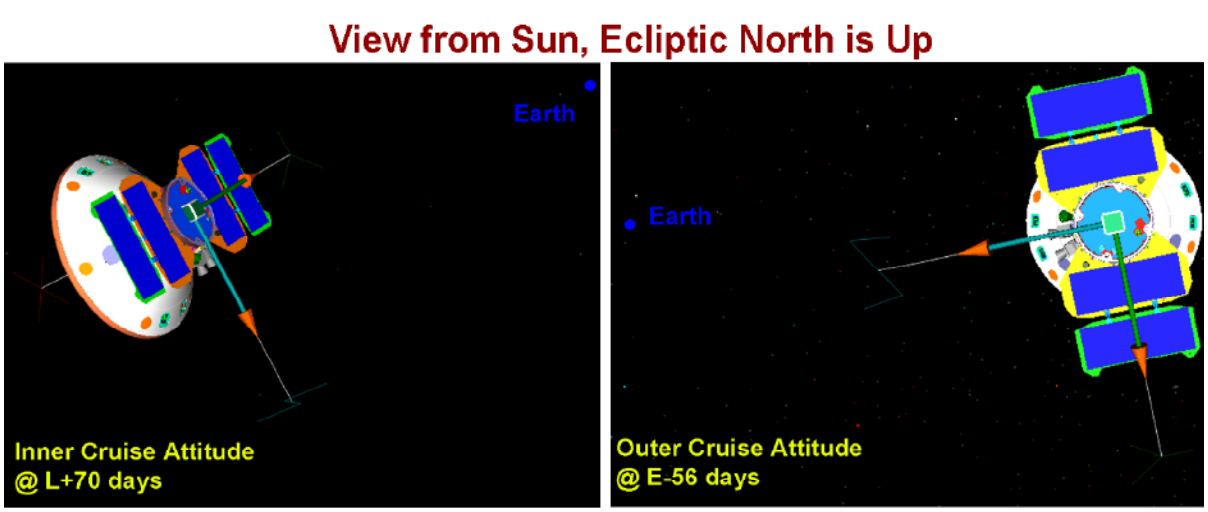

Figure 4. Spacecraft Inner and Outer Cruise Attitude

parallel to the spacecraft -X axis and the solar array normal, pointing in the direction of the cruise stage from the backshell. In order for a TCM to take place along a desired ΔV direction the spacecraft must turn to align the spacecraft -X axis with the desired thrust direction (negative ΔV direction). The TCM is executed by pulsing the TCM thrusters until the desired ΔV is accomplished as measured with the Inertial Measurement Unit (IMU). The spacecraft is three-axis stabilized during the burn using off-pulsing of the TCM thrusters in the Y- and Z-axes, and the RCS thrusters in the X-axis. After execution the spacecraft performs a slew back to the nominal pre-TCM attitude.

The accuracy with which a given maneuver can be executed is a function of the propulsion system behavior and the attitude control system, which maintains the pointing of the spacecraft during thruster firings[1]. Maneuver execution errors are described in terms of components that are proportional to the commanded ΔV magnitude and components that are independent of ΔV magnitude. The maneuver execution error is described as a function of ΔV in Table 1.

Table 1. TCM Error Model

Delta V Magnitude (m/s)	Fixed Magnitude Error (m/s)	Proportional Magnitude Error (%)	Fixed Pointing Error, per axis (m/s)	Proportional Pointing Error, Total (%)
0.04 (minimum burn) <dV < 0.3	0.02 m/s	± 2 %	0.003	± 2 %
0.3 < dV <1.5	0.02 m/s	± 2 %	0.003	± (8/1.2 * |dV|) %
1.5 < dV < 5	0.02 m/s	± 2 %	0.003	± 10 %
5 < dV < 20	*	± 2 %	0.003	± ((- 8/15) * |dV| + 12.667) %
20 <<dV	*	± 2 %	0.003	± 2 %

Slews consist of turns to and from the TCM ΔV attitude. Slews are performed by the RCS thrusters and are implemented as a pulse train designed to achieve a specific turn rate. This thruster activity imparts a ΔV that is predicted and then accounted for in the maneuver design process. The error associated with the slew ΔV prediction could be significant for a maneuver smaller than 0.5 m/s. These errors arise from a number of factors including slew rate, slew size (angle of the slew), execution method and initial attitude with respect to the deadband. The type of slew also depended on spacecraft issues such as telecommunications and thermal constraints (limits to the amount of time spent off-pointed from the Sun). In the approach phase, slews implementation was chosen to minimize the aggregate uncertainty arising from orbit determination, maneuver execution and slew errors[2]

Measurements and Filter Strategy

A. Radiometric Tracking Data

The baseline radiometric data types that were used for Phoenix orbit determination are two-way coherent Doppler, two-way ranging, and Delta Very Long Baseline Interferometry (Delta VLBI) measurements generated by the DSN X-band tracking system or a spacecraft-to-spacecraft UHF system. The first two data types are derived from a coherent radio link between the spacecraft and a receiver at a DSN ground station. Delta VLBI measurements are acquired through the DSN in the form of Delta Differential One-way Range (DDOR) measurements. Spacecraft-to-spacecraft two-way coherent UHF Doppler measurements are generated by a link between the Phoenix spacecraft and a Mars orbiting spacecraft during the surface phase. The schedule of Doppler and range data acquisition Phoenix is shown in Table 2.

Doppler data yields a measurement of line-of-sight spacecraft range rate. During tracking passes in the two-way coherent mode of operation, the DSN tracking system measures Doppler shift by accumulating the cycles of the downlink carrier signal in order to determine the difference between the transmitted and received frequencies. The lower limit for the error of this Doppler observable, using the current DSN system, is established by thermal noise. At typical carrier signal strengths, the thermal noise accounts for about 1.0 mHz for a 10 minute count time, which is equivalent to an accuracy of approximately 0.02 mm/s for X-band (where 1.0 mm/s is equivalent to 56.3 mHz). In practice, Doppler measurements are affected by such things as station elevation, dynamic modeling, ground station location, Earth orientation, ionospheric, tropospheric and interplanetary media (e.g., solar plasma) delays, and spacecraft- and ground-station-related systematic biases. Consequently, the nominal two-way Doppler data accuracy achievable for Phoenix is 0.10 mm/s.

The DSN ranging system constructs an estimate of the range to the spacecraft by measuring the round-trip light time of a radio signal between the ground station and spacecraft. By transmitting range codes with known periods and frequency spectra to the spacecraft, the Sequential Ranging Assembly (SRA) at the ground station determines spacecraft range by using an a priori estimate of the spacecraft range and by correlating the received code with the transmitted code, adjusting for the Doppler shift.

Table 2. DSN Doppler and Range Tracking Schedule

From	To	Type of Support
Launch	Launch + 14 days	Continuous coverage
Launch + 15 days	Entry - 60 days	Three 8 hour tracks/week
TCM - 4 days	TCM + 4 days	Continuous coverage
Entry - 60 days	Entry	Continuous coverage

Range performance correlates with signal strength, and signal strength is measured in terms of the power to noise ratio (referred to as Pt/No). That is, more accurate range measurements result from a strong signal strength and low measurement noise. A tradeoff exists between range accuracy and spacecraft telemetry rate, because adding range modulation to the downlink decreases power available to telemetry, thereby reducing telemetry bandwidth.

The Phoenix telecom system is capable of a ranging noise level of 14 ns (2 m) for a 10-minute integration time at the maximum Earth range for Phoenix during the Cruise phase. To meet this requirement, the ranging measurement must account for un-modeled signal delays. Some representative delays are those occurring within the spacecraft and ground station electronic equipment and delays due to media (e.g., ionosphere, troposphere, and interplanetary media).

B. Delta Differential One-Way Range Tracking Data

The Delta VLBI measurement used for Phoenix is Delta Differential One-way Range (DDOR). DDOR acquire interferometric observations of a spacecraft using pairs of DSN stations. Two DSN stations simultaneously observe a spacecraft followed by simultaneous observations of a reference radio source. These observations measure angular separation between the spacecraft and the reference source. For DDOR, the reference is an extra-galactic radio source. A brief overview of DDOR is presented here (REFERENCE?).

The reference source for DDOR is an extra-galactic radio source (quasar). Quasars are used becuase these sources, being extremely distant, are essentially fixed in inertial space and therefore establish an excellent coordinate system for DDOR. This coordinate system is known as the radio frame. Many quasars (several hundred) have been observed by the DSN and their positions relative to the radio frame have been accurately catalogued. It is the existence of this catalog that makes DDOR feasible.

Table 3. Phoenix DDOR Tracking Schedule

Start Time	End Time	Measurement Frequency
Launch + 36 days	Launch + 42 days	1 measurement/day
Launch + 64 days	Entry - 60 days	1 measurement/week
Entry - 60 days	Entry - 18 days	3 measurements/week
Entry - 18 days	Entry	2- 3 measurements/day (DSN and ESA DDOR)

The DDOR observable is a phase delay time expressed in units of nanoseconds (ns) that is equivalent to an angular separation parallel to the baseline between to two tracking stations. For the DSN, a delay of 1 ns corresponds to about 37.5 nanoradians (nrad) of angular displacement. Knowing the quasar angular position determines the spacecraft's position in plane-of-the-sky. DDOR complements Doppler and range measurements because of its orthogonality to those data types. DDOR measurements combined with range data do not rely on dynamic models to infer all three components of position (unlike Doppler and range alone).

Phoenix DDOR observations are obtained using pairs of DSN complexes. The common pairings are Goldstone-Madrid and Goldstone-Canberra. The Goldstone-Madrid baseline (oriented East-West) primarily measures the right ascension component of the spacecraft. By similar reasoning, the Goldstone-Canberra baseline (oriented Northeast-Southwest) measures the declination component of the spacecraft, along with right ascension. An observation is possible physically when a view period overlap exists between respective stations on a baseline - i.e., when both

stations can view the spacecraft simultaneously. In addition, the spacecraft must transmit a signal with DOR tones activated (sidebands at ¬±19 MHz).

For Phoenix, the DDOR observation consists of an initial spacecraft scan, followed multiple quasar-spacecraft-quasar (QSQ) scan sequences. The initial spacecraft scan lasts 4 minutes and each QSQ sequence takes 30 minutes (9 minute scans separated by a 1 minute slew). A 30-minute pre-observation antenna calibration is also required, which typically takes place prior to the start of the DSN track. Phoenix employed two QSQ sequences in the early part of cruise and three QSQ sequences beginning in late January 2008. Different quasars are used for each QSQ sequence when multiple quasars are close to the spacecraft position in the sky.

For the navigation analysis reported in this document, the DDOR observable acquired at the DSN is assigned a metric data accuracy of 2.5 nrad, and the quasar angular positions are assigned an uncertainty of 1 nrad. The basis for these uncertainties is the experience to date with DDOR performance for Mars orbiters and MER.

A third possible DSN baseline, Madrid-Canberra, represents the longest baseline for the DSN but is rarely scheduled because of the brief overlapping view periods. A similar measurement is observable from stations that are part of the European Space Agency's (ESA) tracking network. NASA and ESA entered an agreement in which Phoenix DDOR observations would be acquired using ESA's New Norcia-Cebreros baseline (Western Australia-Spain). This data was then delivered to JPL and analyzed as part of the Phoenix orbit determination process, but the data was not included in official deliveries. The ESA DDOR observations used two QSQ sequences and were assigned a metric data accuracy of 6.0 nrad due to a narrower recording bandwidth than the DSN.

The schedule of DDOR measurements is shown in Table 3. It is worth noting that three DDOR measurements were acquired each day in the final week prior to Entry. Effectively one-sixth of the time available for contact with the spacecraft in the run up to its most critical event was allocated to DDOR acquisition.

C. Tracking Data Calibrations

Several calibrations are applied to the Doppler and ranging data in order to remove the electronic and transmission media effects mentioned above. The calibrations used in the orbit determination process can be divided into the three categories described below.

The signal delay through DSN ground stations for the ranging system requires calibration. The ground station calibration can be performed to a typical accuracy of about 1 m. The delay through the ranging channel in the spacecraft transponder can be calibrated to an accuracy of 2 m when a constant thermal environment is maintained.

Transmission media include the Earth's troposphere and ionosphere, and the interplanetary media. At each DSN complex, calibrations of the signal delay induced by the local troposphere and ionosphere are estimated from GPS data by the DSN's Tracking System Analytic Calibration (TSAC) Team. These calibrations are expressed as zenith delays for troposphere, and line-of-sight delays for ionosphere. The orbit determination software uses analytical and tabular functions to map the zenith troposphere delays into the actual line-of-sight delays. At present, troposphere delay calibrations (wet and dry) are accurate to 1 cm. The ionospheric delay is calibrated to an accuracy of 55 cm (day) and 15 cm (night) for an S-band signal at zenith. The magnitude of the interplanetary media delay is less than 1 m, is common to all DSN complexes, and generally behaves as a (slowly-changing) bias, so that no external calibrations of this error source were planned. MER encountered interplanetary media effects that acted very similar to atmosphere effects. These varying delays were seen by many spacecraft, which confirmed the interplanetary media cause. The interplanetary media delay was then accounted for with an approximate stochastic model in the orbit determination process that removed the signatures in the data but did not affect the overall solution much.

Earth platform calibrations, also referred to as Earth Orientation Parameters (EOP), are not applied directly to tracking data, but are used within the orbit determination software system to compute transformations of the ground station coordinates used in reducing tracking data. These calibrations include DSN station locations, Earth precession and nutation models, corrections to the shorter-period motion of the Earth's pole, and corrections to variations in UT1. Recent determinations indicate that the DSN station locations are accurate to 3 cm. The precession and nutation models are based on those established by the International Astronomical Union in 1980. Polar motion and timing (UT1) calibrations are developed by the DSN from interferometric observations of extra-galactic radio sources.

The Earth orientation polar motion and UT1 predictions are guaranteed by the DSN to default 1-sigma accuracies of 30 cm and 0.77 ms (~30 cm) for a 1-day prediction, but typical performance is much better. For the Phoenix mission, it has been assumed that the polar motion uncertainties will be in the range of 1-4 cm, and UT1 uncertainties will be in the range of 2-10 cm, depending on the length of the prediction.

D. Filter Strategy and Data Weights

Orbit determination processing is accomplished with a multiple batch consider-parameter filter, incorporating a baseline dataset consisting of two-way coherent Doppler, two-way coherent ranging data, and DDOR measurements. The filter setup for late cruise is shown in the Appendix.

All TCMs contained within the data arc are estimated. Future TCMs (i.e., with respect to a given data cutoff time) are treated in one of two ways. For generating entry delivery uncertainties, the TCM directly after the data cutoff time is considered in the filter at the a priori uncertainty, while any other future TCMs are ignored. For generating orbit determination covariance matrices for maneuver analyses, all future TCMs are ignored, and maneuver execution errors are modeled in the maneuver analysis process.

Spacecraft ACS ΔVs (e.g., spacecraft attitude maintenance) are estimated in the OD filter when these events fall within the data arc. The ΔV from ACS is modeled as a three-axis stochastic scale factor and a three-component stochastic acceleration with zero mean. The scale factor models the short-term randomness of individual thruster firings. The acceleration models the uncertainty associated with a long-term total thrust offset that tends to bias the trajectory. A second, similar stochastic acceleration was used the last two weeks before entry to models the uncertainty associated with short-term randomness of future thruster firings. Additional stochastically estimated parameters include range data biases, and attitude maintenance accelerations. The data biases are estimated during each tracking pass. Moreover, dynamic model margin has been incorporated to account for non-gravitational acceleration mis-modeling.

For navigation analyses the solar pressure uncertainty is modeled using a three-axis scale factor on the total solar pressure acceleration. This strategy is believed prudent, because the alternate choice of increasing the filter's complexity by estimating all five solar pressure components does not elicit any greater insight or accuracy and is insufficient without an accurate spacecraft pointing profile.

The considered parameters consist of Earth orientation parameters, media effects, quasar locations, station locations (a correlated 9x9 error covariance), and the Earth and Mars ephemerides.

The navigation system is required to determine the location of the landing site and the position of the Lander in support of surface operations. The accuracy requirements for landing site position determination are within 30 m after 7 Sols.

The required accuracies are achievable through the use of two-way in-situ Doppler between the Lander and the Mars Odyssey orbiter. Landing site position determination solutions are provided by navigation to assist the Science Team in locating the Lander on the surface of Mars with respect to known landmarks or surface geology.

Two-way in-situ Doppler measurements, collected by the Mars Odyssey orbiter during the short (approximately <10 minute) overflights, will be used to determine the Lander position with accuracies sufficient to assist in Lander surface operations planning. The orbit determination process will incorporate a combined Mars Odyssey, Mars Reconnaissance Orbiter and Lander solution, which will utilize the two-way Doppler measurements collected by the orbiter.

E. Software

JPL has used the ODP/DPTRAJ/MOPS software set for operational support since the mid. It is written primarily in FORTRAN and its heritage dates back to earlier implementations in the early 1960s. This software has been employed to successfully navigation spacecraft to every planet and many small bodies in the solar system.

JPL Navigation management decided in the mid 1990's that new software would be more maintainable and cost effective. A complete rewrite of navigation software named MONTE has been under development for almost 10 years[3]. The software is written in C++ with a Python scripting interface. It supports all functions in the ODP including force models, trajectory propagation, residual and partial generation, filtering & mapping and maneuver design & optimization. Graphical user interface based tools are provided for residual display, data editing, multi-scenario filter/editing runs, case management and solution display & comparison. The software can be used for operations and covariance analysis. Its throughput performance is comparable to the ODP. Extensive on-line documentation is also available, including user reference, formulation, training and search features. The entire software suite is under configuration management to assure the integrity of the software.

Phoenix was the first mission to use MONTE in operations. As such, the Phoenix navigation team developed an operations environment from scratch. In addition, "matching" ODP cases had to be maintained to validate the MONTE solutions and protect against an unexpected MONTE failure during critical events. The same work was also done in the ODP and great discipline was needed to keep the two software sets synchronized. The MONTE cases were considered the baseline and tools were developed to generate ODP inputs, such as stochastic filter update controls, data edits and data weights, from MONTE inputs.

III. Navigation Results

A. Launch Support

Phoenix launch was scheduled on a Boeing Delta II 7925 launch vehicle from Cape Canaveral Air Force Station (CCAFS) Space Launch Complex 17A (SLC-17A). A 22-day launch period was planned, extending from August 3, 2007 through August 24, 2007. A four-day secondary period was added to provide margin in case launch delays by other spacecraft impacted Phoenix launch processing. There are two instantaneous launch opportunities each day. The first on a 93° azimuth and the second, about 40 minutes later, on a 99° azimuth. The injected spacecraft mass was 664.047 kg, significantly less than the pre-launch allocation.

The first day of the launch period was not used due to weather and launch vehicle fueling considerations. Phoenix launched successfully on its first attempt, on August 4, 2007 at 09:26:34 UTC on the 93° azimuth. The spacecraft transmitter was turned on at separation +17 seconds, followed immediately by rate damping and solar array deploy. The spacecraft was programmed to execute up to three "sun search slew / attitude acquisition" sequences to establish knowledge and then slew to the initial acquisition communications orientation. Depending on the separation attitude and tip-off rates, this process could take between 5 and 36 minutes. Note that the spacecraft was above the horizon at Goldstone when the separation and slew events occurred.

The signal was observed at the earliest scheduled time. The spacecraft was launched with coherency-enabled, so uplink of a command to go two-way was not needed. The initial OD solution was complete at 16:00 UTC. It was performed using four hours of two-way Doppler, range, and three-way Doppler from Goldstone, plus three hours of three-way Doppler from Madrid. All data was collected using DSN 34m tracking stations. Angle data and three-way Doppler were also acquired from the ESA tracking station at Kourou as a contingency, but that data was not used on launch day. The relatively simple filter solved for state, a large stochastic acceleration, and range and angle tracking data biases. This accommodated the need for quick analysis and delivery of an early solution. A quick-look solution was performed within an hour of initial acquisition. The first formal solution, including delivery of pointing and frequency predicts, was made one hour before the Canberra tracking station rise (six hours after the quick-look solution).

The Phoenix injection trajectory was intentionally biased to meet planetary protection requirements. The launch aim point was biased approximately 275,000 km away from a Mars impact trajectory to reduce the probability of accidental impact of the launch vehicle on Mars to below 10^{-4}. The hyperbolic injection elements were each within 1 sigma of their respective targets. The achieved B-plane target at Mars was significantly farther away from Mars than the desired aim-point target. The injection error amounted to a 969,000 km miss and an arrival four days, twenty hours late. See Figure 4.

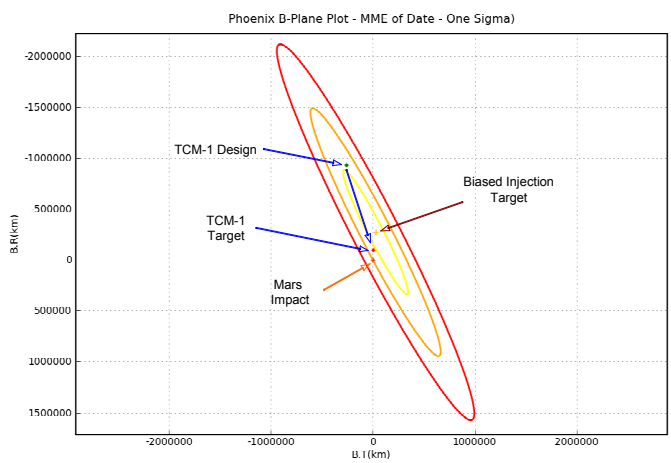

Figure 4. Injection and TCM-1 B-Plane

B. TCM-1

The purpose of TCM-1 was to correct injection errors and to remove a portion of the injection bias. TCM-1 was scheduled to execute at August 10, 2007 18:30 UTC and the tracking data cutoff was five days earlier at August 5, 23:00 UTC. The resulting short tracking data arc (only 32 hours) left little time for the orbit determination team to characterize the veracity of their models, particularly small forces and solar pressure. Following injection, telemetry indicated that the spacecraft was in excellent health. This solution represented our first opportunity to observe the velocity change imparted by the attitude control system.

About 15 thruster-pair firings occurred per hour to maintain the spacecraft within its deadbands, which was significantly higher than the pre-launch predicts of 3 thruster-pair firings per hour. The high rate was a consequence of the fact that Phoenix was still experiencing outgassing. There were a total of 784 thruster firings small forces events in the TCM-1 data arc, in the first 6 hours of the data arc.

Data for the solution included two-way Doppler and range from multiple DSN tracking stations. The filter was enhanced to include a stochastic per-axis small forces scale factor to correct errors in the magnitude of the reported small forces ΔV. A fast, uncorrelated stochastic acceleration was included to account for any out-gassing and mis-modeling in solar pressure. A decaying non-grav acceleration was observed with an initial magnitude of $2.5X10^{-10}$ km^2/sec^2. Finally, a post data arc non-grav acceleration was added to account for the predicted small forces events in the future. This model used accelerations and uncertainties corresponding to the pre-launch deadbanding profile. The planetary ephemeris DE410 was use for this phase of the mission.

Phoenix performed a turn to the inner cruise attitude on August 6, 2007 17:03 UTC, which imparted a ΔV of 1 mm/s. This was after the maneuver design solution had been delivered and represented a small error in the maneuver design. TCM-1 was 18.5 m/s and was successfully executed as planned. See Figure 4.

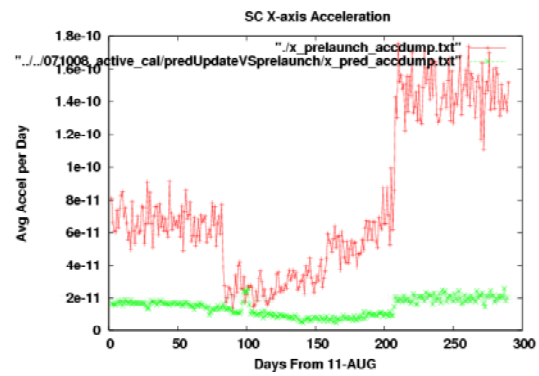

Figure 4a. Prelaunch vs Post ATC S/C X-Axis Acceleration

C. Active Thruster Calibration

The objective of the active thruster calibration was to measure the average RCS thrust vectors with an accuracy of 10% or better[2]. To achieve this objective high rate Doppler tracking and gyroscope data from the spacecraft's IMU were collected during a series of RCS thruster firings. The spacecraft attitude and thruster firing sequence were chosen to provide maximum Doppler observability of the Z-axis component of the RCS thrust vector.

The RCS thruster calibration started on 14 September 2007 17:40 UTC and lasted for 8 hrs. The primary result was that all the estimated magnitudes were 23-56% higher than the magnitude based on the preflight impulse. Furthermore, the magnitudes for RCS thrusters 3 and 4 were 13-23% higher than the magnitude for RCS thrusters 1 and 2. The estimated X and Z components of the thrust direction were larger than the pre-flight X and Z component. The increase in the X-component is consistent with the empirically derived thrust vector for the Mars Polar Lander (MPL) spacecraft, which had the same RCS configuration and spacecraft design as Phoenix. The pre-launch and post active thruster calibration predicted small forces acceleration are shown in Figure 5.

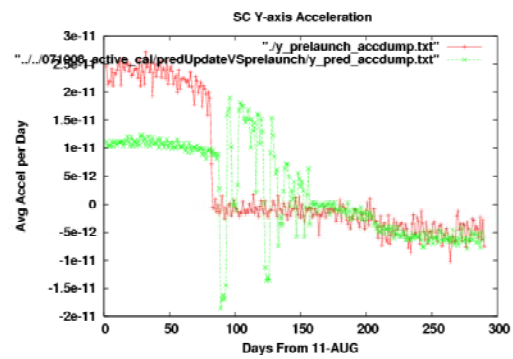

Figure 5b. S/X Y-Axis Acceleration

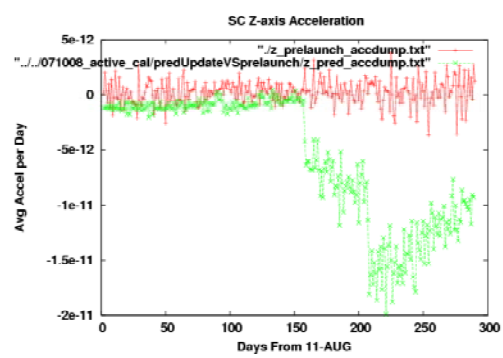

Figure 5c. S/C Z-Axis Acceleration

D. TCM-2

A number of new models were incorporated to improve the orbit determination performance for the design of TCM-2. The results of the active thruster calibration were applied in two ways. First, the acceleration representing future small forces activity was increased in accordance with the active thruster calibration values. Note that the deadbanding plan was not updated, only the magnitude and direction of the acceleration. Secondly, since the reaction control system ΔV that is reported in telemetry is computed on board Phoenix, the on board model was updated to match the active thruster calibration results.

The solar pressure model was also updated at this time. For inner cruise, the spacecraft -X-axis (normal to the solar panels) was offset 53° from sun point. This allowed communication through the low gain antenna while the range to the spacecraft was below 60 million km. The spacecraft-Sun distance was not changing significantly during this phase of the mission. This made it difficult to separate the solar pressure acceleration from the perturbations

caused by the attitude control system. Under these conditions, it was impractical to solve for the specular and diffuse reflectivity coefficients and areas of the five components used in the solar pressure model. Consequently, a bus model was used to estimate errors in the solar pressure model (likely caused by mismodeling of self shadowing in the off-point orientation).

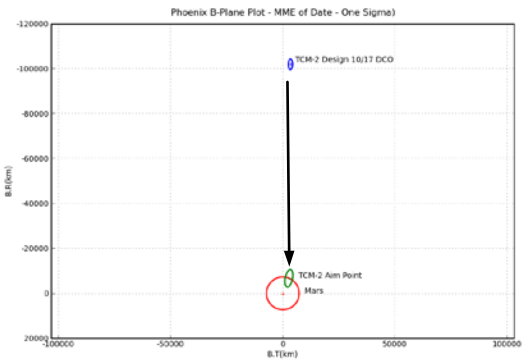

Figure 5. TCM-2 B-Plane

An additional operational complication was added when the spacecraft went into safe mode on October 6, 2007 17:57 UTC. The safe mode event imparted a ΔV of 15 mm/sec, primarily in the spacecraft X direction. The most likely cause of the safe mode event was a high-energy galactic cosmic ray hit. A "warm" reboot was performed to restore spacecraft operations, but there was a concern that the reboot process may have inadvertently written the contents of a non-qualified spare memory to a register. To protect against this possibility, a "cold" reboot was performed on October 16, 2007 16:53 UTC. This imparted a ΔV of 12 mm/s, again in the spacecraft X direction. The "cold" reboot also forced a one-week delay of TCM-2 from the scheduled time of October 17. One benefit of the delay was that additional DSN tracking passes were allocated to help analyze and recover from the safing event. As a result, six days of near-continuous tracking data were available leading into the TCM-2 data cutoff.

The purpose of TCM-2 was to remove the remainder of the injection bias and correct errors in TCM-1. The data cutoff for TCM-2 was October 17, 2007 18:00 UTC. A post-active-thruster-calibration short arc was used with 29 days of data. The maneuver design solution included six DDOR data points. The DDOR data were from three observations (one on the Goldstone-Canberra baseline and two on the Goldstone-Madrid baseline). Each observation consisted of two independent measurements. The observed noise of this data type was 0.013 ns, consistent with the 0.06 ns data weight. The short arc solution was statistically consistent with medium arc (30 days) and long arc (60 days, fitting through the active thruster calibration solutions.

TCM-2 was successfully executed on October 24, 2007 15:00 UTC with a magnitude of 3.61 m/s. Phoenix was on a Mars impact trajectory following TCM-2, and remained so for the rest of the mission. See Figure 6.

E. Passive Thruster Calibration

The objective of the passive RCS thruster calibration was to characterize the thruster activity in the late cruise attitude with the associated tight dead bands and to assess the accuracy of the active RCS thruster calibration result[2]. The original plan for the passive RCS thruster calibration was to put the spacecraft into the late cruise attitude with deadbands of 10 deg in X, 2 deg in Y and 2 deg in Z (10,2,2) for one week in order to characterize the thruster firings and deadbanding characteristics. After the calibration, the deadbands would were planned to be increased to conserve fuel. The nominal mission plan called for the (10,2,2) deadbands to be reinstated at Entry–60 days and remain in effect through the end of the mission. During the passive thruster calibration it was decided to remain in the late cruise attitude deadbands because the increase in propellant usage was small compared to propellant remaining after a relatively small TCM-1 maneuver. It was deemed more advantageous to continue trending the deadbanding in this configuration. The tight deadbands also improved the telecom link margin in outer cruise relative to the nominal mission plan. This extension enabled the estimation of a constant acceleration error in the orbit determination process, which was used to bound the error in the active thruster calibration model.

The passive RCS thruster calibration was started on 15 November 2007 00:00 UTC by tightening the dead bands to (10,2,2). A few days prior the attitude was changed to the late cruise attitude, which caused out-gassing to occur. Out-gassing caused torques resulting in an increase of thruster firings and one-sided deadbanding. The one-sided thruster firings decreased exponentially over time, which was the main symptom of out-gassing. If the passive calibration had not been extended this would have added an extra complication to the analysis.

F. Small Forces Trending

One of the most critical activities performed by the orbit determination team was the monitoring and trending of small forces activity. Records of attitude control thruster firings were accumulated on board and replayed in telemetry during DSN tracking contacts. Reporting latency during continuous tracking was on the order of 45 minutes.

The basic analysis showed the per axis acceleration, reported on the SFF, averged over a one day period versus time. The reported acceleration was compared to the current acceleration used in the orbit determination trajectory prediction (see Figure 7). This gave an immediate graphical assessment of trends in the small forces activity and the quality of the prediction. Similar data was generated for six hour and seven day averages to give better insight into long and short term variability and trends. Another important plot showed the difference in the number of daily thruster firings causing plus and minus rotations about each spacecraft axis. This is essentially a measure of the "one-sidedness" of the attitude control firings. Examination of this plot shows both the magnitude and consistency of the one-sided deadbanding (see Figure 8). The out-gassing-induced attitude control activity is clearly visible in the long-term trending plots. The most unexpected result is both the long term and short term consistency and stability in the small forces thrusting. At the time, this suggested that it should be possible to predict the spacecraft trajectory with a higher degree of accuracy than was anticipated in the pre-launch analysis.

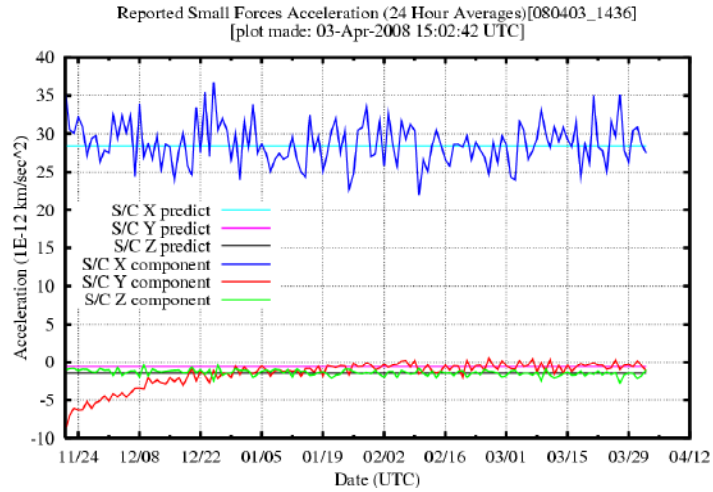

Figure 6. Small Forces Per Axis Acceleration

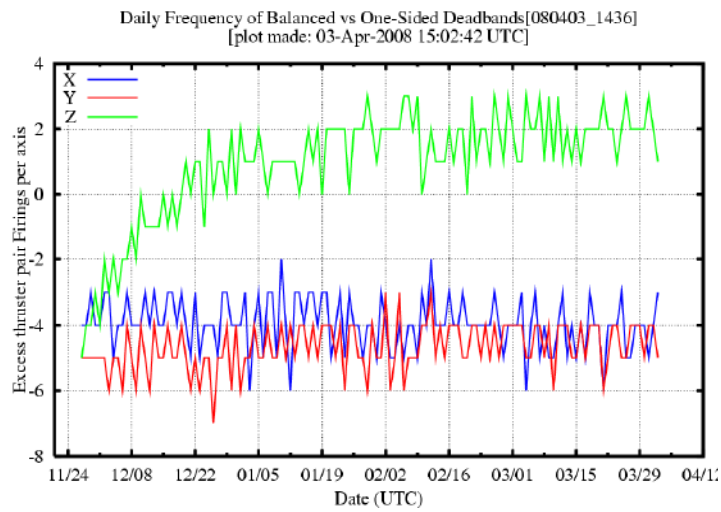

Figure 7. Small Forces One-Sidedness

G. TCM-3

The official beginning of the approach phase was on March 26, 2008. Near-continuous DSN tracking passes were scheduled and the frequency of DDOR measurements was increased to three times per week. At least three tracking passes using 70m stations were scheduled each week to assure that valid range data was acquired and to improve telecommunications link-margin. A number of factors, including the increased data density, contributed to a significant improvement to the orbit determination and trajectory prediction accuracy. The spacecraft slewed to the outer cruise attitude on November 6, 2007 17:02 UTC, with the solar panel normal pointed directly at the sun. This meant that the spacecraft shape was now symmetrical when viewed from the sun. Consequently, the solar pressure acceleration was now radial, except for a minor cross component due to re-radiation effects. The spacecraft-sun range increased from 160 million km to 240 million km between TCM-2 and TCM-3, allowing better separation of the solar pressure and small forces accelerations. A simple overall scale factor was used to estimate the magnitude of the acceleration. This estimate proved consistent enough that the solar pressure model was updated and solar pressure was removed from the list of estimated filter parameters.

The primary error source for the trajectory prediction was the small forces activity in the future. Wide deadbands were planned for this phase of the mission to conserve propellant. The short-term small forces activity was observed to be very consistent. But the large deadbands were susceptible to one-sided deadbanding. Once the one-sided deadbanding was established, the small forces activity would tend to remain in this regimen. The frequency and duration of one-sided deadbanding was not predictable. Consequently, large dispersions were needed in the small forces prediction to bound the chaotic nature of the future deadbanding activity.

Another significant finding of the passive thruster calibration was that the magnitude of small forces activity was very consistent. More importantly, the deadbanding was consistently slightly one-sided and did not switch between different one-sided regimens. The original intent was to leave the spacecraft in the passive calibration attitude for one week and then return to the pre-launch deadbanding profile. A descent engine venting took place on November

26, 2007 17:57 UTC, imparting a ΔV of 3 mm/s. After this activity, the project evaluated the possibility of leaving the spacecraft in the approach deadbands. Propellant margin was available because of the small injection error. The tight deadbands would also result in improved telecommunications link margin following the switch from the low gain to the medium gain antenna on January 8, 2008 13:48 UTC. Another consideration was that significant out-gassing occurred following the turn to outer cruise attitude, which initially corrupted the passive thruster calibration results. Remaining in the tight deadbands allowed small forces trending to begin after the out-gassing had dissipated in mid-December 2007. So the project decided to use some of the propellant margin to improve the predictability of future small forces activity by remaining in the tight deadbands for the remainder of the mission.

Use of the tighter deadbands meant that a priori uncertainty associated with the predicted small forces non-grav could also be reduced. The small forces prediction and its uncertainty were made by applying statistical analysis on the reported small forces activity. Consequently, errors in the reported small forces also manifest themselves as errors in the long-term prediction. Estimation of the error in the reported small forces and correcting the prediction became an important consideration. This was accomplished by estimating a constant per-axis non-gravitational acceleration bias in addition to a fast stochastic acceleration to account for daily variation in pulse counts. The constant bias represents the long-term error in the reported small forces. By propagating this bias to Mars, it essentially corrects the error in the telemetry-based predicted acceleration.

All the above models and refinements were in place in late February 2008. Weekly solutions were performed over this time period. A time history of these solutions showed a small downward drift in the Mars B-Plane, which amounted to a 1.5 sigma shift over six weeks of data processing. The magnitude of the drift exhibited a decaying nature, punctuated by a few larger jumps. This drift corresponded to an underestimate of the magnitude of the X-component of the small forces acceleration. To investigate the nature of this mis-modeling, each of the earlier solutions was rerun using the reported small forces from the final solution. In effect, this eliminated the contribution of the error in the small forces prediction. Mapping the resulting solutions to the B-Plane which were nested and but still exhibited a minor drift.

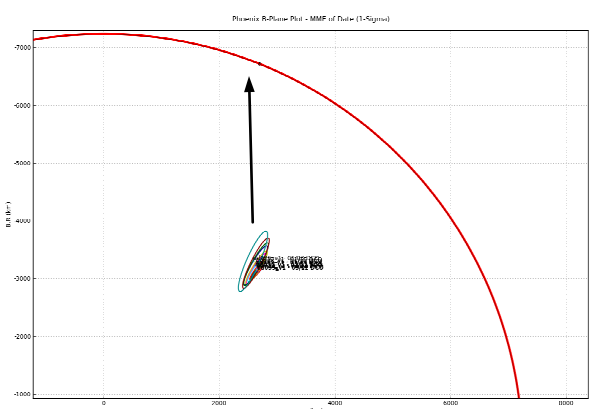

Figure 8. Pre TCM-3 Solution B-Plane History

This indicated that the majority of the drift observed in the baseline solutions was a result of day-to-day variations in the number of pulse-pair firings, not a mis-modeling of the overall small forces predicted magnitude. A small downward shift was noted in the small forces trending data leading into the TCM-3 data cutoff. At the time, it was thought that this would account for the remaining drift in the B-Plane, so the magnitude of the predicted non-gravitational acceleration was adjusted accordingly just prior to the TCM-3 design delivery.

The purpose of TCM-3 was to correct errors in TCM-2 and target the desired landing site. The data cutoff for TCM-3 was April 4, 2008, 18:00 UTC. A data arc, comprising 65 days of tracking was employed which included 45 DDOR measurements. The amount of time allocated for DDOR acquisition was increased by 15 minutes, thus allowing three independent DDOR data points per observation. The trajectory was essentially "ballistic" with no perturbations except attitude control activity (as had been the case since mid December 2007).

The project planned to perform another "cold" reboot on April 21, 2008 13:05 UTC to clear registers prior to entry. The ΔV from this activity was included in the predicted trajectory to improve TCM-3 targeting. The magnitude of the "cold" reboot could be between 0 mm/s and 15 mm/sec with a mean value of 6 mm/s, depending on the attitude of the spacecraft in its deadband control box. The 15 mm/s value was used for the magnitude of the ΔV in the predicted trajectory. Consequently, a smaller execution ΔV would result in the spacecraft being above the target in the Mars B-Plane. This would be a favorable situation if the drift continued because maneuvers performed from above the entry corridor in the Mars B-Plane resulted in smaller slew angles and correspondingly smaller slew errors. The planetary ephemeris DE421 was used for this delivery and the remainder of the mission.

TCM-3 was executed successfully on April 10, 2008 21:00 UTC. The 1.416 m/s maneuver placed Phoenix within the 0.21 degree three-sigma entry corridor. TCM-3 was performed so well that TCM-4 was cancelled. The

ΔV in order to correct the TCM-3 errors was below the minimum that the spacecraft could perform relative to the errors involved. The minimum ΔV that was executable was 0.05 m/s. TCM-4 was originally scheduled for May 11, 2008 04:00 UTC. The effect of TCM-3 is shown in Figure 9.

H. Final Approach Operations

Final approach operations began on May 5, 2008. The rate of acquisition of DSN DDOR measurements increased to two per day. In addition, twelve DDOR measurements were acquired by the ESA Cebreros and New Norcia tracking stations during this time frame.

The tempo of orbit determination support greatly increased and the full spectrum of software tools, which had been under development, were applied for the first time. A multi-function data pre-processing tool was employed to allow delivery of orbit determination products within one hour of the tracking data cutoff. This automated tool removed blunder points from the tracking data and generated per pass data weight commands for each individual pass based on its data noise characteristics. The tool also determined the time of the last delivered data calibrations, including troposphere and ionosphere for Doppler, range and DDOR data and Earth platform parameters. Note that tracking data was often used in the orbit determination solution that was after the time of the latest calibrations (due to latency in the processing of the calibrations). The tool would de-weight tracking data that was un-calibrated. This was particularly important for DDOR data because of its strong influence on the estimated trajectory. The tool would also identify instances when data cutoffs truncated tracking passes. It would then recalculate the per-pass data weight based on the full pass once the data was added to the filter. Finally, the tool would generate commands in MONTE and ODP input formats to assure that the two software sets were using identical data content and weighting. An example of the case management tool B-Plane plotting is shown in Figure 10.

Daily "orbit determination/maneuver design" cycles were performed and round-the-clock support was staffed for the three days leading into the critical designs of TCM-5, and TCM-6. Four different data arcs were maintained leading into the TCM-5 design. The arc lengths included 108, 36, 26 and 6 days respectively. Typically 25 different filter/data scenarios were examined for each tracking arc. These variations included data type, DDOR baseline, data weight, non-gravitational acceleration a priori variations, and TCM a priori cases as appropriate. One ODP solution was usually performed in each orbit determination cycle using the baseline filter scenario.

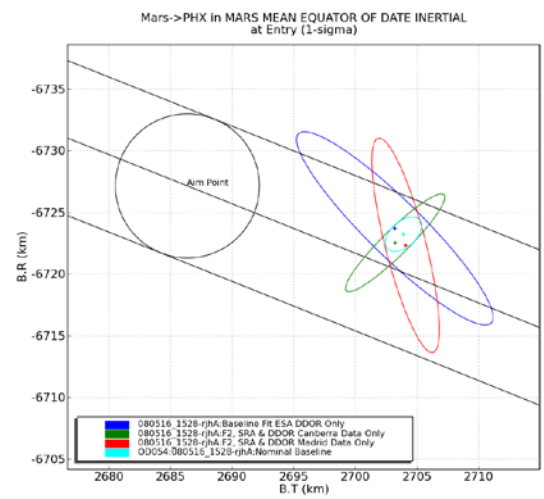

Figure 9. DDOR Data Parameterization

The results of each orbit determination cycle were loaded into a case management tool. The "cm" was a graphical user interface tool driven tool used to analyze solutions. The tool could generate a wide range of plots including B-Planes with entry corridors and target, a history of entry time, flight path angle, and estimated parameters. The "cm" could plot data residuals including pre/post fit residuals for each data type as well as data deleted from the solution. The tool had access to traditional printed output from each solution as well as the ability to compare parameter estimates and their uncertainty from multiple cases. A powerful query function used in conjunction with judicious selection of meta-keys in the orbit determination setup facilitated construction of descriptive keys names. This function was used to rapidly select and display data, using names or combinations of names such as "OD_deliveries", "16-MAY-2008 Daily Data Cutoff", "Long Arc", or "DDOR Madrid Baseline Only", etc.

Navigation status was presented and critiqued by navigation experts for each orbit determination solution and maneuver design cycle before critical TCM deliveries. The case management tool was featured prominently in these "daily shows". The tool allowed visual examination of trends in solution histories and the assessment of the sensitivity of orbit determination solutions to different data and filter scenarios. The "cm" tool could rapidly select new queries for display in response to questions from the navigation experts. Also presented at the "daily show" was a Navigation Data Summary plot. This is a one-page-shows-all graphic representation of the status of the orbit determination system. It included orbit determination data arcs and deliveries, maneuvers and small forces data, tracking data and DDOR points by baseline and calibration latency. This plot was particularly helpful in quickly

identifying late or missing tracking data and calibrations. Finally, the Orbit Determination Web page could be accessed at the "daily show". This site served as a central repository for all orbit determination data and could be access from any workstation on the operational navigation network. This page could be used to view small forces trending and accumulated ΔV plots, graphical representations of media and platform calibrations, procedures, documentation, web based shift logs, and a picture gallery.

I. TCM-5

The purpose of TCM-5 was to correct errors in TCM-3 and target the desired landing site. The new filter baseline scenario included per-pass data weights and an additional stochastic non-gravitational acceleration was added to account for short-term uncertainty in the small forces prediction beyond the end of the data arc. The "cold" reboot was executed on April 21, 2008 with a magnitude of 12 mm/s. This under-burn, relative to the predicted magnitude used in the maneuver design trajectory, had the desired effect of moving the solution 3 km above the entry target and provided some margin for the solutions to drift down toward the desired entry flight path angle.

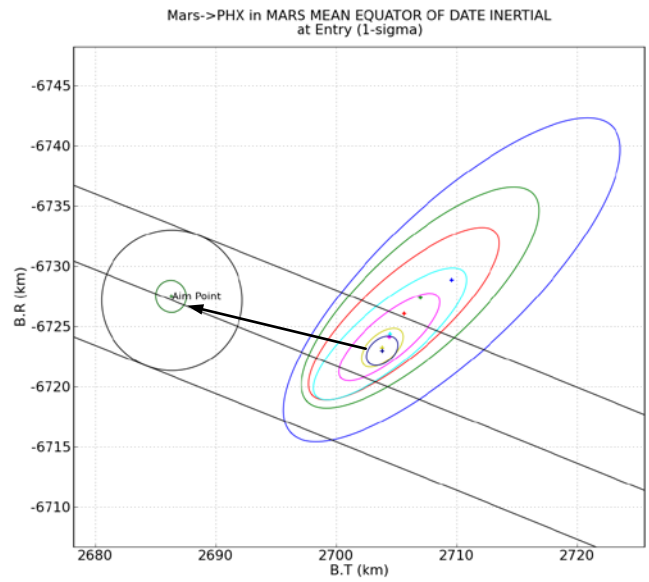

Figure 10. Pre TCM-5 Solution B-Plane History

Daily solutions did indeed begin to converge slowly down in the B-Plane starting the second week of May 2008. Solutions with shorter data arcs were consistently lower in the B-Plane than solutions with longer data arcs. This implied that the correction to the small forces prediction acceleration for the long arc solutions was being constrained to a value that was too small. The small forces trending data confirmed that a 5% increase in the magnitude of the X-component of the small forces acceleration relative to the average in April 2008 had occurred in early May 2008. This value was almost identical to the value used prior to the pre-TCM-3 update. The errors in the acceleration associated with small forces over a variety of averaging times from 0.5 day to 8 days were studied and found to behave like white noise. Thus, a long term average, starting Jan 30, was used for prediction..

The TCM-5 data cutoff was 21 hours prior to its execution. The delivered solution was a 36-day data arc with 111 DDOR points. TCM-5 was successfully executed on May 18, 2008 04:00 UTC. The 0.05 m/s maneuver within the arbitrary six km radius circle centered at the entry target. The maneuver corrected all of the error parallel to the entry corridor but none of the error corresponding to flight path angle (perpendicular to the entry corridor). See Figure 11.

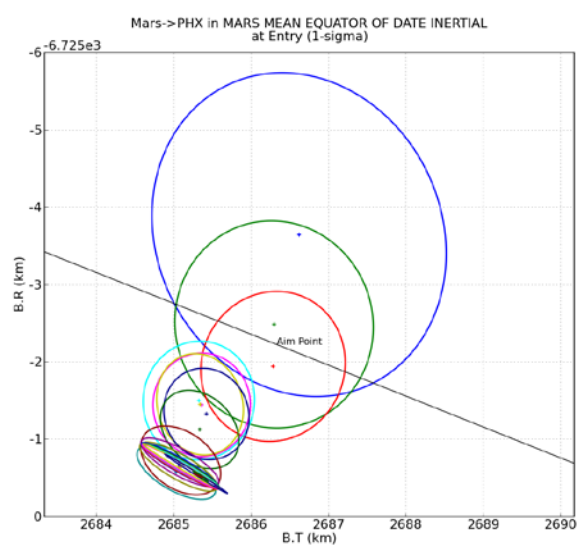

Figure 11. Post TCM-5 Solution B-Plane History

J. Post TCM-5

Immediately following TCM-5, Phoenix satisfied the requirements of the landing site safety criteria. The downward sub-sigma convergence continued, but at this point, time to go was so small that even an extended period of high small forces activity could not push the trajectory out of the entry corridor. On May 22, 2008 the solution made a one sigma downward shift in the B-Plane. This was the first occurrence of a one-sigma shift since modeling had stabilized following TCM-2. The source of the shift was the last two DDOR measurements included in the May 22nd fit. The mean of previous DDOR residuals in the fit was on the order of 0.03 ns. Both the new DDOR residual points were biased 0.06 ns in the fit.

This shifted the solution down in the B-Plane and closer to a surface hazard that the project wished to avoid. Extensive review of the orbit determination setup, calibrations, and DDOR acquisition and correlation showed that there were no errors in processing the data. The bias in the new data was in fact consistent with the expected inherent DDOR data noise of 0.06 ns. DDOR data collected over the next few days verified that the shift was real and solutions converged in the B-Plane. The risk associated with the surface hazard was deemed small so TCM-6 was canceled. Phoenix was only 2.0 km from the entry target when it reached the Mars entry point on May 25, 2008 23:30:57.7 UTC. See Figure 12.

K. Lander Surface Position Determination

The Phoenix Lander position on the surface was determined using radiometric data consisting of in-situ coherent UHF Doppler data between the Phoenix Lander and either the Mars Odyssey orbiter or the Mars Reconnaissance Orbiter (MRO). The orbit geometries of the two orbiters are both nearly polar with about two hour periods affording 10 to 15 minute mutual visibility between orbiter and lander during overflights of the landing site. Given well-determined orbiter trajectories, the Doppler data from a single pass gives strong information about the lander latitude, slightly weaker information about the longitude and much weaker information about the lander altitude. In addition, each pass of UHF data has a starting time known only to several 100 ms. The orbiters' radio reference clocks have stability better than $1.e^{-10}$ s/s, not impacting the Doppler measurement precision of around 1 mm/s which is dominated by multipath. Given these conditions, the lander position can be determined to 500 m accuracy after 3 tracking passes and eventually to a 10 m accuracy after several more tracking passes.

During early landed operations, several coherent tracking passes were scheduled between Phoenix and Mars Odyssey. The data were returned to Earth during subsequent Mars Odyssey to Earth tracking passes. The residuals of these Doppler measurements were processed with a linear least-square filter which included an estimate of the clock epoch for each pass to determine the lander position. Subsequent analysis showed that the quality of the MRO Doppler data was similar to the Odyssey data.

IV. Conclusion

Orbit determination uncertainty, arising from several sources, was expected to be a primary factor limiting the ability to delivery the Phoenix spacecraft to its Mars entry corridor. Project management was well aware of this difficulty. They were willing to invest resources to characterize the performance of the unbalanced thrusters used for attitude control. This was accomplished by executing the Active and Passive Thruster Calibration campaigns. They were aware that the mission operations scenario could have an adverse effect on orbit determination accuracy. Consequently, they chose to operate the spacecraft in a manner that minimized those errors. This was accomplished by placing Phoenix in a sun pointed attitude for late cruise, changing the deadbanding profile in flight to reduce one-sidedness, using propellant margin to extend the period of small forces trending, and selecting maneuver implementations that minimized errors due to slews. Use of DDOR was also extremely important to reducing OD errors. The resultant improvement in orbit determination accuracy allowed delivery of the spacecraft to Mars entry with an error of less than 2 km. The final targeting was accomplished by a maneuver at entry - 8 days. This significantly reduced overall project risk by eliminating the need to perform a maneuver within 24 hours of entry.

Acknowledgments

The authors would like to acknowledge the tireless effort, cooperation and advice of the entire Phoenix Navigation Team and numerous groups whose help made this mission a success. Special thanks for extraordinary support goes to James Border, Peter Kroeger, Peter Tay, Tom Runge, Connie Dang, Jon Streit, Ted Eastman, Todd Ratcliff, Marc Pestana, Teresa Thomas, Wilson Chen, K.J. Lee, Ted Drain, Amy Attiyah, James Evans, Scott Evans,

Michelle Guevara, Rick Sunseri,Joe Wu, members of the Navigation Advisory Group (the NAG, true to their name), Gentry Lee, and our families for putting up with our nonsense.

This work was carried out at the Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California, under contract to the National Aeronautics and Space Administration

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Appendix

Error Source	Estimated?	pre-Approach NAG Baseline (1σ)	Correlation Time	Update Time	Comments/References
Data					
2-way Doppler	Ğ	5.63 mHz	Ğ	Ğ	0.10 mm/s, average of passes that were weighted per pass
Range	Ğ	10 ru	Ğ	Ğ	1.4 m, average of passes that were weighted by pass
ÆDOR	Ğ	0.06 ns	Ğ	Ğ	2.5 nrad, Ref 4
ÆDOR Latency	Ğ	6 hrs	Ğ	Ğ	
ÆDOR Schedule	Ğ	2 per day from TCM-4 to Entry	Ğ	Ğ	
Epoch State					Effectively Infinite
Position (km)	√	100,000	Ğ	Ğ	
Velocity (km/s)	√	1.0	Ğ	Ğ	
Solar Pressure					Not estimated in late cruise, aliases with general non-gravitational acceleration paramters
Solar Pressure Scale Factor	√	0.2	Ğ	Ğ	
Slews (mm/s per axis)	√	38116.0	Ğ	Ğ	-slew uncertainty depended on slew size - full circle slews and slews less than 30 deg used 5 mm/s - large there-and-back slews used 10 mm/s - slew reconstruction uncertainties depended on number of thruster firings
TCM Execution Errors	√	Ref 1	Ğ	Ğ	Spherical per axis
In Arc: Non-gravitational Accelerations (km/s^2)					Estimates the average acceleration error due to solar radiation pressure and small forces ÆV mismodeling
S/C X	√	3.0E-12	Ğ	Ğ	
S/C Y	√	2.0E-12	Ğ	Ğ	
S/C Z	√	2.0E-12	Ğ	Ğ	
Prediction: Long Term Non-gravitational					models long term bias acceleration associated with deadbanding, including thrust magnitude errors, thrust direction errors, and pulse frequency prediction error
S/C X		4.70E-12	10 days	1 day	
S/C Y		8.20E-13	10days	1 day	
S/C Z		3.80E-13	10 days	1 day	
Prediction: Short Term Non-gravitational					models the short term stochastic thrust variations and pulse frequency variations about the average associated with deadbanding
S/C X		1.3E-11	0	1 hr	
S/C Y		2.9E-12	0	1 hr	
S/C Z		1.8E-12	0	1 hr	
Small Forces ÆV per axis scale factor	√	10% per axis	0	1 hr	models short term variations in small forces ÆV
Per Pass Range Bias (m)	√	2	0	per pass	
Mars & Earth Ephemerides		Mars RTN sigmas 9m, 136m, 442m	Ğ	Ğ	
Station Locations		per covariance, Ref 5	Ğ	Ğ	Using latest station location updates and covariance
Quasar Locations (nrad)		1	Ğ	Ğ	
Pole X, Y (rad)		1.50E-08			
UT1 (s)		3.0-e-4			
Media					
Ionosphere Ğ day (cm		65			S-band values converted to X-band in software
Ionosphere Ğ night (cm		15			
Troposphere Ğ wet (cm		4			
Troposphere Ğ dry (cm		1			