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TRAJECTORY DESIGN FOR THE

MARS RECONNAISSANCE ORBITER MISSION

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This paper describes the analysis and design evolution of the Mars Reconnaissance Orbiter trajectory from launch to end-of-mission. The mission uses a combination of propulsive maneuvers and aerobraking techniques to deliver the orbiter to a low-altitude frozen orbit at Mars. The launch/arrival date space for the 2005 Earth-Mars opportunity will be described as well as the particular launch strategy chosen for this mission. Details of the aerobraking profile will be provided. Finally, the Primary Science Orbit will be examined, and the trade-offs between science objectives and orbiter capability will be presented.

INTRODUCTION

In 2005 NASA will launch the Mars Reconnaissance Orbiter (MRO) mission. Its primary mission is to study the surface, sub-surface and atmosphere of Mars and to characterize possible future landing sites for two years (one Mars year). The science payload is a combination of

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several high-resolution instruments in addition to re-flights from the Mars Climate Orbiter (MCO) mission. The suite of six science instruments will allow the spacecraft to conduct observations at higher resolutions and different wavelengths than any previous Mars mission. These instruments will operate in a combination of global and regional surveys and high-resolution targeting (sometimes simultaneously). For example, ground scale per pixel for these high-resolution images will be approximately 30 cm, and images may be as large as 6 km by 30 km (20 Gbits). In addition to its science objectives it provides telecom and navigation relay capabilities for follow-on missions through the UHF navigation and communications experiment. This Relay phase will last another two years. The orbiter also differs from previous missions in its ability to turn to off-nadir targets of interest while maintaining sufficient stability to support demanding instrument requirements.

In order to support these high-quality science observations the orbiter will be placed into a much lower altitude orbit than previously attempted. This Primary Science Orbit (PSO) has a periapsis altitude of approximately 255 km, and an apoapsis altitude of approximately 320 km. The orbit is frozen with periapsis at the south pole, and the near-polar inclination allows nadir observation of targets at –87 degrees latitude. The orbit will be sun-synchronous with a constant local mean solar time (LMST) of 3:00 PM at the ascending node.

The spacecraft will be launched from an intermediate-class expendable launch vehicle (Atlas IIIB-DEC or Atlas V-401) from the Cape Canaveral Air Force Station at the USAF Eastern Space and Missile Range. An in-plane propulsive Mars Orbit Insertion (MOI) maneuver will establish the MRO spacecraft in orbit at Mars after seven months of interplanetary cruise. Approximately six months of aerobraking will be used to reduce the orbit period, following which the orbiter will be delivered into the PSO. The spacecraft, which will be designed and built by Lockheed Martin Astronautics Corporation, has a new bus and solar panel configuration intended specifically to accommodate this extended aerobraking phase. Following the PSO and Relay phases, the orbiter will be raised to a higher orbit in order to provide navigational support to incoming spacecraft through 2015. The Mars Reconnaissance Orbiter Delta-V budget is shown in Table 1.

Table 1: Mars Reconnaissance Orbiter Propulsive Delta-V

Phase	Event	Translational DV
Cruise	TCM's	40 m/s
Orbit Insertion	Mars Orbit Insertion	1010 m/s
	Trim (Period/Inclination)	10 m/s
Aerobraking	Aerobraking Control	33 m/s
	Pop-Up	20 m/s
	Transition	96 m/s
	Inclination Change	18 m/s
Reserves	Unallocated	50 m/s
	Inclination Trim	30 m/s
Primary Science	Orbit Maintenance	19 m/s
Relay Phase	Approach Phasing	20 m/s
	Orbit Maintenance	19 m/s
Extended Mission	Raise to EMO	79 m/s
Total Propulsive DV		**1444 m/s**

LAUNCH / ARRIVAL DATE SPACE

The 2005 Earth - Mars launch space is a demanding opportunity. The launch / arrival date performance contours (or pork-chop plots) in Figure 1 show a large separation between Type I trajectories with an Earth-Mars transfer angle of < 180 degrees and Type II trajectories with transfer angles of > 180 degrees and less than 360 degrees. This is due to relative inclination differences between the Earth and Mars orbits at this opportunity. Several mission constraints were balanced with the interplanetary trajectory possibilities to produce the final cruise trajectory. First, a launch period of at least 21 days to ensure a high probability of launch. Second, the excess hyperbolic velocity (V-infinity) at arrival should be less than 3 km/s in order to keep the total mission propellant to a reasonable level (since this relates directly to the velocity change necessary for capture). Third, the arrival date and nodal orientation must be compatible with the use of aerobraking techniques to establish the Primary Science Orbit. The longitude of the ascending node at arrival must be biased away from 3:00 PM local mean solar time (LMST) in order to provide sufficient time to perform aerobraking with adequate margins.

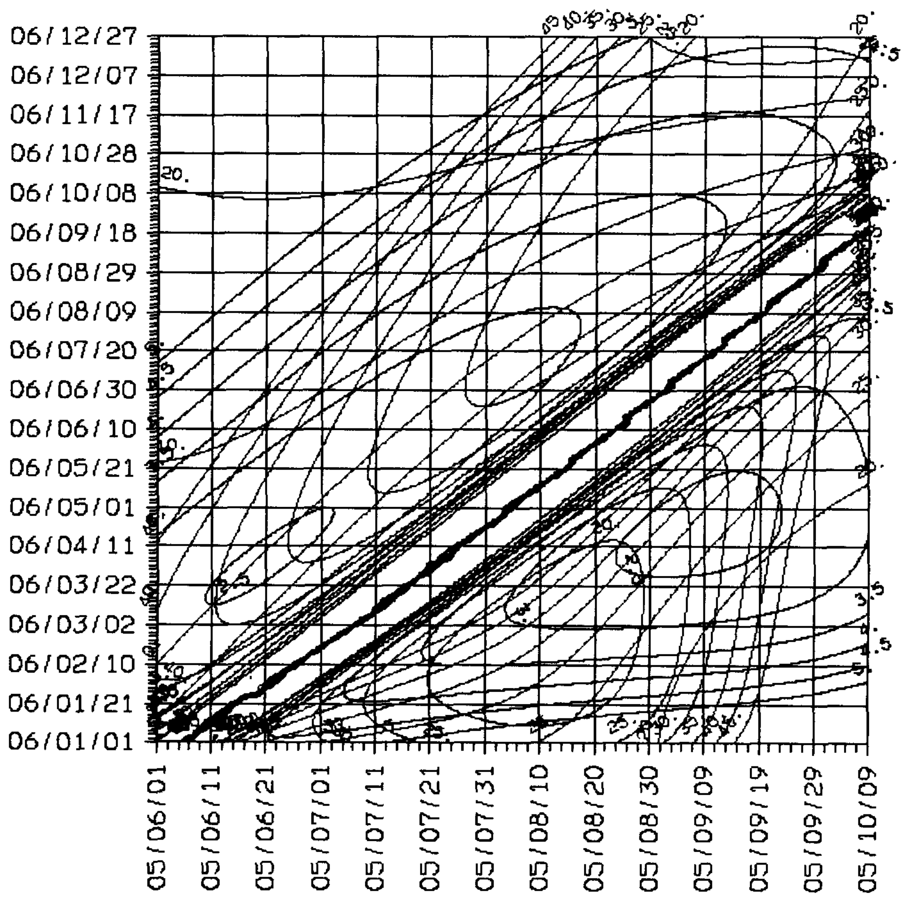

Figure 1: Earth-Mars 2005 Trajectory Performance Contours

The launch targeting strategy must optimize the orbiter dry mass (injected mass minus required propellant mass) while satisfying all these constraints. A common technique is to equalize the C3 at the open and close of the launch period, thereby balancing launch vehicle performance across the launch period. The orbiter dry mass is then a consequence of the required Mars Orbit Insertion ΔV. The MRO launch opportunity does not lend itself to this approach, however, as high declinations of the outgoing asymptote degrade the launch vehicle performance at the open of the launch period. The launch vehicle performance must be balanced by taking into account the penalty for high declinations. This was the guiding approach to launch targeting strategy.

Opportunities such as the 2005-2006 launch / arrival space may be improved by performing a sizeable mid-course propulsive maneuver to change the orbit plane (a broken-plane maneuver [BPM]). MOI maneuver magnitude can vary significantly over the launch period, and this using this available ΔV mid-cruise can allow the injection targets to move to lower C3 values, thereby improving the launch vehicle capability. In particular, this method was used for the early part of the Mars Observer launch period in 1992. A similar technique is the energy augmentation maneuver in which a propulsive maneuver is performed near launch to increase the orbital energy. Both of these methods were investigated to attempt to increase the orbiter dry mass. Unfortunately, the broken-plane maneuvers that emerged required more propellant than was saved. Energy augmentation maneuvers are not useful for the baseline launch period, but it appears that two contingency launch days at the end of the launch period are possible with energy augmentations of up to 100 m/s.

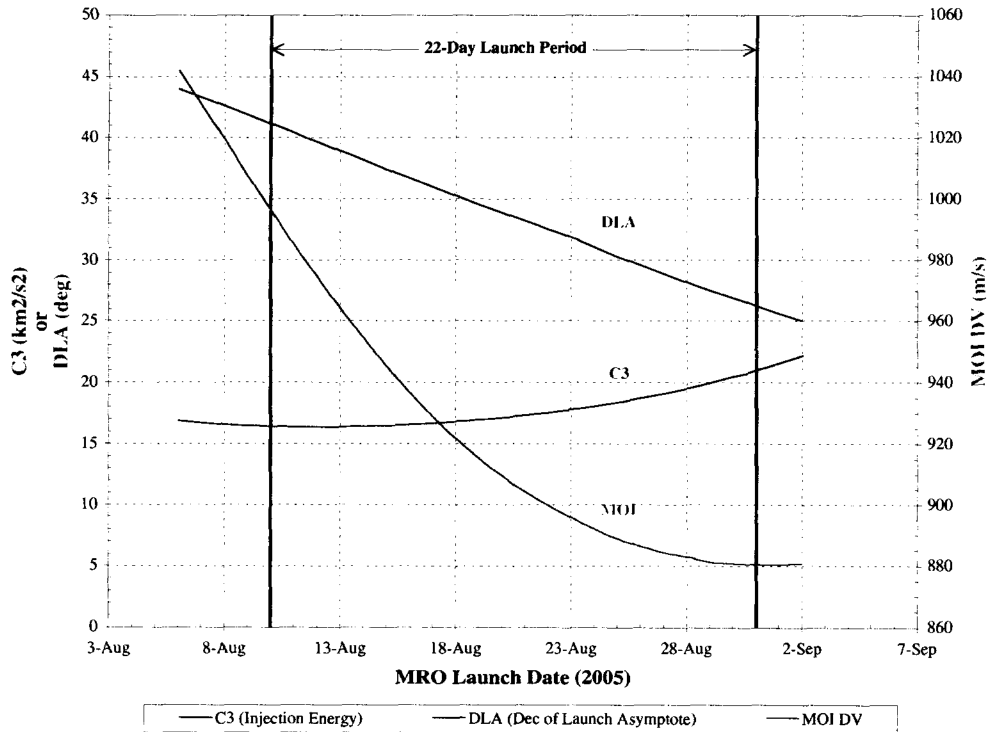

Figure 2: MRO Launch / Arrival Conditions

The baseline 22-day launch period is 10 August through 31 August 2005, with corresponding arrival dates from 10 March to 16 March 2006. The optimal LMST for each launch date was determined, but the LMST varied only slightly. Therefore, the arrival LMST has been fixed at 8:30 PM. The injection conditions are characterized by high launch declinations (DLA > 41 degrees) at the open of the launch period, and high injection energies (C3 > 20 km^2/sec^2) at the close of the launch period. The targeted launch energy (C3), declination of the launch asymptote (DLA) and Mars Orbit Insertion maneuver magnitude (MOI) are shown in Figure 2.

LAUNCH STRATEGY

The Mars Reconnaissance Orbiter will be launched from an intermediate-class expendable launch vehicle (Atlas IIIB-DEC or Atlas V-401) from the Cape Canaveral Air Force Station at the USAF Eastern Space and Missile Range. The dual-engine Centaur upper stage will be used to place the vehicle into interplanetary cruise. The Centaur is an inertially guided, 3-axis stabilized restartable upper stage with a long heritage. Based upon experience it is expected to provide extremely accurate injection, allowing a significant savings in cruise propulsive ΔV. The injected mass requirement for the Atlas IIIB-DEC launch vehicle is 1975 kg. It is expected that the launch vehicle performance will increase; in fact, current analyses show that a capability of greater than 2000 kg exists across the baseline 22-day launch period. This may increase as launch approaches if launch vehicle margins are reduced. The option currently exists to use an Atlas V-401 launch vehicle instead of the Atlas IIIB-DEC. If this option is exercised then launch vehicle capability will likely grow by at least 100 kg.

The current launch strategy is to use one continuous launch window of at least 30 minutes duration on each day of the launch period (10 August to 31 August 2005). The short coast opportunity on each day will be utilized. The launch azimuth and the launch target will be constant throughout the launch window on each day, simplifying the target selection and validation process. The ability of the Centaur upper stage to target directly to the desired interplanetary conditions minimizes the impact of using constant injection targets and the correction at TCM-1 is expected to be less than 1 m/s. Two launch azimuths will be used across the launch period to meet the varying launch declination (DLA) requirements. At the open of the launch period the target DLA exceeds 41 degrees. No direct coplanar ascent is possible at such a high inclination due to range safety constraints, and the launch azimuth for early launch dates will be the maximum southern limit of 108.0 degrees. Late in the launch period the launch azimuth will be more directly due east (95.6 degrees). The boost phase flight profile and first upper stage powered flight will be identical across the daily launch window; the second firing of the Centaur upper stage will perform out-of-plane targeting necessary to match the declination target.

The Atlas boost phase lasts approximately 3 minutes and Atlas-Centaur separation takes place 11 seconds after booster engine cutoff (BECO). Shortly after this the first Centaur ignition will take place (MES1), followed by the payload fairing jettison (PFJ). The first burn is about 5 minutes in duration and is terminated by Main Engine Cutoff 1 (MECO1). The duration of this coast period between Centaur firings is approximately 39 minutes and will be used as a control to reach the desired launch right ascension (RLA). The final upper stage firing MES2 begins a 2.5-minute burn, terminated at MECO2. These events will likely take place during solar eclipse, as well as the Centaur/orbiter separation (SEP) which takes place 220 seconds after MECO2 (depending on the launch date and launch time within the window). The first dependable DSN acquisition is at Goldstone 80-90 minutes after launch (SEP + 30 minutes). Because of this, key orbiter events after SEP such as the solar array deployment and the high-gain antenna (HGA) deployment are unlikely to be observed in real time.

PLANETARY PROTECTION

As a Mars orbiter, the MRO mission is subject to Class III NASA Planetary Protection requirements. In the past, these requirements have been met by establishing minimum probabilities of spacecraft impact on Mars. These probabilities were met during cruise by biasing the Mars encounter points away from the planet while minimizing the total propulsive ΔV penalty, moving the encounter progressively closer to the final encounter.

This aimpoint biasing technique will be used to meet the separate impact probability imposed on the launch vehicle upper stage. The Centaur upper stage must meet a 1×10^{-4} impact probability. This will be achieved by biasing the launch targets at or below this probability. Centaur events such as separation ΔV, the Collision and Contamination Avoidance maneuver, and hydrazine blowdown will be included in the analysis, but are not necessary in order to meet the requirement. The attitudes for these events will be determined by MRO Mission Design in order to reduce the Centaur probability of impact.

INTERPLANETARY CRUISE

The duration of the MRO cruise (injection through MOI) is approximately seven months. Five Trajectory Correction Maneuvers (TCMs) are planned to ensure accurate delivery to the desired Mars encounter. These maneuvers will correct the aimpoint to progressively greater accuracy as more radiometric tracking data is processed.

TCM-1 will be performed at Launch+15 days (L+15). This initial TCM is necessary to correct any injection errors from the Centaur upper stage while the ΔV cost is still within the allocation. This maneuver will also remove the planetary protection bias introduced at injection, currently about 41000 km in the Mars b-plane. TCM-2 is scheduled for Launch+90 days and will correct for maneuver execution errors at TCM-1 as well as any residual orbit determination errors. TCM-3 and TCM-4 are scheduled for MOI-40 days and MOI-10 days respectively. Project requirements specify that the post-MOI capture orbit periapsis altitude shall be 300 km – 50 km, and that the periapsis altitude shall be 300 km – 100 km eight orbits after MOI. TCM-4 will be used to meet these requirements and to minimize the TCM-4 delivery errors (which cost propulsive ΔV post-MOI to correct). The navigation tracking data strategy to support TCM-4 is to obtain continuous DSN tracking from MOI-60 days, with Delta-DOR measurements twice per week. The resulting TCM-4 delivery error is approximately 20 km (3-sigma).

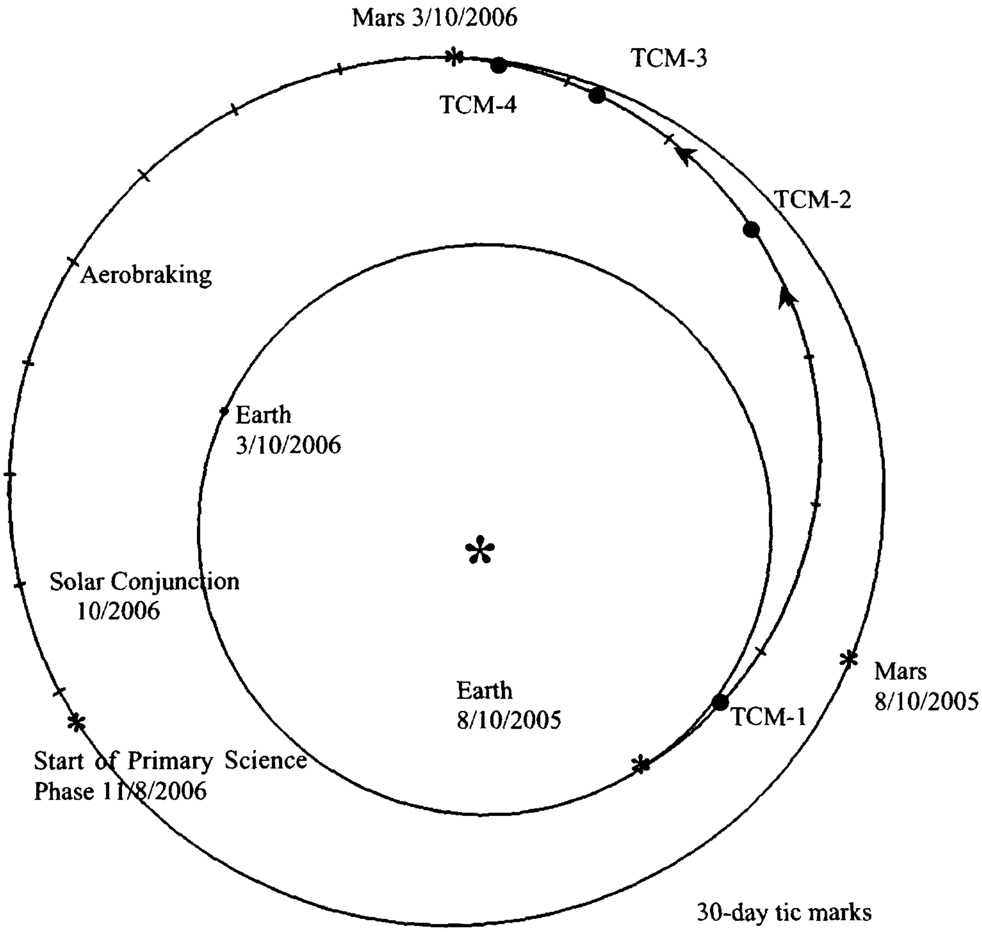

Figure 3: MRO Interplanetary Cruise

Either one of two contingency maneuvers (TCM-5a or TCM-5b) will be performed if necessary. They have been placed in the schedule as contingency maneuvers in case the incoming orbit altitude is unexpectedly low. These maneuvers will be pre-computed to raise the periapsis altitude in 50 km increments; they will not be used to fine-tune the approach trajectory before MOI. TCM-5a is scheduled at MOI-24 hours and TCM-5b is scheduled at MOI-6 hours.

MARS ORBIT INSERTION

The Mars Orbit Insertion maneuver (MOI) will place the orbiter into an elliptical capture orbit about Mars using a constant-rate pitch-over maneuver. The capture orbit period will be 35 hours with a 300 km periapsis altitude. This requires up to 1000 m/s of velocity change (ΔV). As the launch period progresses the arrival V-infinity decreases, and so the ΔV necessary for capture also decreases. The current plan is to constrain the capture orbit period across the launch period rather than keeping the MOI magnitude constant. This simplifies the mission aerobraking plan, and the additional ΔV could be used for Extended Mission operations or to guard against unanticipated atmospheric variability. However a lower capture orbit period would increase the aerobraking margin, i.e., decrease the heating rates on the orbiter components due to atmospheric heating. Whether or not this modification to the capture orbit is necessary will be determined as the mission progresses.

Table 1: Orbit Events During Mars Orbit Insertion (MOI)
(Launch 8/10/05, Arrive 3/10/06)

Event	Date (UTC)	Time from MOI Burn Start	Time from Mars Periapsis
Start MOI Finite Burn	**3/10/06 0:44**	**00:00**	**-15:52**
Mars Periapsis	**3/10/06 1:00**	**15:52**	**00:00**
Enter Shadow	3/10/06 1:00	15:55	00:04
Enter Occultation	3/10/06 1:02	17:27	01:36
Exit Shadow	3/10/06 1:08	23:31	07:39
End MOI Finite Burn	**3/10/06 1:10**	**25:26**	**09:34**
Ascending Node Crossing	3/10/06 1:19	34:47	18:55
Exit Occultation	3/10/06 1:33	48:29	32:37
Duration of MOI Burn (mm:ss)		**25:26**	
Eclipse Duration (mm:ss)		**07:35**	
Occultation Duration (mm:ss)		**31:02**	

The orbiter's all-monopropellant propulsion system is unusual for a planetary mission. Six 170N engines will fire simultaneously for MOI, and thrust vector control will be achieved using six 22N engines in off-pulsed mode. The total effective thrust for these engines will be at least 1065N with a specific impulse (Isp) near 224 sec. Table 1 lists the orbit events of interest during MOI. The MOI maneuver begins 16 minutes before periapsis and is approximately 25 minutes in duration.

Table 1 also describes when the orbiter enters and exits shadow -- the duration of the eclipse is less than eight minutes and does not place any stress on the orbiter power system. The characteristics of the Earth occultation shown in Table 1 are unfortunately also typical. Although the start of the burn and Mars periapsis are visible from the Earth, the orbiter will pass behind Mars late in the maneuver. At this point, the radio signal will be lost for approximately 30 minutes. No navigation or telemetry data will be available during this period to monitor the progress of the maneuver. The orbiter will be out of contact with the Earth until more than 20 minutes after the scheduled completion of the burn, at which time the success of the maneuver will be verified.

This scenario describes the optimal orbit insertion in which the accelerometers are used to command burn termination. A requirement currently exists to ensure successful capture if MOI shuts down due to on-board fault protection (a successful capture orbit is defined for this project as (1) a capture orbit period of approximately 120 hours or less, and (2) periapsis altitude greater than 200 km eight orbits after capture). Recovery from a Single-Event Upset (SEU) would likely require a reboot of the on-board processor, after which point any resulting body rotations would be identified and damped. Slews would be necessary in order to allow the star tracker to determine the current attitude, and more turns would be necessary to place the orbiter in the appropriate orientation to continue the MOI maneuver. A detailed fault analysis by the MRO Flight System has set an upper bound of 14 minutes to complete these tasks.

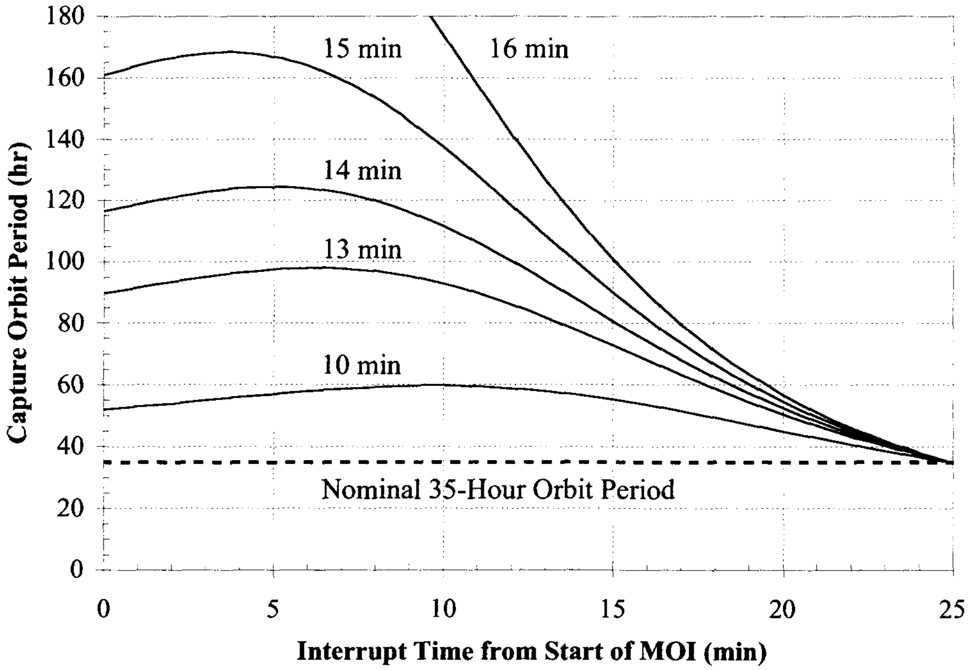

Figure 4: Orbit Period Effects of MOI Interruption

A method to accommodate this burn outage has been chosen to minimize impact on the fault recovery software. The MOI maneuver will begin 2 minutes earlier than optimal at an approximate ΔV cost of 10 m/s in order to accommodate the interrupt. If an interrupt occurs, the on-board software will command the MOI maneuver to restart as soon as the orbiter has recovered. The orbiter will continue the constant-rate pitch-over maneuver until the accumulated acceleration indicates that the MOI ΔV has been achieved. Figure 4 describes the resulting capture orbits as a function of the relative timing and duration of the interrupt.

The most severe effects of an MOI interrupt occur when the fault is near the start of the burn, where the baseline 14 minute interrupt achieves a capture orbit period of 125 hours. In order to accomplish this the thrust vector will be reoriented to the direction at the time of the interruption. This direction has been chosen to simplify the fault protection logic instead of re-calculating the optimal burn direction. It is in fact more efficient to allow the orbiter to turn through the interruption and continue the attitude profile, i.e., more orbit energy change takes place and the direction of the velocity change is more directly opposite to the velocity vector. However, this approach allows the post-MOI periapsis altitude to drop dangerously low (near 60 km). So in this case the slightly less efficient maneuver is preferable.

AEROBRAKING PHASE

The MRO orbiter will use aerobraking techniques to reduce its orbit period, saving over 1200 m/s of ΔV. The operations approach will be similar to those used for the Mars Global Surveyor and Mars Odyssey missions, but there are significant differences. The orbiter has been designed with a high cross-sectional area and is capable of up to 180 days of aerobraking operations. Figure 5 shows the progression of the aerobraking phase as the apoapsis is gradually decreased. Aerobraking operations will commence approximately one week after Mars Orbit Insertion (10 March to 16 March 2006), and the aerobraking termination maneuver (ABX) will take place before solar conjunction (ideally before 20 September 2006; the Sun-Earth-Probe angle will reach 5 degrees on 7 October 2006).

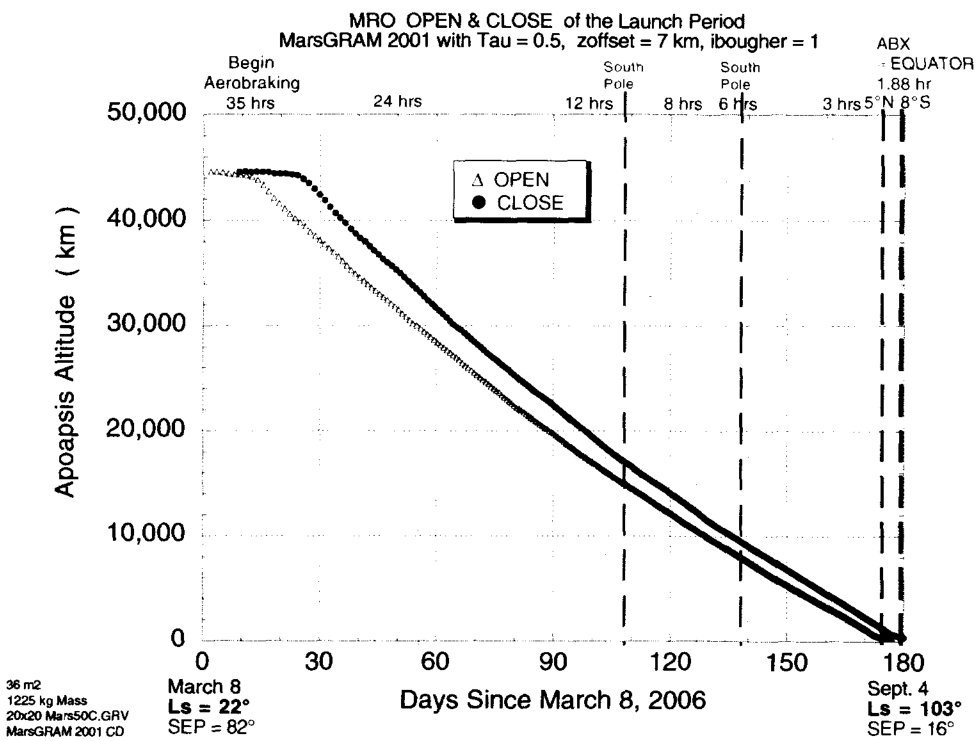

Figure 5: Apoapsis Altitude During MRO Aerobraking

Aerobraking consists of 3 phases: a walk-in phase, a main phase, and a walkout phase. During the walk-in phase, the periapsis altitude of the orbit is slowly lowered by a series of propulsive maneuvers. Since the atmospheric density is not well understood, the exact altitude necessary for aerobraking will be projected from these initial orbits. This initial phase continues until the dynamic pressures and heating rate values required for steady-state aerobraking are established. This will begin the main aerobraking phase, in which large-scale orbit period reduction occurs on each orbit. The dynamic pressure limits on the orbiter will be maintained within predetermined limits by adjusting the orbit periapsis up or down (see Figure 6). The orbits must be carefully monitored during this phase to ensure that the periapsis altitude cannot decay to the point where the orbiter might be damaged by the stresses and heating of aerobraking. The operations team must also track the orbit lifetime (defined as the time it takes the apoapsis altitude of the orbit to decay to an altitude of 300 km). Walk-out begins when the orbit lifetime reaches 2 days. Aerobraking continues during this phase, but the periapsis altitude of the orbit will be controlled to maintain a 2-day orbit lifetime. Aerobraking will cease when the orbiter apoapsis altitude reaches 450 km. At this point, the periapsis altitude will be raised out of the atmosphere by a propulsive maneuver.

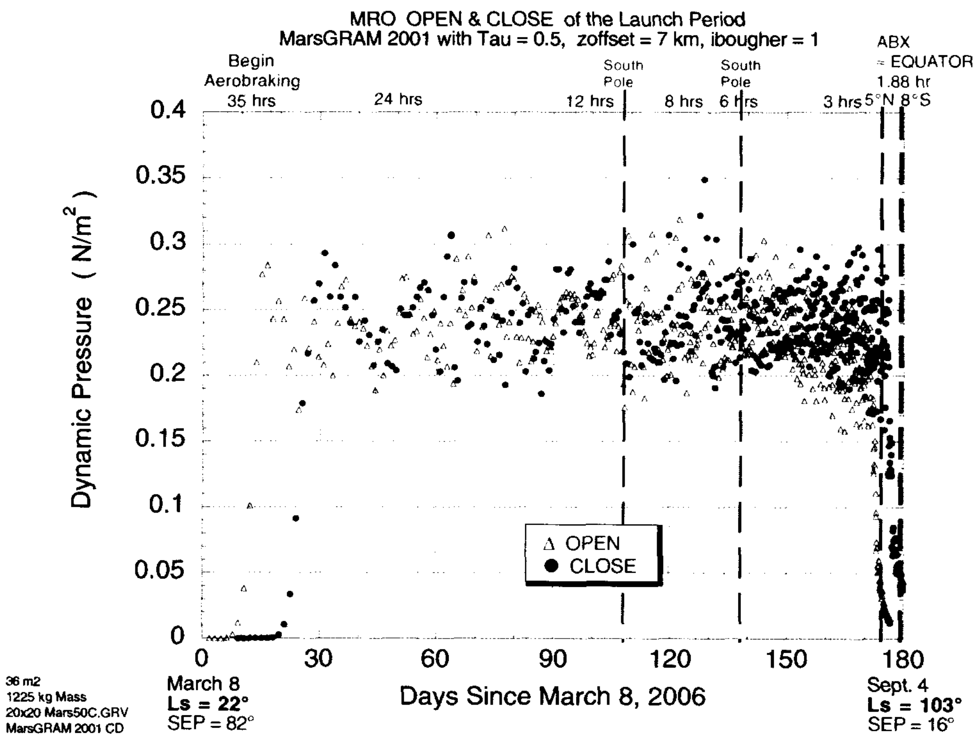

Figure 6: Dynamic Pressure Corridor During MRO Aerobraking

The rate of aerobraking will be determined by the orbit local mean solar time (LMST). The LMST at Mars encounter is constant at 8:30 PM for all arrival dates, and the variation in arrival date through the launch period is only 7 days. Time must be spent in the aerobraking orbit to allow the node to move to 3:00 PM LMST. Aerobraking too quickly would result in a final orbit with a later LMST, violating the science constraints on the Primary Science Orbit (PSO) and causing complications in orbiter operations. Aerobraking too slowly would move ABX closer to the solar conjunction in addition to causing an error in the orbit LMST.

The aerobraking phase has been designed with significant margin to reduce risk to the orbiter and reduce stress on the operations team. The duration of active aerobraking operations is 160 to 171 days (470 to 550 orbits), substantially longer than planned for any previous mission.

However, the heating rate margin for MRO aerobraking is 150% rather than the accustomed 90% to 110%. This margin will account for unexpected atmospheric variability. It should also allow standard 40-hour workweeks for the aerobraking operations team, with personnel continuously available in case of a contingency.

The Primary Science Orbit does not begin immediately after aerobraking is completed. The PSO apoapsis altitude of 320 km is not achievable because of the orbit lifetime requirement, and so a propulsive maneuver in addition to ABX is necessary. The ABX maneuver will establish an elliptical orbit approximately 274 km x 450 km. Argument of periapsis is expected to be in the low northern latitudes and will move southward at a rate of about 6 degrees per day. At the same time the periapsis altitude will move lower due to the same oblateness effects. After about 50 days, the argument of periapsis reaches the south pole and the orbit periapsis will match that of the PSO. At this point a second maneuver (OA-1) will be executed to lower the apoapsis to the target value of 320 km. This passive strategy depends solely on the natural rotation of the argument of periapsis to conserve ΔV. The drawback to this approach is that if the argument of periapsis at ABX is drastically different from the nominal then the wait time might impinge on the solar conjunction period. In this case OA-1 can be safely delayed for one cycle, about 70 days, and performed on the opposite side of solar conjunction. Although this does not affect the total data return, alternate approaches are being considered to avoid this "once-around" situation.

PRIMARY SCIENCE PHASE

The PSO is designed to provide low-altitude observation opportunities with a desirable groundtrack repeat pattern while minimizing atmospheric drag effects. A sun-synchronous orbit is desirable to allow repeatable observations, with similar lighting conditions, of any particular site . The method of achieving this has undergone significant changes since the initial proposal.

During the early phase of the project, the possibility of taking high-resolution observations near 200 km altitude proved to be very attractive to some science investigators. Other experiments desired wider ground coverage at high altitudes. An elliptical orbit with minimum periapsis altitude of 200 km and maximum apoapsis altitude of 430 km was selected to balance these conflicting requirements. In this preliminary science orbit the precession of the argument of periapsis provided periapsis altitudes equivalent to a frozen orbit at 200 km by 265 km (periapsis/apoapsis). As analysis progressed, however, this orbit proved to be undesirable from both science and engineering viewpoints. The orbit period is nearly resonant with Mars rotation, so the groundtracks move very slowly across the surface, and some regions of Mars would be not be visited for many months. All the revisit opportunities for a particular site would be clustered in a short period of time. Another characteristic of this orbit was that the lowest altitudes (200 km at south pole) were frequently above shadowed areas and thus unsuitable for imaging. This nullified a driving requirement for the elliptical orbit. Perhaps the strongest argument against the elliptical orbit arose as the atmospheric effects were analyzed. As previously noted, the orbiter has been designed with a high surface area to facilitate aerobraking. This is obviously a disadvantage when the desire is to minimize drag effects in orbit. Operating at these low altitudes would have required articulation of the solar panels or the addition of larger momentum wheels to manage aerodynamic torques on the orbiter. Both of these were deemed undesirable.

A new Primary Science Orbit was designed in order to meet the project science requirements and avoid active aerotorque management. This was achieved with a frozen orbit with periapsis altitude near 250 km and apoapsis altitude near 320 km. The method by which this orbit was chosen will be described in the following paragraphs. The orbit has a near-polar inclination of 92.7 degrees and argument of periapsis at the south pole. The orbit is sun-synchronous with a constant local mean solar time of 3:00 PM with respect to the orbit ascending node. The viewing characteristics of this orbit are improved; the fraction of time spent below 300 km altitude and above the sunlit surface is greater than the initial 200 x 430 km orbit.

Groundtrack repeat characteristics determined the precise orbit elements for the PSO. The driving requirements were (1) allow rapid "mapping" of the surface in a few days, (2) provide short-term ground site revisit opportunities using the off-nadir targeting capability, and (3) space groundtracks less than 5 km apart at the equator over a long period of time. The idealized groundtrack is one which has equally-spaced equatorial crossings across the entire planet. In practice, this cannot be achieved because of longitude-dependent variations in the gravity field. However, these idealized repeat patterns still indicate families of desirable orbits.

The gross characteristics of the groundtrack repeat pattern are determined by the ratio of the orbit period to the length of the Mars synodic day (sol), or conversely

$$Q = \text{Sol Duration} / \text{Orbit Nodal Period} \tag{1}$$

$$Q = \text{Number of Orbits} / \text{Number of Sols} = R / D \tag{2}$$

When R and D are integers this indicates that the groundtracks will begin to repeat after D sols, i.e., the groundtrack for Orbit R+1 will be identical to that for Orbit 1. The distance between R equally-spaced groundtracks at the equator is simply

$$Req / R \tag{3}$$

where Req is the Mars equatorial radius. Short-term revisit opportunities were defined as groundtracks approximately 100 km apart at the equator (equivalent to a 20-degree off-nadir spacecraft turn at 300 km altitude). Using Equation (3), a short-term repeat spacing near 100 km requires at least 15 sols.

Table 2 shows the characteristics of several candidate orbits that meet this short-term spacing. Useful science orbits for this mission need periapsis altitudes greater than 200 km to avoid aerotorque management, and less than 300 km to allow high-resolution imagery to take place. Q-values in the range of 13.0 to 13.4 correspond approximately to semi-major axes from 3700 km to 3625 km. Based upon experience, defining a reference eccentricity of 0.01 for these orbits provides a good estimate of the periapsis altitude. These reference periapsis altitudes are also described in Table 2 (with respect to the polar radius). It is evident from Table 2 that many possible orbits exist in this range.

Table 2: Groundtrack Characteristics of Candidate MRO Science Orbits

D (Sols to Exact Repeat)	R (Orbits to Exact Repeat)	Q = R / D			Nodal Period (sec)	Mean Semi-Major Axis (km)	Reference Periapsis Altitude (km)	Groundtrack Spacing (km, at Equator)
		Decimal	Fractional					
15	196	13.06667	13	1/15	6794	3686	273	108.9
15	197	13.13333	13	2/15	6759	3673	260	108.3
15	199	13.26667	13	4/15	6692	3649	236	107.2
16	209	13.06250	13	1/16	6796	3687	274	102.1
16	211	13.18750	13	3/16	6732	3663	250	101.1
16	213	13.31250	13	5/16	6668	3640	228	100.2
17	222	13.05882	13	1/17	6798	3687	274	96.1
17	223	13.11765	13	2/17	6768	3676	263	95.7
17	224	13.17647	13	3/17	6737	3665	253	95.3
17	225	13.23529	13	4/17	6707	3654	242	94.8
17	226	13.29412	13	5/17	6678	3644	231	94.4
17	227	13.35294	13	6/17	6648	3633	220	94.0

The Q = 211/16 orbit was selected in order to keep the groundtracks near 100 km without driving up the required period of time from 16 to 17 sols. In addition, it is not close to resonance with Mars rotation (Q = 13) and has a periapsis altitude that is acceptable to the flight system. Figure 7 shows a subset of the groundtracks over this 16-sol period. The initial orbit (Orbit 1, Sol 1) is shown at the lower right, and the next orbit is 27.3 degrees west (commonly called the fundamental interval). Note how the groundtracks fill the fundamental interval over the first 6 sols from west to east, then repeat this two more times. The effect of this is that global scans can be obtained every 5 to 6 sols, or less than once per week, and sites of interest may be revisited on the next global pass (subject to constraints such as surface lighting conditions and other science activities).

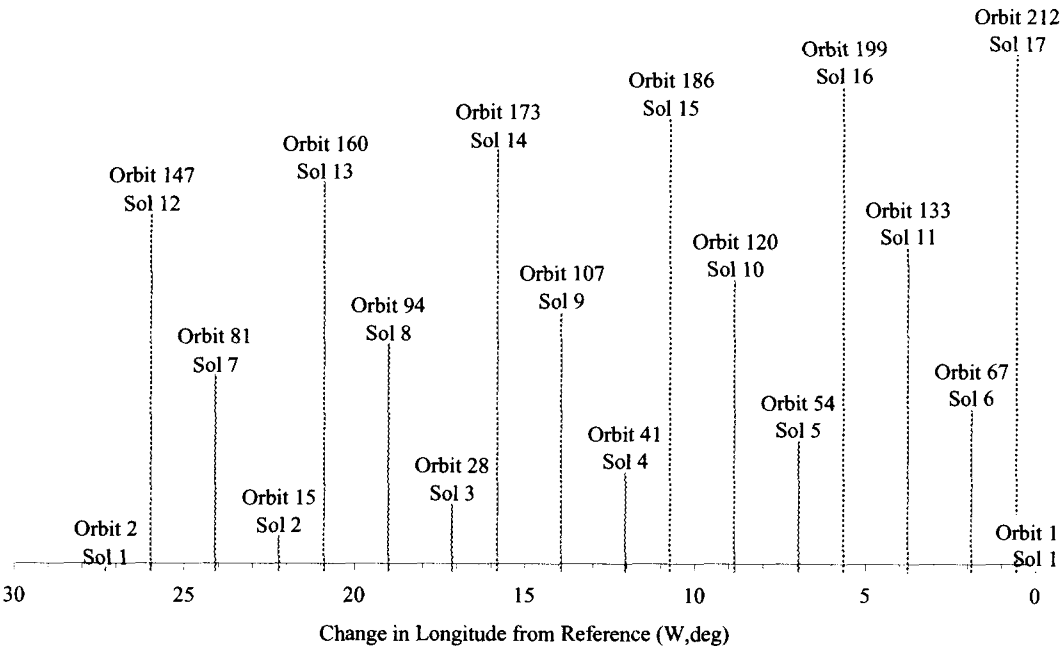

Figure 7: PSO Short-Term Groundtrack Repeat Pattern

The final requirement, that the groundtracks be separated by less than 5 km over some long period of time, was fulfilled by slight modifications to the value of Q in much the same manner. This entailed choosing a suitably long number of sols and orbits which approximated the ideal value of 211/16. The value of Q chosen to completely define the science orbit is Q = 4602/349 = 13.18625. After 349 sols and 4602 orbits, the groundtracks at the equator will be spaced 4.6 km apart. Careful examination of Figure 7 shows that the groundtrack on Orbit 212 does not fall precisely over the initial groundtrack, nor are the groundtracks all spaced equally. This is due to the change in Q to establish the eventual close spacing.

These precise groundtracks cannot be flown exactly because of the presence of major perturbations such as atmospheric drag and gravity field perturbations. Atmospheric drag will decrease the orbit period, pulling the groundtracks west. Initial studies also indicate that significant drag effects take place throughout the entire orbit, not simply at periapsis as is typically assumed. This is bolstered by the fact that the MRO apoapsis altitude is lower than either the Mars Odyssey or Mars Global Surveyor periapsis altitudes. Angular momentum desaturation events will also take place approximately every 48 hours, but these thruster firings will be balanced and will impart little velocity change to the orbiter. The ΔV magnitude of corrections for these effects will be small, and must be balanced carefully against the total accuracy requirement of the propulsion system. Propulsive events on the spacecraft must meet a 3-sigma error requirement of 0.02 m/s in ΔV magnitude, indicating that a velocity change of 0.05 m/s is a reasonable lower limit. Preliminary studies indicate that maintaining the orbit semi-major axis with maneuvers of this magnitude is possible. Orbit Trim Maneuvers (OTMs) would be triggered by this minimum ΔV, and maneuvers could take place every 2 to 6 weeks depending on the atmospheric density.

The use of bio-burden techniques to meet NASA Planetary Protection requirements has probably removed the need for a high-altitude "quarantine orbit." An extended mission in the PSO would require about 50 kg of propellant per Mars year for attitude control and orbit maintenance, and none of this propellant exists in the MRO ΔV Budget. Therefore, at the end of the Primary Science Phase and Relay Phase in 2010 a higher-altitude orbit will probably be necessary to meet the mission requirement to operate until the end of 2015. One concept currently under consideration would be to raise the orbit 20 km below the Mars Global Surveyor orbit. At this altitude there would be no possibility of impact with MGS. This option would require about 50 m/s of ΔV, which has been accounted for in the ΔV Budget.

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