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Aerobraking at Mars: the MGS Mission

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Aerobraking at Mars: the MGS Mission

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The Mars Global Surveyor mission, scheduled for launch in November 1996, will employ aerobraking as a means reaching a low altitude, sun-synchronous mapping orbit. 'I'his technique is necessary to minimize the cost of the mission. The dry mass of the spacecraft (about 660 kg) and the capability of the Delta 11 launch vehicle (about 1060 kg to Mars) do not permit an all-propulsive transfer to the desired orbit. Aerobraking in the MGS mission presents several challenges not faced by the Magellan acrobraking mission, The requirement to reach a 2 pm node, sun-synchronous mapping orbit puts a limit on the time available for acrobraking and drives the spacecraft thermal design. The atmospheric densit y encountered from orbit to orbit is expected to vary more than was experienced at Venus due to several factors: 1) the changing topography below the periapsis pass duc to the planet rotation, 2) the atmospheric changes due to winds and dust storms, and 3) the irregular Mars gravity field. in addition, the Mars atmosphere models are less accurate than the Venus models were at the time of Magellan acrobraking. As a result the spacecraft and mission design accommodates a 90% orbit-to-orbit atmospheric density change. To be consistent with a low-cost flight team, operations during the acrobraking mission phase will be planned to be as routine and repetitive as appropriate for safe operation of the spacecraft.

Introduction

The Mars Global Surveyor spacecraft will be launched from Cape Canaveral in November 1996 aboard a Delta 11 launch vehicle. Following a 1 O-month cruise to Mars, the spacecraft will be inserted propulsively into a 48-hour elliptical orbit whose line of nodes is near the day/night terminator. The spacecraft will not carry enough propellant to go propulsivel y to the desired low, circular mapping orbit, but will use repeated passes through the atmosphere to remove orbital energy.

During the first month in Mars orbit, during the walkin phase, the periapsis altitude will be lowered (via apoapsis maneuvers) into the atmosphere in four steps to the desired main phase altitude. The majority of the orbital energy removal occurs during the main phase which lasts for 82 days, assuming launch occurs at the beginning of the launch period. For a launch occurring at the cud of the launch period, the main phase must be completed in 69 days. The duration of acrobraking is determined by the science requirement to reach a 2 pm local mean solar time, sun-synchronous mapping orbit.

After the acrobraking main phase, the periapsis altitude is gradually raised during the 20-day walkout phase in order to maintain a 3-day orbit lifetime to allow time to correct any anomalies that may occur. If the spacecraft were to enter a safing condition during this phase there will be sufficient time to take corrective action and perform a periapsis raise maneuver.

Acrobraking is terminated with a maneuver which raises the periapsis out of the atmosphere and places it near the approximate mapping orbit altitude. After this maneuver there is a 26-day period during which the gravity field rotates the line of apsides from the equatorial region to the south pole. Finally, a maneuver at periapsis lowers the apoapsis altitude, completing the transfer to the mapping orbit.

The flight team, through the on-board sequence, will configure and orient the spacecraft for entry into the atmosphere each orbit. The sequence is executed via initiation of a reusable command script which is resident on-board. The periapsis time must be predicted to within about four-minute accuracy in order for the

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spacecraft to achieve the desired drag attitude. If the correct attitude is not sequenced at the correct time, precious attitude control propellant will be wasted fighting the aerodynamic torques. In addition to the acrobraking sequence, the spacecraft attitude control software requires an on-board ephemeris in order to maintain the proper nadir attitude throughout the drag pass.

Based on atmospheric data from **Mariner and Viking** missions at Mars, it has been concluded that a short-tenn or orbit-to-orbit density variation of 70% should be accommodated in the spacecraft and mission design, An additional 20% density variation due to periapsis altitude perturbations due to the gravity field must also be accommodated. In order for the spacecraft to be able to accommodate this total 90% density variation it was necessary to add "drag flaps" at the ends of the two solar arrays. These flaps increase the projected area and allow the spacecraft to operate at a lower mean density.

Aerobraking Baseline Trajectory

Several days after capture in Mars orbit, the periapsis will be lowered from the nominal capture orbit periapsis altitude of313 km to the upper edges of the atmosphere (139 km), where the heating rate is approximately 0.01 W/cm^2. The periapsis will be lowered in steps (see Fig. 1) because of uncertainties in the atmospheric density of Mars, especially at the acrobraking altitudes. The spacecraft will spend approximately four orbits at each intermediate density level in order to characterize the densit y and density variability. The targets for each of the watkin steps will be based on what is observed at the previous level.

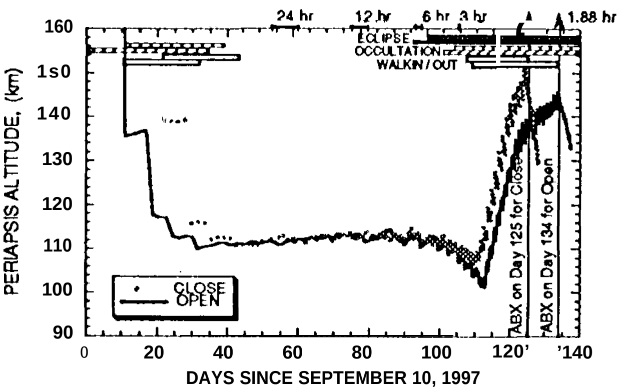

Fig. 1 Periapsis altitude

Figs. 1-4 show time histories of key trajectory parameters. These figures show the parameters for trajectories corresponding to the opening and closing launch dates of November 5 and 25, 1996, respectively. For these cases, the Mars orbit insertion dates are September 11 and 22, 1997, respectively.

After periapsis is lowered to the desired altitude for the main phase of aerobraking, small propulsive maneuvers will be required to keep the heating (see Fig. 2) and drag forces within acceptable limits. If the drag is too small, the orbit will not reach the required 2 pm mapping orbit. If the drag is too large, the spacecraft will be vulnerable to random atmospheric fluctuations which temporarily increase the density and aerodynamic heating rate to unacceptable levels.

The apoapsis altitude decreases almost linearly during the main phase of aerobraking. As the apoapsis nears the final, nearly circular value, the rate of decrease is reduced because the periapsis altitude is raised in order to maintain a 3-day recovery time in the event of an anomaly which could prevent the next periapsis raising maneuver. This end phase of acrobraking is called the walkout phase because daily periapsis raise maneuvers are required to maintain the 3-day margin, and the periapsis altitude gradually walks out of the atmosphere. When the apoapsis reaches the desired (450 km) altitude, the periapsis altitude is raised to the approximate mapping orbit value.

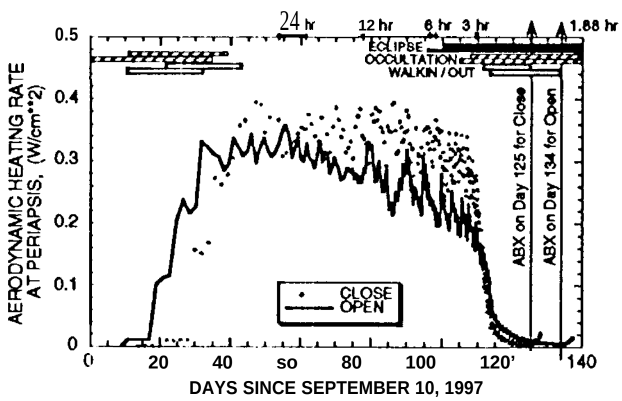

Fig. 2 Peak heating rate

The required mapping orbit has a descending node (at the equator) which is at 2 pm relative to the fictitious mean Sun. The node of the 48-hour orbit does not move much because the spacecraft does not spend much time very close to the planet where the gravitational perturbations are largest. As the orbit shrinks, the gravitational perturbations become larger and larger, As the orbit approaches the final mapping orbit, the nodal rate approaches the sun-synchronous value required by the mapping orbit (see Fig. 3).

Similar to the node, the argument of periapsis (the angle measured from the ascending node at the equator to the periapsis of the orbit) moves slowly at first and then more rapidly as the orbit shrinks. The periapsis begins at 30 deg north latitude, mows up past the north pole and then down the other side of the planet, which is on the far side from the Sun (see Fig. 4).

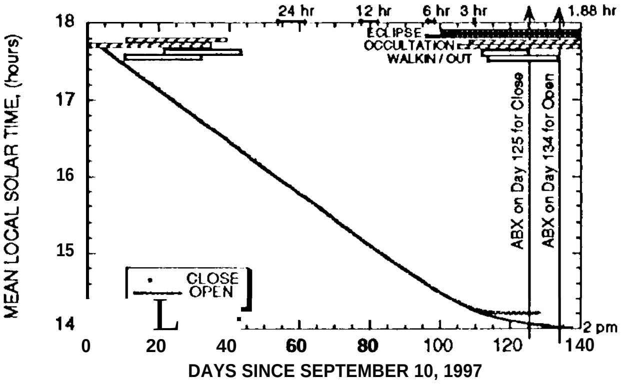

Fig. 3 Descending node local mean solar time

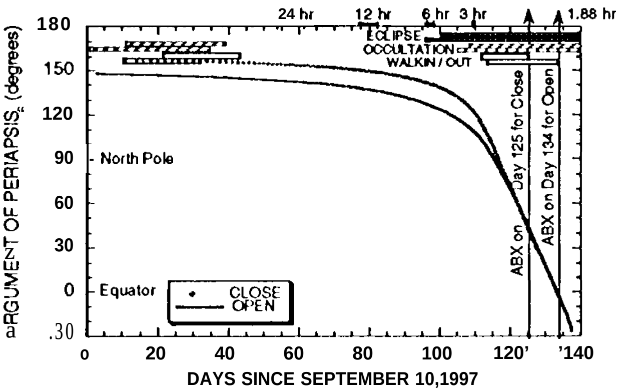

Fig. 4 Argument of periapsis

The aerobraking trajectory simulation models an average atmospheric density from the MarsGRAM model[1] which depends on latitude, longitude, dale, altitude, F10.7 flux, terrain and dust. The fluctuations in the simulated densities (apparent in the heating rate curves in Fig. 2) are caused by periapsis altitude perturbations duc to the gravity field alone. Atmospheric variability is accounted for by putting margin between the average simulated values and the values which would cause the spacecraft to overheat. The aerodynamic heating rate is half the density times the velocity at periapsis cubed. The actual heat transferred to the spacecraft depends on how efficiently this aerodynamic heating is transferred to the spacecraft, which depends on flow characteristics around the spacecraft, as well as the molecular interactions as the molecules collide with the spacecraft.

The time spent in the atmosphere (drag duration) is a very small fraction of the orbit for most of aerobraking, only about 7 minutes for most of aerobraking. As the orbit becomes more circular, the time in the atmosphere increases. Using a dynamic pressure (one half the density times the velocity at periapsis squared) of 0.0015 N/m^2 to define the start of the drag pass provides adequate margin for momentum accumulation during the drag pass and for attitude control authority.

The change in orbit period per orbit is of interest to the navigation design. When the orbit period is large, the drag causes large (5000 s) changes in the orbit period each orbit. If the uncertainty in the atmosphere is only 10%, then the uncertainty in the next predicted time of periapsis will be 500 s. Larger atmospheric fluctuations, up to 70%, will be accommodated by the navigation design by using tracking data after each drag pass to predict a new periapsis time for the next orbit. The predicted periapsis time triggers the on-board sequence of events and must be uplinked to the spacecraft each orbit when the orbit period is more than a few hours.

Although the orbit period decreases rapidly at first, the periapsis velocity changes very slowly. As the period shrinks, there are more orbits per day, so the velocity begins to decrease more rapidly. As the orbit approaches the final circular orbit, the drag duration atso increases, which coupled with the increasing number of orbits, rapidly increases the daily change in the periapsis velocity. Even during the walkout phase where the heating rates and peak drag forces are much smaller, the velocity still changes significantly. Of the 1200 m/s change in the periapsis velocity, 250 m/s occurs during the walkout phase.

Mars is relatively close to the Sun as viewed from the Earth at arrival, and moves closer as it heads toward conjunction. The capture orbit is nearly face-on to the Sun, and gradually moves toward the 2 pm orientation, where it will be locked into a sun-synchronous orbit. When Mars begins to cast its shadow on the orbit near the end of aerobraking, the eclipse durations increase as the orbit becomes more edge-on to the Sun. When the spacecraft is in the "tail-first" aerobraking attitude, little or no solar power is available. Near the end of aerobraking where the orbits periods arc smallest, the drag pass is not always centered on the eclipse and the total time without solar power becomes a significant fraction of the orbit, During this power limited period, the transmitter will be turned on for shorter periods to maintain an adequate power margin.

Aerobraking Configuration

The spacecraft alternates between the "tail-first" orientation (Fig. 5) during the drag pass and "Array Normal Spin" orientation to enable telecommunications during the rest of the orbit. The velocity vector is close to the -Z axis during the "tail-first" drag pass. This attitude protects the instruments which are mounted on the +Z from most of the aerodynamic heating. The solar arrays are oriented so that the incoming atmospheric molecules strike the back sides, thereby protecting the solar cells themselves. The arrays are swept back 30° with the panels placed against hard stops which prevent aerodynamic forces from backdriving the gimbal motors. The resulting configuration is aerodynamically stable in that the aerodynamic forces will tend to center the vehicle with the velocity lined up with the -Z axis. This orientation places the propulsion end of the spacecraft into the velocity direction.

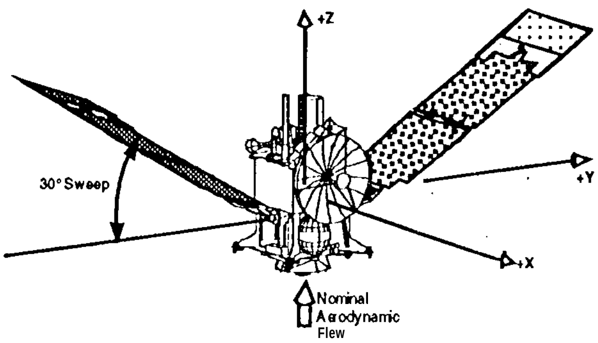

Fig. 5 Spacecraft configuration

The solar arrays are rotated 90 deg about the Y-axis using the inner gimbals to put the active sides toward the Sun (the +X side of the spacecraft) during the Array Normal Spin phase of each orbit. The outer gimbals maintain the 30 deg angle to minimize gimbal movement. The High Gain Antenna (HGA) boresight (+X) is pointed at the Earth and the spacecraft rotates Once every 100 minutes so that the celestial sensor assembly (CSA) can detect different stars in order to update the attitude knowledge. The attitude control software updates the inertial attitude reference by identifying the scanned stars with an on-board star catalogue, maintained by the ground. Attitude control is via the reaction wheels. Autonomous momentum management is triggered near the drag pass via momentum threshold limits, for which the reaction control system (RCS) thrusters are fired to reduce the system momentum.

In order to perform aerobraking passes in the most efficient manner, the spacecraft attitude control is switched to thruster control with a wide (±15 deg) pointing error deadband during significant drag phases. This allows the spacecraft to free oscillate around the aerodynamic null, thereby using minimal propellant. The spacecraft is turned to the "tail-first" attitude just prior to switching to thruster control and is turned back to the Array Normal Spin attitude just after switching back to reaction wheel control. The time spent on thruster control is equal to the expected time in the atmosphere (drag duration) plus a margin for periapsis timing error (currently five minutes) on both sides of the expected drag pass (see Fig. 6). Periapsis timing errors during the drag pass have the potential to be large enough that the resulting attitude error would create aerodynamic torques larger than the reaction wheels could counteract. The spacecraft fault protection would interpret the failure to maintain the commanded attitude as a hardware failure and would take unnecessary and unwanted actions, including switching to thruster control with a tight pointing deadband.

The current best estimate for some of the properties of the spacecraft during the aerobraking phase are as follows: Mass = 745 kg (wet), Area = 17.04 m^2 (projected area for drag, including flaps), Weighted Drag Coefficient = 1.95, Heat Transfer Coefficient = 0.80, Thermal Accommodation Coefficient = 0.95.

During a normal aerobraking pass, the solar panels are the component which will be pushed closest to the thermal limit. The back side of the solar panel will take the impacts of the atmospheric molecules and will be slightly hotter than the front (solar cell) side of the panel. The nominal aerodynamic heating rate changes slowly with each day due to a different aspect angle to the Sun and Mars and a drag duration which grows larger as the orbit becomes more circular. The nominal heating rate is approximately half the heating rate which would cause the temperature constraints to be violated, with most of the remaining margin being allocated to potential atmospheric blooming.

Because it is the most thermally strenuous, the spacecraft development team has used trajectory data for the last day of the launch period for their design studies. Several factors might increase these margins. If the spacecraft is launched closer to the beginning of the launch period, the required aerodynamic heating will be less, so there will be more margin in the design. The delta-V at Mars orbit insertion could be increased, which would decrease the peak heating rates. If the random atmospheric variability is less than expected, the peak heating rates will be less. If the local solar time requirement is changed from 2 pm to 1:45 pm, more time would be available for acrobraking and the heating rams will be lower. If more molecules bounce off the spacecraft in a specular fashion, less heat will be transferred to the spacecraft and thermal margins will be larger.

Orbital Timeline Analysis

Two spacecraft events are sequenced in parallel. The first event is called the drag pass and occurs each orbit near periapsis. The second event is called an aerobrake maneuver (ABM) and is performed as required (up to a maximum of once per day) at apoapsis to raise or lower the periapsis altitude.

A typical drag pass event is illustrated in Fig. 6. In the following description the "predicted start of drag" time includes a navigational timing uncertainty of five minutes. Twenty minutes prior to the predicted start of drag, the catalyst bed heaters for the RCS thrusters are turned on and allowed to warm up. Eight and one half minutes prior to the predicted start of drag, the command & data handling (C&DH) subsystem is configured to record 2 kbps engineering telemetry onto a solid state recorder (SSR). Normally one minute later, the transmitter is turned off to conserve power. The spacecraft is next commanded to initiate a reaction wheel turn to the initial drag pass attitude, in which the spacecraft -Z axis, is aligned along the velocity vector and the -X axis is pointed at Mars. Seven and one half minutes is allocated to acquire this attitude. After acquiring the initial drag pass attitude, attitude control maintains a nadir orientation through the use of an on-board Mars ephemeris, with the spacecraft rotating about the +Y axis (once per orbit) to keep the -Z axis close to the velocity vector and the +X axis pointed at Mars. Star processing is disabled through the drag pass, with the inertial reference propagated using the gyros.

One minute after initiation of the spacecraft turn, the +/-Y solar army gimbal drives are enabled and commanded to orient the panels as required for the drag pass. Both solar arrays are rotated simultaneously 90 deg about their inboard gimbal, so that the cell side of the army is swept back 30 deg towards the spacecraft +Z body axis. This orientation keeps the backside of the arrays into the aero flow, while providing an aerodynamically stable attitude. Three minutes is allocated for the solar array orientation.

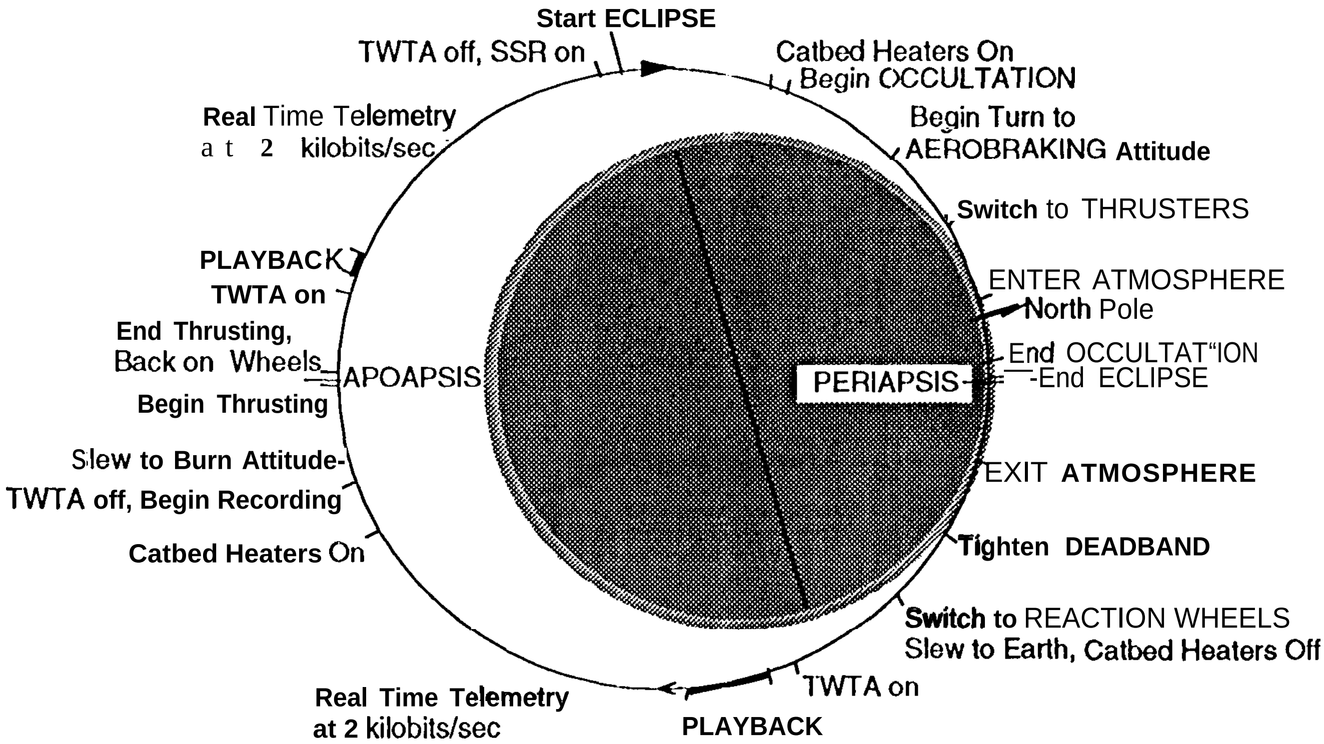

Fig. 6 Spacecraft events around a typical aerobraking orbit

At the predicted drag start time, the spacecraft is commanded to thruster control mode (with the reaction wheels held at their current speeds) and the attitude control deadbands opened up to prohibit excessive hydrazine expenditure during the drag pass. Angle of attack is controlled to within 15 deg. At the predicted time of periapsis, the reaction wheels are commanded to approximately zero speed, to dump the spacecraft momentum. At the end of the predicted drag period plus 5-minute navigation timing uncertainty, the attitude control deadbands arc tightened up to reduce the spacecraft body rates below the limits for reaction wheel control authority. Five minutes arc allowed for this rate damping period. At the end of this period, control is switched back to reaction wheels and the spacecraft commanded back to the normal cruise attitude control mode described previously. Additionally star processing is re-enabled, allowing the inertial reference to be updated oucc the cruise attitude and spin are reestablished. Onc minute later the solar arrays arc commanded back to their cruise positions. After allowing time for the acquisition of the cruise attitude, the transmitter is turned back on and a minimum 5-minute real-time 2 kbps engineering telemetry downlink period is initiated to give a snapshot of the current spacecraft health. Following the real-time period, the SSR is commanded to play back the engineering telemetry recorded during the drag pass.

The relative timing bctwccn the drag pass events are adjusted periodically to fit the shrinking orbit. As the orbit period decreases, the drag duration increases. Consequently, the amount of time that the solar arrays arc not receiving solar input and that the spacecraft bus load is supported by the batteries increases as well. Additionally, when the orbit period decreases below six hours, the orbit becomes eclipsed from the Sun by Mars for up to a maximum of 41 minutes per orbit. The phasing between the eclipse portion of the orbit and the drag pass determines the total off-Sun duration per orbit. 10 maintain adequate margins on tbc orbital energy balance and state of charge of the batteries, the transmitter is turned off at the start of eclipse period and remains off until the spacecraft is pointed back at tbc Earth and out of eclipse.

A typical ABM is also illustrated in Fig. 6. The flight software maneuver control parameters (e.g. burn direction, dc.sired delta-V magnitude, backup burn duration timers) are first loaded into the flight soft ware maneuver control task compool. The catalyst bed heaters for the RCS thrusters arc warmed up 20 minutes prior to the bum. One minute prior to the turn to the burn attitude, recording of the 2 kbps engineering telemetry is initiated on the SSR and the transmitter is turned off to conserve power. The spacecraft is commanded to the bum attitude 12.5 minutes prior to the burn. Onc minute later the solar arrays are commanded to their maneuver positions, in which they are rotated '90 deg about their inboard gimbal to place them against the hard stops during tbc burn. By two minutes prior to the burn start, the spacecraft will have acquired the desired bum attitude. At this time the inertial measurement unit (IMU) is commanded to the accelerometer output format. A fcw seconds prior to the bum, the RCS thrusters are armed. At the burn start time the attitude control software is commanded to the maneuver control mode.. The maneuver control task tires the delta-V thrusters until tbc desired delta-V has been sensed by the accelerometers. The bum attitude is maintained by on-pulsing the non delta-V thrusters and off-pulsing the delta-V thrusters as required. In the event of an accelerometer failure, a backup bum timer will terminate the burn. The sequence of commands to reconfigure and reorient the spacecraft to the normal cruise mode is as described above for the drag pass event.

Spacecraft Margins

Two areas of the spacecraft have required special attention duc to aerobraking heating loads, the solar arrays and tbc HGA. Minimum temperature margin is 15° C bet wccn flight allowable temperature and qualification temperature for normal operations in aerobraking. The solar arrays consist of an aluminum honeycomb and composite face.sheet assembly. The drag flaps are constructed of graphite composite struts wrapped with 1 mil carbon impregnated Kapton. A specific pattern of white paint and Kapton tape protects kading surfaces of the panels against asymmetric heat transfer rates experienced in transitional flow. Aerobraking temperature predictions show the array facesheet-core bond temperature reaching 165° C as compared the 218° C adhesive limit. Materials and processes on the front and rear of the solar arrays bad been changed to resist higher temperatures caused by the original safing orientation. The prc.sent sating orientation heats the same side (non-cdl side) of the panels as normal aerobraking resulting in large temperature margins on the cell-side surface. The arrays have been qualified to 200 cycles at 190" C peak temperature to simulate the aerobraking environment, The HGA reflector is constructed of a composite facesheet trended to a composite core. The reflector and rim arc cured at 177° C. Multilayer insulation has been added to the rim and back side of the reflector and a two roil, germanium coated, Kapton radome has bccu added to prevent overheating of the antenna in the acrobraking mission phase. Although the current safing orientation does not require a radome it will remain to keep the 1 IGA warm during cold thermal conditions. This assembly will be qualified to a temperature of 150° C to provide acceptable margins for the mission.

Single string thruster control is used for nominal acrobraking (dual string for Z axis to provide a couple torque). The worst case angular rates expected are 4 deg/s as compared to 15 deg/s before gyro saturation occurs. In the event of safing (wherein the spacecraft will flip over from Array Normal Spin attitude to the aerodynamically stable drag attitude), dual string thruster control is used to maintain safing flip-over rates below 6 deg/s. Single string thruster control will maintain flip-over rates below gyro saturation of 15 deg/s in several cases. Fuel usage for normal acrobraking and one safing event is estimated to be 11 kg of hydrazine as compared to an allocation of 13.7 kg. The current sating orientation has resulted in five times lcss fuel usc for sating control over the previous baseline. The current sating orientation will also result in drag pass fuel mass dynamics to provide the required 0.5 square inch surface contact between fuel mass and the propellant management device sump.

Attitude knowledge must also be maintained sufficiently during the drag pass to return to Earth point without star acquisition. The attitude control mode for the drag portion of an orbit is CSA back-up mode. In this mode the spacecraft will experience a knowledge drift of 0.25 deg after 50 minutes. Attitude knowledge after the drag pass needs to be better than 0.5 deg to quickly restart Array Normal Spin mode and locate stars. Forty percent margin exists on attitude knowledge for the longest period expected on CSA back-up mode.

Each outboard solar array drive is forced against a static hard stop during normal acrobraking and can not be forced out of configuration. In the. event of a spacecraft safing event, aerodynamic torques prior to the ftip-over, will act to push the outboard drives away from the mechanical hard stops and must be resisted by the gimbal drive mechanisms. The outboard drive is overtorqued and begins to move at an equivalent dynamic pressure of 0.88 N/m*. Spacecraft aerodynamic characteristics will guarantee that the flip-over is complete at 0.2 N/m^2 to provide a 440% margin of safety in the event of a 90% atmosphere. The inboard solar array gimbal drive atso experiences aerodynamic torque duc to the 25 deg rotational position in sating. The inboard overtorque margin is 214%. worst case, for the inboard drives.

Power margin is a concern for the acrobraking orbits where the period is less than two hours and the drag pass orbital position immediately follows solar eclipse. Energy balance is not achieved on orbits which contain an ABM to maintain the desired periapsis altitude. Eventually, the short orbit and long drag pass do not provide sufficient battery recharge time and the energy balance approaches 50 watt-hours on the final few orbits of acrobraking. Nickel-Hydrogen battery state of charge (SOC) drops below 70% fOr a significant period of the acrobraking mission phase. However, battery life will not be degraded to less than the three year design lifetime of the mapping and relay mission. SOC is estimated to be as small as 45% during an ABM. Power margin was a consideration in the selection of the current safing orientation. Even with the off-sun time in the safing drag pass energy balance is always positive in each orbit and SOC remains greater than 70%.

Navigation Design

The primary acrobraking requirements and constraints placed on navigation design areas follows: a) predict the time of periapsis passage (Tp) to 225 s, b) predict the periapsis distance (Rp) to 1.5 km and c) due to spacecraft and orbital constraints, tracking data shall not be acquired within 0.5 hours of periapsis. Eight acrobraking cases were identified bawd upon the spacecraft's orbital period; one each during walkin and walkout and six during the main phase of acrobraking. For each case, a determination was made as to how well the above requirements could be satisfied. In addition, these results were important for establishing an operational plan and work-schedule.

The procedure used to simulate acrobraking navigation operations was as follows: a) acquire one orbit (two orbits during the latter stage of the main phase) plus several hours of post-periapsis doppler tracking data, b) apply an accuracy of 0.2 mm/s for data counted or averaged over 60s, c) analyze this data using a weighted least-squares algorithm (with a priori information) and solve for the spacecraft's state,

Table 1. Tp and Rp uncertainties **for six cases during the main phase of aerobraking.**

EPOCH AND ORBITAL PERIOD (HRS)	PREDICTED UNCERTAINTIES PERIAPSIS	Tp (S)	Rp (KM, 1 SIGMA)
10/10/97	P2 *	5.6	0.02. / 0.06
40.0	p3	ER	0.10/0.11
11/04/97	P2 *	1.9	0.07 / 0.20
24.0	P3	ER	0.10 / 0.21
11/30/97	P2 *	0.3	0.02 / 0.05
10.0	p3	375.	0.05 I 0.07
12/14/97	P2	0.2	0.02 / 0.07
6.0	p3 *	151.	0.03 / 0.08
	P4	439.	0.04 / 0.10
12/29/97	P4	42.2	0.04/0.11
3.0	P5 *	126.	0.06 / 0.13
	P6 *	255.	0.12 / 0.18
01/01/98	P5	57.7	0.08 / 0.21
2.42	P6 *	112.	0.10 / 0.21
	P7 *	185.	0.10 / 0.21
	P8 *	275.	0.10 / 0.22

* Navigation can predict n (1,2,3) periapses ahead; ER = exceeds requirement.
Note that the Rp uncertainty corresponds to a) nominal gravity field (25th degree and order) coefficient uncertainties (col. 1) and b) an increase in the coefficient uncertainties by a factor of three (co]. 2).

Generally this consisted of orbital elements at the epoch of the analysis and model parameters for atmospheric density, gravity field coefficients, spacecraft thrusting around periapsis and an average acceleration to account for angular momentum desaturations (AMD) and solar radiation pressure. This analysis provided the reconstructed knowledge of the spacecraft's orbit. In addition, these results were propagated forward in time. Thus predictions of the time-of-periapsis-passage, orbit elements and their corresponding uncertainties were generated. With this information, the navigation capability could be compared to the above requirements leading to the development of an operational navigation strategy.

The results of simulations carried out for each of the main phase cases are summarized in Table 1. For example for case 1 (i.e. 40.0 hours orbital period), the predicted accuracy of the second periapsis-passage (P2) time is 5.6s and the radial distance (Rp) is 0.02 km. Note the following: a) P1 (first periapsis) refers to the orbit during which recking data were acquired, b) the Tp uncertainty is driven by the atmospheric density - a 70% orbit-to orbit density variation or uncertainty was used, c) the. Rp uncertaint y is mainly due to the gravity model uncertaint y - it is a one-sigma result and d) the effect and uncertainty associated with an aerobraking maneuver (ABM) was not included in these results. For this case, the prediction of the time of periapsis-passage for P2 satisfies the requirement but the P3 prediction does not-

Based upon these results, navigation can develop a workable operations strategy which will satisfy the requirements. For example, for case 1, navigation will predict Tp and Rp only one orbit ahead in order to satisfy the requirements. However for the fifth case (i. e. orbital period of 3.0 hours), two orbits of doppler data shall be acquired (this includes P1 and P2) and accurate and timely predictions shall be made for P5 and P6. Note that the 225 s requirement is in effect, however occasionally exceeding it up to a limit of approximatel y 260 s can be tolerated at the expense of a small amount of propellant for at titude control.

Ground Operations

The flight team responsibilities during aerobraking include spacecraft commanding (through use of stored commands and real-time commands) to perform the required orbit events, navigation orbit determination and atmospheric modeling, and routine spacecraft health monitoring. The aerobraking operations design is based on the Magellan operations.

There are two critical spacecraft orbital events which must be commanded by the ground in order to execute aerobraking successfully. The first spacecraft event is performed every orbit around periapsis to configure and orient the Spacecraft fOr entry into the atmosphere. This event is executed via initiation of a reusable command script which is resident on-board and stored in the mission phase script area of onboard memory. This drag pass command script is generated and updated by a ground resident command block called AEROBRAKE, which resides in the sequencing software. The on-board drag pass sequence is initiated via "start script" or trigger commands which are loaded as stored sequence command files as frequently as needed to meet navigation periapsis timing error prediction requirements. In addition to the sequence of trigger commands, the spacecraft attitude control software requires a spacecraft ephemeris in order to maintain the proper nadir attitude throughout the drag pass. The Navigation Update Process is used for executing the drag pass event each orbit.

Associated with the AEROBRAKE block are specific parameters which allow the relative timing between the events to be adjusted as the orbit period decreases. Periodically, the block will be rerun on the ground to adjust the block timing and the resulting sequence reloaded on the spacecraft into the appropriate reusable script area. The Underlying Sequence Update Process is the process used to accomplish this.

The second critical spacecraft orbital event is the propulsive maneuver at apoapsis which is executed as needed 10 maintain the required periapsis altitude. The command script is generated by a ground resident command block called ABM and stored on-bard the spacecraft similar to the drag pass script and is initiated via a real-time "start script" trigger command. The ABM Decision and Implementation Process determines when an ABM will be triggered.

Atmosphere Modeling and Observation

Waves from various sources cause the atmospheric density to vary as a function of time, which means that the density from orbit to orbit is somewhat unpredictable. This short term orbit to orbit random variability is accommodated by targeting for a mean density which is only half the density which would cause the solar panels to overheat. Unfortunately, there is very little actual data at the typical aerobraking altitudes. Models are being used to make up for this lack of data. Global circulation models of the lower atmosphere are being extended upwards and upper atmosphere models are being extended downwards. Aerobraking will take place at the boundary between these two models.

The other atmospheric concern is the sudden change to the mean atmospheric density that occurs when then. is a large global dust storm, The worst case predicted by the MarsGRAM model of the Mars atmosphere is on the order of a 400% increase in less than a week, Since the spacecraft is designed to withstand only a 90% increase, propulsive maneuvers to raise periapsis will be required to keep temperatures below critical levels if a large global dust storm occurs during aerobraking.

The question is "How do we decide if a maneuver is required?" The expected random fluctuations could mask the steady increase from a global dust storm in the densities inferred from the effects of the drag on the spacecraft. The primary means for inferring the atmospheric density is for the orbit determination process to *reconstruct the* density required to produce the observed changes in the orbit. When the orbit period is large, there are two days between samples and only a handful of samples during the week it takes a dust storm to encircle the planet. Densities can also be inferred from spacecraft solar panel temperatures, accelerometer measurements, and possibly from the attitude oscillations. Except for the accelerometer measurement, which contains data about the structure of the atmosphere along each drag pass, these methods tend to produce only a single "averaged" measurement per orbit. Measuring changes in the structure may help us to separate randomly occurring waves from global changes.

Several other options for detecting dust in the atmosphere are under study. Todd Clancy has been very successful in detecting dust storms on Mars using a ground based measurement of the CO absorption band using one of the antennas at Kitt Peak. Observations will be made this year when

Mars is at the same Longitude of the Sun as it will be for the aerobraking phase two years later. Further measurements will be made during the aerobraking phase to alert the project of the presence of a global dust storm. The major problem with this option is that wc must compete for time on the Kitt Peak antenna, so wc do not expect to obtain the daily measurements wc desire for all of aerobraking. Since the spacecraft will be making several samples of the atmospheric density per day near the end of aerobraking and will be better able to detect changes in the mean density, wc put a higher priority on using the Kitt Peak observations earlier in the aerobraking main phase.

Another Earth based option is to use the Hubble space telescope to image Mars and watch for obscuration caused by dust in the atmosphere. In addition to the problem of time availability on the Hubble, the main phase of aerobraking falls in the Sun exclusion zone for Hubble. Thus Hubble will only be useful during the walkin phase.

The Pathfinder lander will be on the surface of Mars taking measurements as part of its extended mission when MGS arrives at Mars. The Pathfinder atmospheric pressure measurements will be especially useful in that the Viking lander data showed that global dust storms have a detectable signature in the diurnal pressure measurement. The Pathfinder camera may also be useful for observing dust in the atmosphere and measuring the optical depth of the atmosphere.

The science instruments on the MGS spacecraft can also be used to watch for the development of dust storms on Mars. Use of the instruments will be limited to observations which can be built into the sequence and made repetitively over many days requiring little attention from the flight team. The Mars orbital camera can image a large fraction of the planet immediately following each drag pass. Such an image with the wide angle detector will take in more than half the planet, of which at least half will be illuminated.

The thermal emission spectrometer (TES) can also detect dust in the atmosphere. When the spacecraft is maneuvered to image Mars, the TES can take data at the same time. The TES will also take measurements of Mars from the Array Normal Spin mode. Because the TES boresight is orthogonal to the ANS spin vector, it sweeps out a plane in space which usually intersects with Mars at some point each rotation. Furthermore, the TES scanning mirror can he used to sweep the TES boresight back and forth above and below the orthogonal point in order to obtain a wider swath as the sensor sweeps across the planet.

Other options arc under study for using the Mars horizon sensor or the magnetometer to infer information about the atmosphere.

Conclusions

A plan for using aerobraking trajectories to reach a low altitude, near circular orbit about Mars has been developed. The spacecraft design is compatible with this plan. Efforts arc underway to develop ways to obtain early warning on atmosphere density changes and therefore minimize risk to the spacecraft. By using aerobraking to reduce propellant requirements, mission costs can be dramatically reduced through the use of smaller launch vehicles.

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References

[1]Justus, C. G., "Mars Global Reference Atmospheric Model for Mission Planning and Analysis", *Journal of Spacecraft and Rockets,* Vol. *2$, No. 2,* March-April 1991, pp. 216-221.

'1.yens, D. "1'., et. al., "The Magellan Venus Mapping Mission: Aerobraking Operations", 44th Congress of the International Astronautical Federation, Graz, Austria, October 16-22, 1993. paper IAF-93 -Q.4.409