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MARS SAMPLE RETURN ORBITAL RENDEZVOUS DETECTION METHODS

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Navigation trades while in pursuit of a rendezvous with a sample capsule in orbit around Mars are described. The rendezvous operation is described in terms of far-field and near-field detection. Passive optical detection is compared with active capsule detection using a radio beacon. Passive methods are, to first order, unconstrained by time while a radio beacon link is time-limited. A possible activity to make use of orbiter plume interactions is explored.

INTRODUCTION

NASA has investigated many designs and case studies for retrieving Mars surface samples[1]. The work described in this paper assumes that the samples to retrieve have been placed inside a canister and subsequently carried into Mars orbit with a Mars Ascent Vehicle (MAV). It also assumes a low-thrust, electric propulsion retrieval spacecraft is in Mars orbit, awaiting the orbiting sample (OS). After the OS capsule reaches orbit, the orbiting spacecraft (Orbiter) begins a four-step operation. First step is to find the passive (or mostly passive) OS. Subsequent steps would be: OS rendezvous, OS retrieval, and Earth return. This paper describes navigation design concepts for the first two steps (known as far-field and near-field detection). Also, while in the near-field regime, a potential dust mitigation method is proposed.

FAR-FIELD NAVIGATION

The assumed hypothetical rendezvous scenario is as follows. MAV launches OS into a polar orbit similar to Orbiter's, except OS lags Orbiter and is injected 29.15 km lower. The MAV-OS separation time is 11-JAN-2030 12:19:32.4 UTC. The injection point defines orbit true anomaly. The line-of-sight distance between Orbiter and OS at separation is 2300 km. The OS then drifts towards Orbiter, eventually passing underneath. The orbit is a Sun-synchronous, 1-sol (12-orbit), repeating ground track, optimized with a 95x95 gravity field. Figure 1 illustrates the launch sequence. Note that "S1" and "S2" in the figure refer to MAV's stage 1 and stage 2. Figure 2 shows Orbiter's ground-track.

Orbit parameters of Orbiter and OS are summarized in
Table 1. Respective Orbits for Attempting Rendezvous
(IAU Mars Pole)

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Orbital Parameter	Orbiter	OS
Altitude (km)	508.15	479
Semi-major Axis (km)	3897.65	3868.5
Eccentricity	1e-3	1e-3
Inclination (deg)	92.7	92.7
Longitude of Node (deg)	90	90
Argument of Periapsis (deg)	0	0
True Anomaly, at separation (deg)	0	0
LMST at Node	9:54	9:54

Table . The physical parameters of the vehicles are summarized in Table 2. Note that atmosphere drag on OS was ignored (initially) because its ballistic coefficient is significantly larger than Orbiter's (1000 kg/m^2 versus 200).

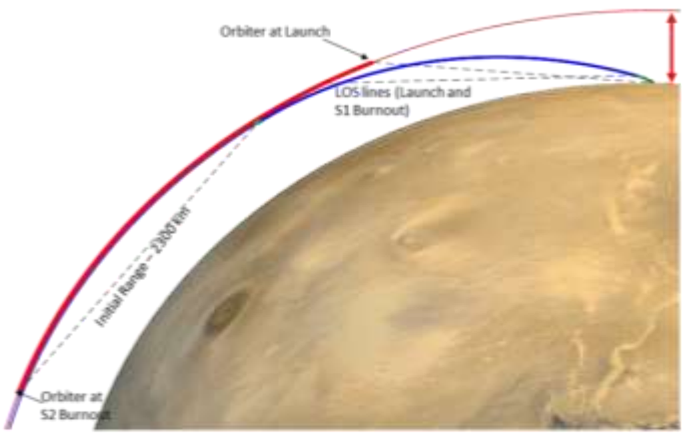

Figure 1. Orbiter and OS relative positions

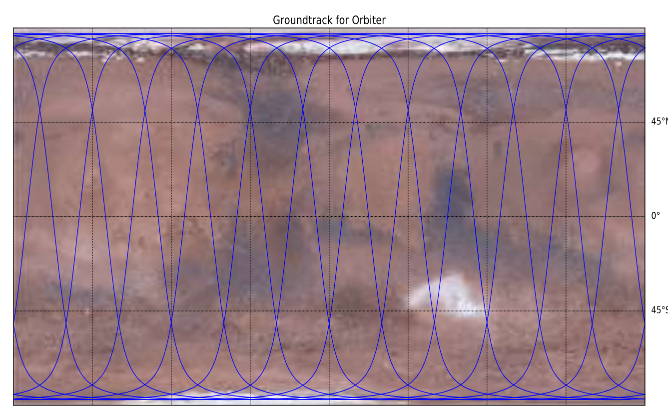

Figure 2. Orbiter ground-track

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Table 2. Physical Parameters of Rendezvous Vehicles

Feature	Orbiter	OS
Mass (kg)	2000	100
Area (atm cross-section) (m^2)	5	0.06
Reaction Control System	yes	no
T=0.9 N (x8), Isp=290 s		
Ion Propulsion System	yes	no
T=0.22 N, Isp=4200 s		
Power	solar	battery*
MarsGRAM atmosphere	drag-yes	drag-no
*only for radio frequency beacon		

Passive Optical Detection

Earth-based radiometric tracking in conjunction with on-board optical imaging was found to be a powerful data combination for rendezvous operations. For Orbiter tracking, we assume a near-continuous Deep Space Network two-way doppler signal (*i.e.* when Mars does not occult the link). OS is “tracked” by inter-spacecraft optical images acquired by Orbiter. There are constraints on spacecraft-based imaging. OS needs to be visible to Orbiter’s cameras (i.e. illuminated by the Sun) and also not occulted by Mars. Additionally, imaging is unavailable if OS transits beneath Orbiter (*i.e.* OS passes between Orbiter and the surface of Mars). This latter assumption is conservative and can be revisited in future analyses. Figure 3 shows tracking availability overlaid with geometric events for this analysis.

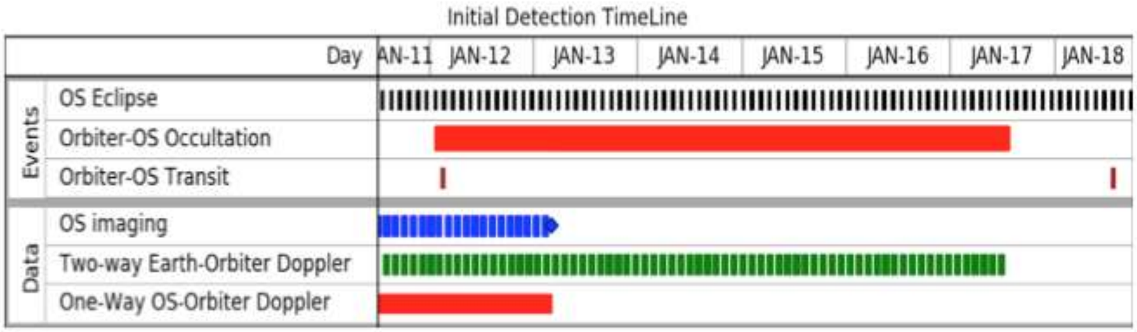

Figure 3. Tracking schedule and geometry events

OS Imaging. For *initial* detection of OS, again applying conservatism, we presumed ground-based processing of the imaging telemetry (not on-board processing). Appealing to JPL's experience with past optical navigation processing, a 5-hour turn-around time was assumed for processing the initial set of downlinked images *i.e.* the first pictures from Orbiter after OS is released from MAV (5-hours between the optical telemetry hitting the ground and the finalized OS ephemeris ready for uplink to Orbiter). Acquisition frequency is 1 image per every 5 minutes.

The five-hour processing time is applied for only the first two hours of downlink (wall clock time). Thereafter the process becomes 'smarter' with processing time expected to decrease to two-hours, yielding a single processed image every 10-minutes, subject to the constraint that every hour-and-a half, OS goes into Mars' shadow for 38 minutes (OS eclipse).

Occultations between Orbiter and OS occur when Mars comes between the two, so no OS imaging occurs when their relative positions are in this geometry. For the two orbits we've selected, the vehicles remain in-view two days for every 4.5-days out-of-view. So to prevent a possible lost-in-space situation, the OS orbit should be determined before OS enters its first occultation. Thus the observability cadence is 48-hours in-view versus 108-hours out-of-sight. In actuality, after the MAV launch, OS could disappear from view as soon as 12-hours after separation.

OS Orbit Determination. Figure 4 shows OS position uncertainty starting from its initial configuration (release from MAV) and illustrates the rapid improvement of knowledge over time as navigation data from Orbiter is received and processed. Doppler and optical image points are shown on the Xaxis (dark red for doppler, blue for images). Shadowing events show up as gaps in the data. Note that doppler tracking of Orbiter is nearly continuous and therefore is available before OS imaging begins. The Appendix summarizes our model's error sources and data accuracies.

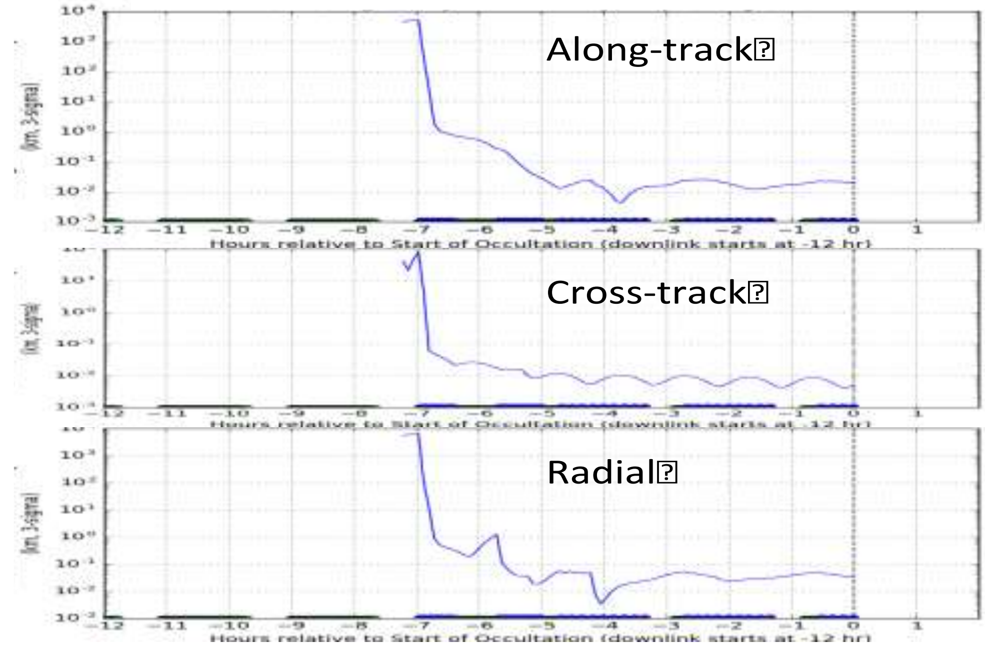

Figure 4. Initial OS position uncertainty, post MAV launch

The wait for the first OS orbit solution after MAV separation is approximately 5.25-hours (including light-time). Subsequent navigation data improves the OS orbit solution. Steady-state knowledge is achieved within 4-hours of the first solution. Therefore even an 8-hour initial processing delay can be accommodated, given a worst-case occultation scenario. After achieving steady-state knowledge, the OS orbit is known well-enough for orbit-matching activities on Orbiter to begin.

Under nominal conditions OS is expected to be detected and pin-pointed within a single workday, starting from a plane-of-sky uncertainty of thousands of kilometers. If an anomaly or scheduling mishap prevents orbit-matching from commencing immediately, OS can be re-acquired following an occultation as knowledge of its orbit degrades slowly. (Atmospheric drag is not a significant perturbation source, contributing less than 1 m downtrack offset to OS after four days.) Figure 5 shows the increase in OS position uncertainty while in occultation. After four days, the OS position uncertainty is less than 1 kilometer and therefore is recoverable within a single image frame of the narrow angle camera on Orbiter.

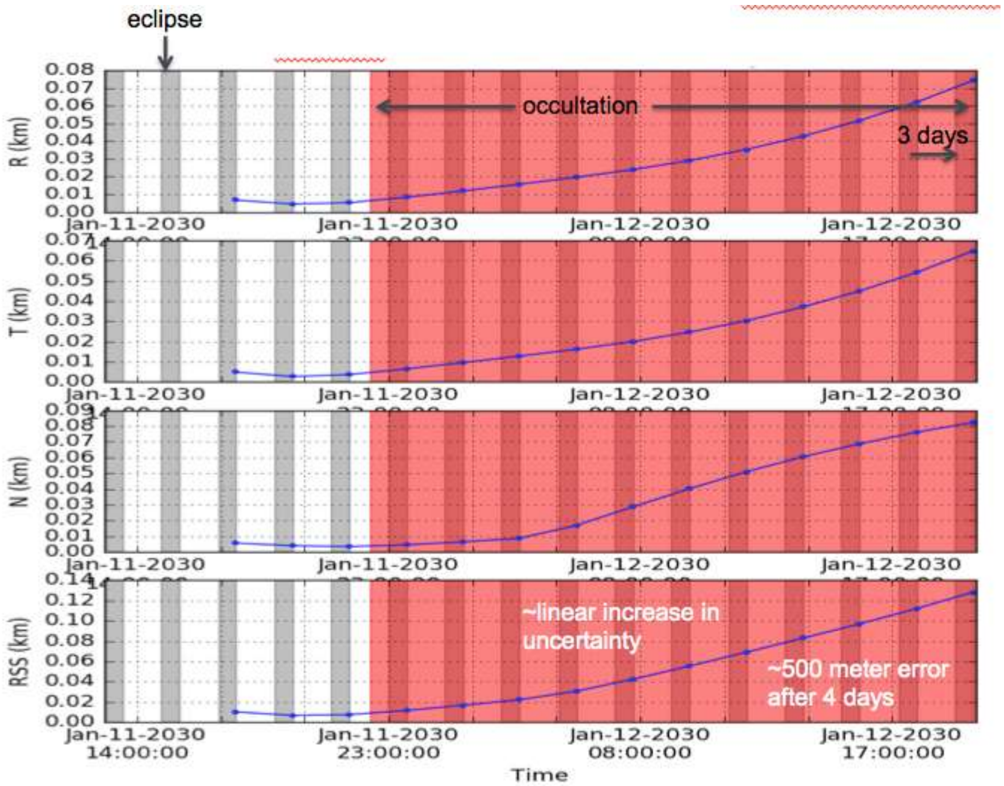

Figure 5. OS orbit knowledge degradation while in occultation

Radio Frequency Detection

An active Radio Frequency (RF) beacon onboard OS was examined to determine if it could, on its own, improve rendezvous operations with Orbiter. An RF beacon has several advantages over optical methods. These include relaxing the need for OS illumination and the availability of omnidirectional tracking. On the other hand, drawbacks include additional mass and power requirements.

The beacon study was performed with Monte[4], JPL's operational navigation software. Standard analysis techniques were used, with additional tool development to analyze spacecraft-to-spacecraft one-way doppler data. The analysis was automated to sweep over ranges of duty cycle and oscillator stability. In all, 1,984 figures of merit were computed and over 3 GB of data generated.

With a COTS oscillator, the frequency stability of a one-way RF beacon on OS does *not* provide precision navigation, so therefore it would not be a good substitute for orbiter imaging. An RF beacon does not even provide meaningful enhancement to imaging because, over time, the changing geometry increases the usefulness of optical angular measurements but RF data remains unchanged. See Table 3 for details.

Table 3. RF Beacon / Optical Rendezvous Summary

	Position Uncertainty Exit from 1st Eclipse	Position Uncertainty Exit from 2nd Eclipse

Sensor Types	RSS Position	Semi-Major Axis	RSS Position	Semi-Major Axis
1-Way RF Beacon Only	30-110 km	9-30 km	31 km	5 km
Optical + 1-Way RF Beacon	1-7 km	0.3-2.0 km	4.5 - 39 m	1 - 11 m
Optical Only	1-7 km	0.3-2.0 km	4.5 - 39 m	1 - 11 m
2-Way Doppler Only	<2 km	<5 m	52 m	Sub-meter
Optical + 2-Way Doppler	10's m	Sub-meter	10 m	Sub-meter

NEAR-FIELD NAVIGATION

The near-field regime is a transition region. In this paper it is defined to be a range between the Orbiter and OS of 10 km and 50 m. Distances less than 50 m are called the terminal phase, also known as retrieval phase, where the stalking spacecraft mates with OS. (The terminal phase is not described here.)

The near-field approach to OS is described in detail by Riedel[2,3]. Those references describe a procedure to close on the capsule within a distance of say, 50 m. Having reached that limit however, Orbiter and OS's relative positions will separate without active control because of the marked differences in their ballistic coefficients.

In this paper we're interested in exploring the capability of a low-thrust spacecraft to maintain a fixed standoff distance from OS – the station-keeping problem. With the ballistic coefficients we've noted above, the station-keeping ΔV control authority needed to maintain a 50 m separation under nominal conditions is the equivalent of 1 cm/s per day.

Station-keeping and Dust Mitigation

Of analytical interest is the opportunity to use the orbiter's low thrust plume to remove potential residual dust from OS and/or its immediate vicinity. While this is not known to be a requirement for the mission, the astrodynamic consequences of "pluming" are addressed in this work.

Pluming Assumptions. Eclipse season (worst-case orbit geometry: orbit is edge-on with respect to Sun). Orbiter is located 50 meters behind OS, at an equal radius. For this analysis, a constant thrust plume is initiated by Orbiter's ion propulsion system on a generic day at 12:19. Orbiter's plume is directed downtrack with an intial force on OS equal to 12 micro-newtons, imparting an acceleration on OS of 1.0 micron per second per second. Adam Nelessen developed a plume model based on the NEXT engine, shown in Figure 6[5]. Note: the OS pluming work was discontinuous in time with respect to the far-field analysis (*i.e.* the pluming study happened several months later). During that passage of time, parameters and assumptions for rendezvous operations changed. Specifically, Orbiter's mass increased to 3000 kg and OS mass decreased to 12 kg. Those assumptions are applied here to the near-field analysis (but were not retroactively applied to the far-field analysis nor to the nominal station-keeping ΔV budget). The ballistic coefficients used for the pluming analysis are: Orbiter = 275 kg/m^2, OS = 120 kg/m^2.

Pluming Results. The force from the plume on OS over time is shown in Figure 6. It's not constant for two reasons: 1. when passing into Mars' shadow, pluming ceases until the vehicles re-emerge into Sunlight, and 2. perturbation by the plume changes OS dynamics.

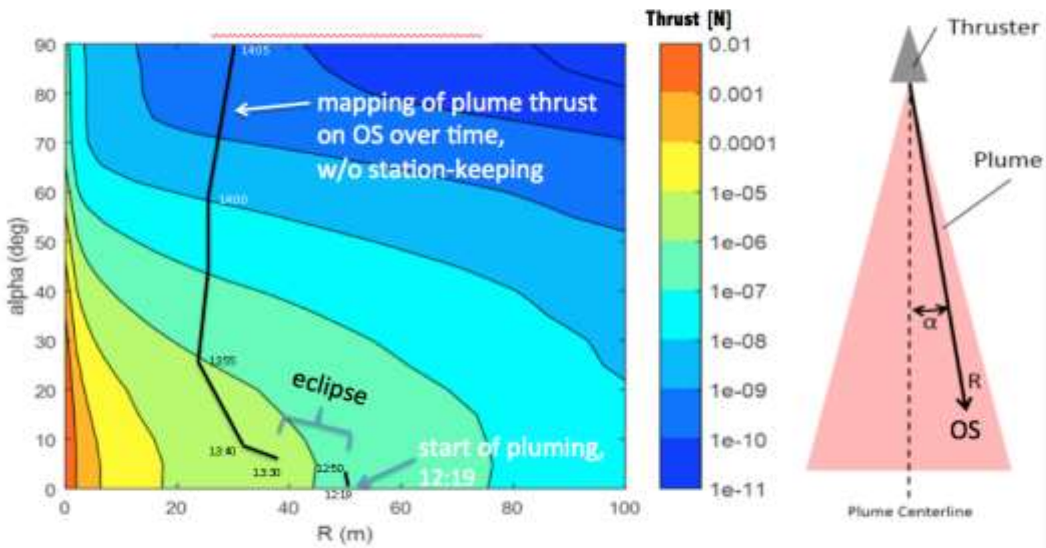

Figure 6. Plume Thrust Model

After one-hour of pluming, OS increases its semi-major axis (SMA) by 5 meters. Two-hours of pluming increases SMA by 38 meters. (Recall, Orbiter simultaneously maintains its *a priori* orbit using RCS thrusters.) Figure 7 shows what happens after two-hours of thrusting. The figure plots relative differences between Orbiter and OS in a cartesian Euler-Hill frame.

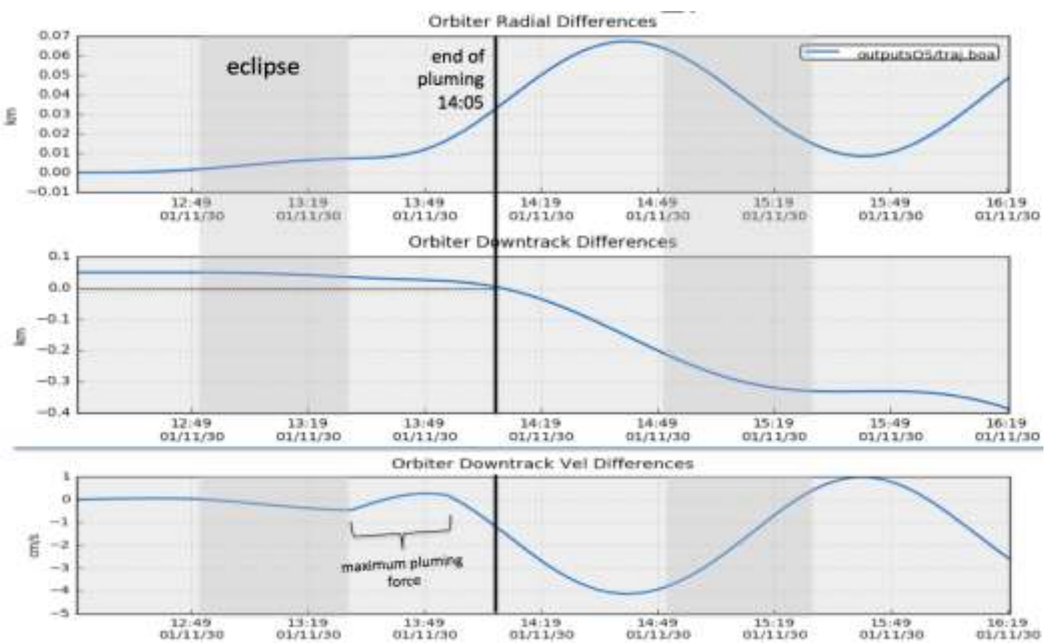

Figure 7. 2-hour Plume: Radial & Downtrack position; Downtrack velocity

Little relative change occurs in the first half-hour. Both vehicles have the same orbit radius and remain separated by 50 m. When the first eclipse is entered, pluming stops, but by the end of the eclipse discernable changes in the vehicle's relative positions are evident. Pluming effectively ends 33 minutes after emerging from the eclipse because OS passes overhead (above Orbiter), as can be seen in the middle plot of Figure 7. The plume accelerates OS, raising its altitude and thereby slowing it, thus allowing Orbiter to overtake OS (despite Orbiter's station-keeping activity).

Pluming Summary. Two cases are described in Table 4. In the first case Orbiter maintains its *a priori* orbit characteristics (does not care what OS is doing), and in the second case Orbiter maintains a constant 50-meter separation from OS while pluming it. In the latter case Orbiter actively responds to OS relative movement.

Case 1 plumes OS for 64 minutes, requiring an Orbiter ΔV of 1 m/s to station-keep (using the RCS system described in Table 2). Case 2 maintains a constant stand-off of 50 meters and a constant force of 1.2 μN. Pluming is active for 78 minutes (2-hours wall clock-time).

Table 4. OS Pluming

Case	Length of Pluming	SMA perturbation	Orbiter-OS Separation	Comments
1	64 min	38 m	OS passes overhead after 1¾-hrs, 35 m miss	Orbiter offsets plume thrust w/ RCS thrusters (maintains present orbit) RCS ΔV~1 m/s
2	78 min	10 m	maintains 50 m horizontal spacing	Orbiter offsets plume thrust w/ RCS thrusters & maintains relative position wrt OS, RCS ΔV~1 m/s (ΔSMA=10 m requires only 2 mm/s of ΔV)

SUMMARY

A notional Mars Sample Return rendezvous procedure has been described, emphasizing the far-field and near-field regimes. Two far-field methods for detecting the orbiting sample capsule are passive optical and radio beacon. Each has advantages, but the optical technique is less massive (easier for MAV to launch) and more robust (no need for batteries).

When in the near-field regime and waiting for the terminal phase to begin, under special circumstances Orbiter could point its ion-engine nozzle toward the orbiting sample and illuminate it with the beam. Two options were examined. 1. Orbiter could: Maintain its *a priori* orbit using station-keeping techniques. After two-hours of pluming, the orbiting sample increases its semi-major axis by 38 meters, rises in its orbit, and passes overhead and behind Orbiter. Or 2. Orbiter could: Actively follow the orbiting sample while pluming it and maintaining 50 meters of separation. After two-hours of pluming, the orbiting sample and Orbiter both increase their semi-major axes by 10 meters. The orbiting sample's semi-major axis increases more in the first case than the second because as it rises the distance between vehicles decreases, allowing the plume to impart greater force on the orbiting sample capsule as it draws near (as long as $\alpha < 90$ degrees).

The information presented about potential Mars Sample Return is pre-decisional and is provided for planning and discussion purposes only.

ACKNOWLEDGEMENTS

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APPENDIX

Rendezvous Operation Uncertainties

Error Source	*A priori* Error 1σ	Comments
MAV	50 km, 2 m/s	MAV location at OS release, spherical.
Orbiting Sample	50 km, 2.1 m/s	Epoch state after release, spherical.
Orbiter	20 m, 5 cm/s	Epoch state, spherical.
Atmosphere drag	25%	Orbiter only (atmosphere drag area = 5 m^2).
Solar pressure	20%	Orbiter only (solar pressure area = 62 m^2).
Momentum unloading	0.6 mm/s	Twice daily, spherical.
Gravity field (15x15)	MRO95a covariance	5X MRO95a used in analysis.
2-way doppler tracking	0.1 mm/s	Continuous when in-view.
Narrow Angle Camera	4.65 μrad	= 1 arc-second.
Target centroid	0.2 pixel	Typical centroid error is 0.1 pixel; using 0.2 pixel for conservatism.
RF duty cycle	1%, 70-s period	0.7 seconds on, 69.3 seconds off.
OS oscillator stability	1.0e-07	Performance range of Micro-computer Compensated Crystal Oscillator (MCXO)
OS oscillator frequency bias	1 kHz	Result is insensitive to bias error.
Orbiter oscillator stability	1.0e-12	Ultra-Stable Oscillator (USO).

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