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MARS MICROMISSIONS

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Abstract

A Mars micromission launches as an Ariane 5 secondary as early as November 1, 2002. One possible mission is a comm/nav micromission orbiter. The other possible mission is a Mars airplane. Both missions are enabled by a low-cost, common micromission bus design. The future of Mars micromissions has at least two micromissions per Mars opportunity in cooperation with CNES. Other destinations besides Mars are possible by small changes and technology infusion in the micromission system design detailed here. Micromissions enable a new class of science investigations and comm/nav orbiters due to low-cost, focused design.

1. Mars Micromissions Architecture

The current Mars Architecture covering missions through 2005 endorses micromissions as a way to increase scientific access to Mars for low cost. Starting with the 2003 Mars opportunity, a micromission launches to Mars as a secondary payload on the Ariane 5 in cooperation with the French space agency CNES. The decision concerning the first 2003 micromission type is not yet made. This first micromission will either be a Mars airplane arriving on Dec 17, 2003 (the centennial of the Wright brothers' historic flight at Kittyhawk, North Carolina) or the first communications/navigation orbiter for the Mars Network[5].

Beginning in 2005, at least two Mars micromissions (consisting of some combination of probe carriers, science orbiters, or communications orbiters) launch per Mars opportunity. The Mars Micromissions Project supplies the multi-purpose spacecraft to accomplish these missions for a per mission cost after the first mission in 2003 of less than $50M. A basic tenet of this low cost is that each mission shall not make major changes to the multi-purpose micromission spacecraft. And, only missions which fit within the capabilities of the spacecraft should be attempted.

Other science missions (for example, Venus, Mercury, asteroids, or small bodies) are possible with minimal changes to the common Mars micromission spacecraft bus. Technology demonstration missions in Earth orbit or interplanetary space could potentially use the micromission spacecraft. It is anticipated that many missions that fit the micromission requirements envelope will use this new capability.

Since there are many possible uses for the basic multi-purpose micromission spacecraft, US launch vehicles could potentially support micromissions. This capability hinges on the development of a secondary launch capability similar to the Ariane 5. A primary launch is also possible for a micromission. However, the added cost of this option could make a primary launched micromission much less attractive.

2. Mission Design

Micromissions utilize a secondary launch to Geosynchronous Transfer Orbit (GTO). The primary payload is two geosynchronous communications satellites. After flying around the Earth-Moon system for 1 - 6 months (see Figure 1 on the next page), the spacecraft swings by the Moon to obtain the correct Earth flyby geometry. After the powered Earth flyby, the spacecraft cruises to Mars (or other planetary target) just as a dedicated launch planetary mission would.

Launch

The launch site is Kourou, French Guiana. The launch period is from November 1, 2002 to May 1, 2003. A particular launch date depends upon the primary mission that the micromission is carried with. On a given launch date, the most probable launch time is midnight local time. A midnight launch results in noon apogee time. However, significant variations can occur to this launch time. The actual time of launch is

[5]"Mars Comm/Nav MicroSat Network", R.C. Hastrup et al, 1999 USU/AIAA Smallsat Conference, paper SSC99-VII-5.

governed by the primary communications satellite geometry constraints and operations considerations.

The local time of launch should not vary by more than +/- 3 hours around midnight. Further insight into compatibility of the micromission with the primary payload launch conditions is given in Figures 2 and 3 on the next page. Initial micromission manifesting will not occur until one to two years before launch. Final micromission manifesting will not occur before approximately one year before launch.

Figure 2 on the next page shows the early portion and Figure 3 shows the later portion of the 6 month 2002 Micromission launch period. The total delta-V indicated is for the deterministic Earth-Moon trajectory corrections only. Navigation delta-V requires an additional propellant allocation. Note that micromissions will not be able to choose the launch date, so the propellant subsystem must be able to accommodate the maximum delta-V required. Currently, Rosetta will be launching on an Ariane 5 in the January/February 2003 time frame. As a result, a 6 month launch period must be maintained to ensure adequate probability of successful launch. This 6 month launch period requires from 3 to 7 major delta-v maneuvers depending upon the launch date. Earlier launch dates require more burns and lunar flybys to align the Earth escape direction to target Mars.

The ASAP5 structure carries the micromissions to GTO. After primary payload separations, the micromissions deploy. In order to minimize the time spent passing through the Van Allen radiation belts, the spacecraft per-

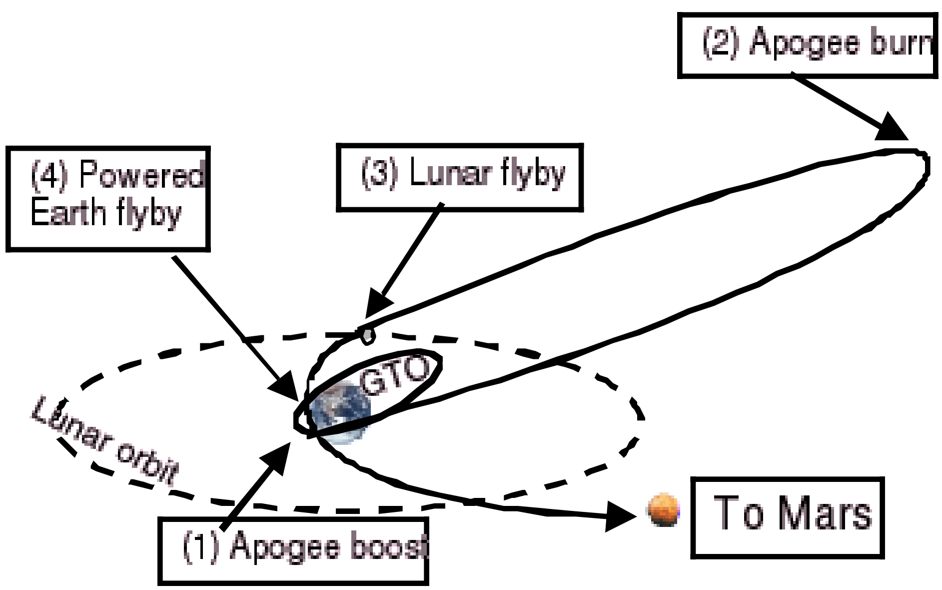

Figure 1 shows a simplified view of the 3 burn Earth-Moon system micromission trajectory. Not shown is the post-GTO series of orbits in the 1 to 5 day orbit-period range. This simplified view is not to scale. The lunar phasing orbit ranges in period from 10 to 60 days, depending upon launch date and time. For a 5 burn case, an additional lunar flyby (with additional required delta-V) is added to move the line of apsides towards the desired trans-Mars injection direction. For a 7 burn case, two additional lunar flybys (with additional required delta-V) are added above the 3 burn trajectory.

forms a series of perigee maneuvers as soon as possible to place the spacecraft in a 1 to 5 day period Earth orbit. Assuming up to 10 days spent in GTO leads to 40 passages through the Van Allen belts[6].

Cruise to Mars

During the cruise to Mars, the majority of the time is spent with the solar arrays Sun pointed. One to three statistical maneuvers clean up any errors from the powered Earth flyby. During cruise, it is expected that occasional checks will be made of the payload.

[6]The total radiation dose expected for the probe carrier mission is about 22 krad and the total radiation dose for the comm/nav orbiter mission is about 27 krad for a November 1, 2002 launch.

Figure 4 on the right shows the Earth to Mars Type 1 trajectory for the 2003 opportunity. The arrival date for the airplane is December 17, 2003. This date is the minimum delta-V case for the required trans-Mars injection on May 31, 2003. The minimum delta-V arrival date for an orbiter required to do a substantial Mars Orbit Insertion is December 26, 2003.

Probe Deployment

As the spacecraft approaches Mars, navigation uncertainties decrease. Sometime between one and seven days from atmospheric entry, the probe release sequence starts. The spacecraft will achieve a quiescent state to minimize probe deployment error. Then, probe release occurs. The minimum time between probe releases, if there is more than one probe, is 30 minutes.

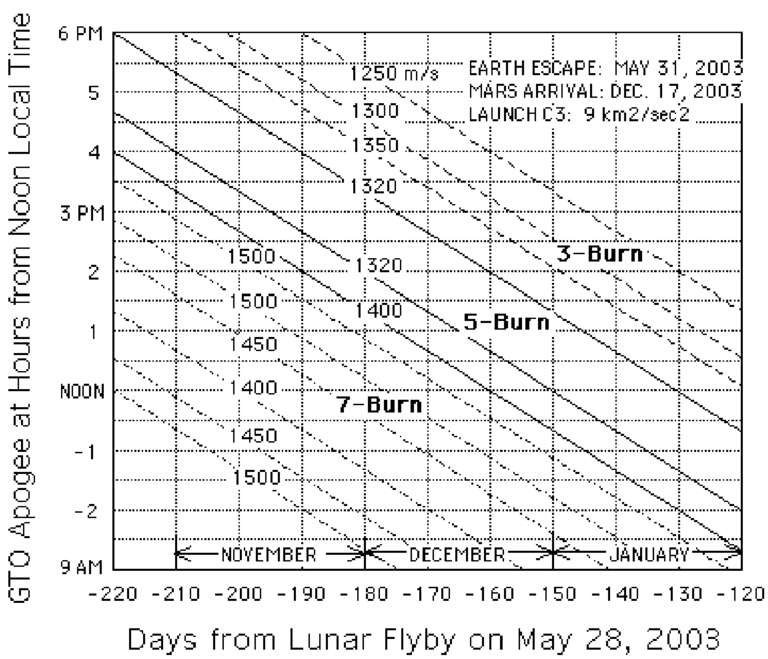

Figure 2 shows the Launch Period Performance from November 2002 — January 2003. The total delta-V indicated is for the deterministic Earth-Moon trajectory corrections only. Navigation, Mars deflection, and any Mars orbit insertion (for orbiters) delta-V requires an additional propellant allocation of from 150 to 1200 meters per second for the 2003 opportunity.

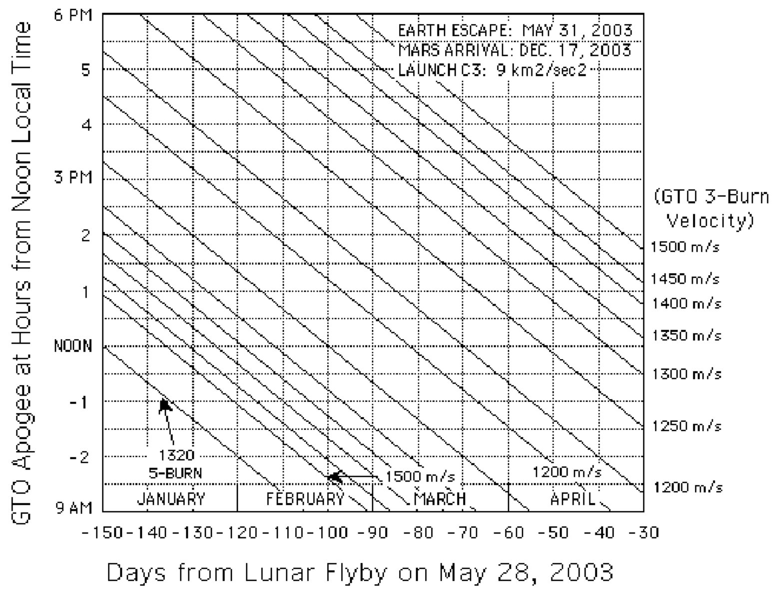

Figure 3 shows the Launch Period Performance from February to April, 2003. The total delta-V indicated is for the deterministic Earth-Moon trajectory corrections only. Navigation, Mars deflection, and any Mars orbit insertion (for orbiters) delta-V requires an additional propellant allocation of from 150 to 1200 meters per second for the 2003 opportunity.

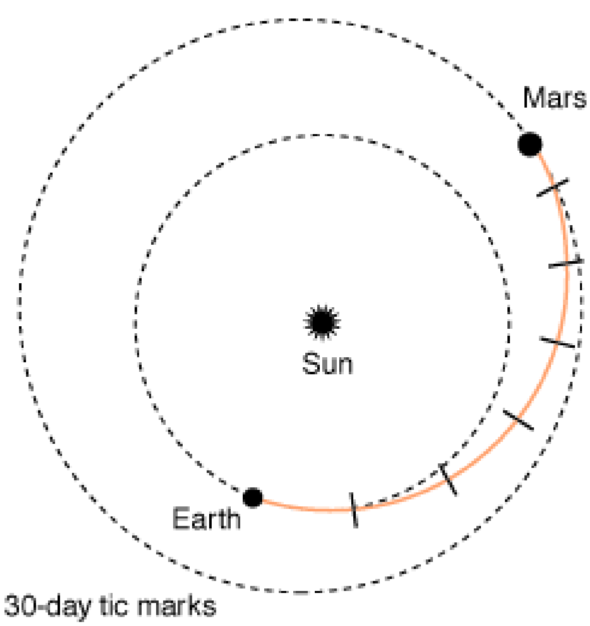

Figure 4 shows the 2003 opportunity Type 1 Earth-Mars Cruise Trajectory

Post-Deployment Activities for the Airplane Mission

After airplane probe deployment, the micromission spacecraft does a delta-V maneuver (or series of maneuvers) as soon as possible. The purpose of this maneuver is two-fold: one, to retarget the spacecraft from an impact trajectory with Mars, and two, to adjust the spacecraft trajectory so that airplane relay of data is possible. This enables the spacecraft to become the prime data return relay link with the aircraft.

The spacecraft will be pointed at the center of the airplane flight error ellipse while the airplane is transmitting data. After collecting data during the airplane flight, the spacecraft relays this data back to Earth via the Deep Space Network over several weeks.

Com Orbiter Mars Orbit Insertion (MOI) and Aerobraking Phases

After the cruise phase, a series of small navigation targeting maneuvers are performed. These maneuvers occur between 30 to 1 day before MOI. At MOI, a large burn puts the spacecraft in a loosely captured, retrograde Mars orbit. The period of the orbit is 3 sols with a periapsis of 250 km. The retrograde orbit is necessary since long eclipses of almost 7 hours can occur with prograde orbits. This orbit is maintained until such time that investigations at Mars (Mars Lander/Rover 2003, Mars Aircraft Mission, tracking of the Mars Sample Return orbiting canister) no longer need to relay data. Then, the spacecraft aerobrakes down to its operational orbit over a period of about 4 months. After reaching operational orbit, minimal orbit maintenance is required.

Total Required delta-V for the Probe Carrier and Com Orbiter 2002 Missions

The micromission spacecraft mass consists of approximately 1/3 dry mass and 2/3 propellant. The majority of the propellant is for the large deterministic maneuvers required to properly escape the Earth - Moon system from GTO. This unusually large percentage of propellant allows the spacecraft to accomplish a focused mission as a secondary payload.

The micromission spacecraft must be able to provide large trajectory correction delta-V's as well as much smaller attitude control authority over the life of the mission. Table 1 on the next page provides the current scenario for the location and magnitude of the deterministic delta-V's as well as the total delta-V required for the 2002 launch opportunity.

Future Mars Mission Considerations

The 2005 launch opportunity requires a total delta-V of 1900 meters per second for the Probe Carrier and 3000 meters per second for the Communications or Science Orbiter. The design of the multi-purpose micromission spacecraft for the 2003 mission launch shall either be able to accommodate the added propellant load for the 2005 mission or include sufficient design flexibility to allow infusion of lightweight technologies to meet the 2005 delta-V requirements at a small incremental cost. For the required deterministic total delta-V characteristics of 2003, 2005, and 2007 see Table 2 below.

Table 1: ΔV Required for Future Mars Micromission Opportunities

Mars Opportunity	Probe Carrier ΔV (m/s)	Orbiter ΔV (m/s)
2003	1650	2700
2005	1950	3000
2007	1850	2700

For missions beyond 2003, design flexibility and technology infusion will be important for allowing an increased payload mass allocation.

Table 2: Δ-V Magnitude and Placement for 2003 Airplane and Com Orbiter Micromissions

Required Trajectory Correction	ΔV (m/s)	Comments
1st burn to leave GTO	250	Increases apoapsis altitude
2nd burn	250	2nd burn of phasing orbit
3rd burn	250	3rd burn of phasing orbit
Apoapsis 1	40	1st loop apoapsis
Earth flyby 1	60	1st loop Earth flyby
Apoapsis 2	40	2nd loop apoapsis
Earth flyby 2	60	2nd loop Earth flyby
Apoapsis 3	40	3rd loop (final) apoapsis
Trans-Mars injection at Earth	500	Injection to Mars
Total Earth-Moon Nav ΔV	100	
Total Earth-Mars Cruise Nav	20	
Post-airplane probe release ΔV	40	Required to miss Mars and relay data
Total Probe Carrier ΔV	**1650**	Below this line is Com Orbiter only
Mars Orbit Insertion	900	3 sol orbit, gravity losses incl, 250 km periapsis
Aerobraking pop-up	10	Contingency to temporarily stop aerobraking
To 800 km circular orbit	160	
Orbit maintenance	20	For life of com orbiter mission
Total Com Orbiter ΔV	**2700**	Does not include post-probe release ΔV

3. Multi-Purpose Micromission Spacecraft System Design and Technology

The Ariane 5 Structure for Auxiliary Payloads (ASAP5) allows single or dual slot launches of 100 - 200 kg. Planetary missions currently require the dual (or "twin") configuration due to ΔV requirements to escape the Earth. The micromissions are unusually shaped due to volume and dual attachment constraints imposed by the ASAP5 structure. See Figure 5 on the next page for a view of the ASAP5 qualification structure. The qualification structure shown was built to qualify both micro and mini capabilities. Consequently, only the smaller attachment points located around the outside of the structure apply to the micromissions.

A multi-purpose spacecraft is appropriate for micromissions. Many of the requirements of a probe carrier, a science orbiter, and a com/nav orbiter are met by a multi-purpose, 3-axis stabilized, bi-propellant spacecraft (see Table 3 below for a high level summary of requirements). Of course, a multi-purpose micromission spacecraft cannot satisfy all potential Mars mission types. Judgement must be exercised to ensure that the multi-purpose spacecraft changes very little from mission to mission. Small changes will allow the recurring cost of the spacecraft to be kept to a minimum.

Table 3: Multi-Purpose Micromission Spacecraft Requirements

System Requirements	Specification
Storage lifetime ("tanks dry")	3 years
Pointing control (3 sigma per axis)[a]	+/- 1 degree
Pointing knowledge (3 sigma per axis)	+/- 0.1 degree
X-band uplink Bit Error Rate	$1x10^{-5}$
X-band dowlink Bit Error Rate	$1x10^{-6}$
Command rate (emergency @ 2.7 AU)	7.8125 bits/sec
Command rate (operational @ 2.7 AU)	125 bits/sec
Engineering data rate (emergency @ 2.7 AU)	10 bits/sec
Minimum transmission time (at maximum power)	8 hours
Safe mode survival time (minimum)	14 days

a. Except during aerobraking where the spacecraft is passively stable.

Figure 5 shows the ASAP5 Qualification Model at Matra Marconi Space in Stevenage, England (photo courtesy of CNES, Arianespace, and Matra Marconi Space).

JPL worked feasibility studies since June of 1998. Before that time, JPL submitted four Discovery proposals using ASAP5 capabilities for Mars missions. The goal of these studies was to investigate the applicability of the ASAP5 "twin" capability for Mars missions. As the studies matured, it became evident that the multi-purpose spacecraft could be kept simple, reliable, and low cost. At the same time, the science community and Mars Network realized that micromissions could open up access to Mars and other exciting planetary destinations as well as provide a low cost way to build a com/nav network at Mars.

JPL initiated five industry studies in early 1999. The message of these industry studies was that the concept (including cost, schedule, and multi-purpose capabilities) of Mars micromissions is viable. Currently, JPL is in the process of selecting an industrial contractor to build the first multi-purpose micromission spacecraft. This process will result in a contractor selection in September, 1999.

The JPL and industry studies resulted in several technical conclusions:

- mass will be the primary driver (the current launch mass limit is 220 kg).
- telecommunications from the surface of Mars and to Earth are challenges.
- future opportunities beyond 2003 require the infusion of new technologies to reduce mass.

The JPL feasibility studies developed a point design shown in artist's concept below in Figures 6 and 7. The unusual shape results from having to fit in one quarter of the ASAP5 ring (see Figure 8). One design driver is that as much mass as possible must be placed over the each of the two ASAP5 attach points. This leads to the two bi-propellant tanks being placed directly over the attach point. There must also be room to carry one large probe of about 80 cm diameter, two smaller probes of about 65 cm diameter, or 3 to 4 probes of 40 cm diameter (similar to the Deep Space 2 Mars probes currently going to Mars aboard the Mars 98 lander). A com/nav orbiter dictates volume for an 80 cm diameter high gain X-band antenna for relay of data back to Earth and larger propellant tanks for the increased delta-V. Science orbiters require a high gain antenna and volume for science instruments as well as the larger propellant tanks.

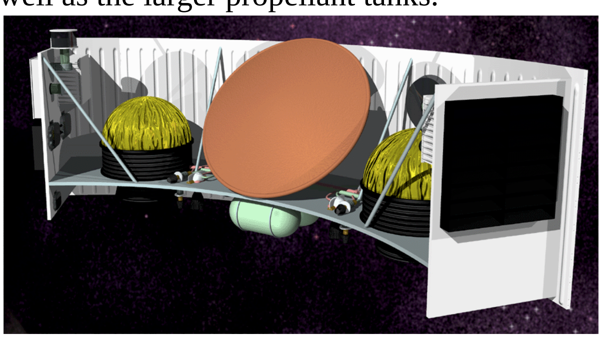

Figure 6 shows an artist's concept of the probe carrier. The object in the center is an example of one large probe. The tanks are to either side of the probe. The tank below the probe is a pressurant tank for the bi-propellant system. The end structural pieces contain louvers for thermal control and spacecraft electronics. The white "corrugated" structure behind the probe is the backside of the solar array structure.

Figure 7 shows an artist's concept of the com/nav orbiter. The tanks are in the same location as figure 6. The end structural pieces contain louvers for thermal control and spacecraft electronics. The "corrugated" structure behind the probe is the backside of the solar array structure. The circular object is the deployed high gain antenna. It rotates upward from between the propellant tanks to deploy on one hinge.

Figure 8 shows a rough layout of the Ariane 5 during a micromission launch. The artist's concept shows the upper stage and fairing as the fairing begins to fall away. The lower part contains the ASAP5 structure with secondary micromission mounted on top. The two primary payload communications satellites are stacked on top. This is preliminary artwork from a micromission animation that will be finished in the fall of 1999.

As discussed earlier, future Mars opportunities beyond 2003 require low-mass, high performance technologies. Preliminary analyses yield the following key technology development areas:

- low mass, high performance bi-propellant propulsion systems
- low mass power systems
- low mass surface/atmosphere probe communications
- low mass structures
- in-situ comm/nav orbiter capabilities
- ballute aeroassist for orbiter capture

These technology areas insure that micromissions continue to adapt to focused science missions. A key challenge of technology infusion is to keep the cost of the program low while increasing capabilities over time.

4. Micromissions to Other Destinations

Small changes to the Mars micromission multi-purpose spacecraft design enable missions to other destinations such as Venus, Mercury, Near-Earth and Main-Belt Asteroids, the Moon, and the Earth-Sun Libration points. Missions to these targets can take advantage of a Mars micromission spacecraft "production line" which can provide a low-cost spacecraft bus. Since the modifications for these missions are small, the additional cost beyond the recurring micromission spacecraft cost is also small. First time modifications add approximately $10 million to the cost of the spacecraft, with the recurring cost for spacecraft with the same design changes being about $1 million.

The principal modifications to the basic Mars micromission spacecraft are for missions to Venus or Mercury and involve modifying thermal properties. For both targets, additional thermal blankets for solar insulation are required and a fraction of the solar cells in the solar array are replaced with Optical Surface Reflectors (OSRs) to maintain array temperatures. In addition, for a Venus orbiter, additional radiators are added to compensate for Venus thermal loading and an actuator is added to point the HGA. For Mercury missions a fold-out solar panel is added to maintain solar array temperatures by allowing backside radiation to deep space and off-normal array pointing.

The delta-V requirements for Venus are similar to those for Mars, while a probe delivery to Mercury using a Venus Gravity Assist (VGA) requires approximately 800 m/s more delta-V than a probe delivery to Mars. The larger delta-V requirements for Mercury missions result in significantly lower payload capability for Mercury probe missions and preclude a Mercury orbiter mission. Missions to Main-Belt asteroids, by utilizing a VGA, have similar delta-V requirements to Mars probe carrier missions. Examples of Main-Belt asteroid opportunities include a July/August 2002 launch to Ceres or Pallas and a March 2004 launch to Juno or Vesta.

Table 4 on the next page summarizes the micromission spacecraft payload capabilities for some possible missions and launch opportunities. In addition to the missions listed, the Mars micromission spacecraft could be utilized to demonstrate space technologies, such as solar sails. As these examples show, the micromission spacecraft bus can provide a cost-effective capability to perform a wide variety of missions.

Table 4: Mars Micromission Capabilities to Other Destinations

Destination	Launch Year	Payload to Approach [a] (kg)	Payload to Orbit (kg)
Mars	2003	42[b]	6
	2005	40 - 45	5 - 10
	2007	45 - 50	10 - 15
Venus	2002	30 - 40	0 - 5
	2004	40 - 50	0 - 5
	2005	45 - 55	5 - 10
	2007	50 - 60	10 - 15
Mercury	2002	10 - 20	None
	2004	0 - 10	None
	2005	5 - 15	None
Main Belt Asteroids	Same As Venus	30 - 50	None
Near-Earth Asteroids	Any	50 - 60	None
Moon	Any	80 - 90	30 - 50
Earth - Sun Libration Points	Any	Not Applicable	60 - 70

a. Includes Entry, Descent, and Landing Hardware

b. for the Mars Airplane Probe, 3 kg carried on the spacecraft for probe in-situ data relay to probe carrier

5. Summary

Mars micromissions begin with a launch as early as November 1, 2002. This mission will be either a Mars airplane or a com/nav orbiter. Either mission provides challenges such as limited launch mass and volume, a flexible and lengthy launch period, cost containment, and providing capabilities useful for future micromissions. If micromissions live up to expectations, a new era of low cost international planetary exploration will begin. Lastly, many promising mission concepts to destinations other than Mars can benefit from the low cost, multi-purpose micromission spacecraft.

6. Acknowledgments

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