Interplanetary Small Mission Studies[i]

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Abstract--Small missions can play a large role in future robotic space exploration. While these missions cannot accomplish the vast scope of science objectives achieved by large missions such as Mars Sample Return or Cassini, they offer opportunities to explore smaller, but pertinent, science goals for significantly reduced total mission cost.

The Jet Propulsion Laboratory's Advanced Projects Design Team (Team X) has conducted several mission studies to explore the feasibility of scientifically significant small interplanetary missions. These mission studies encompassed various targets (Mars, Earth's Moon, Venus, the Sun) using several scientific payloads (radar, imagers, radiometers). These missions can also perform other functions such as probe/balloon delivery or communications relay for landed missions. The studies considered a range of secondary payload launch vehicle options. This paper will highlight the results from these studies and discuss how the concurrent engineering environment of Team X lends itself to pre-phase A concept investigations.

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1. INTRODUCTION

Interplanetary missions provide a vast array of exciting mission options, but require large commitments in terms of key resources (schedule, budget, etc.). Performing small-scale interplanetary missions on a secondary payload platform can achieve many of the same objectives that make interplanetary missions so attractive while reducing the resource commitments. Finding *feasible, scientifically significant* missions that fall into this "interplanetary small mission" class was challenging and required flexibility in mission/science objectives, technical design choices, and "level of acceptable risk."

As launch vehicle providers (both foreign and domestic) explore options for increased numbers of secondary launch opportunities, the possibilities for small missions increase. The ability to launch small interplanetary missions for a fraction of the cost is a primary benefit of secondary missions. Secondary missions can also be used to divide up larger missions into a number of smaller missions. This allows project risk to be spread over a wider array of missions, as losing one spacecraft out of several does not constitute a total project loss. Instruments that are flown together are not always optimized for the same orbit. Therefore, splitting science instruments between spacecraft also allows for more precise scientific targeting. These and other benefits provide valuable options for mission planners.

The goal is to present several viable options for small "secondary payload class" missions, explain how these missions differ from traditional primary missions, and show specific metrics which demonstrate the attractiveness of the small mission platform.

Design Envelope

The design envelope itself was quite challenging. Most launch vehicles provide only very small volume and mass allocations for secondary payloads. In addition, reducing

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available secondary launch opportunities, thereby increasing the level of confidence in choosing a secondary launch. While costs for secondary launch opportunities are a fraction of those for primary launches, the adage "you get what you pay for" applies. Secondary payloads cannot demand launch slips, orbit insertion parameters, etc. These are factors at the discretion of the primary launch vehicle customer. With this in mind, Team X chose to pursue a small design envelope of 500kg total mass, inside a cylindrical volume of approximately 1.5m high, 1.0m diameter. Figure 1 shows the configuration of this envelope within a launch vehicle shroud.

Launch Vehicle Selection--Team X obtained information from Space Operations International (SOI) to better understand secondary payload launch vehicle envelopes [1]. The major goal was to study mission concepts that could fit within several launch vehicle secondary payload slots, again increasing the number of possible launch opportunities.

The following diagram illustrates the chosen envelope, which fits within published secondary payload allocations for several launch vehicles. As shown, the spacecraft would fit inside a standard launch vehicle adapter, underneath the primary payload.

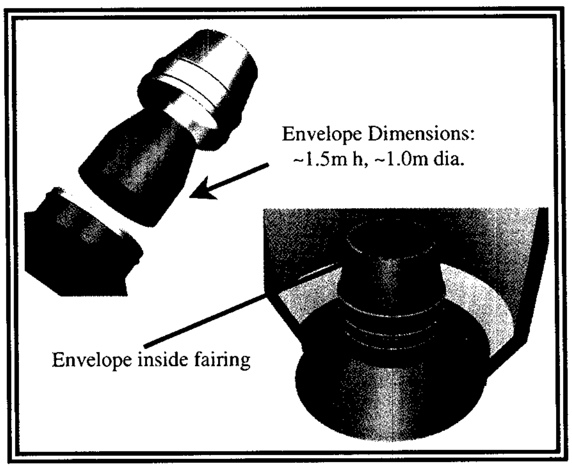

Figure 1 - Secondary Payload Envelope

Cost Cap Considerations--A specified cost cap usually bounds studies performed by Team X. This limit aids mission architects in selection of technologies (is the added cost of new technology worth it?), redundancy/spares approach, and development schedule. While cost is not the only important factor, it is a primary driver. One of the central goals when conducting these small interplanetary mission studies was to reduce the cost cap for a "typical" interplanetary mission by 50%. This factor of reduction was chosen to offset the scaling back of science objectives required to fit within the micromission design space.

Study Goals & Processes

Team X provides a unique, concurrent engineering environment to allow for rapid, yet comprehensive evaluation of pre-phase A mission concepts. The Team X environment was an ideal venue for these small interplanetary mission studies. A dedicated team X session was conducted for each study, allowing for evaluation of system-level trades on a case-by-case basis. Individual study goals (such as science investigation objectives, etc.) could be considered and stacked up against the overall mission concept goals, which were defined similarly for all studies.

Study Goals--Small missions can play a unique role in future robotic space exploration. While these missions cannot accomplish the vast scope of science objectives achieved by large missions such as Mars Sample Return or Cassini, they offer opportunities to explore smaller, but pertinent, science goals for significantly reduced total mission cost.

The overall goals for this series of studies were defined clearly as the following:

- Show feasible small interplanetary mission concepts, to be launched on a secondary payload platform, that reduce total mission cost by *at least* a factor of 2.
- Preserve scientifically significant payloads, despite the limited design envelope, to show viability of the interplanetary mission concept.
- Investigate mission concepts considering a range of interplanetary targets including Earth's Moon, Mars, and Venus.
- Document specific metrics to weigh the value of these small mission concepts against (1) other small mission concepts, and (2) traditional interplanetary mission concepts.

Concurrent Engineering/Study Process—This paper presents work from studies conducted over several weeks. In a traditional, sequential design process, performing this number of studies in just a few weeks would likely be impossible. Team X, however, implements a concurrent engineering approach to mission concept evaluation and pre-phase A design. Within a very short period of time, Team X can consider, implement, evaluate, and recommend acceptance/rejection of numerous ideas, with a relatively high level of fidelity. The format of Team X vastly increases the efficiency of the design process, and made the work performed for these small interplanetary mission studies possible. Figure 2 highlights the advantage of a concurrent engineering process [2].

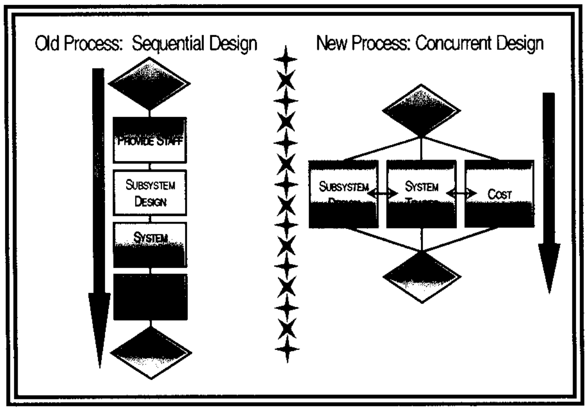

Figure 2 - Concurrent Engineering Process

Team X is comprised of experienced engineers from a range of subsystems working in parallel to develop and evaluate a spacecraft system-level design. Each subsystem engineer is an "expert" in his or her dedicated field (such as cost estimation, telecom hardware, mission design, programmatics, etc.) and brings considerable technical prowess to the team. This aids in rapid system-level trade evaluation since each team member can speak intelligently for his or her subsystem. Figure 3 shows how Team X is organized.

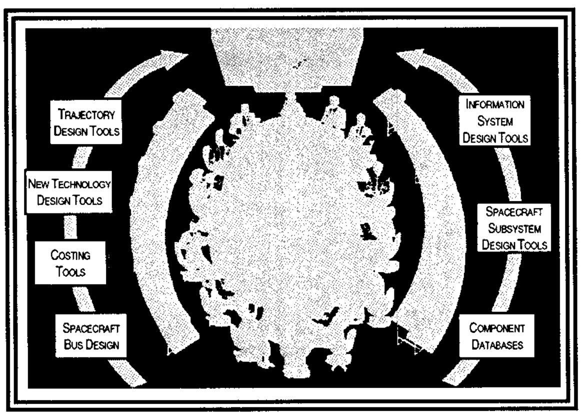

Figure 3 - Team X Organization

Within the Team X framework, the studies conducted during this investigation were performed following a standard Team X study flow—save a few exceptions. Typically, for interplanetary studies, single-string systems are found to be too risky (given the high cost). However, there are significant recent examples of singel string interplanetary missions such as Mars Pathfinder and Deep Space 1. Due to the significantly lower cost associated with these micromissions, a higher level of risk was acceptable. Given this assumption, subsystem designers had more freedom to select "risky" solutions when considering trades. Details of such decisions will be highlighted in the following section.

2. MISSION STUDIES

The objective of this work was to provide a database of studies considering the micromission platform. Multiple studies were conducted for several different targets—the subset discussed herein covers the majority of the candidate targets, and highlights the major issues associated with each dedicated target. Some targets were more challenging than others, and presenting them together allows quick comparison between mission scenarios.

The first micromission study conducted was a lunar orbiter mission to the Earth's Moon. This mission was designed to support current science interest in the search for water on the Moon, and show that significant advances in this arena can be achieved with a small mission. The next small mission study presented here is a Venus Probe carrier—designed to deploy a probe upon flyby at Venus (an attractive option for in-situ science). Finally, two studies are described which consider Mars as a target. This destination was by far the most challenging in terms of "fitting" the micromission concept. The two studies highlighted are a Mars balloon mission designed to showcase an innovative science concept (the Mars balloon) on a secondary payload platform, and a Mars Science Orbiter--designed to recover lost science from the Mars Climate Orbiter failure.

Lunar Science Orbiter

Earth's Moon is an interesting target. With scientific discoveries by recent lunar missions such as Clementine and Lunar Prospector, the opportunities for valuable science are enticing. This mission has major science goals focused on the search for, and mapping of, water on the Moon. The design team chose a unique approach for achieving this objective, and the final design reflects an inexpensive, yet viable method for water detection/mapping.

Mission Overview--The Lunar Science Orbiter Micromission follows a standard micromission trajectory—this means launch as a secondary payload to geosynchronous transfer orbit (GTO), followed by a series of propulsive maneuvers (escape, lunar transfer, lunar orbit insertion) designed to achieve a 100 km circular lunar orbit. To position the spacecraft for optimal science gathering, a final propulsive burn is used to establish a 15km x 200km orbit with periapsis at the Moon's south pole. The spacecraft spends one month in this orbit, then performs a maneuver to reorient the orbit so that periapsis occurs over the north pole. All primary science goals can be achieved in this two-month operational science period.

Science Objectives--The primary science goal of this mission is the detection of water at the lunar poles. Water is a necessary resource for human life, and detection of large quantities on Earth's Moon would lend support to ideas for future human colonization of the Moon. If a sufficient lunar water supply exists to support a human settlement, the process of establishing that settlement would become much less complicated.

Upon detection of water in the lunar polar regions, the mission will go on to map the concentrations of water in the pertinent regions. In addition, the instruments will gather information related to the quantity of water found at each pole. The mapping of the polar surface will be done using ice-penetrating radar. The spacecraft will also carry a small imager with a resolution of 10m to correlate with radar data.

Major Design Trades--During the course of the design sessions for this mission several design trades were considered. Major trades that had significant impacts on the final design are discussed below.

The first major trade considered concerned the primary science mission. The science mission called for mapping of the lunar surface, an objective that could be achieved with both high-resolution imaging and radar instruments. The original baseline carried a high-resolution imager, which required illumination of dark spots (craters) on the lunar surface. This was to be accomplished with a large deployable reflector. This design proved much too complicated and far too expensive for a micromission-class scenario. The elements required to support a high-resolution imaging system placed heavy burdens on the Attitude Control System (ACS), and were ultimately too massive. A simpler radar instrument was suggested in place of the imager. The radar instrument did not degrade the science, and made the mission much more viable for a micromission platform. The final design incorporated the radar instrument.

Another major trade came in the propulsion system. Most micromissions studied by Team X have required a bi-propellant propulsion system due to the large ΔV usually required for interplanetary missions launched as secondary payloads. A mono-propellant system is significantly more desirable for small missions because it is more than 50% cheaper than a bi-prop system. The original baseline included a bi-prop system (inherited from a history of previous micromission studies) to minimize the mass of the propulsion system. As the session progressed, however, mass did not emerge as a design driver. The spacecraft was well under the launch vehicle capability of 500kg, even with the standard Team X dry mass contingency of 30%. To take advantage of cost savings, a mono-prop system was implemented in the final design. This change consumed most of the "extra margin" against the launch vehicle, but still maintained some margin in addition to the 30% contingency.

Finally, there was a trade considering the desired frequency band for the Telecom system, X-band or S-band. The S-band system was found to cost half as much as an X-band system but involved a higher risk element due to questionable availability of S-band ground system hardware during the timeframe of this mission. X-band was chosen even though it was more expensive, as the risk of a loss of S-band capability outweighed the extra cost associated with the X-band system.

Final Design Mass/Power Summary--

Table *1* summarizes the flight system designed during the Team X session for the Lunar Science Orbiter Micromission. The table shows that the design has margin against the launch vehicle allocation even with the 30% dry mass and power contingency. A large percentage of the spacecraft mass is propellant, which would be lower with a bi-prop system, but would still represent a majority of the spacecraft mass. This table represents a feasible design from a mass, power, and mission design perspective.

Discussion of Study Results--The Lunar Science Orbiter Micromission was a successful example of the small interplanetary mission concept. With a small but significant science investigation, this mission fit within the design envelope. Physical configuration had to be considered, as fitting within the provided volume envelope was challenging. This volume restriction dictated that the antenna dish be folded back at launch and the spacecraft's longest points be in the center of the spacecraft. This mission fits in the launch vehicle envelope from both a mass and volume perspective. Figure 4 shows the candidate configuration for this mission concept. The CAD drawing shows the body mounted solar arrays along with the delta-v thrusters.

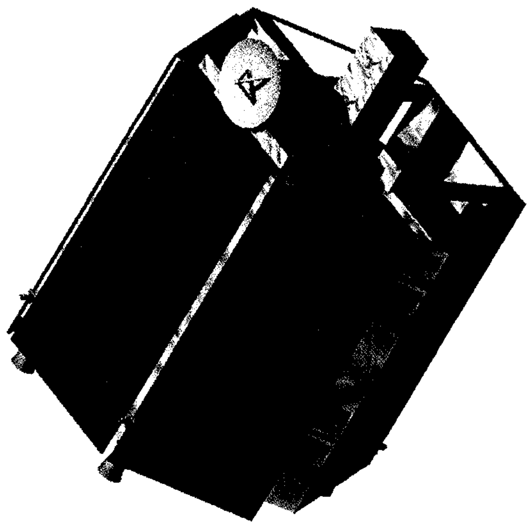

Figure 4 - Lunar Science Orbiter Configuration

All the science goals presented are met by this spacecraft with a *40% lower* cost than would be possible with a traditional primary launched spacecraft.

Table 1 - Lunar Science Orbiter Mass/Power Summary. This is the standard Team X system sheet. The upper portion contains system level requirements. Note that Radiation Total Dose is based on behind 100 mils of Al with no RDM added. Science BER is the required Science Bit Error Rate. The technology cutoff is the year in which Phase C/D begins, all technologies must be at Technolgy Readiness Level (TRL) 6 by this point. The data storage requirement is based on the amount of storage needed to hold all the data if a telecom pass is missed. Note, systems bookkeeps five power modes they are reassessed for each mission. And For S/C mass means that the propellant system has been sized for that spacecraft wet mass.

Lunar Science Orbiter Micromission

Orbiter Element

SYSTEMS WORKSHEET
Analyst: Matt Johnson
Start Date: 8/8/00

Stabilization - cruise	**3-Axis**		Pointing Direction - cruise	**Sun**				
Stabilization - science	**3-Axis**		Pointing Direction - science	Near Nadir				
					Max probe sun distance	**1**	AU	
Pointing control	**515**	arcsec	Radiation Total Dose, krad	**10**	Inst Data Rate	**256**	kb/s	
Pointing Knowledge	**200**	arcsec	Science BER	**1.00E-06**	Data Storage	**8.0**	Gb	
Pointing Stability	**60**	arcsec/sec	Redundancy	**Single String**				
Determined by:	**Radar**				Maximum Link Distance	**400000**	km	
			Technology Cutoff	**2003**	Return Data Rate	**100**	kb/s	

	Mass Fraction	Mass (kg)	Mode 1 Power (W) *Science w/o Transmit*	Mode 2 Power (W) *Transmit w/o Science*	Mode 3 Power (W) *Eclipse (45 min TBR)*	Mode 4 Power (W) *TCM (20min)*	Mode 5 Power (W) *Launch (2hrs)*	NASA TRL *Today*
Payload								
Instruments	9.5%	12.3	26.0	0.0	10.0	0.0	0.0	8
Payload Total	9.5%	12.3	26.0	0.0	10.0	0.0	0.0	
Bus								
Attitude Control	2.4%	3.1	18.3	15.2	15.2	16.2	10.8	5
Command & Data	5.9%	7.7	35.1	35.1	18.1	18.1	18.1	8
Power	12.1%	15.7	15.7	13.9	11.0	13.8	14.2	6
Hydrazine Prop	22.8%	29.4	9.5	9.5	9.5	131.8	33.5	9
Structure	28.4%	36.8	0.0	0.0	0.0	0.0	0.0	6
S/C Adapter	5.8%	7.5						
Cabling	7.0%	9.1						
Telecomm	2.8%	3.7	13.6	28.6	13.6	28.6	28.6	0
Thermal	3.2%	4.1	19.0	19.0	19.0	19.0	19.0	6
Bus Total		117.0	111.1	121.2	86.3	227.5	124.2	
Spacecraft Total (Dry)		129.3	137.1	121.2	96.3	227.5	124.2	
Mass/Power Contingency		38.8	41.1	36.4	28.9	68.3	37.3	
Spacecraft with Contingency		**168.0**	**178.3**	**157.6**	**125.3**	**295.8**	**161.5**	
Hydrazine	63.9%	297.2	For S/C mass =	471		Delta-V1	2150.0	m/s
Spacecraft Total (Wet)		**465.3**						
L/V Adapter		29.0						
Launch Mass		**494.3**						
Launch Vehicle Capability		**500.0**	**Secondary**		Launch C3	0		9
					Fairing type	standard		
					Fairing dia., m	1		
Launch Vehicle Margin		**5.7**	**1.1%**					

Contingencies	Mass	Power
Instruments	30%	30%
Other	N/A	N/A
S/C, dry	30%	30%

Venus Probe Carrier

Venus is about the same size as Earth, and is a compelling target due to its proximity, mysterious evolution, and harsh atmospheric and surface environments. Venus has a thick, poisonous atmosphere of carbon dioxide and sulfuric acid. The immense heat and pressure at its surface make this planet completely uninhabitable. From 1990 to 1994, NASA's Magellan spacecraft used a distinct radar instrument to reveal Venus's surface during its mission. To extend Magellan's measurements with an in-situ investigation, a small probe mission to this Earth neighbor was chosen as the desired baseline. The intent is to capitalize on the small mission platform (inexpensive, yet capable) for exciting planetary science.

Mission Overview--The Venus Micromission Probe will use a trajectory similar to that discussed in the Lunar Orbiter discussion above, starting at GTO and progressing through the Earth-Moon system. However, it swings by the moon and through a powered Earth flyby to send it on its way to Venus. There are two elements to this mission, a probe and a carrier. On approach to Venus the carrier uses a spin table to spin-stabilize the probe before separation. The probe then descends through the Venus atmosphere, and the carrier lifetime ends after separation (probe delivery is its only mission). The probe must survive only until it reaches an atmosphere of 1 bar. Its most significant measurements are taken at this pressure, which occurs at approximately 50km above the surface. Science data taken below this altitude is not required and will be considered an added mission benefit.

Science Objectives--The atmospheric composition of Venus has sparked much curiosity since its volatile, corrosive nature was first discovered. To help answer questions of Venus' early history, a science objective of measuring the noble gases in the Venutian atmosphere was established as the primary goal. Precise measurements of these gases will provide significant clues to the origin and evolution of the Venus atmosphere. In addition, it gives clues to early outgassing characteristics, which sheds light on the planet's origin, as well as that of the entire solar system. A secondary science goal for this mission is to measure the concentration of other atmospheric gases.

Major Design Trades--With any probe mission one of the biggest problems is sending the data back to Earth. Several communications issues have to be considered. First, if the probe is not visible from Earth at atmospheric entry, then a data relay is required. Given the selected launch date for this mission, a trajectory analysis showed that the Earth would be in view of the probe during the entire entry event. Second, data volume can present a problem. A large data volume would require a high data rate, which is impossible for a direct link to Earth. The instruments used for this mission do not require high data volume, so a direct link to Earth was not impossible. Mission complexity is greatly increased when a data relay is added, as this usually requires orbit insertion of the spacecraft, thereby significantly increasing propellant sizing. In this case, however, an orbital relay would be undesirable due to the short life of the probe. Combining these factors, a direct telecom link between Earth and the probe was selected.

As was the case with the lunar mission, this mission exhibited a large positive launch margin using a bi-propellant propulsion system. A future trade (not worked during the design session) could consider the feasibility of a mono-prop system within the constraints of the micromission launch envelope. The design shown in the tables on the following pages represents a bi-prop system. This was chosen to minimize total mission cost by reducing system mass (bi-prop has a higher I_{sp} and is therefore a less massive option in most cases). However, a cost trade could also be worked to see if the lower cost of the mono-prop system would offset the cheaper launch. While these trades were discussed as options, the time constraints of the Team X session for this concept did not allow either to be fully considered. Future work on this concept would include these trade studies.

Initial probe release was also studied. There was a trade performed concerning spin-stabilization of the spacecraft. If the entire spacecraft were spinning, the dedicated probe spin platform would not be necessary. The cruise stage would simply spin itself up to give the probe sufficient rotation to stabilize itself during atmospheric entry. Spin-stabilizing the bus was deemed too complex for this mission, and would add significantly to the cost of analysis and systems engineering. This cost impact was large enough to override the cost savings from less expensive ACS hardware associated with a simple spinner.

Final Design Mass/Power Summary—Table 2 and Table 3 summarize the Venus micromission carrier spacecraft and probe, respectively. The tables show that the design has substantial margin against the launch vehicle allocation even with the 30% dry mass and power contingency. A trade considering a mono-propellant system should be considered to reduce system cost and complexity since the launch margin is so high. These summaries showcase another feasible, secondary launch, interplanetary micromission design.

Discussion of Study Results—The following diagrams show the probe, and its location on the spacecraft, as well as the secondary payload envelope. Note that the probe is mounted on a spin platform—this could be unnecessary pending the spin-stabilization trade.

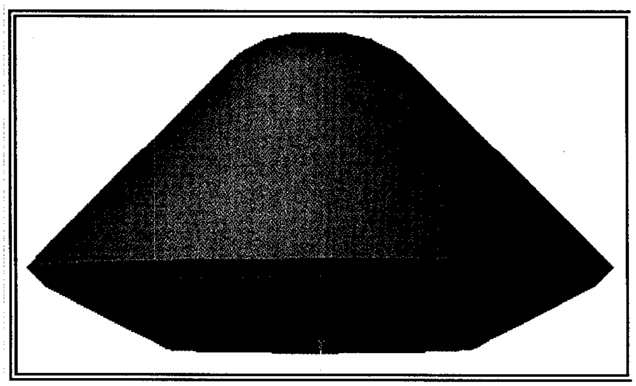

Figure 5 - Venus Micromission Probe

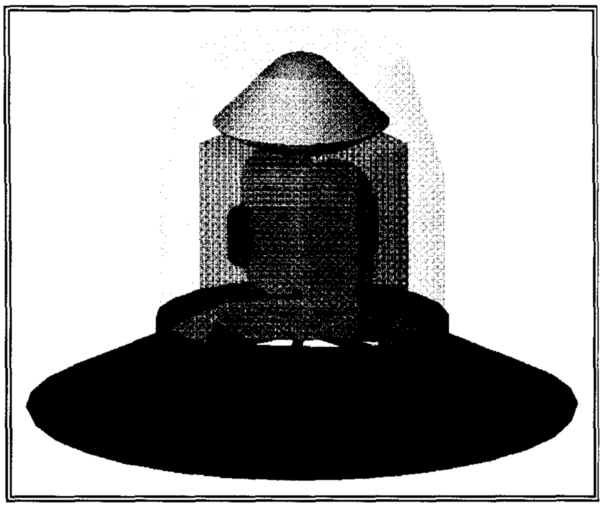

Figure 6 - Venus Micromission Inside Launch Vehicle

The Venus Probe design adequately achieved the desired science goals. The probe will send the required data to Earth, and provide some additional aero-entry data that could be used for future missions requiring atmospheric entry. Future studies need to be done to consider the possibilities of using a mono-propellant system, and verifying the outcome of the initial spin-stabilization trade. These are primarily cost trades and could be performed if cost became a significant issue. This design represents a "best case scenario" for the Venus probe concept, but a less expensive, less complicated design could very well meet the requirements. These trades will eventually be worked through as demands for smaller, more focused missions increase.

Table 2 - Venus Micromission Probe, Carrier Spacecraft Mass/Power Summary

Venus Micromission Probe 9-00
Cruise

SYSTEMS WORKSHEET
Analyst: Matt Johnson
Start Date: 9/29/00

Stabilization - cruise	3-Axis		Pointing Direction - cruise	Sun-Earth				
Stabilization - science	3-Axis		Pointing Direction - science	N/A	Mission Duration	1.3	years
					Max probe sun distance	1.0	AU
Pointing Control	3600	arcsec	Radiation Total Dose, krad	20	Spacecraft Data Rate	2.0	kb/s
Pointing Knowledge	1800	arcsec	Science BER	1.00E-06	Data Storage	8	Mb
Pointing Stability	60	arcsec/sec	Redundancy	Single String			
Determined by:	Probe Release				Maximum Link Distance	0.76	AU
			Technology Cutoff	2002	Return Data Rate	0.25	kb/s

	Mass Fraction	Mass (kg)	Mode 1 Power (W) *Probe Relay*	Mode 2 Power (W) *Probe Release*	Mode 3 Power (W) *TVI-TCM*	Mode 4 Power (W) *Cruise*	Mode 5 Power (W) *Launch*	NASA TRL *Today*
Payload								
Probe	38.4%	71.9	0.0	0.0	1.6	1.6	0.0	
Payload Total	38.4%	71.9	0.0	0.0	1.6	1.6	0.0	
Bus								
Attitude Control	1.9%	3.6	24.6	24.6	24.6	24.6	24.6	5
Command & Data	2.8%	5.2	17.6	17.6	17.6	17.6	17.6	5
Power	6.8%	12.8	16.0	16.0	34.0	16.0	14.9	6
Propulsion	12.5%	23.4	23.2	23.2	162.0	23.2	23.2	4
Structure	23.8%	44.6	0.0	0.0	0.0	0.0	0.0	6
S/C Adapter	2.4%	4.6						
Cabling	4.3%	8.1						
Telecomm	2.7%	5.1	43.2	43.2	43.2	43.2	43.2	0
Thermal	4.3%	8.0	14.1	14.1	14.1	14.1	5.8	6
Bus Total		115.3	138.6	138.6	295.5	138.6	129.3	
Spacecraft Total (Dry)		187.2	138.6	138.6	297.1	140.3	129.3	
Mass/Power Contingency		34.6	41.6	41.6	88.6	41.6	38.8	
Spacecraft with Contingency		221.8	180.2	180.2	385.7	181.8	168.0	
Propellant & Pressurant	37.6%	133.8	For S/C mass =	355		Delta-V1	1400	m/s
Spacecraft Total (Wet)		355.6						
L/V Adapter		29.0						
Launch Mass		384.6						
Launch Vehicle Capability		500.0	Secondary		Launch C3	80		9
					Fairing type	standard		
					Fairing dia., m	?		
Launch Vehicle Margin		115.4	23.1%					

	Contingencies	
	Mass	Power
Instruments	0%	0%
Other	N/A	N/A
S/C, dry	30%	30%

Table 3 - Venus Micromission Probe, Probe Mass and Power Summary

Venus Micromission Probe 9-00
Probe

SYSTEMS WORKSHEET
Analyst: Matt Johnson
Start Date: 9/29/00

Stabilization - cruise	**Spin**		Pointing Direction - cruise	**Sun**			
Stabilization - science	**Spin**		Pointing Direction - science	**Nadir**	Mission Duration	**1.3**	years
					Max probe sun distance	**1**	AU
Pointing control	**Passive**	arcsec	Radiation Total Dose, krad	**20**	Instrument Data Rate	**3**	kb/s
Pointing Knowledge	**N/A**	arcsec	Science BER	**1.00E-06**	Data Storage	**0.4**	Mb
Pointing Stability	**N/A**	arcsec/sec	Redundancy	**ngle-String**			
Determined by:	**TBD**				Maximum Link Distance	**0.76**	AU
			Technology Cutoff	**2002**	Return Data Rate	**0.33**	kb/s

	Mass Fraction	Mass (kg)	Mode 1 Power (WHr) Science w/ Telecom (20 min)	Mode 2 Power (WHr) Telecom (20 min)	Mode 3 Power (WHr) Entry (10 min)	Mode 4 Power (WHr) Sep Cruise (4 hrs)	Mode 5 Power (WHr) Cruise	NASA TRL Today
Payload								
Instruments	17.3%	6.2	2.7	0.4	0.2	4.4	1.1	6
Payload Total	17.3%	6.2	2.7	0.4	0.2	4.4	1.1	
Bus								
Attitude Control	2.0%	0.7	0.0	0.0	0.9	0.0	0.0	5
Command & Data	1.5%	0.5	1.4	1.5	0.7	16.2	0.0	4
Power	11.0%	3.9	9.2	8.9	8.8	4.0	0.1	6
Structure	32.3%	11.6	0.0	0.0	0.0	0.0	0.0	6
S/C Adapter	1.5%	0.5						
Cabling	7.8%	2.8						
Telecomm	21.5%	7.7	65.5	65.5	65.5	0.0	0.0	0
Thermal	5.2%	1.9	0.8	0.8	0.3	10.0	0.0	6
Bus Total		29.6	76.9	76.8	76.1	30.2	0.1	
Spacecraft Total (Dry)		35.8	79.6	77.1	76.3	34.6	1.2	
Mass/Power Contingency		10.8	23.9	23.1	22.9	10.4	0.4	
Spacecraft with Contingency		**46.6**	**103.5**	**100.3**	**99.2**	**44.9**	**1.6**	
Probe Mass Total w/o Entry System		**46.6**						
Heatshield, Backshell, parachute		25.3						
Probe Carrier + LV Adapter Mass		312.7						
Launch Mass		**384.6**						
Entry System Mass		**71.9**						
Launch Vehicle Capability		**500.0**	**Secondary**					9
Launch Vehicle Margin		**115.4**	**23.1%**					

Contingencies	Mass	Power
Instruments	30%	30%
Other	N/A	N/A
S/C, dry	30%	30%

Entry System Dia.	0.85	m
Drag Coefficient	1.1	
Ballistic Coefficient	115.2	kg/m^2
Launch C3	0	
Fairing type	standard	
Fairing dia., m	?	

Mars Balloon

NASA's science goals with respect to Mars are focused on the investigation of past, present, and future environments that may support life/show evidence of past life. To achieve these goals, there is considerable attention paid to in-situ science investigation. Several remote sensing spacecraft have traveled to Mars and returned valuable data to Earth. Mars Pathfinder performed a rover mission on the surface of Mars, which gave scientists a much better understanding of the Martian surface. These past missions have opened the door for smaller, more focused missions that can concentrate on a specific discovery. The Mars balloon concept discussed below is one such mission, with a focus on a recent Martian discovery—the possibility of water on the surface of Mars.

Mission Overview--This micromission delivers a balloon to Mars. It also uses the Earth-Moon tour to make the interplanetary transfer possible given a secondary launch. After a final powered Earth flyby, the spacecraft travels to Mars. The mission consists of two elements, a balloon and a probe carrier. The probe carrier serves as a delivery vehicle only, and does not undergo orbit insertion for telecom relay. The balloon is deployed from the carrier vehicle for descent through Mars' atmosphere inside an entry package consisting of an aeroshell and backshell. The balloon reaches its final orbital altitude of 2km through a series of entry jettisons and deployments (see Figure 7). The mission is designed to place the balloon near the south pole, at least below 75° S latitude. The balloon is required to operate for 5 days, but has a goal of 10 days.

Science Objectives—Mars is the planet in the solar system most similar to the Earth. If Mars truly is similar to the Earth, then understanding its evolution could help predict future conditions on Earth. Central to investigating similarities between Earth and Mars is the search for water. Finding water on Mars will help scientists understand the potential for life elsewhere in the universe, understand the relationship to Earth's climatic change processes, and provide useful resources for future human exploration. Recent images delivered by Mars Global Surveyor have shown evidence of water once existing on the surface of Mars. This has increased the desire for missions that can characterize significant portions of the Martian surface. This balloon micromission will further the understanding of the southern polar region, and continue the search for water on the surface of the red planet.

The balloon will carry a three-head imager, a neutron spectrometer, and pressure and temperature sensors. The three-head camera has narrow-, medium-, and wide-angle imagers. The best imaging resolution of these imagers is 20cm (for the narrow-angle imager). The neutron spectrometer is used to measure hydrogen concentrations. The temperature and pressure sensors are designed to serve as simple real-time weather monitors. These instruments support the primary science objective—the search for water. In addition, this science instrumentation will perform a study of the fine Martian topographical structure in the southern highlands.

Major Design Trades--This was a straightforward design, with very little space for major design trades. The chief constraint for a balloon is that the mass of the gondola must be kept as small and light as possible. The balloon itself gets much bigger with each additional bit of mass added to the gondola. In addition to that, each kilogram added to the entry system requires more than an additional kilogram of propellant to push it to Mars. A Mars micromission would be very difficult, if not impossible, to accomplish without some key assumptions/design choices which are outlined below.

The tight mass constraints (and penalty for large propellant requirements) drive the telecom design to use a relay satellite for sending data back to Earth. This mission assumes that sufficient orbital assets would be in place for data relay. The balloon uses the UHF frequency band for its uplink to the relay satellite.

Choosing a mission designed to orbit near the south pole is advantageous, as a large majority of time is spent in the sun. This allows solar cells the ability to generate the majority of the power, thereby reducing the size of battery required.

Final Design Mass/Power Summary--Table 4 and Table 5 show the mass and power summary information for the Mars Balloon Micromission system elements. Note that the final design has a comfortable mass margin against the secondary payload allocation including the 30% dry mass and power contingency used for pre-Phase A design. Notice how much more propellant is needed to deliver a probe to Mars—a greater mass was delivered to Venus with less propellant. This shows the great challenge that micromissions to Mars represent. The spacecraft requires more propellant mass than it can deliver in the form of dry mass. This presents a very steep slope that makes the requirements for a viable Mars micromission very imposing. This will become more apparent in the next section when an orbital Mars mission is discussed. Note that even with the large propellant required to deliver the spacecraft to Mars, an orbit insertion would require an even greater increase in propellant mass.

Discussion of Study Results--Team X was able to design a mission that helps further the search for water on Mars. This mission will provide very detailed images of portions of the southern polar region of Mars. It will also give details of the concentration of hydrogen in that same area. The balloon mission described is shown in the following figure. This highlights the entry/descent of the balloon after deployment from the carrier spacecraft.

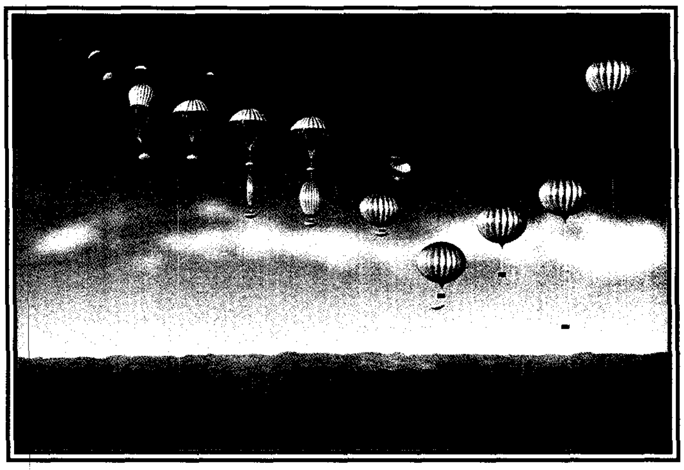

Figure 7 - Mars Balloon Mission

Employing some reasonable assumptions and an innovative mission design, the Mars Balloon can significantly enhance scientific understanding of an interesting region of the red planet. The secondary launch platform enables this mission for a very low cost *and* allows for an important science investigation to be conducted.

JEAM-X

Table 4 - Mars Balloon System Mass and Power Summary

Micromission Balloon 6-00

SYSTEMS WORKSHEET — **Balloon (and Entry System)**

Analyst: Matt Johnson
Start Date: 6/28/2000 Directory: C:\data\

Entry System Stabilization **spin-stabilize** Pointing Direction - after release **Nadir**
Stabilization - Balloon operations **passive, nadir pointing**

Pointing control	**3600**	arcsec	diation Total Dose, krad	**22**	Balloon Ops	**10.0**	days
Entry System Pointing Knowledge	**n/a**	arcsec	Science BER	**1.00E-06**	Instrument Data Rate	**1**	Mb/s
Pointing Stability	**n/a**	arcsec/sec	Redundancy	**single-string**	Data Storage	**1**	Gb
Determined by:					Data Rate to Orbiter	**16**	kb/s
			Technology Cutoff	**2003**			

		Mass (kg)	Mode 1 Power (W) *Day Science*	Mode 2 Power (W) *Day Science + Telecom*	Mode 3 Power (W) *Night*	Mode 4 Power (W) *EDI*	Mode 5 Power (W) *Sep from Carrier*	NASA TRL *Today*
Payload	**Balloon Lifted**							
Instruments	*Note 1*	1.2	1.6	1.6	0.0	0.0	0.0	4
Payload Total	**1.0**	1.2	1.6	1.6	0.0	0.0	0.0	
Gondola	**lass of items retained**							
Attitude Control	0.200	0.2	0.000	0.000	0.000	0.0	0.6	2
Command & Data	0.557	0.6	1.4	1.4	0.5	0.5	0.5	2
Power	0.919	0.9	0.3	0.4	0.0	0.0	0.1	6
Structure	1.361	1.4	0.0	0.0	0.0	0.0	0.0	4
Gondola/Heatshield interface	0.300	0.3						
Cabling	0.633	0.6						
Telecomm	0.900	0.9	0.1	0.4	0.1	0.1	0.1	6
Thermal	0.700	0.7	0.0	0.0	0.0	0.0	0.0	6
Gondola Total (without Payload)	**5.570**	5.6	1.8	2.1	0.6	0.6	1.2	
Gondola Total (with Payload)	**6.570**	6.8	3.4	3.7	0.6	0.6	1.2	
Mass/power contingency	**1.671**	2.03	1.0	1.1	0.2	0.2	0.4	
Balloon diameter = 13.44	**8.5**	8.5	Contingency included					
Balloon Container		0.8	Balloon Container 10% of balloon					
Inflation Pressurant (Hydrogen)	0.8	0.8						
Inflation System		8.7						
Inflation System (contingency)		2.6						
Total with Contingency	**17.6**	**30.3**	**4.4**	**4.9**	**0.8**	**0.8**	**1.6**	
Deorbiting Propellant		0.0	r S/C mass =	0				
Total Delivered Balloon Mass (Wet)		**30.3**						
L/V Adapter		0.0						
Heatshield, Backshell and Parachute		**15.2**						
Total Entry Mass (for 1 Balloon)		**45.5**						

Contingencies	Mass	Power
Instruments	30%	30%
Other	N/A	N/A
S/C, dry	30%	30%

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Table 5 - Mars Balloon Carrier Spacecraft Mass and Power Summary

Micromission Balloon 6-00
Probe Carrier

SYSTEMS WORKSHEET

Analyst: Matt Johnson

Start Date: 6/26/00 Directory: Team X-Cost:MJ:Micromissions:UN ESA Demo:UN ESA Demo MM_EXCEL:Prc

Stabilization - cruise	**3-Axis**	Pointing Direction - cruise	**Earth**		
Stabilization - science	**3-Axis**	Pointing Direction - science	**N/A**	Mission Duration	**1.0** years
				Max probe sun distance	**1.7** AU
Pointing control	**720** arcsec	Radiation Total Dose, krad	**22**	Engineering Data Rate	**1** kb/s
Pointing Knowledge	**360** arcsec	Science BER	**1.00E-06**	Data Storage Capability	**1500** Mb
Pointing Stability	**120** arcsec/sec	Redundancy	**Single String**		
Determined by	**Telecom**			Maximum Link Distance	**1.8** AU
		Technology Cutoff	**2003**	Return Data Rate	**1** kb/s

	Mass Fraction	Mass (kg)	Mode 1 Power (W) *Probe Relay*	Mode 2 Power (W) *Probe Release*	Mode 3 Power (W) *TCM*	Mode 4 Power (W) *Cruise*	Mode 5 Power (W) *Launch*	NASA TRL *Today*
Payload								
Probe	25.4%	45.5	0.0	12.0	12.0	12.0	12.0	5
Probe Release Mech.	3.8%	6.9	0.0	0.0	0.0	0.0	0.0	
Payload Total	29.2%	52.4	0.0	12.0	12.0	12.0	12.0	
Bus								
Attitude Control	1.8%	3.2	19.0	19.0	19.0	19.0	19.0	5
Command & Data	1.8%	3.2	11.5	9.5	9.5	9.5	9.5	4
Power	8.5%	15.3	12.7	14.0	32.7	14.0	13.3	6
Propulsion	11.9%	21.3	43.2	43.2	162.0	43.2	43.2	4
Structure	29.1%	52.1	0.0	0.0	0.0	0.0	0.0	6
S/C Adapter	3.1%	5.5						
Cabling	6.1%	10.9						
Telecomm	3.7%	6.7	33.8	33.8	33.8	33.8	11.8	6
Thermal	4.8%	8.5	13.3	13.3	13.3	13.3	6.7	6
Bus Total		126.7	133.5	132.8	270.3	132.8	103.4	
Spacecraft Total (Dry)		179.1	133.5	144.8	282.3	144.8	115.4	
Mass/Power Contingency		38.0	40.1	39.9	81.1	39.9	31.0	
Spacecraft with Contingency		**217.1**	**173.6**	**184.7**	**363.4**	**184.7**	**146.5**	
Propellant & Pressurant	52.3%	237.6	For S/C mass = 471		Delta-V 1	2100.0 m/s		
Spacecraft Total (Wet)		**454.7**						
L/V Adapter		29.5						
Launch Mass		**484.3**						
Launch Vehicle Capability		**500.0**	**Secondary**		Launch C3 0			4
					Fairing type standard			
					Fairing dia., m ?			
Launch Vehicle Margin		**15.7**	**3.1%**					

Contingencies	Mass	Power
Probe	0%	0%
Other	N/A	N/A
S/C, dry	30%	30%

Mars Science Orbiter

Of central importance to future exploration of Mars is understanding the climate. This Mars Science Orbiter micromission will utilize the secondary payload platform to perform valuable atmospheric science (global mapping, seasonal measurements, etc.) at Mars. Mars Climate Orbiter (MCO) carried an impressive science instrument package designed to monitor weather patterns at Mars. This mission concept is a re-flight of the MCO instruments, important for the recovery of lost data upon MCO failure.

Mission Overview--This mission uses the GTO to the planets secondary payload interplanetary trajectory used for all the micromissions discussed herein. After the Earth-Moon maneuvers, this spacecraft will travel to Mars. At Mars the orbiter uses chemical propulsion for orbit insertion around the planet. The orbiter captures into a 24 hour orbit, then uses aerobraking to descend to its parking orbit at 400 km, inclination of 90°. The mission is required to survive one Martian year in its final science orbit.

Science Objectives--Ever since the loss of the Mars Climate Orbiter, scientists have clamored for recovery of the very important science that was lost. This mission is meant to re-fly the MARCI and PMIRR mk-II from MCO and recover that lost science. The MARCI carries both a medium- and a wide-angle imager. The PMIRR is a nadir scanning atmospheric sounder designed to vertically profile atmospheric temperature, dust, water vapor, and condesate clouds to quantify surface radiative balance. These instruments will give scientists a much better picture of Martian weather patterns.

Major Design Trades—One way to cut cost and mass for a mission is to use a thruster-based reaction control system (RCS). Reaction wheel based systems cost more and are generally more massive. The ACS engineer was asked to evaluate the use of an RCS system for this spacecraft. This choice worked for almost all pointing requirements in this mission. However, there was a tight yaw pointing requirement of 23 arcsec. Over the course of a three-year mission this would have led to an RCS fuel usage in excess of 100 kg. Based on this alone, an RCS pointing system was ruled out and reaction wheels were used in the design.

Another system level trade discussed was the use of a Ka band telecommunications system. A Ka band system allows for higher data rate transmission than a comparable (mass-wise) X band system. Ka band is more expensive, but in this case it was worth consideration due to the tight mass constraints. Ka band would work for all nominal data relay modes. However, when the spacecraft goes into safing mode it can no longer use its high gain antenna. This causes a problem because most telecom systems cannot communicate with the Deep Space Network (DSN)'s 34 m Beam Wave Guide antennas (the link is not closed). Most emergency modes are sized to use the low gain antenna to the DSN's 70m antennas. However, the 70m antennas are not scheduled for Ka band use in the time frame considered. Thus X band was chosen, necessitating a 1m high gain antenna. This took some creativity to fit into the launch configuration.

Battery type/size selection is often a system design driver. Most deep space missions baseline a Nickel Hydrogen (NiH) battery. This is chiefly because NiH batteries are the best alternative for long-life missions requiring a large number of cycles. From a mass perspective, the best option is a Lithium Ion battery, which does not support long-lived missions. The mission considered here had a four-year mission life: two years in cruise, two years on orbit. Due to this moderate mission lifetime, Lithium ion batteries were a borderline (risky) choice. The mass limitations drove the design team to accept this risk, and select Lithium Ion batteries for the final design.

Final Design Mass/Power Summary—The spacecraft designed for this mission is summarized in Table 6. It shows a launch mass that is slightly above the launch vehicle capability. However, since this includes the 30% dry mass contingency, the spacecraft mass is close enough to merit further study. Also note that this is a single string design. It is assumed that a micromission can absorb the risk that this presents because of its relatively inexpensive nature. The configuration below shows the major components of the spacecraft within the launch envelope.

Discussion of Study Results—This is an important science mission that is nearly accomplished within the constraints presented by this study. This mission could be launched as a primary payload, but the lower cost of the secondary opportunity helps offset the extra risk that this mission presents. This mission also shows some of the limitations elemental to micromissions. For instance, this mission could not be flown in the 2005 opportunity for Mars where the delta-V is 350 m/s higher. Also, using a single string spacecraft for a four-year mission is inherently risky. It is left to mission architects to decide what risk is acceptable based on mission cost and science return.

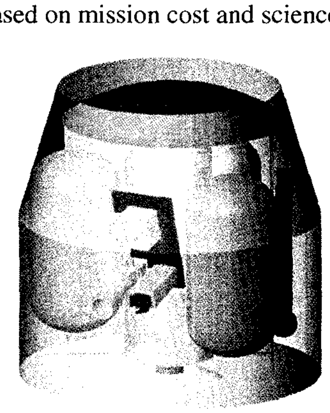

Figure 8 - Mars Science Orbiter Micromission

Team-X

Table 6 - Mars Science Orbiter Spacecraft Mass and Power Summary

MCO Reflight uMission Orbiter 10-00

MCO Reflight Orbiter

SYSTEMS WORKSHEET

Analyst: Matt Johnson

Start Date: 10/31/00 Directory:

Stabilization - cruise	**3-Axis**	Pointing Direction - cruise	**Sun-Earth**		
Stabilization - science	**3-Axis**	Pointing Direction - science	**Mars**	Mission Duration	**4.0** years
				Max probe sun distance	**1.7** AU
Pointing control	**23** arcsec	Radiation Total Dose, krad	**24**	Instrument Data Rate	**2.0** Mb/s
Pointing Knowledge	**11** arcsec	Science BER	**1.00E-06**	Data Storage	**1000** Mb
Pointing Stability	**34** arcsec/sec	Redundancy	**Single String**		
Determined by:	**Telecom**			Maximum Link Distance	**2.7** AU
		Technology Cutoff	**2005**	Return Data Rate	**5** kb/s

	Mass Fraction	Mass (kg)	Mode 1 Power (W) *Day Science*	Mode 2 Power (W) *Eclipse*	Mode 3 Power (W) *MOI*	Mode 4 Power (W) *Aero-brake*	Mode 5 Power (W) *Launch (3 hours)*	NASA TRL *Today*
Payload								
MARCI, PMIRR	5.2%	7.4	16.0	9.0	2.0	2.0	2.0	5
Payload Total	5.2%	7.4	16.0	9.0	2.0	2.0	2.0	
Bus								
Attitude Control	7.2%	10.3	34.7	34.7	37.4	30.9	19.6	5
Command & Data	4.3%	6.1	15.6	11.6	11.6	11.6	11.6	5
Power	11.9%	16.9	21.5	14.1	24.3	18.6	14.8	6
Propulsion	16.7%	23.8	18.7	18.7	55.8	18.7	18.7	0
Structure	31.9%	45.6	0.0	0.0	0.0	0.0	0.0	6
S/C Adapter	4.5%	6.5						
Cabling	7.0%	9.9						
Telecomm	5.3%	7.6	73.2	27.4	73.2	73.2	73.2	0
Thermal	6.0%	8.5	6.9	6.9	6.9	6.9	3.5	6
Bus Total		135.3	170.6	113.4	209.3	160.0	141.4	
Spacecraft Total (Dry)		142.7	186.6	122.4	211.3	162.0	143.4	
Mass/Power Contingency		42.8	56.0	36.7	63.4	48.6	43.0	
Spacecraft with Contingency		**185.5**	**242.6**	**159.1**	**274.6**	**210.6**	**186.4**	
Propellant & Pressurant	60.8%	288.2	For S/C mass =	471		Delta-V1	2850.0	m/s
Spacecraft Total (Wet)		**473.7**				Contingencies		
L/V Adapter		29.0				Mass	Power	
					Instruments	30%	30%	
Launch Mass		502.7			Other	N/A	N/A	
					S/C, dry	30%	30%	
Launch Vehicle Capability		**500.0**	Secondary		Launch C3	0		9
					Fairing type	standard		
					Fairing dia., m	?		
Launch Vehicle Margin		**-2.7**	**-0.5%**					

3. CONCLUSIONS

Secondary launch opportunities can be synergistic with innovative small interplanetary mission concepts in providing exciting scientific opportunities. Use of the GTO micromission trajectory enables such missions for a fraction of the cost of traditional interplanetary missions. While the science payloads are not as all-encompassing or capable as their traditional counterparts, the return they offer is *valuable, cost-effective* science.

The work presented here shows viable, realistic options for small interplanetary missions to a wide range of targets, with a large scope of meaningful science investigations. Some of the most substantial results compiled after completion of these studies are discussed below.

Cost Analysis

The primary goal in terms of cost was to reduce total projected mission cost by a factor of 2. While specific cost numbers are not presented here due to proprietary data concerns, each of the micromissions studied during this effort did show tremendous reduction in projected total mission cost. All missions had estimated costs 50% lower (+/- 10%) than their traditional, primary launch counterparts. The sponsors of this work were pleased with these results, as scientifically significant, cost-effective secondary launch missions were the goal. The specific missions discussed in this paper had total projected mission costs near the desired cost cap.

Candidate Mission Types

As projected, numerous targets were assessed for potential mission concepts. While only a few are presented herein, studies were conducted considering varied science missions to small bodies (near-Earth asteroids, comets, etc.), Mars, Venus, and Earth's Moon.

In general, the exercise was a success. Each of the candidate targets produced at least one viable mission concept that (a) fit within the launch vehicle secondary payload envelope, (b) were accomplished for a significantly lower cost than possible for a primary launch, and (c) provided valuable scientific data return. The secondary launch opportunities were implemented successfully in the Team X design environment, and emerged as an attractive option worthy of further study for small interplanetary science missions.

Launch Vehicle Options

The Advanced Projects Design Team had experience prior to these studies with secondary payload opportunities on the Ariane V (Ariane 5 Structure for Auxiliary Payloads, ASAP5) [4]. These studies extended the database of secondary launch slots by considering secondary envelopes within the Atlas and Delta vehicles [1]. An overall result of the many studies conducted is a set of real launch options that take advantage of extremely low-cost secondary opportunities. The cost savings alone is a very attractive asset to this concept, and combined with the innovative GTO to the Planets trajectories [2], enables interplanetary missions at a fraction of the cost of current missions.

4. Future Work

The success of this exercise spurs the continuation of similar concept investigations. Team X will consider several new micromission studies in the coming year, as well as revise previous studies (some trades are still to be assessed on concepts addressed in this paper) to reflect lessons learned throughout the process. The study customers (the Discovery and Mars Program Offices at JPL) are interested in the outcome of these studies, and frequently suggest new and exciting science investigations to "audition" on the secondary payload platform.

The long-term goal is to develop one or more of these interplanetary micromissions beyond the pre-phase A level, and demonstrate the feasibility of the concept on a flight project. Small missions *can* play a large role in future robotic space exploration. The studies conducted to date are intended to support the cause for small missions by helping to define what their future role in interplanetary science investigation could be.

5. References

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6. Acknowledgements

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The work discussed in this paper would not have been possible without the hard work of the members of Team X. Their subsystem expertise is a credit to each of their individual divisions, and a proud representation of the best of JPL teamwork.

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7. AUTHOR BIOGRAPHIES

Jennifer M. Owens served as a Systems Engineer and Cost Analyst for Team X during the micromission studies discussed in this paper. She is now a part of the Flight Systems Section, Systems Engineering division at JPL, and serves as a spacecraft systems engineer for the GRACE mission, an Earth Systems Science Pathfinder mission to be launched in October 2001. Jennifer has a B.S. in Aerospace Engineering from the University of Colorado, Boulder and an M.S. in Aeronautics and Astronautics from Stanford University.

Matthew B. Johnson is a Systems Engineer and Design Tool Lead for the Advanced Projects Design Team (Team X) at JPL. He also serves as the liaison between Team X and the Mars Program Office at JPL. He is a member of the Earth-Like Planets and Microspacecraft Architecture Group, Mission and Systems Architecture Section, Systems Engineering Division. Matthew graduated from the California Institute of Technology with a B.S. degree in Mechanical Engineering and a B.A. in Math-Physics from Whitman College.