Mars Aeronomy Explorer (MAX): Study Employing Distributed Micro-Spacecraft

R. Shotwell, A. Gray, P. Illsley, M. Johnson, R. Sherwood, M. Vozoff, J. Ziemer
Jet Propulsion Laboratory, California Institute of Technology
4800 Oak Grove Drive
Pasadena, CA 91109
gray@jpl.nasa.gov

Abstract— an overview of a Mars Aeronomy Explorer (MAX) mission design study performed at NASA's Jet Propulsion Laboratory is presented herein. The mission design consists of ten micro-spacecraft orbiters launched on a Delta IV to Mars polar orbit to determine the spatial, diurnal and seasonal variation of the constituents of the Martian upper atmosphere and ionosphere over the course of one Martian year. The spacecraft are designed to allow penetration of the upper atmosphere to at least 90 km. This property coupled with orbit precession will yield knowledge of the nature of the solar wind interaction with Mars, the influence of the Mars crustal magnetic field on ionospheric processes, and the measurement of present thermal and non-thermal escape rates of atmospheric constituents. The mission design incorporates alternative *design paradigms* that are more appropriate for—and in some cases motivate—distributed micro-spacecraft. These design paradigms are not defined by a simple set of rules, but rather a way of thinking about the function of instruments, mission reliability/risk, and cost in a systemic framework.

TABLE OF CONTENTS

1. INTRODUCTION

The Mars Aeronomy study presented involves significant mission architectural changes over current state of practice missions. The architecture is based on distributed micro-spacecraft(Andrew – might be better to emphasize these are focused spacecraft and not so much micro) which enable significant improvements in science gathering and an improved science per cost profile. Development of such an architecture requires distributed system design philosophies and benefits from a systemic or macro approach to analysis and synthesis—e.g. function and reliability/risk are analyzed over the system of spacecraft. The architecture also facilitates design and manufacturing paradigms not employed with current state of practice for deep space missions. The architecture, focused-spacecraft technology, associated analysis/synthesis methods for mission design, and distributed paradigms are all motivated by the goal of accomplishing improved functions for science data gathering for less cost.

We identify the fundamental characteristics of technology, systems reliability, mission and spacecraft design paradigms and processes, and culture that are aligned with or result in *low cost* and high *macro-functionality*. We define spacecraft re-occurring cost to be low relative to those of historical JPL and NASA (deep-space) missions. We define high macro-functionality or macro-performance to be mission functionality–not necessarily instrument or spacecraft functionality–that achieves the science objectives of a stated mission. Obviously a comprehensive analysis of any one of technology, reliability, design paradigms, and culture is a tremendous undertaking and beyond the scope of any single study. However, the study seeks to identify and analyze characteristics or properties of these elements most critical in achieving improved science per cost when distributed micro-spacecraft are in the mission design space. From this analysis and characterization design principles are synthesized.

The Mars Aeronomy mission architecture developed offers hitherto unobtainable science gathering capability and has been designed using a system trade framework of cost-for-performance that utilizes the underlying design principles/processes and guidelines developed as part of the study. The architecture incorporates the use of distributed focused-spacecraft with simple deployment and operations. The mission cost estimate is significantly less than that of a previous JPL Team X study option utilizing a single monolithic spacecraft with less overall science gathering capability; however we acknowledge that there can not be a fair comparison for at least two reasons. First, the original

IEEEAC paper #1400

Team X study is over two years old, and second, the science instruments, and hence science data types, are not identical. Table one summarizes the cost and science gathering capability of the previously proposed and newly proposed Mars Aeronomy mission. It should be recognized that there is a long list of assumptions underlying each mission and the precise methods by which each mission gathers new science, while largely overlapping, are not identical.

Table 1 - Cost and Science Comparison

Mission Options	Total Cost	Mission Function	Mission Duration
Conventional Mars Aeronomy	~\$428M-\$588 M	2.7 giga-bit science data	2 years
Distributed Mars Aeronomy	~\$550 M	81.2 giga-bit science data	2 years

The result of the analysis and trade study contained herein is the mission architecture conceptually illustrated in Figure 1; there are a total of ten micro-spacecraft. The ten are comprised of three types or groups. There are two each of Group 1 and 2 spacecraft, both with the majority of their design, manufacture, launch, and operational characteristics fitting the accepted definition of micro-spacecraft; however, there are a small number of characteristics that necessarily fall outside this definition and these are addressed herein. Group 3 spacecraft are defined by the most modern definition of micro-spacecraft with strong emphasis on low mass/volume and very low complexity in design, manufacturing, launch, and operations.

Finally, the study incorporates a summary of micro-spacecraft technologies appropriate for the cost and performance profile of the proposed Mars Aeronomy mission and many other distributed spacecraft missions.

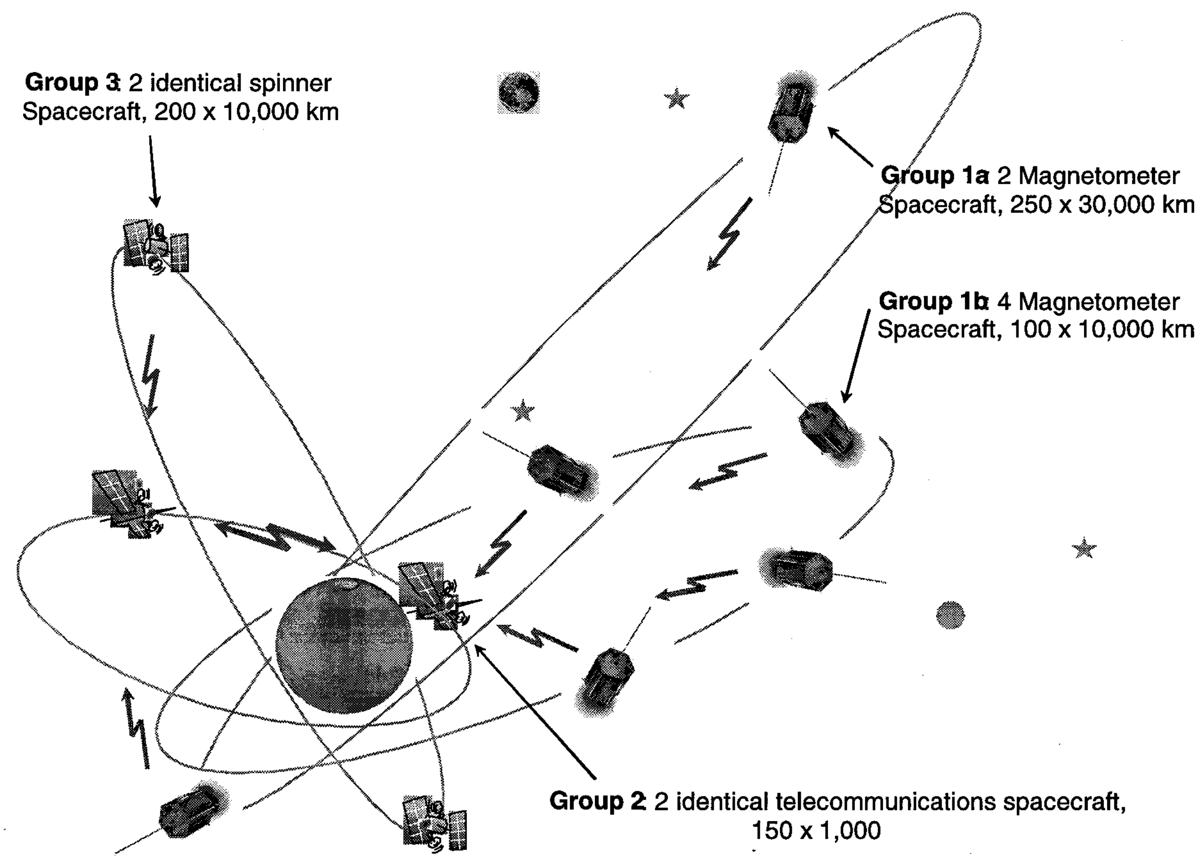

Figure 1 – Conceptual micro-spacecraft architecture

2. STUDY PHILOSOPHY

The design of distributed spacecraft systems has principles (e.g. trading one resource for another) fundamentally in common with the design of single spacecraft systems. However, many of these principles may be applied using different overall paradigms. When compared with a single spacecraft, a distributed spacecraft system is a system of systems, with each individual system spatially separated from all others. The design and implementation of a single spacecraft system is a complex and resource intensive undertaking; to design a system that is a composite of such systems would appear at first to be substantially more so. We argue that this is not necessarily the case unless the same design philosophies or paradigms for single spacecraft systems are applied "brute force" to distributed spacecraft systems. Furthermore, with appropriate distributed design paradigms overall system improvements are often possible in more dimensions (e.g. the complexity, cost and function dimensions) as compared with the design paradigms of single spacecraft system. These improvements are possible because with distributed design and implementation the number of ways in which the dimensions may be traded is often increased as compared with single spacecraft system design and implementation. This introduces the need for new paradigms or models that guide the application of such distributed system trades.

Every trade-framework must have metrics in order to judge between competing concepts or designs. The direct metrics are cost and functionality. Furthermore, application of the framework requires detailed knowledge of science objectives that may require the functionality offered by multiple spacecraft and at a minimum do not preclude the use of multiple spacecraft to accomplish the required functionality.

The analysis begins by identifying and analyzing characteristics of technology, high reliability systems, design paradigms, and culture that are conducive to low cost and/or high functionality. This is accomplished in order to "pare-down" the trade space and to start the trade/analysis and design framework with a foundation of elements aligned with overall objectives of the mission design and realization. It should be expected that most characteristics of the four elements that align with low cost would conflict with high functionality.

Table 2 indicates a primary and generic characteristic of each element that is aligned with low cost, and one that is aligned with high functionality. Note that the characteristics listed were determined to be the most important–given the criterion outlined–by the design team. With this table as a first step we proceed to conceptually analyze the relationship between these characteristics in more detail. Figure 2 illustrates the conceptual relative cost and functionality of each element on a hi-low scale.

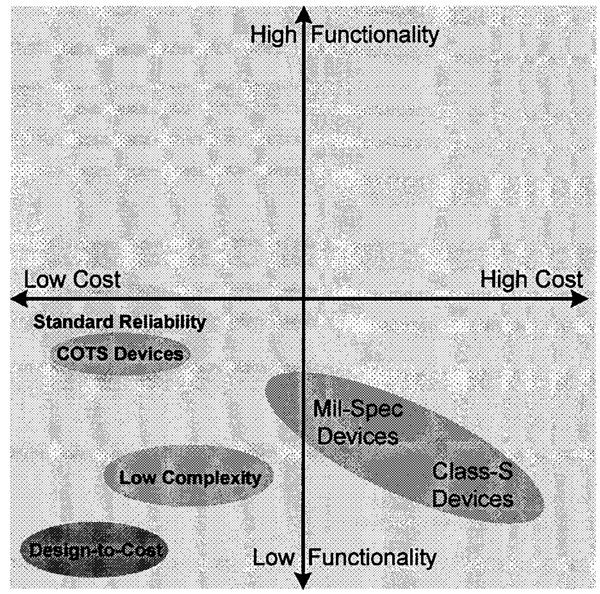

Figure 2 – Functionality versus Cost

With appropriate system design the cost and functionality characteristics of a single spacecraft may be re-mapped at the system level. Such a re-mapping is conceptually illustrated in Figure 3.

As a simple example we define the mission to be reliable as long a single spacecraft does not fail over the lifetime of the mission (ie-do not consider graceful degradation). Furthermore we assume the probability of system failure is the probability that all spacecraft fail. The probability of success of a multiple (homogeneous) N-spacecraft constellation with the reliability of a single spacecraft = P_S is given by:

$$P_{SC} = (P_S)^N + \sum_{K=1}^{N-1} \frac{N!}{K!(N-K)!}(P_S)^{N-K}(1-P_S)^K,$$

with N equal to the number of spacecraft. Of course this equation assumes that the probability of failure for each spacecraft is independent of all others (there is not a systemic problem such as a defective spacecraft bus that would impact all spacecraft). This independence assumption is critical in the following example; however it should be understood that this assumption is often not valid. Determining the probability of failure for micro-spacecraft and the inter-dependence of these probabilities is a critical aspect of distributed design. Figure 4 illustrates a simple example of a high system reliability with less stringent spacecraft reliability ($P_S = 0.7$).

One Spacecraft Conventional Approach	Primary "Notional" Characteristics	
Mission Design Elements	**Low Spacecraft Cost**	**High Spacecraft Functionality**
Technology	COTS and Standardization	Custom
Reliability	Standard reliability	High reliability
Design Paradigms	Low Complexity	High number of options
Culture	Design-to-cost	Design-to-performance

Table 2 – Notional characteristic of the four fundamental elements

Also, this analysis assumes that the primary objectives can be achieved with only 1 of each type of spacecraft

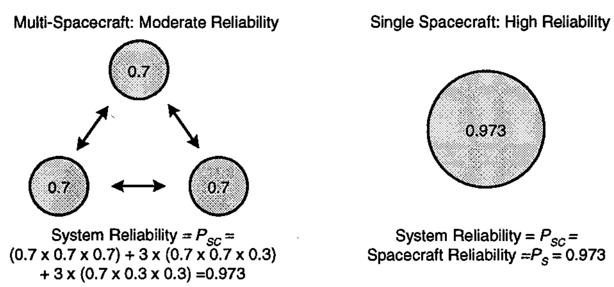

Figure 4 – System reliability with one versus many spacecraft as a function of spacecraft reliability (failure independence assumed)

functioning over the duration of the mission and cn tolerate a failure of one of the redundant spacecraft.

It is common that costs go up nonlinearly with reliability; in distributed design reductions in overall cost are possible due to the non-linear relationship between cost and reliability. However, determining the actual cost curves as a function of reliability is very challenging, and in this case was not practical given the schedule/resources available to perform the study. Nevertheless the recognition of this relationship guided system design in a direction that would otherwise not have been possible; the study team attempted to design a mission architecture that pushed the design to the edge of what is feasible/possible given current technology in order to understand in more depth benefits and limitations of these paradigms. The results of these design efforts are summarized in section 4.

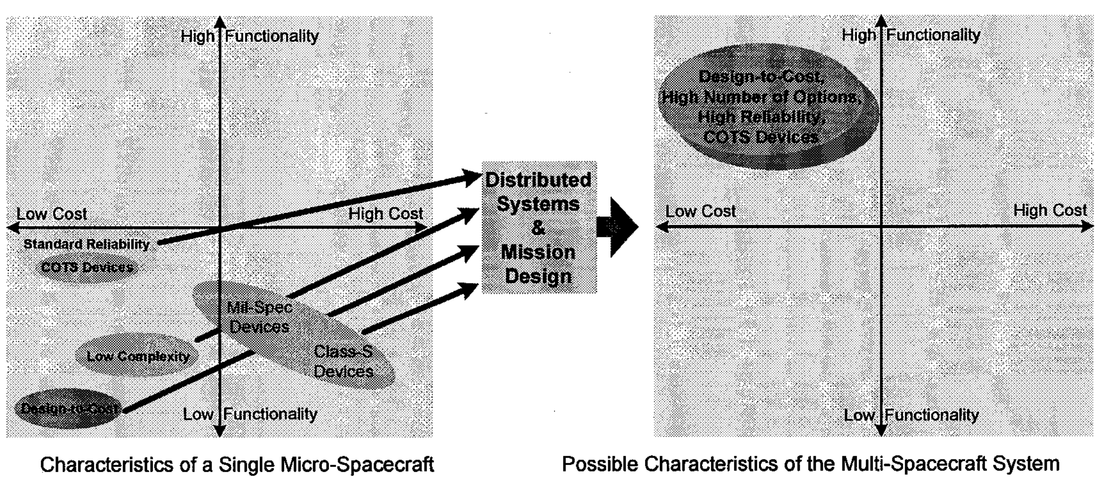

Figure 3 - Conceptual illustration of characteristic remapping through distributed design

3. Science Objectives and Instruments

The objectives for a Mars Aeronomy mission involve gaining an understanding of the following processes and characteristics of the Martian upper atmosphere and ionosphere:

- Spatial variations of the constituents of the upper atmosphere and ionosphere (i.e. functions of altitude, local time, latitude, longitude, season, solar cycle).
- Energy budget for the ionospheric plasma populations (heat sources and energy loss processes).

- Dynamics of the atmosphere (energy and momentum transport processes).
- Sources of the night side ionosphere (e.g. transport versus local production).
- The solar wind interaction boundaries (magnetic pile up region and boundary).
- Temporal variation of atmospheric constituents, due to solar extreme ultraviolet (EUV) variability and solar wind events.
- Existence and influence of crustal magnetic fields on ionospheric structure and dynamics.

In order to achieve the science objectives, the following must be performed:

1. Measurements of the characteristics (i.e. composition, density, temperature, and winds) of the neutral upper atmosphere and ionosphere as a function of altitude to the lowest possible altitudes (greater than or equal to 90 km).
2. Measurements of the characteristics (i.e. ion and neutral composition, density, and temperature) of the upper ionosphere and ionosphere/solar wind interaction region.
3. Map of the crustal magnetic field to highest resolution possible (less than 150 km altitude).
4. Measurement of the diurnal variation of the upper atmosphere and ionosphere throughout the entire range of altitudes.
5. Measurement of these variations over the widest possible range of latitudes consistent while obtaining good diurnal coverage and resolution.
6. Measurement of the effects of changing season and solar activity level upon the atmosphere and ionosphere.

The instruments the design team chose to obtain these measurements are:

1. Langmuir Probe
2. Magnetometers (2)
3. Plasma Wave Detector
4. Ion/Neutral Mass Spectrometer
5. Fabry-Perot Interferometer
6. Radio Science (for occultations not gravity)
7. Retarding Potential Analyzer and Ion Drift Meter
8. Plasma and Energetic Particle Analyzer

These eight instruments will be grouped in four units as follows: 1. (Ion/Neutral Mass Spectrometer, Ion and Electron Detectors, Langmuir Probe), 2. (Magnetometers and Plasma Wave Detector), 3. Fabry-Perot Interferometer and 4. Radio Science. The four groupings are considered moderate in complexity.

All of the instrument technologies required by the goals of the Mars Aeronomy mission are well developed and have been flown on previous missions. No prioritization has been specified for the core payload on the basis that all eight instruments contribute key measurements and all are essential to the mission objectives. In both the aeronomy and solar wind interaction disciplines most scientific questions are resolved only through the synergistic combination of observations by multiple instruments.

The upper atmosphere is addressed by four instruments, one in situ (the neutral portion of the Neutral and Ion Mass Spectrometer) and two remote (Fabry- Perot Interferometer and radio science). The ionospheric instruments are four in number, three in situ (the ion portion of the Neutral and Ion Mass Spectrometer, retarding potential analyzer and ion drift meter, and Langmuir probe) and one remote (radio science). Finally, the solar wind interaction is examined by 3 instruments, all in situ (plasma + energetic particle analyzer, magnetometer, and plasma wave analyzer). In many cases the instrument capabilities extend outside of these main regions (e.g., the ion mass spectrometer and Langmuir probe can make same measurements in the shocked solar wind, while all of the solar wind interaction instruments can make observations in the ionosphere), but this categorization reflects the regions where observations from each instrument will make their major contributions.

Strategic Instrument Groupings

The Aeronomy mission uses a wide variety of instruments with several different needs. Several factors can be used to discern how best to group the set of instruments. Using microsat technology allows this mission to separate these instruments into groups in ways that would have been impossible before. The first factor used to decide how to divide up the instruments is their orbit needs. For example, there are several instruments that take measurements in the solar wind and others that would be wasted at those high altitudes. Another discerning factor are the pointing requirements levied on a spacecraft by the instruments. For example, we have several instruments that do not require any pointing control from the spacecraft, but a few have pointing control requirements on the order of 1 degree. Another discerning factor is that some instruments create high voltages to make their measurements, creating a different strain on the bus on which they reside. These factors led the architecture to the following three instrument groupings around which the spacecraft and mission architecture as designed in the study. The ability to categorize instruments and platforms allows optimization of functionality and minimization of competing requirements which drive costs.

Group 1 (1-a and 1-b)

The first group consists of the magnetometer, the PWA and the LP. All three of these instruments take measurements as low as possible and as high as possible, both in the Mars atmosphere and out in the solar wind. In addition, none of the instruments listed here have any pointing control requirements, leading to the possibility of a simple spinner or otherwise passively controlled vehicle. These instruments also want to be mounted away from the spacecraft bus.

Group 2

The FPI and NIMS instruments make up the second group. Both instruments only take measurements at low altitudes. They also have pointing control requirements that lead to the need for a 3-axis controlled bus. Because of these pointing control requirements it also makes sense to group these instruments with the radio science/direct to earth telecom systems. It is likely that the telecom systems will have tighter pointing requirements than the instruments, so in this case the pointing requirements from the instruments do not even drive any bus requirements.

Group 3

The third group consists of the PEPA and RPA/IDM. Both instruments need to take measurements at both low and high altitudes. Another reason for this grouping is that these are the two instruments that create high voltages for their measurements. It is likely that fields created by these instruments could cause problems for the magnetometer measurements leading to a natural division. However, these two instruments are different in their pointing requirements. The PEPA has none, leading toward the possibility of a simple spinning spacecraft. The RPA/IDM wants to point along the velocity vector, a requirement met easiest by a 3-axis controlled spacecraft. Ideally the RPA/IDM would have been grouped with Group 2 to meet its pointing requirements, but the need to take measurements at high altitudes rules out that grouping. This led to this grouping and the design changes to the RPA/IDM requiring multiple sensors and accurate pointing knowledge. This is the least ideal of the three groupings, but with some innovations it can be made to work, even as a simple spinner.

Alternative Groupings

This mission has earlier been conceived of as operating with two spacecraft carrying all these instruments. This architecture has always been troublesome because of the orbit requirements of the instruments involved, but before distributed architectures were considered, no other implementation was envisioned. Using smaller, less capable but less expensive satellites this mission can avoid these problems.

This leads to questions about what other distributed architectures are possible. One option is to move to two groups. As mentioned, the RPA/IDM has pointing control requirements that lead to a 3-axis stabilized bus, so this instrument would be combined with the FPI and NIMS on the group 2 bus. The PEPA would then be combined with the magnetometer, PWA and LP on a spin controlled bus. There are two major concerns with this architecture. The first is that the RPA/IDM needs measurements at very high altitudes, where the FPI and NIMS are wasted. The second is that the PEPA does use high voltages, which complicates the magnetic field measurements. In most other respects this is an attractive grouping, but these problems do not exist with the current architecture.

Another option is to group these satellites strictly by their orbit needs. This leads to the combination of groups 1 and 3 from above, while leaving Group 2 as is. This leads to the same voltage/magnetometer combination conflict discussed above. In addition, the boom required for the spacecraft would cause problems with the fields of view required by the RPA/IDM sensors. These issues can be resolved, but they complicate the design of the spacecraft—perhaps unnecessarily, and will be more costly--given the proposed 3 group architecture. However, these alternative cases do allow for one less spacecraft design possibly reducing cost of the overall architecture. Another factor considered in the overall architecture was complexity. From a design, test and operational perspective, increased complexity, often measured by number of instruments and/or number of operating modes, leads directly to increased cost in a highly non-linear fashion. Other architectures are also possible, but the design team proceeded with the study using the 3 group architecture discussed above as it appeared to have the most promise of all possibilities identified, minimized individual complexity and optimized functionality. The development is now summarized.

4. MISSION ARCHITECTURE DEVELOPMENT

The system engineering and primary sub-system trades are founded on the following two hypotheses:

- Science per dollar increases with the number of spatially and temporally distributed micro-spacecraft spacecraft up to an unknown number of spacecraft.
- Distributed spacecraft may provide comparable or greater overall mission reliability with no more cost than a single monolithic spacecraft.

There is a long list of assumptions and criteria that must be satisfied for these hypotheses to be true; a list that has been partially presented in the introduction with more comprehensive assumptions provided in the reference literature. The potential benefits of multiple micro-spacecraft missions include:

- Increased science coverage
- Increased optimization of orbits
- More optimized instrument utilization due to orbit specific instrument placement. This avoids the often inefficient deployment and utilization of instruments when multiple instruments have *mutually exclusive spatial/temporal criteria.*
- Increased system-level reliability
- Low cost development through manufacturing and experience curve effects (assuming proper design and manufacturing methods are employed).

We performed system engineering analyses and trades that resulted in maximizing these benefits. The primary system engineering results are summarized as follows:

1. Science and instrument deployment criteria dictate the separation of instruments into clusters as discussed in section 3. This led to the three spacecraft types.
2. Next it was determined that four orbits were required based on science requirements.
3. Science pointing requirements led to different attitude control architectures as discussed in [1]. This also led to the addition of sensor heads on the RPA/IDM to meet measurement requirements.
4. Attitude determination and control was designed to increase autonomy using atmospheric drag and spin stabilization to control Group 1 and 3 spacecraft respectively.
5. Trades were made between commandable and simple state machine architectures. Simple state machines were used were possible. Attitude control was traded off with solar array size leading to larger area body mounted panels on Group 1 and 3, and deployed arrays on Group 2.
6. Propulsion systems where added only where necessary to save mass and cost, on the carrier stage and Group 2 spacecraft. Group 1 was designed without propulsion, and Group 3 uses only small solid propellant devices for spin-up and Vacuum Arc Thrusters (VATs) for nutation control.
7. Based on initial data volume estimates long-haul telecommunications require medium or high gain antennas, thus leading to the decision that long haul-telecommunications be placed on the spacecraft with highest instrument point requirements.
8. The number of spacecraft required in each orbit is a complex trade involving not only science coverage but a host of other factors, many of which are influenced by underlying criteria for validity of one or both of the hypotheses stated above, such as reliability.
9. The use of UHF for crosslink communications allows the use of low-gain omni-directional antennas and low-cost transceivers.

   Power was traded with telecom design, reducing the need for substantial array or battery size by only transmitting at short ranges (<5000 km)
10. Despite an emphasis on simplicity, one-way links were rejected in favor of a highly asymmetric 2-way protocol due to dramatic power savings with minimal other penalties.
11. The 2-way protocol chosen for the crosslinks is kept extremely simple and does not allow for commanding of the Group 1 & 3 spacecraft. This fulfills the minimum functional requirement for this crosslink without leaving the door open for significant cost growth and requirements creep.
12. The crosslink design allows the use of two different methods for crosslink ranging and POD: one-way Doppler and 'ping-pong' time-of-flight. This study shows that requirements can be met using the former. Either technique can be implemented using one or both of the Group 2 spacecraft. Further analysis should be performed.
13. The X-band transponder for communications with Earth was sized to allow download of the whole constellation's data volume in < 6 hours every

day. This keeps HGA dimensions similar to the host spacecraft size and avoids levying excessively tight pointing requirements.

Resulting Spacecraft Highlights

Figure 4 illustrates block diagrams of the three types of spacecraft and the carrier with high-level descriptions a follows.

Group 1a-b – (6 spacecraft total)

- Reliable science gathering for low cost
- Simple state machines
- Experience effects in design and manufacture for low cost
- Size/mass/quantity optimized for cost and science return with constraints including the launch vehicle and requirements of other spacecraft

Group 2 – (2 spacecraft total)

- Mission reliability
- Science gathering reliability
- Experience effects in design and manufacture for low cost
- Size/mass/quantity optimized for cost and science return with constraints including the launch vehicle and requirements of other spacecraft

Group 3 – (2 spacecraft total)

- Reliable science gathering for low cost
- Simple state machines
- Experience effects in design and manufacture for low cost

Size/mass/quantity optimized for cost and science return with constraints including the launch vehicle and requirements of other spacecraft.

Advanced technologies used in the three spacecraft design include:

- Command & Data Handling processors based on field programmable gate arrays (FPGAs)
- Telecommunications hardware
- Micro-propulsion components
- Power technologies
- Commercial micro-spacecraft buses
- Technical methods for up-screening commercial-of-the-shelf products.

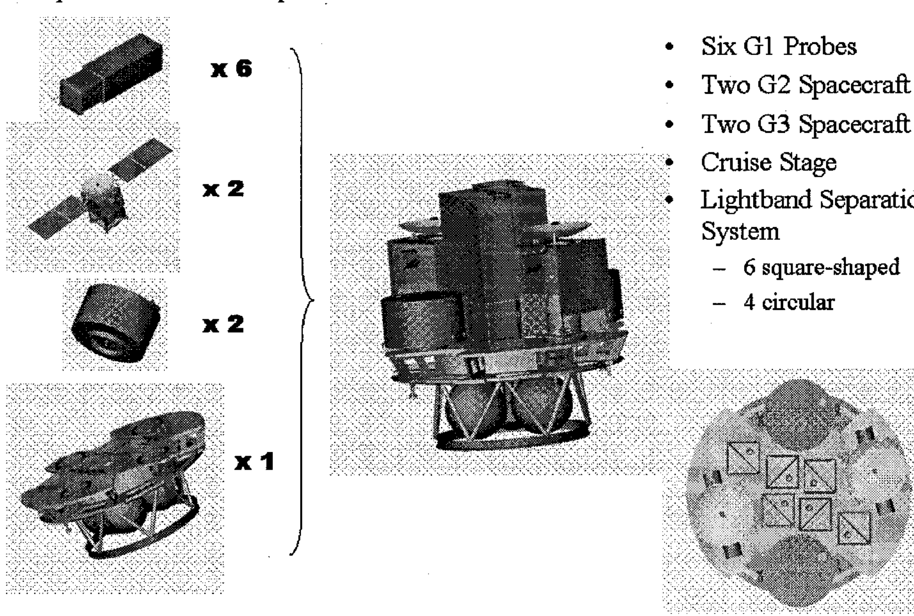

Figure 4 - Block diagrams of the three spacecraft types and the cruise stage

Mission Design

Mars Micro-Aeronomy mission is a multi-vehicle composite mission. It utilizes multiple vehicles that are designed for specific science functions and will operate in orbits that are ideal for those specific functions (see Figure 1). The entire mission will launch on a standard Delta IV 4040-12 with a 1575-4 Payload Attach Fitting (PAF) and will follow a Type II trajectory to Mars. Upon arrival, it will perform a standard MOI burn and capture into a high, elliptical 36 hour orbit. From there, the mission deviates from conventional single-spacecraft orbit insertion.

There are 10 individual vehicles each mounted onto a centralized propulsive carrier stage for launch. Three classes of vehicle are used for this mission and are shown below. There are 6 Group 1 spacecraft, 2 Group 2 Spacecraft and 2 Group 3 spacecraft. As shown below, these vehicles will package into the standard 4040-12 shroud with clearance.

The cruise stage is comprised of a main deck, a high performance dual mode bi-propulsion system, a power bus and interfaces to all 10 microspacecraft. All microspacecraft are mounted on the zenith deck and the entire propulsion system is mounted on the nadir deck. Cruise arrays are mounted on the deck and face in the nadir direction.

The Group 2 spacecraft are the central nodes for the assembly and perform the control and operations of the cruise stage. All spacecraft connect to a common power bus and communications bus through the cruise stage. Power from the cruise stage is used to supply the Group 2 node in command mode for the duration before the spacecraft are all deployed.

5. COST

The design team applied multiple costing tools, in addition to a grass roots estimate, to develop mission cost estimates for Phases A through E. The cost tools used are the parametric cost model (PMCM) typically used by Team-X at JPL and the Small-Sat cost model developed by the Aerospace Corporation, COCOMO II for software, and Advanced projects design team instrument cost model.

JPL's Parametric Mission Cost Model (PMCM) was used to validate the grass-roots estimate developed by the design team. PMCM models science, mission operations, spacecraft, management, and engineering cost. The model has been developed from many pre-phase A studies by JPL's Advanced Projects Design Team (APDTICM). The model has been validated against recent JPL missions (e.g., Mars Odyssey, Stardust, and Genesis).

The Small Satellite Cost Model, SSCM, is a parametric cost model which runs on any Microsoft Excel-supported platform. The latest version, SSCM Edition 2002 Pro Version, estimates the development and production costs of a small satellite bus for Earth-orbiting or near-planetary spacecraft.

SSCM is a parametric cost model which is the end-result of over ten years of study at The Aerospace Corporation. Its development was motivated by the observation that traditional cost models, based on larger civil and military systems, often times tended to over-estimate the development costs of modern, smaller satellites (post-1990, under 1000 lbs.). Initiated as an in-house study of small satellites and their capabilities in 1989, the development of the Small Satellite Cost Model has benefited from the efforts of many hours of data collection, normalization, and analysis, and remains one of the most relevant and credible cost models today for performing cost estimates of small spacecraft.

The total estimated cost for the distributed MAX mission as indicated earlier is ~$550 M in fiscal year 2003 dollars. This estimate does not include any technology development that might be required in the development of a real mission (e.g. the development of micro-transceivers for Group 1 spacecraft). It is important to note that this estimate is based upon a long list of assumptions that are not possible to comprehensively document here, and that the fidelity of estimates obtained with a limited study schedule and budget is lower than that which could be obtained if the hypothetical mission were in pre-phase A development. The exact "breakdown" (ground operations costs, launch costs, etc.) of the cost estimate requires a level of detail that is beyond the scope of this work.

6. CONCLUSION

An overview of a Mars aeronomy mission design based on micro-spacecraft was presented. The design consists of ten micro-spacecraft orbiters launched on a Delta IV to Mars polar orbit to determine the spatial, diurnal and seasonal variation of the constituents of the Martian upper atmosphere and ionosphere over the course of one Martian year. The estimated mission cost is ~$550 M and offers improved science gathering capability and total science data return over a previously studied Mars aeronomy mission design based on conventional spacecraft. An overview of the guiding design principles and framework utilized by the design team in the development of MAX was also presented.

REFERENCES

[1] JPL internal publication, Team-X report ID#: X-578

[2] R. Shotwell, A. Gray, P. Illsley, M. Johnson, R. Sherwood, M. Vozoff, J. Ziemer, "Mars Aeronomy Exploerer (MAX): Study Employing Distirbuted Micro-Spacecraft, JPL internal technical report, September, 2003.

BIOGRAPHY

Robert Shotwell *is the Project Systems Engineer for the Phoenix Mars Lander scheduled for launch in August, 2007. Prior to Phoenix, as a senior Systems Engineer Robert led various internal study teams and micro-spacecraft development groups, as well as deep space mission studies in the MSA section at JPL. He spent 8 years prior to this in the Propulsion and Advanced Propulsion groups at JPL, and has delivered extensive flight hardware for missions such as Mars Pathfinder, Deep Space 1, SRTM and SIRTF. He was a co-founder of the Eureka team and remains as a mentor, advisor and reviewer for the team. Robert earned his MS in Astronautics at USC in 2003, and a BS in Aerospace Engineering at Texas A&M University in 1995 (Magna Cum Laude).*

Andrew Gray *is a group supervisor of the Advanced Signal Processing Projects group in the Communications Architectures and Research Section, and co-coordinator for the forward-looking Eureka team that resides in the Mission and System Architecture Section at the Jet Propulsion Laboratory (JPL). Andrew earned his MBA and PhD in electrical engineering from the University of Southern California in 2004 and 2000 respectively, his MS in electrical engineering from the Johns Hopkins University in 1997, and his BS in electronics with minor in mathematics in 1994.*