Cyclical Visits to Mars via Astronaut Hotels

Phase II Final Report

Global Aerospace Corporation

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Abstract

Global Aerospace Corporation has developed a revolutionary concept for an overall interplanetary rapid transit system architecture for human transportation between Earth and Mars which supports a sustained Mars base of 20 people circa 2035. This comprehensive study includes the analysis, options development and design of the entire transportation infrastructure including orbital mechanics, in situ resources utilization systems, crewed and robotic cargo vehicles, planetary transportation nodes, technology identification, and costing. The baseline design architecture relies upon the use of small, highly autonomous, solar-electric-propelled space ships, we dub *Astrotels* for **astro**naut ho**tels** and hyperbolic rendezvous between them and the planetary transport hubs using even smaller, fast-transfer, aeroassist vehicles we shall call *Taxis*. Astrotels operating in cyclic orbits between Earth, Mars and the Moon and Taxis operating on rendezvous trajectories between Astrotels and transport hubs or *Spaceports* will enable low-cost, low-energy, frequent and short duration trips between these bodies. In addition, we have compared Mars transportation architectures namely Aldrin Low-thrust Cyclers, Stopover and Semi-cycler architectures. This concept provides a vision of a far off future that establishes a context for near-term technology advance, systems studies, robotic Mars missions and human spaceflight. This concept assists the NASA Enterprise for Human Exploration and Development of Space (HEDS) in all four of its goals, namely (1) preparing to conduct human missions of exploration to planetary and other bodies in the solar system, (2) expanding scientific knowledge (3) providing safe and affordable access to space, and (4) establishing a human presence in space. Key elements of the baseline Aldrin Low-thrust Cycler architecture are the use of:

- Five to six month human flights between Earth and Mars on cyclic orbits,
- Small, highly autonomous human transport vehicles or *Astrotels*,
    - Cycling between Earth and Mars
    - Solar Electric Propulsion for orbit corrections
    - Untended for more than 20 out of 26 months
    - No artificial gravity
- Fast-transfer, aeroassist vehicles, or *Taxis*, between Spaceports and the cycling Astrotels,
- Low energy, long flight-time orbits and unmanned vehicles for the transport of cargo,
- i*n situ* resources for propulsion and life support
- Environmentally safe, propulsion/power technology

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1 Introduction

This is the Final Report for Phase II of NIAC Universities Space Research Association Subcontract No.: 07600-59 for the development of the concept of Cyclical Visits to Mars via Astronaut Hotels. This report incorporates some Phase I results and Phase II Interim Report material. This comprehensive architecture study includes analysis, options development and design of the entire Earth-to-Mars transportation infrastructure including orbital mechanics, in situ resources utilization systems, crewed and robotic cargo vehicles, planetary transportation nodes, technology identification, and costs. In addition three transportation architecture concepts are examined including the baseline Aldrin Low-thrust Cyclers, Stopover Cyclers and Semi-Cyclers. For two of these architectures, the baseline Aldrin Low-thrust Cyclers and the Stopover Cyclers, detailed life-cycle costs (LCC) are estimated and compared. Furthermore, for these two architectures a number of subsystem mass and cost trades are performed and analyzed, namely, solar vs. nuclear power generation, the use of *in situ* resources for propellants, and the cost of launch services.

In this section we summarize the baseline concept for Cyclical Visits to Mars via Astronaut Hotels and we discuss the potential significance of this concept to NASA.

The primary objective of this concept is to provide low-cost, frequent access to Mars for scientists and explorers by means of cyclic visits to and from Mars using new concepts for interplanetary transport vehicles. Such a concept will have significant implications on our ability to understand one of Earth's nearest neighbors and our preparedness for future visits to other planetary bodies.

The concepts envisioned by the baseline systems architecture have a potential role to play in the expedition phase of Mars exploration. The application of these orbit and systems concepts in the expedition phase of Mars exploration may serve to reduce overall mission development costs and improve overall mission reliability and safety. Once launched into cycling orbits Astrotels can orbit indefinitely as long as they are periodically maintained, improved and supplied with orbit correction propellants. In addition, the result of embracing such a mission concept early in an expedition phase means that a permanent inhabitation phase of Mars is all the more closer.

Finally this interplanetary rapid transit concept provides a framework and context for future technology advance and robotic mission exploration. If one can envision an optimized interplanetary transportation systems architecture, then one can take steps today that will enable it. These steps could include establishing key technology goals to insure technology advance meets the future need. Other steps include embarking on robotic pathfinder missions to explore Mars, Phobos and the Moon and to search for *in situ* resources that are useful in any transportation systems architecture. For example, there is the high potential for the existence of water on the Moon, within Phobos and at the Martian North Pole. It is clear that the existence of water, or even just hydrogen, could have a dramatic impact on future plans and technology development for near-Earth exploitation and Mars exploration. Water broken down into its component molecular states of oxygen and hydrogen is rocket propellant. Hydrogen could be combined directly with oxygen for propulsion as with the current Space Shuttle. Alternatively, hydrogen could be combined with carbon to make methane, a more easily stored form of chemical energy. Past robotic missions have yet to resolved the issue of water at any of these bodies listed above. Unfortunately, there are also no planned missions to resolve the uncertainties at this time. A concept for an Earth-to-Mars transportation system could generate the interest and excitement necessary to get such missions off the ground.

1.1 Cyclical Visits to Mars via Astronaut Hotels

In 1985 the National Commission on Space (NCOS) published their plans for the future of space exploration, which included support to a sustained Mars base [National Commission on Space, *Pioneering the Space Frontier*, Bantam (1986)]. The NCOS plan assumed the existence of a sustained Mars base of 20 humans circa 2035, which required significant support in the form of crew replacement and cargo. The NCOS Mars base was supported by the use of large (>460 metric tonnes [mt]) interplanetary space ships for transporting humans and their material back and forth between the planets originally conceived by W. M. Hollister at MIT ["Castles in space" *Acta Astronautica*, 1967]. In addition, an entire support infrastructure was envisioned that includes human, cargo and propellant transfer vehicles, transport hubs and propellant manufacturing plants [K. Nock and A. Friedlander, Elements of a Mars Transportation System, *Acta Astronautica*, Vol. 15, No. 6/7, pp. 505-522, 1987].

The baseline Mars transportation system architecture concept being developed by Global Aerospace Corporation uses small, highly autonomous, solar-electric-propelled space ships, we dub *Astrotels* for **astro**naut ho**tels**, for transporting humans to and from Earth and Mars on cyclic orbits between these planets. Human transfer between planetary Spaceports and Astrotels is by means of hyperbolic rendezvous trajectories using new, even smaller, fast-transfer, aeroassist vehicles called *Taxis*. Figure 1-1 illustrates one concept for an Astrotel along with a Taxi docked at one end.

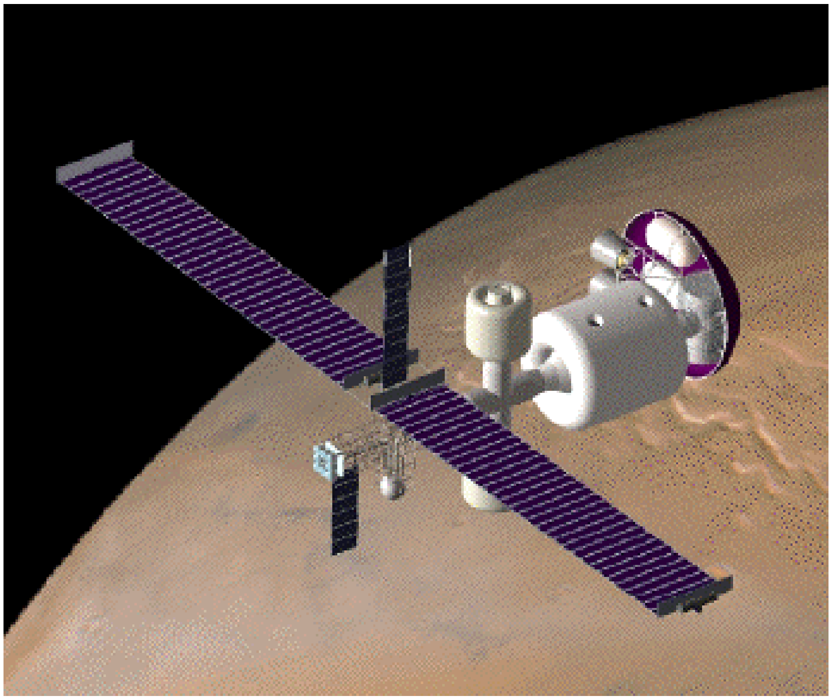

Figure 1-1 Astrotel and Taxi Concepts

These basic systems combined with other elements of the Mars transportation infrastructure and a new analysis of the celestial mechanics and aeroassist options will enable low life cycle costs, low–energy, frequent and short duration trips between these bodies. Figure 1-2 illustrates a schematic of the overall concept for regular human visits to Mars via an Astrotel concept that uses cyclic interplanetary orbits. The innovative design architecture being developed by Global Aerospace Corporation departs from the concepts in the mid-1980s in several fundamental ways. The goals of this new work are listed in the next section.

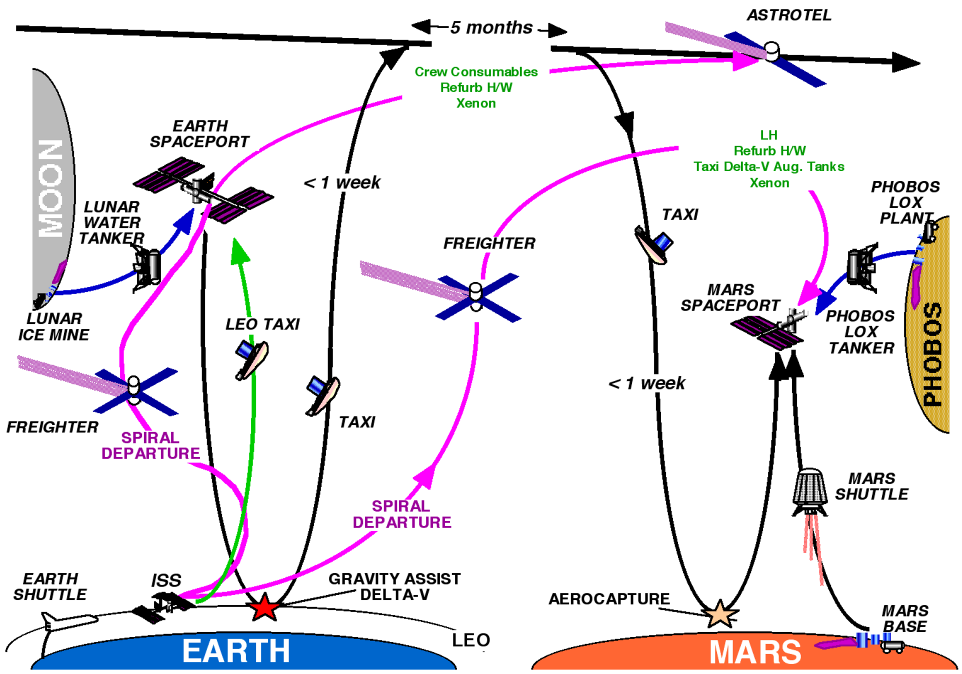

Figure 1-2 Mars Transportation Architecture Schematic

1.2 What are the Goals of this Work?

The primary goal of this work is to make Earth-to-Mars interplanetary transportation for a Mars Base a reality as soon as possible. Specific goals in support of a permanent Mars Base are listed below:

- Reduce crew in-space time
- Reduce vehicle sizes
- Rely on environmentally safe power generation
- Reduce propellant requirements
- Reduce reliance on Earth materials and hardware
- Eliminate need for new, expensive class of rockets and launchers
- Offer a concept that could accelerate the timeframe for permanent inhabitation of Mars

1.3 What Makes This Concept Revolutionary?

First, the baseline transportation systems architecture assumes the use of highly autonomous on–board systems to a) reduce the number of crew and b) their occupation time of the transfer space ships to only five months in interplanetary space. Recent experience with untended space flight on the Russian Mir and the construction phase of the International Space Station make it clear that crew are not essential to maintain support systems. Reducing the size of crew and reducing the duration of their time spent in space reduces the size of the space vehicle and its complexity and the amount of logistics supporting the daily needs of the crew. In addition, by eliminating crew on long flight legs, we eliminate the need for additional Taxis for return to Spaceports thus reducing the number required by one half. Because these Taxis are not carried on these long trajectory legs, Astrotel propulsion requirements are therefore reduced.

Second, in previous plans, a means to generate artificial gravity was required due to the lengthy crew stay time (up to 2 1/7 years). The Mir experience, Russian (one year) and US (Shannon Lucid's 6 month flight), indicates 6 months of zero-g are clearly tolerable. When transit times are reduced to no more than 5 months, artificial gravity may not be necessary thus reducing mass, complexity and risk.

Third, in past planning, conventional propulsion has been envisioned for the crew transport space ships using a Taxi's rockets. We are proposing instead to use solar electric propulsion for the periodic course corrections that are required for Astrotels (major corrections will generally occur during untended periods). Utilization of low-thrust Solar-powered Ion Propulsion reduces propellant mass requirements by a factor of 9. The cost in propellant mass for conventional chemical propulsion for course corrections for the large 460-mt vehicle over 15 years is more than 173 mt (more than twice our entire proposed Astrotel vehicle!). If we combine the interplanetary vehicle size reductions with SEP, the total reduction in propellant required for the Astrotel in 15 years is less by a factor of ***sixty***! This reduction has a tremendous mass and cost multiplying effect since all this propellant must also be mined, manufactured and stored, transported to the Spaceport and injected onto high-energy trajectories required for rendezvous with the Astrotels. See Table 1-1 for a comparison of several NCOS and preliminary NIAC study results including propellant requirements. As they are developed, evolutionary improvements in propulsion technologies will further reduce propellant requirements, but they probably will not change the fundamental architecture explored in this study.

Table 1-1 Comparison of NCOS and NIAC Study Results

Item	NCOS Study	NIAC Study	Improvement Factor
Cyclic Transport Vehicle Size, mt	460	70	7
Total 15-year Propellant and Consumables, mt	34,335	2,011	17
Lunar LOX Production Rate, kg/day	4,014	73	55
Phobos LOX Production Rate, kg/day	1066	189	6
Primary Power Generation Mode	Nuclear	Solar	--

Finally, in previous planning, all cargo except certain propellants needed at Mars, went via the same large crewed interplanetary space ship. The implication was that a lot of propulsive energy was being expended on hardware and supplies that could take a lot longer to get to Mars without detrimentally effecting the operation of the base.

All of these departures from the plans originally envisioned by the NCOS result in significant reductions in mass requirements and, therefore, they have enormous implications to overall energy requirements of a Earth-to-Mars transportation system. Reduced energy requirements impact the design of other elements of the transportation infrastructure and the cost of their development and operations. Since this new concept for support of a future Mars base results in a significant reduction in operations cost over previous concepts, a Mars base could be much closer to reality. In fact, elements of this concept could be implemented at the very beginning of Mars exploration insuring that the first humans to Mars begin the permanent inhabitation of this our nearest, most hospitable neighbor.

The key elements of the baseline concept for an Earth-to-Mars interplanetary rapid transit infrastructure are listed below:

- Cycler orbits between Earth and Mars that enable fast, frequent transfers between these planets
- Small, human transport space ships, or Astrotels, on cycling orbits between planets,
- Orbital Spaceports at the planets
- Very small, fast, hyperbolic transfer vehicles, or Taxis, between Spaceports and Astrotels.
- Propellant and life support *in situ* resource manufacturing plants
- Cargo vehicles that utilize low-energy, long-flighttime orbits to transport propellant and low value cargo to and from planets
- Shuttles to and from Spaceports and planetary surfaces

1.4 What is this Concept's Significance to NASA

This concept can assist NASA in addressing enterprise goals.

1.4.1 Enterprise for Human Exploration and Development of Space (HEDS)

HEDS has four goals addressed by this revolutionary concept, namely:

- Preparing to conduct human exploration missions to planetary and other bodies in the solar system,
- Expanding scientific knowledge,
- Providing safe and affordable access to space, and
- Establishing a human presence in space.

The proposed concept supports the first HEDS goal by providing a means to expand human exploration to Mars and by providing a transportation architecture that could be put in use to explore other planetary bodies, potentially near-Earth and Main Belt asteroids. The second HEDS goal is supported by this concept by enabling frequent, short visits to Mars by scientists. Opportunities for extended direct and teleoperated field science (e.g. geology) by scientists at

Mars will swiftly expand scientific knowledge of the planet and increase our understanding of its similarities and differences with our own planet. This transportation architecture supports the third HEDS goal by offering transport to and from Mars at an expected very low life cycle cost. High life cycle costs will limit Mars exploration by Apollo-like expeditions. If life cycle costs can be significantly reduced, permanent exploration and inhabitation of Mars can be argued as being cost effective. This concept very clearly supports the fourth HEDS goal by contributing to the establishment of a permanent human presence on the planet Mars. Finally, this concept could also provide future direction to NASA regarding flight system technology development that could set the stage for Mars expeditions in the future.

1.4.2 Space Science Enterprise (SSE)

This concept also supports an important goal of NASA's Space Science Enterprise (SSE), which is to "*pursue space science programs that enable and are enabled by future human exploration*". The mission and system architecture concept proposed by Global Aerospace Corporation assists NASA's SSE in their planning of future robotic exploration missions necessary to establish the key resource utilization technologies. In addition, there is the potential use of autonomous Astrotels in exploration of the main belt asteroids that needs to be explored. Small robotic vehicles could be deployed from each Astrotel for main-belt asteroid exploration when the Astrotels are at the farthest from the Sun at about 2 AU or alternatively they could be deployed at Mars approach where they could use gravity assist to raise perihelion. Such a concept could enable a series of very low-cost, main-belt asteroid, sample return missions.

1.5 What Will Mars Transportation Architectures Cost?

There are two types of costs that are important to discuss: non-recurring and recurring. Non-recurring costs include advanced studies, advanced technology development, flight and operations systems development and the cost of initially launching this hardware to space. Recurring costs include flight operations, development of refurbishment and upgrade hardware, and the cost of launching this hardware, new crews and any needed propellants into space. In Section 9 we summarize the results of several trade studies that were carried out using our Mission Architecture and Model Analysis (MAMA) tool. Trade studies included Stopover and Aldrin Low-thrust architectures, solar vs. nuclear power generation, the variation in launch costs, and whether or not *in situ* resource utilization (ISRU) is employed.

The results indicate that the total architecture life-cycle costs for solar power generation options range from $109.4B to 132.4B. The Aldrin Low-thrust architecture, with solar power generation and ISRU, has the lowest recurring cost (~$2.8B per year), though its total life-cycle cost (LCC) is ~$8B higher than the Stopover architecture. The lower LCC of the Stopover architecture is due primarily to the development of fewer vehicles. While the Stopover recurring costs are somewhat higher than the Aldrin cycler, ~$2.1B more over 15 years, if launch costs are as high as $10,000/kg, the difference in recurring costs grows to ~$12B. LCCs, assuming nuclear power generation, are much higher than for solar power generation ($204-297B), but this high cost may be partially driven by adverse scaling and refurbishment assumptions. Finally, we observe that assumed launch costs, at least between $2,000 and $10,000 per kg, can play a major role in determining which option is best.

2 Concept Development Summary

2.1 Summary of Phase II Tasks

Phase II of the Cyclical Visits to Mars via Astronaut Hotels Development includes the following tasks as originally planned and described.

2.1.1 Task 1 Cyclic Orbit and Celestial Mechanics Concepts Research

The Aldrin Cycler will be redesigned, if possible, so as to eliminate large propulsive maneuvers. An extensive search will be made for new cyclers and semi-cyclers and to explore the feasibility and benefit of Earth-Moon cycler concepts. New theoretical techniques will be employed to examine all possible free return trajectories. Low-thrust trajectories (using Solar Electric Propulsion) will be designed. The feasibility of employing a lifting body to eliminate propulsive maneuvers will be examined in concert with the aero-assist task. Promising patched-conic trajectories will be optimized for minimum propulsive maneuvers.

2.1.2 Task 2 Advanced Aero-assist Technology Studies

An end-to-end aero-assist model will be developed to simulate the controlled aerocapture of the taxi vehicle at Mars, with special emphasis on the highest approach speed cases where it is difficult to create the necessary centripetal force by aero means alone. This model will be used to investigate promising aero-assist modes for a Mars transportation system including the use of tethered aero-bodies for aero-gravity assist, propulsive burns during aerocapture, advanced altitude and g-load control techniques, and the use of ballutes to improve aero-assist performance.

2.1.3 Task 3 In Situ Resource Systems Concepts Development

This task will support the Mars Transportation system architecture development by providing technical expertise and by carrying out preliminary systems design trade studies of competing resource utilization options. We will develop in situ resource utilization system models for propellant production on the Moon, Phobos, Mars' surface and associated space propellant depots. We will develop system designs (system drawings, configurations, masses, power requirements, etc.) for resource operations and plants. We will emphasize the problems of excavating and processing materials in the near-zero–g, vacuum environment of Phobos. Recommendations on future robotic exploration goals will be provided as they relate to future in situ resource utilization.

2.1.4 Task 4 Develop and Assess Options for Mars Transportation Systems Concepts

We will refine the Astrotel and the Taxi life support, radiation shielding and structural designs. We will carry out additional system design definitions of the Earth and Mars Spaceports, the SEP Cargo Freighters, the Lunar Water and Phobos LOX Tankers, the LEO and Mars Shuttles.

Design and interface requirements of these vehicles will to be identified; computer-aided designs developed; and refurbishment, repair and upgrade needs refined. We will assess the subsystem commonality between vehicle systems to reduce overall life cycle costs.

The options for cyclic orbit designs, transportation node location, and ISRU need to be assessed in terms of design impacts to vehicles and surface systems. Options data for vehicles, systems, cyclic orbits, transportation nodes, subsystems technologies and ISRU will be generated and will be input into the MAMA system so trade studies of these options can be carried out and options compared.

The requirements on several, to date, peripheral elements of the Mars transportation infrastructure needs to be understood. These elements include Earth to LEO space shuttles, LEO space stations, launch vehicles, and a possible Lunar Base. We need to project the non-Mars transportation infrastructure ahead in time in order to assess its potential implication to a Mars transportation architecture development.

2.1.5 Task 5 Mars Astrotel Model Analysis (MAMA) Development and Life Cycle Costing

MAMA. Integrate the element requirements and supporting analyses and the system integrator (MAMA SI) modules. Incorporate a structured approach for updates. Revise input/output formats to facilitate assessment of Astrotel mission and design alternatives. Output formats shall include high-level and lower-level tabular results and graphical summaries. Incorporate, if feasible, the capability to simultaneously evaluate multiple design options to greatly facilitate conducting trade studies.

Life Cycle Costing. Early in Phase II, cost references used for each transportation flight system sub-element shall be reviewed/revised to ensure the most analogous cost reference is being used for each subsystem/component. Develop an enhanced estimating methodology including cost estimating relationships for lower-level Operations WBS elements based on operations cost drivers, versus using only Development costs. Identify potential cost saving approaches and develop methods to assess their impact. Incorporate, if feasible, a set of costing inputs to capture cost savings approaches and transportation architecture design options within MAMA.

2.1.6 Task 6 Identify Pathways to Architecture Development

The pathways to the future implementation of a Mars transportation architecture and its potential impact on near-term space development will be explored with NASA robotic and human Mars program planners and managers.

To facilitate discussion with NASA planners and program managers we will 1) invite key NASA personnel to a briefing to present the results of our Phase I study, 2) work with NIAC to ensure that key NASA personnel are invited to NIAC conferences, 3) encourage NIAC to hold a conference at NASA JSC, and 4) travel to NASA facilities to engage NASA personnel in discussions on the potential benefit of embracing a long range Mars transportation architecture.

2.1.7 Task 7 Planning and Reporting

A more detailed Phase II plan will be developed at project initiation. Monthly status reports, a mid-term report and a final report shall be written. We shall participate in and present status reports at the fall NIAC Fellows Conference in Atlanta, GA and at the NIAC Annual Meeting in Washington, D.C. held in 2001 and 2002. GAC will support a site visit by the NIAC Director at the GAC facilities in Altadena, CA, if deemed appropriate by the NIAC Director. Copies of all briefings, presentations or professional society technical papers pertaining to the Phase II study will be provided to NIAC.

2.2 Summary of Work Accomplished

This section provides a concise summary of the work accomplished during the Phase II effort. A more detailed description of the work follows in later sections. The progress made during Phase II is illustrated by this summary of accomplishments:

March 2001

- Prepared and released RFPs to Purdue University, the Colorado School of Mines, and SAIC.
- Received a consulting agreement from Michael Duke
- Began detailed planning for Astrotel Phase II
- Began preparing a paper for the Space Studies Institute, High Frontier Conference

April 2001

- Prepared a paper for the May 7-9, 2001 Space Studies Institute (SSI)/Princeton University, High Frontier Conference
- Received and reviewed subcontract proposals from CSM, SAIC and Purdue
- Began negotiating CSM, SAIC and Purdue subcontracts
- Continued detailed planning for Astrotel Phase II
- Prepared and delivered March monthly report.

May 2001

- Present a paper at the May 7-9, 2001 Space Studies Institute (SSI)/Princeton University, High Frontier Conference
- Prepared presentation material for NIAC Annual Meeting at NASA/ARC June 5-6, 2001
- Colorado School of Mines (CSM) and Purdue placed on contract
- Received, reviewed and commented on Purdue's first subcontract communication
- Continuing negotiations with SAIC on its subcontract
- Continued detailed planning for Astrotel Phase II
- Prepared and delivered April monthly report.

June 2001

- Gave presentation on concept at NIAC Annual Meeting at NASA/ARC June 6, 2001
- Began detailed simulations of Taxi and Mars Shuttle aero-assist trajectories.

- Began looking at low thrust optimization of cyclic orbits.
- Prepared material for Astrotel Team Meeting at Colorado School of Mines (CSM) on July 13, 2001.
- Continuing negotiations with SAIC on its subcontract.
- Began preliminary conceptual drawings of a cryogenic fuel depot satellite.
- Continued detailed planning for Astrotel Phase II.
- Prepared and delivered May monthly report.

<u>July 2001</u>

- Attended and gave presentations to Astrotel Team Meeting at Colorado School of Mines (CSM) on July 13, 2001.
- Developed preliminary list of system options to be studied in Phase II
- Computer simulations of Phobos excavator begun at Colorado School of Mines
- SAIC placed on subcontract.
- Continued detailed planning for Astrotel Phase II.
- Prepared and delivered June monthly report.

<u>August 2001</u>

- Prepared briefing material for Dr. Cassanova for meetings with NASA Associate Administrators
- Interview and animations for Swedish Education TV – "University TV"
- Compare reference architecture costs with architecture using Purdue optimized low thrust trajectories
- Initiated detailed Astrotel-Taxi-Spaceport hyperbolic rendezvous and phasing study
- Colorado School of Mines completed cryogenic fuel depot satellite and surface facility conceptual drawings and prepared summary of findings
- Initiated visit to NASA JSC for discussions on Astrotel work.
- Continued detailed planning for Astrotel Phase II.
- Prepared and delivered July monthly report.

- **<u>September 2001</u>**
- Prepared briefing on Astrotel concept to NASA JSC Exploration Office (Task 7),
- Began development of hyperbolic rendezvous software model (Task 1),
- Identified lunar orbit as a prime alternative transportation node (Task 1),
- Identified candidate additional mission applications for a high Earth orbit transportation node (Task 1),
- Completed Taxi radius trade study (Task 2),
- Colorado School of Mines revised propellant storage depot systems (Task 3),
- Began identifying major transportation architecture options to be modeled, analyzed and compared (Task 4),
- Initiated monthly Astrotel Team telecons (Task 7),

- Continued detailed planning for Astrotel Phase II (Task 7), and,
- Prepared and delivered August Monthly Report (Task 7).

<u>October 2001</u>

- Astrotel concept presented as headline story on Space.com (Task 7),
- Generated a complete set of trajectories for the ballistic stopover missions for years 2011 to 2024. (Task 1),
- Colorado School of Mines (CSM) began dust analysis on low-g bodies (Task 3),
- CSM prepared presented a paper "Conceptual Design of ISRU Propellant Storage Depots" at the 3rd Annual Space Resources Roundtable (Task 3),
- Decided on set of major transportation architecture options to be modeled, analyzed and compared (Task 4),
- Continued detailed planning for Astrotel Phase II (Task 7), and,
- Prepared and delivered September Monthly Report (Task 7).

<u>November 2001</u>

- A GAC hyperbolic rendezvous paper was been accepted for the Astro Conference in San Antonio Jan. 27-30, 2002 (Task 1),
- Colorado School of Mines (CSM) researched and completed a decision matrix of excavation alternatives for the Phobos low-gravity environment (Task 3),
- Visited NASA JSC to brief Exploration Office on Astrotel concept and have discussions on future funding opportunities (Task 6),
- Continued detailed planning for Astrotel Phase II (Task 7), and,
- Prepared and delivered October Monthly Report (Task 7).

<u>December 2001</u>

- GAC began preparing a paper “Hyperbolic Rendezvous for Earth-Mars Cycler Missions,” to be presented at the AAS Space Flight Mechanics meeting in San Antonio Jan. 27-30, 2002 (Task 1),
- Colorado School of Mines (CSM) final presentations for Senior Design Groups working on the carbothermal reactor for the Phobos environment (Task 3),
- Purdue University submitted an abstract titled “A Low-Thrust Version of the Aldrin Cycler,” for the AIAA Astrodynamics Conference, August 2002,
- Began Mars Shuttle system design activity, (Task 4),
- Continued detailed planning for Astrotel Phase II (Task 7), and,
- Prepared and delivered November Monthly Report (Task 7).

<u>January 2002</u>

- Began preparing for Astrotel Site Visit scheduled for February 7, 2002 (Task 7)
- Presented paper “Hyperbolic Rendezvous for Earth-Mars Cycler Missions,” at the AAS Space Flight Mechanics meeting in San Antonio Jan. 27-30, 2002 (Task 1),
- Began detailed aero-assist analysis of Mars Shuttle vehicle (Task 2), and,
- Began detailed system design study of Mars Shuttle vehicle (Task 4),

February 2002

- Continued detailed aero-assist analysis of Mars Shuttle vehicle (Task 2),
- Continued detailed system design study of Mars Shuttle vehicle (Task 4),
- CSM completing their Final Technical Report to GAC,
- Continued preparations for providing data for the new MAMA model (Tasks 1 & 5),
- Hosted NIAC Site Visit at GAC World Headquarters in Altadena, CA on February 7, 2002 (Task 7), and,
- Completed and delivered the Phase II Interim Report (Task 7).

March 2002

- Continued to generate MAMA trajectory data especially as it relates to cargo freighter low thrust trajectories (Task 1),
- Continued refinement of Crew Module design in order to support Mars Shuttle design efforts (Task 4)
- Continued Mars Shuttle ascent analysis and carried out further Taxi Mars aerocapture studies (Task 2)
- Completed the February Monthly Report, (Task 7).

April 2002

- Continued to generate MAMA trajectory data especially as it relates to cargo freighter low thrust trajectories (Task 1),
- Continued refinement of Crew Module design in order to support Mars Shuttle design efforts (Task 4)
- Continued Mars Shuttle ascent analysis (Task 2), and
- Completed the March Monthly Report, (Task 7).

May 2002

- Began generating a full 30-year sequence of Astrotel cargo freighter low-thrust trajectories (Task 1),
- Continued refinement of Crew Module design in order to support Mars Shuttle design efforts (Task 4)
- Continued development of conventional and Phobos ISRU Mars Shuttle aerobrake concept
- Continued Mars Shuttle ascent analysis (Task 2), and
- Completed the April Monthly Report, (Task 7).

June 2002

- Continued generation of a full 30-year sequence of Astrotel cargo freighter low-thrust trajectories (Task 1),
- Completed refinement of Crew Module design in order to support Mars Shuttle design efforts (Task 4)

- All required inputs and outputs for all the MAMA modules have been identified and are being incorporated into the database (Task 5),
- Continued Mars Shuttle ascent analysis (Task 2), and
- Completed the May Monthly Report, (Task 7).

July 2002

- Completed generation of a full 30-year sequence of cargo freighter low-thrust trajectories (Task 1),
- Continued development of Mars Shuttle aerobrake concepts (Task 3),
- Completed versions of all lower-level MAMA modules (Task 5),
- Continued Mars Shuttle ascent analysis (Task 2), and
- Completed the June Monthly Report, (Task 7).

August 2002

- Began defining the thruster on/off periods for the cargo freighter low-thrust trajectories (Task 1),
- Presented a paper on Stopover trajectories at the AIAA Astrodynamics Conference in Monterey, CA (Task 2),
- Continued development of Mars Shuttle aerobrake concepts (Task 3),
- Completed draft versions of all lower-level MAMA modules (Task 5), and
- Completed the July Monthly Report, (Task 7).

September 2002

- Continuing development of algorithms for delta-V calculations for MAMA (Task 1),
- Developing optimal techniques for alignment of Spaceports for Taxi departures (Task 1)
- Completed Mars Shuttle Ascent analysis (Task 3),
- Continued development of Mars Shuttle aerobrake design (Task 3),
- Completed the August Monthly Report, (Task 7).

October 2002

- Developed an optimal techniques for alignment of Stopover cycler for planet departures (Task 1)
- Completed Generation of Cargo Freighter Timeline data (Task 1),
- Continued development of Mars Shuttle deployable aerobrake design (Task 3),
- Completed the September Monthly Report, (Task 7).

November 2002

- Analysis of Mars and Astrotel Cargo Freighter low-thrust trajectories (Task 1)

- Began update of Aldrin Cycler trajectory data incorporating new Spaceport location and departure phasing geometry (Task 1),
- Began design revision of Taxi vehicle including augmentation tankage configuration (Task 3),
- Completed the October Monthly Report, (Task 7).

December 2002

- Generate Earth-Mars Stopover trajectory data (Task 1)
- Continued design revision of Taxi vehicle including augmentation tankage configuration (Task 3),
- Began generation of Stopover version of MAMA (Task 5),
- Was invited by the Advanced Systems Office of the NASA Office of Space Flight to attend its Transformational Space Concepts and Technologies (TSCT) for Future Missions January 14-16, 2003 (Task 6)
- Completed the November Monthly Report, (Task 7).

January 2003

- Modify delta-Vs for MAMA for Earth-Mars Stopovers (Task 1)
- Continued design revision of Taxi vehicle including augmentation tankage configuration (Task 3),
- Completed development of Stopover architecture concept, (Task 4),
- Completed generation of Stopover version of MAMA and cost module (Task 5),
- Attended Transformational Space Concepts and Technologies (TSCT) for Future Missions Meeting, January 14-16, 2003 (Task 6)
- Began generation of presentation for Space Technology and Applications International Forum (STAIF), (Task 6)
- Completed the December Monthly Report, (Task 7).

February 2003

- Generated Stopover and no ISRU MAMA versions (Task 5),
- Began drafting Final Report, (Task 7).

March 2003

- Completed MAMA tool development and begun several cost trade studies, (Task 5),
- Completed and sent preliminary draft of ICES paper to session chair, (Task 7), and
- Continued writing Final Report, (Task 7).

2.2.1 Task 1 Cyclic Orbit and Celestial Mechanics Concepts Research

Purdue University's research in cyclic orbit and celestial mechanics concepts included attempted redesign and re-optimization of the Aldrin Cyclers, search for new cyclic orbit opportunities using the Tisserand Graph technique to locate all possible free return trajectories, and development of low-thrust trajectory optimization techniques. Unfortunately, Purdue's redesign and re-optimization of the ballistic Aldrin Cyclers did not result in finding any better trajectories. In addition, the search for new cyclic orbit opportunities using the Tisserand Graph technique between Earth and Mars was not successful. Most of Purdue's research efforts were directed at developing and adapting GALLOP, a low-thrust trajectory optimizer, for application to optimizing Aldrin Low-thrust trajectories. In Phase I, SAIC investigated the use of low-thrust burn arcs on Aldrin Cycler orbits that require delta-V maneuvers to keep the cycle repeating. SAIC optimized the low-thrust burn arc for each orbit but not for the entire 15-years. Purdue's goal was to optimize the entire 15-year sequence of orbits altogether instead of one orbit at a time as was done by SAIC in Phase I. If this could be done, one could then trade Mars V-infinity (directly related to Taxi delta-V) for low-thrust orbit delta-V by the Astrotel. Purdue used the data generated in Phase I to understand the relationship between Astrotel V-infinity at Mars and required Taxi propellant for rendezvous. One 15-year orbit case studied (actually beyond Astrotel limits) resulted in lower Mars V-infinities equivalent to a reduction of about 114 mt of Taxi fuels for Mars departure along with 17 mt of auxiliary tanks required to carry this propellant. This reduction came at the expense of an additional 10 mt of low thrust propellants for the Astrotel. MAMA studies reported that this was a favorable trade indicating that Mars V-infinity can be successfully traded for Astrotel delta-V. At the end of their funded studies, Purdue had made considerable progress in this direction, but were still having difficulties with position and velocity mismatches and keeping the power levels consistent with the Astrotel constraints. Since, their studies ended under GAC funding, Purdue has continued research in this area on their own and have made progress in obtaining 15-year optimized trajectories. At the time of this report, however, they had not yet developed a technique to constrain flyby conditions in order to minimize Mars V-infinity. Using conventional low-thrust software codes, SAIC has assisted GAC in validating the Purdue low-thrust augmented Aldrin Cyclers orbit analysis in order to ensure that low-thrust burn requirements were within Astrotel Ion Propulsion System (IPS) power limitations.

GAC's Dr. Paul Penzo began studying the Astrotel-Taxi-Spaceport hyperbolic rendezvous and phasing problem at Earth. The geometry and trajectory options for departure from circular orbits were studied. New insights from this work have had a dramatic effect on the choice on Spaceport location. Lunar Orbit Radius (LOR) was selected as a preferred Spaceport location due to the ease of rendezvous rather than the Earth-Moon L_1 point that was selected in Phase I. Dr. Penzo submitted an abstract on "Hyperbolic Rendezvous for Earth-Mars Cycler Missions" to the San Antonio AAS Flight Mechanics Meeting January 2002. Detailed trajectory states and encounter times were received from SAIC for the Basic Aldrin cycler, and an interface program has been written to use this data to provide inputs to a program completed to compute the 3-burn (and three dimensional) transfer from an arbitrary circular orbit to hyperbolic rendezvous. We also began to identify non-Mars transportation architecture mission applications for a high Earth orbit transportation node.

A complete set of optimized trajectories has been generated for the ballistic stopover missions (Earth to Mars and Mars to Earth). These have been used to develop three scenarios depending on how propellant tanks are allocated. These data clearly show that there is a trade-off between propellant tank sizing, tank loading and flighttime.

Finally, SAIC generated a full 30-year sequence of Astrotel and Mars Cargo Freighter low-thrust trajectories including thrust periods, stay times, and planetary spiral durations.

2.2.2 Task 2 Advanced Aero-assist Technology Studies

GAC's Purdue co-op (summer 2001 and 2002), working closely with Dr. McRonald, used a planetary aerocapture model in order to generate Taxi and Mars Shuttle aero-assist simulations. For Mars and Earth Taxi aerocapture trajectories, we found that the initial angle of attack (AOA) has little effect on the start of the aerocapture maneuver. We decided on constant AOA throughout the aerocapture maneuver since fixed AOA is the traditional, more conservative strategy and there seems to be no particular advantage to AOA variation. Instead, roll modulation of the lift vector is being used to control the aerocapture process. Using the prototype elliptical raked cone vehicle shape of NASA TM 58264, an analysis was carried out of the variation of mass expected if the Taxi radius were changed keeping constant the forebody shape and the aerocapture deceleration and flight time. The result was that the vehicle flies level at a new altitude and at a new level of stagnation point heating when the vehicle size is changed but there is very little difference in subsystem design. In addition, we studied the benefits of thrusting during the aerocapture maneuver in reducing the total g-load on the vehicle. Delta-Vs and corresponding propellant loads were calculated. Aerocapture trajectories at Earth were evaluated for vehicles returning to LEO from lunar orbit radius.

Mars Shuttle entry trajectories were analyzed in detail. Parameters of interest are peak heating rate, time-integrated stagnation point heating, and peak g-load, as a function of the entry angle and vehicle ballistic coefficient. These parameters impact structural mass, drag area and heat shield mass. Newtonian lift (C_L) and drag coefficients (C_D) were evaluated for several variants of the Mars Shuttle prototype vehicle. Ascent trajectories of the Mars Shuttle were evaluated to determine delta-V losses as a function of the ballistic coefficient of the vehicle (a function of cross-section area and shape), the thrust/mass ratio of the motors, path angle of the ascent, and tankage factor. Mars Shuttle landing trajectories were studied in order to understand the requirements on vehicle design including number and size of rocket engines.

2.2.3 Task 3 In Situ Resource Systems Concepts Development

The Colorado School of Mines developed concepts for those elements of the Astrotel concept that involve the use of resources derived from materials on the Moon, Mars and Phobos. This includes consideration of excavation and extraction systems for production of water, oxygen and hydrogen and the translation of these concepts into spreadsheet formats for use with MAMA that will describe the Astrotel scenario and will form the basis for analysis of performance and cost for the architecture. In Phase I, the work was limited to spreadsheet modeling, based on previous studies of mining and extraction systems. In Phase II, CSM integrated recent design studies of Mars, Moon and Phobos excavation and extraction systems. The design work for Mars and lunar extraction systems was supported by other funding mechanisms; the Phobos excavation and extraction systems had not previously been studied in detail, so emphasis has been placed on those systems for this study.

Phobos is a low gravity body. The work conducted in Phase II has addressed the problems of excavation of regolith under these environmental constraints by analyzing various conventional excavation techniques and identifying techniques that are particularly suited to the Phobos environment. Models for excavation systems are adapted for the low gravity environment and the environmental limitations used to determine constraints on the mass, power and excavation rates of systems designed to be small in scale. These systems, while potentially testable using robotic exploration missions of typical small scale (a couple hundred kilograms in mass), are expected to be able to perform excavation at rates that are compatible with the Astrotel scenario, though it may take several excavators to provide the required excavation rate.

Phobos is assumed to be of dehydrated carbonaceous chondrite composition. Studies are performed that evaluate the chemical separation of oxygen from Phobos regolith by carbothermal reduction, based on previous analyses of carbothermal reduction of lunar regolith. Methods were studied to deal with the expected difficulties due to the high sulfide content of carbonaceous chondrite material, and to optimize the production of oxygen as a function of mass and power requirements for the system.

The Astrotel scenario requires propellant production facilities on the Moon, Mars and Phobos and capabilities to produce and/or store and transfer propellants on those bodies as well as the Mars and Earth spaceports.

In a trade study analyzed with a spreadsheet model, a comparison of the production of oxygen from Phobos regolith was conducted under assumptions that (1) liquid oxygen is produced on Phobos and transferred to the Mars spaceport; (2) regolith is mined on Phobos and transported to the Mars spaceport for extraction of oxygen there, in cases where the space port is near in DV to Phobos and where the spaceport is significantly distant in ΔV from Phobos. In addition, the case where Phobos regolith contains 10% bound water (i.e., the regolith is not dehydrated) was considered to address the potential importance of discovering water on Phobos.

Finally, the ISRU spreadsheet-based model was re-evaluated to ensure proper scaling for the Stopover architecture, which utilize considerable more propellants than the Aldrin Low-thrust Cycler architecture.

2.2.4 Task 4 Develop and Assess Options for Mars Transportation Systems Concepts

Options were generated for the Mars Transportation System in almost every element of the architecture. Major transportation architecture options have been identified as a first step in defining the major MAMA format. Cyclic orbit options initially considered included: low-thrust Aldrin cycler, basic ballistic Aldrin cycler, low-thrust 78-mo. semi-cycler, ballistic 78-mo. semi-cycler, low-thrust stopovers, and ballistic stopovers. After evaluating the degree to which these orbit options meet the architecture requirements, only three options remained. Note that neither the Semi-cycler or Stopover trajectory options are truly cyclic trajectories since they require stops at Earth and/or Mars. No other cyclic trajectory option met or came close to the minimum crew flight duration requirement of 6 months, hence they were not included. From this set of orbits, major transportation architecture options were selected as a first step in defining the major MAMA format. These architecture options include:

1. Low-thrust Aldrin Cycler- Current Baseline
2. Ballistic 78 mo. Semi-cycler
3. Ballistic Stopovers

These options were compared with respect to vehicle needs and types, node locations and resource needs in order to assess the implications on the set of infrastructure developed for the reference architecture. The following comparison table was constructed in order to summarize the required elements of each architecture option. The Stopover trajectories are included in the MAMA analysis in Phase II so that the Aldrin Cyclers can be fairly compared with at least one competitive and highly touted option.

Table 2-1 Major Architecture Options and their Implications of Infrastructure

Transportation Architecture		Reference Low-thrust Aldrin	High-thrust Semi-cycler	High-thrust Stop-overs
Cycler Orbit Type		Modified Aldrin Cyclers that use low-thrust propulsion to achieve required apsidal rotation to continue the cyclic flyby sequence of Earth and Mars. (Phase I study reference)	Three synodic period (78 month) orbits between Earth and Mars with flybys of Earth and Taxi-based LOX/LH propulsion stops at Mars, into Phobos orbit radius.	Type I, 180 day transfers to and from Earth and Mars with use of LOX/LH propulsive capture at both planets. Capture into Phobos orbit radius at Mars.
Crew Transport Duration		5 months	6 months to/from Mars	6 Months Transfers. If desired, trade off between flight-time, propellant loading and tank sizing.
Transport Nodes	LEO	Low Earth Orbit Space Station.		
	Earth/Mars	Earth Spaceport, Lunar orbit radius; Mars Spaceport, Phobos orbit radius.	Earth Spaceport, Lunar orbit distance; Use Astrotel at Mars, Phobos orbit radius.	Use Astrotel at Earth, Lunar orbit; Use Astrotel at Mars, at Phobos orbit radius.
	Astrotel	Astrotel Vehicles, Low-thrust	Astrotel Vehicles, LOX/LH Propulsion	
Inter-node Transport	LEO to Earth Spaceport	Taxi Vehicle, Aero-assist, LOX/LH		
	Earth Spaceport to Astrotel	Taxi Vehicle, Aero-assist, LOX/LH		None
	Mars Surface to Orbit	Mars Shuttle, Crew & Cargo, Aero-assist, LOX/LH Propulsion		
Cargo Transport	Earth to Orbit	Earth to Orbit Cargo Launcher		
	LEO to Earth Spaceport	Astrotel Cargo Freighter, Low-thrust	Mars Cargo Freighter, Low-thrust	None
	LEO to Astrotel	Astrotel Cargo Freighter, Low-thrust	None	
	LEO to Mars	Mars Cargo Freighter, Low-thrust	Mars Cargo Freighter, Low-thrust	

All the architectures assume use the lunar or Phobos orbit radius as a planetary node. The reason for this node location is that in all the architectures we expect to utilize the resources of the Moon and Phobos thus necessitating close proximity to those resources. In addition, a very high elliptical orbit node at Mars significantly increases the size of the Mars Shuttle fuel load on the surface of Mars and thus the overall size and mass of the vehicle. An alternative to the larger Mars Shuttle is a Mars Spaceport at Phobos radius and the need for a Taxi vehicle of some sort to transport crew between the Semi-cycler or Stopover Astrotels. The addition of vehicles would require additional development costs which is counter productive to the desire to reduce total life-cycle cost hence the lower orbit nodes near resources.

We also carried out a number of detailed and/or refined system design studies for the Mars Shuttle, the Common Crew Module (used in the Mars Shuttle and the Taxi), and the Mars departure Taxi with augmentation tanks. The results of detailed aero-assist studies (see Task 2) were factored into the Mars Shuttle design. The Mars Shuttle concept uses a deployable aeroshell concept for entry that is retracted for Mars surface launch.

2.2.5 Task 5 Mars Astrotel Model Analysis (MAMA) Development and Life Cycle Costing

In Phase II, SAIC developed a tool to support trade study analyses of Mars Astrotel Concepts – referred to as the Mission Architecture and Model Analysis (MAMA) system. MAMA is used to identify technologies and approaches with potential for high leverage, develop and compare high level metrics based on results from in depth analyses, and to capture requirements for all life cycle cost elements for a given architecture design. MAMA is actually a collection of multiple lower-level modules, each addressing a different aspect of the Mars Astrotel architecture. The modules include 11 "Vehicle" types, 2 "Transportation Nodes", 5 "Other Elements", and 9 "Support Analyses". Each module has tailored input/output templates along with identified ties to the other MAMA modules. MAMA includes database functions designed to collect all inputs/outputs for a given architecture design to facilitate trade studies. Outputs include a detailed equipment list for all flight elements and life cycle costs to a detailed WBS covering all phases from start of advanced technology development through 15 years of mission operations. In Phase II, three high-level architecture types were analyzed and modeled including Aldrin cyclers with ISRU, Aldrin cyclers without ISRU, and Stopover cyclers with ISRU. Also, for any given concept, the primary power sources used for major flight elements can use either solar arrays or a nuclear reactor.

2.2.6 Task 6 Identify Pathways to Architecture Development

In Phase II we identified potential non-Mars transportation architecture mission applications for a high Earth orbit transportation node. There is interest at NASA in large space observatories that operate either in very high Earth orbits or in heliocentric orbits just outside the Earth's sphere of influence in order to reduce the thermal interference caused by the Earth and/or the Moon. Today, when the Hubble Space Telescope requires servicing, the Space Shuttle is sent into LEO and astronauts carry out the required refurbishment, repair and upgrade of its subsystems. For the expensive space observatories of the future in very high Earth or nearby heliocentric orbits, there is a question of how to make these expected service calls. One possibility is to return the observatory to a high Earth orbit, crewed service facility by means of low-thrust propulsion tugs. Once at the service facility astronauts can do the needed servicing. Such a facility could be located at a future Earth Spaceport transportation node in high Earth orbit in order to reduce the recover time and the propulsive requirements. In addition to science observatory servicing, such servicing operations could provide valuable experience in operations beyond LEO that could be of importance to a future Mars transportation system. In this manner near-term space objectives could be met while at the same time creating a framework for a future transportation architecture.

GAC traveled to NASA JSC in November 2001 and June 2002 to brief the Astrotel Concept to elements of the Exploration Office. Interest was expressed in low-thrust trajectory shaping of cyclic orbits, the substantiation of the need for ISRU, and the integration of the Space Launch Initiative costs into the study. Assistance was offered and contact provided for scaling habitability systems. GAC discussed several opportunities for NASA to leverage ongoing NIAC efforts. Several opportunities for NASA leveraging are listed below.

Trajectory Studies

- Expand search for new low-thrust-augmented cyclic orbits
- Extend transportation node options and hyperbolic rendezvous options
- Application of transportation nodes to observatory servicing

ISRU Activities

- Expand modeling and prototyping of excavation and extraction systems
- Broaden ISRU resource and system options
- Develop resource exploration & demonstration mission concepts

Aeroassist Studies

- Develop advanced aeroassist guidance algorithms
- Assist in the development of demonstration mission requirements

MAMA System and Cost Model

- Develop user-friendly, portable Java-based tool
- Expand applicability to other complex system designs

In January 2003, we were invited to attend and give a presentation on the Astrotel concept at the HEDS Transformational Space Concepts and Technologies (TSCT) for Space Missions technical interchange meeting at Caltech in Pasadena, CA.

2.2.7 Task 7 Planning and Reporting

GAC and its subcontractors participated in conferences, prepared technical papers, prepared and participated in briefings to NASA and were the subject of media interviews and articles on the NIAC advance concept. These are listed Section 10. In addition, GAC wrote several monthly reports to NIAC summarizing the technical progress of the work.

3 Key Conceptual Design Definitions, Assumptions and Requirements

This section contains a brief description of system definitions, key concept assumptions and requirements for a transportation system for astronauts between Earth and Mars. These conceptual design requirements are levied on the design of the operational system. The requirements in this section provided conceptual design requirements for use in the conceptual design phase of Phase II of the NIAC Astrotel study.

3.1 Definitions

3.1.1 Mars Base

The Mars Base is a permanent human surface station and research center for scientific study of Mars and its environs. It consists of human habitats and life support systems; warehouses; science laboratories; any resource mining, storage and distribution facilities; shuttle landing, operations, refueling, refurbishment and launch facilities; and human and robotic surface mobility systems and their refurbishment facilities.

3.1.2 Astrotels

Astrotels is a contraction of the words **Astro**naut Ho**tels**. An Astrotel is a crew habitat for fast human trips between Mars and Earth.

3.1.3 Spaceport

Spaceports are collection points for the arrival and distribution of humans, cargo and propellants destined for transport to planet or natural satellite surfaces or to cycling Astrotels. Spaceports shall support crew needs during the crew stopovers between interplanetary Taxis and planetary Shuttles. Spaceports perform delta-V maneuvers required for station-keeping, provide protection to the crew from solar flare, or solar proton events (SPE), and cosmic ray radiation, provide electrical power to its subsystems, are highly autonomous, and are capable of refurbishment and autonomous resupply.

3.1.4 Taxi

Taxi is a small, self-contained, crewed, aeroassist spaceship for fast human transfers between Spaceports and Astrotels. Taxis employ aeroassist technology within a planetary atmosphere to reduce orbit energy thus facilitating orbit capture. Taxis utilize propulsion systems to escape planetary Spaceports, to rendezvous with Astrotels, to depart Astrotels and to rendezvous with Spaceports.

3.1.5 Shuttle

Shuttles are crewed aerospace vehicles for human and time-critical cargo travel between Spaceports and space stations or planetary surfaces. The characteristics of these vehicles may be quite different for Earth and Mars application.

3.1.6 Lunar and Phobos Propellant Tankers

These unmanned vehicles are required to transport raw material (Lunar Water Tanker or Phobos LOX Tanker) to Spaceport fuel depots where the materials are processed into propellants and/or the propellants are stored.

3.1.7 Cargo Freighter

Freighters are unmanned cargo vehicles for transporting bulk materials between planets, Astrotels and Spaceports. Characteristics of Freighters may vary, however they are expected to be made from the same modular building blocks. Freighters are reusable.

3.1.8 Propellant Augmentation Tanks (PATs)

Propellant augmentation tanks (PATs) provide additional propulsion capability to vehicles when delta-V requirements exceed the base vehicle capability. For example, when Taxi delta-V requirements exceed the Taxis capability (about 3.4 km/s) for Mars departures, PATs are added to the Taxi. PATs are not planned to be reusable. In some cases, the very large Mars delta-Vs, PATs must include additional rocket engines in order to reduce burn time and thus gravity losses.

3.1.9 In situ Resource Production Plants

These include planetary and orbital facilities where propellants are created from indigenous materials, e.g. oxygen, water, etc. or are processed e.g. electrolyzed, liquefied and stored. Potential locations for such plants include the Martian surface, Phobos and the Moon. Potential locations for processing plants include the Earth or Mars Spaceports.

3.2 Transportation System Assumptions

There are both assumptions that govern this conceptual design study of an Earth/Mars transportation system. A number of assumptions have been made that form the basis for the conceptual design study. These assumptions include the basic timeframe of Mars Base operations, number of people in a Mars Base, the technology horizon for the study, cargo transport assumptions, Spaceport locations, and propulsion and power systems.

3.2.1 Timeframe of Mars Base

The timeframe for a sustained Mars Base is assumed to be circa 2035.

3.2.2 Mars Base

It is assumed that the full Mars base crew is 20 human beings and many robots. Half the crew is assumed rotated to Earth on each opportunity. This means that there are times when there are only 10 people on Mars for short periods (2-4 months). In addition, it is assumed that all food and life support consumables required at Mars Base are grown and processed at or near the base.

3.2.3 Lunar Base

It is assumed that a sustained Lunar Base exists for its own scientific rationale. The Mars transportation requirements imposed on a Lunar Base are therefore only incremental in nature.

3.2.4 Technology Horizon

For the purpose of this study, the technology horizon will be 2010. All technology is assumed to be at NASA Technology Readiness Level (TRL) equal to 9 by 2010. This means that this study will not employ technologies if they are not projected to be at TRL-9 by the end of 2010. This early date, as compared to the 2035 date for a sustained Mars Base, insures that the systems and architecture concepts developed in this study could be utilized for near-term missions to Mars and that there is a clear near-term pathway to infrastructure development.

Typically, technology advances despite plans or the lack thereof. It is therefore desirable to have a system architecture that is robust to reasonable technology advance, which means that when technology evolves, the system architecture can take advantage of that advance without wholesale alteration of the system architecture.

3.2.5 State of Scientific Knowledge

It is assumed that the currently envisioned Mars robotic science exploration program will be carried out through the return of one or more samples from Mars by 2012. These missions will provide fundamental data for Mars surface mission planning and propellant production.

3.2.6 Space Station

A Space Station in LEO is a key transportation node of any Mars transportation infrastructure. It is the place where Earth cargo will be collected for transport to Mars and the Astrotels and where crew will be transferred between LEO and the Earth Spaceport.

3.2.7 Robotics and Automation

Extensive use of robotics and automation are assumed throughout the entire Mars transportation system architecture. Robots and other automated machines carry out equipment monitoring, fault protection, cargo handling, refueling, periodic machine maintenance, *in situ* resource system operation and many others activities.

3.2.8 Cargo Transport

Bulk cargo such as propellants, PATs, refurbishment hardware, and other non-time critical cargo are sent to Mars, Planetary Spaceports, and Astrotels by means of reusable Cargo Freighters that fly low-thrust trajectories to and from their targets. It is assumed that cargo does not need to be transported from Mars to the vicinity of the Earth. Robotic off-loading of cargo is assumed, thus human crews are not required to be at the target vehicle at the time of arrival of Freighters.

3.2.9 Spaceports

The Mars Spaceport is assumed not to have a permanent human crew. The Earth Spaceport may be crewed only if required to support activities in Earth-Moon space such as the Lunar Base. Temporary crews on their way to or from Mars will carry out maintenance at the Mars Spaceport.

3.3 Key Requirements

The key requirements imposed on the various Mars transportation infrastructure elements are delineated below. More detail requirements can be found in the Phase I Final Report.

3.3.1 Interplanetary Cyclic Orbits

The minimum planetary flyby altitude for the conceptual design studies shall be 200 km. The maximum transfer time between Earth and Mars shall be less than 6 months. It is desirable to minimize transfer times (5-6 months).

3.3.2 Spaceports and Astrotels

3.3.2.1 Crew

The Spaceports and Astrotels shall generally not be tended between crew rotation phasing. During tended phases it shall accommodate a crew of 10 for periods of time up to 10 days for Spaceports and 180 days for Astrotels. Crew support shall include life support; dormitory, kitchen, health, and recreation facilities; interplanetary and local communications; and computational capability.

3.3.2.2 Radiation Protection

Radiation sensors and radiation shielding to an equivalent level of 30 cm of water shall be required to protect the crew against natural radiation including galactic cosmic rays (GCR) and solar proton events (SPE). Radiation shielding can be incrementally increased over time in order to improve the shielded environment.

3.3.2.3 Modularity and Interfaces

Spaceport, Astrotel and Freighter power, propulsion and cargo hold systems shall have maximum commonality. The Spaceport Astrotel shall be capable of autonomous docking with the Taxis, Freighters, and planetary Shuttles.

3.3.2.4 Power System

The power system shall support all crew power requirements of 30 kW, continuous, plus propulsion power requirements. Renewable energy storage capability shall provide emergency minimum crew power requirements for a period of 8 hours.

3.3.2.5 Autonomy

The Spaceport and Astrotel shall be highly autonomous to reduce crew workload during crewed mission phases, maintain needed subsystems during uncrewed mission phases and enable autonomous resupply that may occur during untended phases.

3.3.3 Taxis

Taxis shall support crew needs during the very short transit (<10 days) between Spaceports and Astrotels. In addition, Taxis must perform delta-V maneuvers, perform aeroassist maneuvers within planetary atmospheres, navigate autonomously during all maneuvers, provide protection to the crew from solar flare, or solar proton events (SPE), and provide electrical power to its subsystems. There shall be one basic vehicle type for Earth and Mars application.

3.3.3.1 Crew

The maximum crew size for the Taxis shall be 10 people for flight duration of less than 7 days. Life support, sleeping areas, food, and communications shall be provided. “Apollo-like” space accommodations shall be provided.

3.3.3.2 Aeroassist

The Taxis shall be capable of aeroassist orbit capture at Earth and Mars.

3.3.3.3 Radiation Protection

Radiation sensors and radiation shielding to an equivalent level of TBD [10] cm of water shall be required to protect the crew against natural radiation including major SPEs.

3.3.4 Cargo Freighters

Cargo freighters are uncrewed transporters of cargo. They use slow, low-thrust trajectories and therefore require long trip times. There shall be two vehicle types. One vehicle shall be dedicated to transport cargo from LEO to the Earth Spaceport and to the Astrotel while the other vehicle shall be designed to transport cargo from LEO to the Mars Spaceport.

3.3.4.1 Reusability

Both Cargo Freighters shall be reusable after repair, refurbishment and upgrade upon return to LEO at Earth.

3.3.4.2 System Interfaces

Freighters must be capable of autonomous docking and cargo transfer with Astrotels or Spaceports.

3.3.5 Lunar Water Tanker

The Lunar Water Tanker shall be an uncrewed, reusable vehicle that can be fueled either on the Moon or at the Earth Spaceport. Lunar Water Tanker shall transport water from the lunar surface to the Earth Spaceport and returns empty to the lunar surface. The vehicle shall be highly autonomous in order to carry out water loading, propulsion maneuvers and autonomous water transfer at the Earth Spaceport.

3.3.6 Mars Shuttle

The Mars Shuttle shall support a crew of 10 during the very short transit (<1 day) between the Mars Base and the Mars Spaceport. In addition, the Mars Shuttle must perform delta-V maneuvers, perform re-entry and landing maneuvers within the Martian atmosphere, navigate autonomously during all maneuvers, provide electrical power to its subsystems and carry RRU cargo from the Mars Spaceport to the Mars Base.

3.3.6.1 Propulsion

The propulsion subsystem shall be capable of carrying out all navigation trajectory corrections and major orbit shaping delta-Vs up to 5.1 km/s, which is required for the transfer from the Mars Base to the Mars Spaceport. The Mars Shuttle shall be required to maintain space storable propellants at cryogenic temperatures at the Mars Base utilizing external power supplies.

3.3.6.2 Major System Interfaces

The Mars Shuttle must be capable of autonomous docking with the Mars Spaceport. In addition, the Mars Shuttle shall be interfaced directly into the Mars Base propellant manufacturing plant and act as an element of its storage system.

3.3.7 Propellant and Resources Plants

3.3.7.1 Mars Surface Plant

The Mars surface propellant plant shall produce all propellants required for the Mars Shuttle to reach the Mars Spaceport.

3.3.7.2 Phobos Plant

The Phobos Plant shall produce sufficient LOX for operation of the Taxi and the return of the Mars Shuttle to the Mars surface.

3.3.7.3 Mars Spaceport

The Mars Spaceport shall have facilities to receive and store LOX and LH propellants.

3.3.7.4 Lunar Water Mine

The Lunar Water Mine shall excavate and extract sufficient water from Lunar soils to operate the Lunar Water Tanker, Taxis and the LEO Shuttle vehicles and to produce LH for transport to the Mars Spaceport.

3.3.8 Propellant Augmentation Tanks (PATs)

The Propellant Augmentation Tanks (PATs) enable staging of the Taxi vehicles in order to accommodate natural celestial mechanic variations in Mars-to-Astrotel orbit injection velocities over a 15-year cycle of operation. PATs shall be designed to be fully integrated with the basic Taxi vehicle design. PATs shall be designed to be jettisoned after the propellants within them are used. It is possible that these tanks could be used for storage of propellants at the Spaceport.

4 Architecture Design Descriptions

4.1 Introduction

Three Mars transportation architectures have been studied during Phase II. However, most of the effort has been in the development of the Aldrin Low-thrust Cyclers architecture. Two architectures, Aldrin and Stopover, were developed to the point that cost trade studies could be carried out using the MAMA tool described in Section 9. This section briefly describes the orbits that are key to the architecture designs; the common architecture elements including Mars Base, power and propulsion systems, crew module, and Mars Shuttle; and the Aldrin low-thrust and the Stopover cycler architectures

4.2 Mars Base

We continued to use the key elements of the circa 2035 permanent, sustained Mars Base concept [National Commission on Space, *Pioneering the Space Frontier*, Bantam, 1986] as the logistical driver for our study of an innovative transportation system.

The level of capability envisioned at the Mars Base supports significant surface activities in the areas of science exploration, resource surveys, life-cycle maintenance, propellant production, and materials processing and fabrication. These activities will take place at one or two fixed-site facilities on Mars and on distant traverses from the base. Such operations will require a high degree of mobility, appropriate levels of automation with efficient man-machine interfaces, and they require crews that combine the need for individual specialization with job sharing abilities. A crew complement of 20 on the Martian surface will carry out these activities; the resident population at any time could fluctuate substantially from the average depending on the phase of the crew rotation cycle dictated by the interplanetary transportation orbit options. The Mars surface is assumed to be continually inhabited thereby requiring staggered crew rotations and, thus, overlap between “experienced” and “fresh” personnel. Figure 4-1 illustrates one base concept and illustrates an example equipment list (where mt is metric tonne or 1000 kg).

The Mars Base is nearly self-sufficient and it maximizes its use of in situ resources with minimal replenishment from Earth. Robotics and automation activities are focused on in situ resource, refurbishment, repair and upgrade (RRU), power generation, and life support monitoring functions. The environmental control and life support systems are regenerative to a large degree but not entirely closed. Life support gases and water will be extracted from the soil and atmosphere as needed. Agriculture, in greenhouses, and aquaculture will supply plants and perhaps animals for food. Propellants for mobility systems on the surface and in the atmosphere and for rocket transportation between the Mars Base and the Mars Spaceport are created in situ. The entire Mars Base requires delivery of about 280 metric tonnes (mt) of hardware to the surface in the build up phase and about 50 mt of RRU hardware every 15 years. To support the Mars Base a means of transporting crews and RRU equipment between the planets is needed. It is the crew and logistical support to this base that is the driver for this Mars transportation system architecture.

Mars Base Systems	# of Units	Unit Mass, mt	Total Mass, mt
Life Critical Systems			
Habitat	4	38.5	154.0
Washdown facility	2	0.9	1.8
			155.8
Mission Support Systems			
120 kW Power Source (solar array @100W/kg)	2	1.2	2.4
Power Management, Distribution and Maintenance	2	0.3	0.6
Energy Storage (NRFC packages)	2	1.1	2.2
Suitup/Maintenance Facility	2	1.8	3.6
Pressurized Transporter	3	9.1	27.3
Open Rovers	3	1.0	3.0
Inflatable Shelter w/Airlock	10	0.5	5.0
Communication Satellites	3	0.8	2.4
Crane	2	5.0	10.0
Trailer	2	2.0	4.0
			60.5
Science and Exploration Systems			
Base Laboratory	2	13.6	27.2
Mobile Laboratory	3	9.1	27.3
200 m Drill	1	2.3	2.3
10 m Drill	3	0.1	0.3
UAV	3	0.3	0.9
Robotic Rovers	10	0.2	2.0
Weather Station	5	0.2	1.0
Survey Orbiters	2	0.8	1.6
			62.6
Total			**278.9**

Figure 4-1 Mars Base Concept and Equipment Inventory

In addition to the Mars surface facilities, the overall Mars base infrastructure includes (1) *in situ* resource production plants on the Martian surface and at Phobos, (2) a Spaceport near Phobos serving as a work station and a transportation hub, (3) a LOX Tanker to transport LOX from Phobos surface to the Mars Spaceport's propellant depot, and (4) a Mars Shuttle transport operating between the Martian surface and the Mars Spaceport.

4.3 Orbits

4.3.1 Cycler Orbits

Cycler orbits are resonant or near resonant trajectories between celestial bodies. Cycler orbits can be designed to enable sustained human interplanetary transportation through regular encounters with Earth and the target planet or between Earth and the Moon. Several interplanetary cyclic orbit concepts have been developed over the last two decades to support sustained Mars operations, however the reference Mars transportation architecture assumes the use of low-thrust modified Aldrin Cycler Orbits. Aldrin Cycler orbits have a period that is approximately equal to the Earth-Mars synodic period (26 months) and, when the line of apsides is rotated by gravity assist methods (average of about 51.4° each orbit), will enable Earth-to-Mars and Mars-to-Earth transfers every 26 months. Aldrin Cycler orbits come in two types, an Up Escalator and a Down Escalator orbit. The Up Escalator has the fast transfer occurring on the Earth to Mars leg while the Down Escalator is just the reverse. Figure 4-2 illustrates (a) both Up Escalator and a Down

Escalator orbit geometry, (b) example 15-year propagation of the outbound or Up Escalator Aldrin Cycler, and (c) example low-thrust orbit rotation augmentation maneuver. When two Astrotels are used, an Aldrin Cycler provides relatively short transit times (~5 months) and regular transit opportunities. However, the planetary encounters occur at high relative velocities and typically, impose harsher requirements on the Taxi craft than other cyclers. Also, as illustrated in Figure 4-2 (b), the Aldrin Cycler requires a substantial orbit correction on 3 out of 7 orbits to maintain the proper orbit orientation. Shown are the impulsive or ballistic delta-Vs that would be required to maintain the proper orbit alignment. The reference architecture replaces this ballistic delta-V by a low-thrust arc using a solar electric powered, ion propulsion system (IPS). A low-thrust version of this orbit correction is illustrated in Figure 4-2 (c). These corrections are required because of limitations of flyby altitude during the gravity assist. In the case shown a 200-km flyby altitude constraint has been imposed. Mathematically correct, but impractical subsurface flybys eliminate the need for corrections.

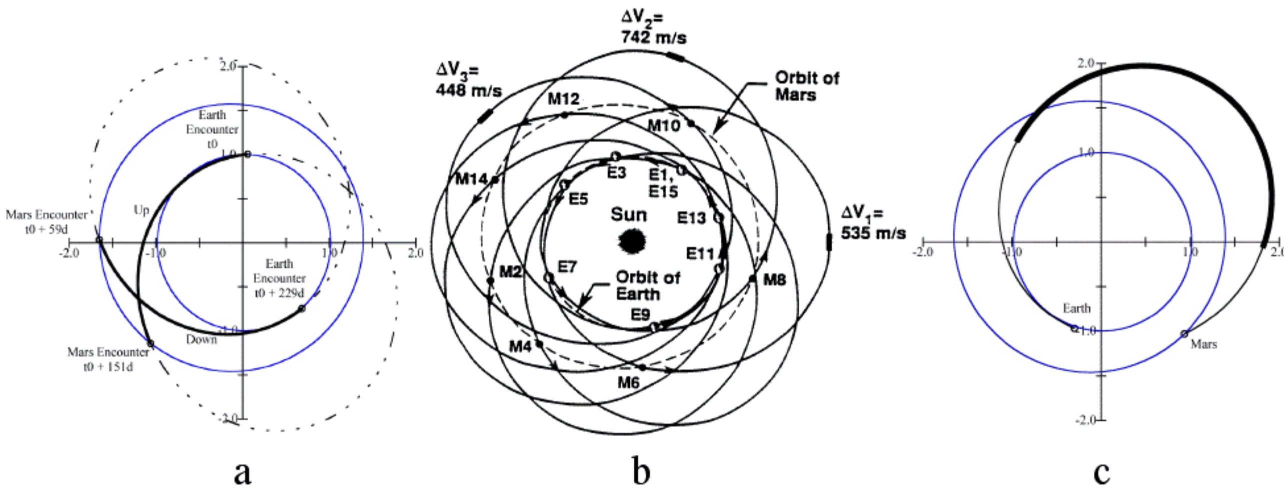

Figure 4-2 Aldrin Cycler Orbits: (a) Up and Down Escalators, (b) a 15-year Ballistic Up Escalator Sequence, (c) Example low-thrust orbit rotation augmentation maneuver

4.3.2 Reusable Cargo Vehicle Orbits

Figure 4-3 illustrates the outbound low-thrust orbit for the Astrotel Cargo Freighter. The black trajectory is the path of the Cargo vehicle. For one example opportunity studied, the total flight duration was 1311 days. This flight time included 966 days from LEO to the Astrotel and only 335 days back to LEO, including all planetocentric spirals. Total propulsion on-time was 608 days and there was an Earth turnaround time of 304 days before departure to the Astrotel again. Two Astrotel Cargo vehicles operate at a time re-supplying each Astrotel every other opportunity.

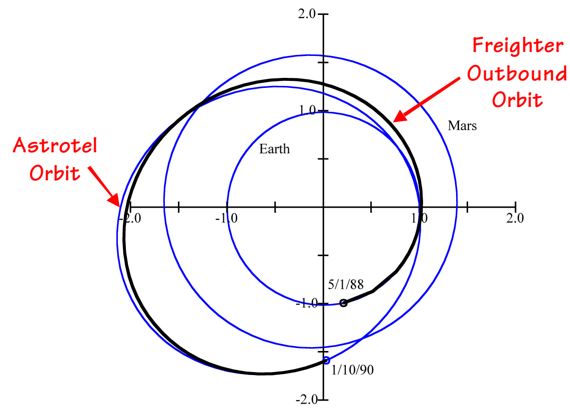

Figure 4-3 Astrotel Cargo Outbound Orbit

For one example studied, the flight time from LEO to Mars was 753 days (including a 263 day Earth escape spiral and 90 Mars capture spiral). A stay time of 178 days at the Mars Spaceport days was assumed followed by flight time of 459 days back to LEO (including a 22 day Mars escape spiral and 37 day Earth capture spiral). Total round-trip flight time from LEO is 1390 days. Total propulsion on time is 733 days and there is an Earth turnaround time of 140 days before departure to the Mars Spaceport again. Two Mars Cargo vehicles operate in the architecture, one leaving Earth for Mars at every opportunity.

4.3.3 Hyperbolic Rendezvous

On those Astrotel flybys which are followed by a short flighttime (~5 month) to Earth or Mars, hyperbolic rendezvous is carried out by Taxi vehicles to transfer crew from Spaceports, in high planet orbit, to Astrotels. Flighttime from Spaceport to Astrotel is constrained to just 7 days in order to minimize risk of crew injury due to space radiation since the Taxi vehicle only has minimal shielding (equivalent to 10 cm water). A three impulse departure sequence is employed to minimize delta-V. This sequence includes a moderate delta-V to lower periapsis, the major escape burn at the planet periapsis, and finally the small rendezvous burn at the Astrotel on the hyperbolic trajectory. The features of departure from a high planet orbit (Lunar orbit radius at Earth and Phobos orbit radius at Mars) and a 3-impulse transfer, offers several advantages over alternatives. These advantages include the ability to incorporate launch periods into the departure planning for the cost of modest amounts of delta-V and the relative ease of re-positioning the Spaceport to optimum positions well in advance of Taxi departure. Further discussion of hyperbolic rendezvous can be found in Section 5.4.

4.3.4 Spaceport and Propellant Depot Locations

In Phase I, the Earth Spaceport location was assumed to be at the Earth-Moon L-1 point. In Phase II we found that L-1 was not an ideal location for an Earth Spaceport due to the need for proper phasing of the Spaceport location and the Astrotel flyby. In general, for the L-1 Earth Spaceport location, proper phasing required long flighttime or large delta-Vs, both undesirable. On the other hand, with the Earth Spaceport at Lunar orbit radius (LOR) and allowed to re-position itself in longitude to provide optimum alignment, only a few meters per second of IPS delta-V are required assuming months are available for re-positioning. For example, the position of the Earth Spaceport can be altered by 60° in longitude for only about 20 m/s over a 3 month period. Placing the Earth Spaceport at LOR makes both Earth and Mars Spaceport locations and departure problems and analysis similar.

4.4 Aldrin Low-thrust Cycler Architecture

This section describes the baseline Aldrin Low-thrust Cycler architecture.

4.4.1 Overview

The following chart summarizes the current overall architecture for the Aldrin Low-thrust Cyclers Mars transportation system. The overall reference architecture has changed little from that developed during Phase I, though research and design work has occurred on orbits, aero-assist, ISRU, Taxi vehicle, and the Mars Shuttle. A major change from Phase I is the locating of the Earth Spaceport at Lunar orbit radius (LOR) as opposed to L-1 in order to facilitate Astrotel and Taxi Phasing, hyperbolic rendezvous and to help provide a reasonable launch period.

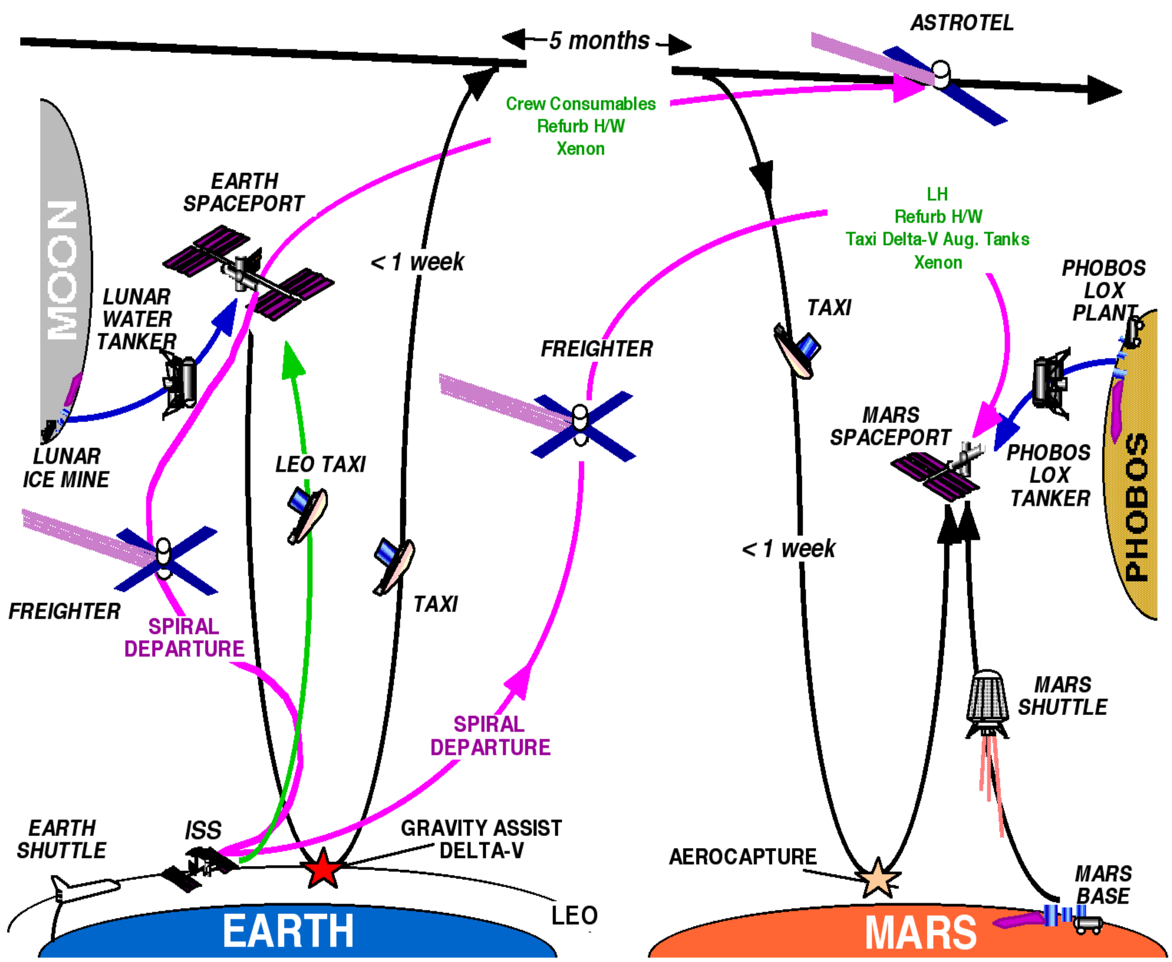

Figure 4-4 Aldrin Low-thrust Cyclers Mars Transportation Architecture Schematic

The overall transportation systems architecture currently consists of a Mars Base; a Lunar Base; two Astrotels, one each in the Aldrin Up and Down Escalator orbits (Down Escalator Astrotel not shown); two Spaceports one at Earth at Lunar orbit radius (LOR) and the other at Mars at Phobos orbit radius (POR); a Space Station in LEO (shown as the ISS) that is supplied by an Advanced Space Shuttle and a low-cost heavy lift launcher; a LEO Taxi to transport crew to and from the Space Station to the Earth Spaceport; propellant manufacturing facilities on the Moon (Ice Mine), at the Earth Spaceport (water electrolysis, LOX/LH liquefaction and storage), Phobos (LOX Plant), Mars Spaceport (LOX/LH storage) and Martian surface (Sand Dune Water Mine, LOX/LH production and storage); two Taxis transporting crew to and from Astrotels and Spaceports; a Mars Shuttle to carry crew and refurbishment cargo to the surface of Mars; a Lunar Water Tanker and Phobos LOX Tanker to carry resources to the Spaceports; and at least four SEP-powered Cargo Freighters (two shown) to transport consumables, refurbishment hardware and Xenon to the Astrotels and refurbishment hardware, Taxi propellant augmentation tanks and LH to the Mars Spaceport.

The following table summaries the key features of the various vehicles in terms of vehicle type, propulsion type, current design heritage, purpose, location or nodes serviced, delta-V capability, reusability and dry mass.

Table 4-1 Aldrin Low-thrust Cycler Mars Transportation Systems Summary

Systems	System Type	Propulsion	Primary Purpose	Location
Vehicles and Surface Systems				
Astrotel	Space	IPS	Crew Transport to/from Earth and Mars	Mars/Earth
Escape Pod	Aero-Space	SRM	Crew Astrotel Emergency Escape to Planet	Astrotel to Planets
Earth Spaceport	Space	IPS	Crew & Cargo Transfer/Storage & Propellant Production & Storage	Lunar Orbit Radius
Mars Spaceport	Space	IPS	Crew & Cargo Transfer/Storage & Propellant Storage	Phobos Orbit Radius
Taxi	Aero-Space	LOX/LH	Crew Transport to/from Spaceports and Astrotels	Astrotel/Spaceports
Mars Cargo Freighter	Space	IPS	Fuel, refurb cargo to Mars Spaceport	LEO/Mars Spaceport
Astrotel Cargo Freighter	Space	IPS	Consummables, Fuel, Refurb Cargo to Astrotel	LEO/Earth Spaceport/Astrotel
Space Station	Space	Shuttle	Crew, Cargo, Propellant Transfer	LEO
Space Shuttle	Aero-Space	SRM/LOX/LH	Crew to LEO Space Station	Earth Surface/LEO
HLLV or Magnum	Surface L/V	SRM/LOX/LH	Consum, LH2 , Refurb cargo to LEO	Earth Surface/LEO
LEO Shuttle	Aero-Space	LOX/LH	Crew, Consum, LH2 , Refurb cargo to LEO	LEO/Earth Spaceport
Lunar Water Tanker	Space	LOX/LH	Lunar water to Earth Spaceport	Lunar Surface/Earth Spaceport
Mars Shuttle	Aero-Space	LOX/LH	Crew and refurb hardware to/from Mars Surface	Mars Surface/Mars Spaceport
Phobos LOX Tanker	Space	LOX/LH	Propellant transport to Mars Spaceport	Phobos/Mars Spaceport
Mars Base	Surface	-	Crew accomodation and science	Mars Surface <30 Lat
Lunar Base	Surface	-	Science and support water mine	Near Lunar Pole
in situ **Resources Systems**				
Lunar Water Mine	Surface		Mine Lunar ice and extract water	Lunar Pole
Earth Spaceport Water Elect/Cryo/Strg	Space		Electrolyze Lunar water, liquefy and store LOX/LH	Earth Spaceport
Phobos LOX Plant	Surface		Mine regolith, extract, liquefy and store LOX	Phobos surface
Mars Surface Water Plant	Surface		Mine regolith, extract, liquefy and store LOX/LH	Near Mars Base
Mars Spaceport LOX/LH Storage	Space		Store Phobos LOX and Earth LH2	Mars Spaceport

4.4.2 Astrotels

This Mars transportation system architecture concept uses small, highly autonomous space ships, we dub *Astrotels*, for transporting humans to and from Earth and Mars on cyclic or near-resonant orbits between these planets. Human flight time each way is reasonably short, between 5 and 6 months. Key elements of these ships are that they are highly autonomous and transport only human and other high value cargo, use highly efficient solar electric IPS, and do not require artificial gravity. Current Astrotel design requires 70 mt including IPS, radiation shielding and a planetary escape pod. Reducing its mass significantly reduces the total propulsive energy budget required for course corrections to the 2,780-kg propellant required for all major corrections over 15 years. The IPS consists of eight 50-cm diameter ion engines each requiring 17.2 kW of power and producing thrust at an I_{sp} of 5000s. Propulsion and crew power is supplied via a 160 kW concentrator solar array with multi-component, mechanically stacked solar cells. The 70 mt mass includes a habitability module for a crew of ten. The size and volume of this system provides a crew volume of about 6 times that available to today's Space Shuttle crew. The astronaut living space is a three-story structure patterned after the ***TransHab*** module that has been under study by NASA. Figure 1-1 is a schematic of one concept for an Astrotel that is approaching Mars. The two smaller modules between the TransHab and the solar array are cargo bays. The Astrotel Cargo Freighter autonomously delivers all cargo to the Astrotel contained within a standard cargo bay. These are pressurized modules to facilitate crew unloading of consumables and RRU hardware. Once emptied the cargo bay could be discarded or used to provide added crew volume, perhaps for zero-g squash courts and other recreational uses. The entire Astrotel requires 70 mt of hardware to be delivered to the cyclic orbit in the build up phase and about 60 mt of crew consumables, RRU hardware, and IPS propellant every 15 years. A detailed mass breakdown of the Astrotel is provided in Section 8.

4.4.3 Spaceports

Two Spaceports are envisioned, one for Mars and the other located at Earth. Spaceports are collection points for the arrival and distribution of humans, cargo and propellants destined for transport to planet or natural satellite surfaces or to cycling Astrotels. In addition because of the nearly 100% availability of solar energy, Spaceports are ideal locations for propellant processing, liquefaction, and cryogenic storage.

Spaceport design and operation are based on Astrotel design heritage. Crew stay times are limited in order to minimize effects of zero-g. Crew maintenance is minimized by maximum application of autonomy in order to shorten stay times. Station-keeping, orbit corrections, orbit-phasing delta-Vs are performed by the same solar powered IPS envisioned for the Astrotels. Spaceport design heritage is based on the design of Astrotels except without the large power and propulsion subsystems.

4.4.4 Taxis

Taxis provide transportation between Spaceports and Astrotels. In order to minimize propulsive energy use, Taxis use advanced aeroassist technologies for planetary orbit capture. Aerocapture takes maximum advantage of planetary atmospheric drag to slow the vehicle on its approach from planetary space. The initial sizing of the Taxi vehicle was carried out during Phase I. Key assumptions are:

- Minimal radiation protection (equivalent to ~10 cm polyethylene surrounding crew module) for the crew is provided since transfer times to/from the Astrotels could be 7-10 days
- No cargo is transported to the Astrotel by the Taxi except crew
- 10-15% of the entry mass is aeroshell
- LOX/LH propulsion system at I_{sp} of 460 s and thrust of 60,000 lbs./engine
- Fuel cell energy storage, no solar array power source

- Propellant tank augmentation (expendable drop tanks and in some cases additional engines) are required at Mars

The nominal Taxi system aeroshell design is an elliptical raked cone (see Phase I Final Report). Taxis utilize LOX/LH propulsion to escape planets and place them and their crew onto hyperbolic rendezvous trajectories with the interplanetary orbiting Astrotels. Figure 4-5 depicts a Taxi departing the Earth Spaceport with the Moon in the background and a Taxi during aerocapture at Mars arrival. This figure illustrates the crew module, propellant tanks, rocket engines (in their deployed position), and the aeroshell. Propellant capacity of the basic Taxi vehicle is 20.6 mt. Rendezvous time to Astrotels would be measured in days in order to reduce the duration of crew time in the expected cramped quarters. Crew volume is comparable to what the Apollo astronauts had on their flights to the Moon and back. In the Mars arrival figure the Taxi is viewed from 50 km above Mars during aero-cruise. Note the rocket engines are in their stowed position in order to minimize altering the air flow. During this time the tanks are almost empty, containing only the propellant necessary to rendezvous with the Mars Spaceport, in a Phobos radius orbit, after aerocapture. The crew module is shown in *see-through* mode so one can observe the crew g-seats, which rotate in order to accommodate the different g-load direction during aerocapture as opposed to g-load direction during propulsive thrust maneuvers.

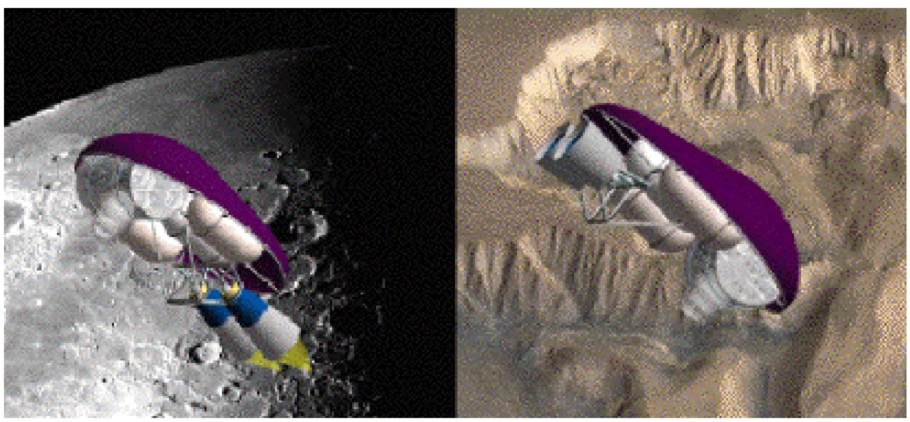

Figure 4-5 Taxi Leaving the Earth (left) and Aerocapture at Mars (right)

The usable volume of the crew module is roughly comparable to that of the Apollo Command Module, or about 2-3 m^3 per crewmember. The crew module has 10 crew pods, which includes a rotating g-seat or hammock, communications and computer panels, and hygiene systems. At least two crewmembers have additional Taxi control and monitoring equipment adapted to their g-seat. Crew hammocks rotate in order to accommodate the varying acceleration vector during aerocapture maneuvers and very different acceleration vectors during staged propulsive rocket burns. Maximum g-load is nominally 5. The crew module is cylindrical in shape and surrounded by about 3-cm thick polyethylene radiation shield to protect the crew during major solar particle events. This level of shielding is less than suggested thickness; on the other hand there is no current recommendation. This is an area for further study.

At Mars the departure delta-Vs are significantly larger than at Earth due to the higher V-infinity of the Astrotel Down Escalator orbit as it passes Mars (Phase I Final Report). For these larger delta-Vs the Taxi escape maneuver must occur in stages at Mars. Two stages are required for 3 of 7 opportunities, where the delta-V is less than 6.7 km/s, and three stages are required for the other 4 opportunities, where the delta–V can reach up to 10.5 km/s. Staging is accomplished by use of expendable propellant augmentation tanks (PATs). Propellant requirements for the Taxis have been estimated during Phase I. Taxi propellant requirements drive the propellant production system requirements at Mars and Earth.

Two Taxis operate in the Mars transportation architecture. In a typical sequence a Taxi departs Earth and 7 days later rendezvous with the Up Astrotel for a 5-month trip to Mars. Less than 7 days before Mars arrival, the Taxi departs the Astrotel and redirects its trajectory to a very close periapsis to enable Mars aerocapture. After aerocapture, the Taxi, now in orbit, rendezvous with the Mars Spaceport, where it docks. This Taxi remains docked to the Mars Spaceport as the crew departs the Spaceport on the Mars Shuttle toward the Mars Base. About 2.3 years later, the next crew ascends to the Mars Spaceport via the same Mars Shuttle and eventually boards the same Taxi which then departs the Mars Spaceport to rendezvous with the Down Astrotel for its 5-month trip back to Earth. Total Taxi mission duration is an average of 2.8 years from Earth departure to return. Once back at the Earth Spaceport, major refurbishment and upgrades are planned including possible replacement of aeroshell components. The entire Taxi vehicle with consumables but without fuel is 15.5 mt. About 4 mt of RRU hardware is required every 15 years for each Taxi. Figure 4-6 illustrates the current three-engine (shown retracted), three-stage Taxi configured for a high-delta-V Mars escape maneuver.

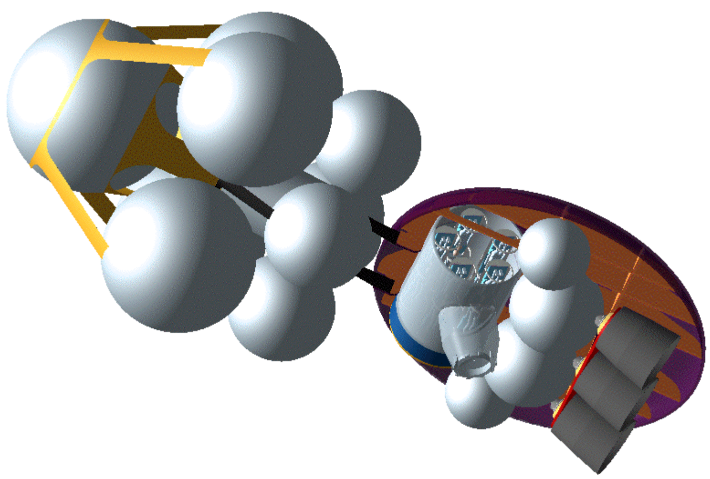

Figure 4-6 Three Stage Taxi Preparing to Depart Mars

4.4.5 Mars Shuttle

The Mars Shuttle transports a crew of 10 to and from the Martian surface base and the Mars Spaceport near Phobos. The Mars Shuttle supports crew needs during the very short transit (~3 hours) between the Mars Base and the Mars Spaceport. In addition, the Mars Shuttle carries out delta-V maneuvers, performs aero-entry and landing maneuvers within the Martian atmosphere, navigates autonomously during all maneuvers, provides electrical power to its subsystems and carries RRU cargo from the Mars Spaceport to the Mars Base. The Mars Shuttle is designed to travel only between the Mars surface and the Mars Spaceport at Phobos.

An early Mars Shuttle design version, shown in Figure 4-7, is a low lift/drag (L/D) ratio design with a deployable 20-m diameter aerobrake deployed before entry and landing. At take-off, the aerobrake is stowed to reduce atmospheric cross-section and minimize drag. The low lift/drag ratio design offers reduced mass, ease of fabrication, reduced cost and growth accommodation over higher L/D designs. Though the vehicle is sized for carrying up to 10 mt of cargo to the Mars Base, the Mars surface cargo requirement is only about 7.8 mt, thus indicating considerable reserve exists in the design.

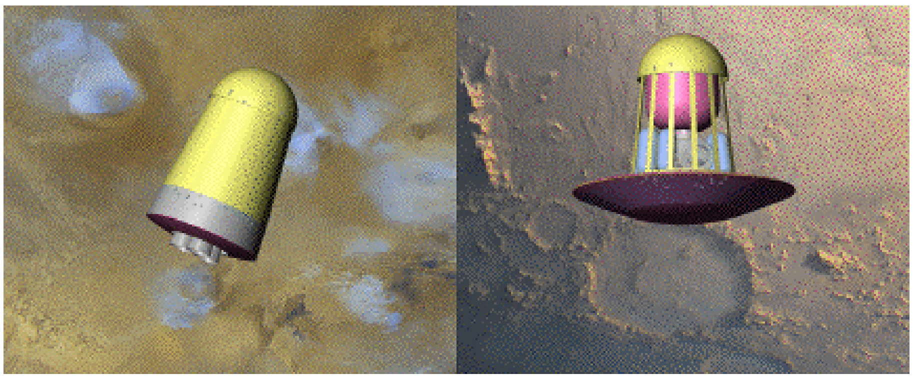

Figure 4-7 Mars Shuttle Launching from Surface (Left) and at Mars Entry (Right)

4.4.6 Lunar Water Tanker

The Lunar Water Tanker (LWT) is a reusable vehicle that can be fueled either on the Moon or at the Earth Spaceport. It transports water from the lunar surface to the Earth Spaceport where it refuels and then returns empty to the lunar surface. The LWT is not crewed thus minimizing additional infrastructure. The key power and LOX/LH propulsion elements are in common with other elements of the Mars transportation architecture in order to minimize development and recurring cost. The maximum payload of the LWT is expected to be about 19 mt of lunar water. Leaving the Moon for the fuel depot somewhere at lunar orbit radius, the required delta-V is 2200 m/s, which requires a propellant load of about 16 mt of LOX/LH. The departure delta-V propellant is assumed to be produced totally on the Moon. Once delivered to L-1, some of the water is processed into LOX/LH for the LWT to return to the Moon. Since the LWT is empty leaving lunar orbit radius, only about 5 mt of LOX/LH propellants (processed Lunar water) are required to return to the Moon. The net materials left at the fuel depot at lunar orbit radius is about 8 mt of LOX/LH for use as propellants and some excess oxygen. Total water produced at the Moon is 37 mt every 2 1/7 year cycle.

4.4.7 Integrated Propellant and Cargo Use Profiles

In this section we summarize the overall propellant and cargo masses that move through nodes of the transportation architecture over a 15-year period. Vehicle propellant, crew consumable and RRU mass requirements over 15 years have been summarized and collected in MAMA at their eventual transportation node in the following tables. All masses are kilograms.

Table 4-2 Cargo Mass Summary

Location------------->	At LEO	At Earth Spaceport	At Mars Spaceport	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Users						
Taxis (2)						
Propellant						
LOX		102,935	619,612			
LH		14,705	88,516			
Tot		117,641	708,128			
Water required at Spaceport		132,346				
Augmentation Tanks	88,047		88,047			
Taxi Refurb Mass	23,258	23,258				
LEO Shuttle (1)						
Refurb Mass	*23,258*					
Mars Shuttle (1)						
Propellant						
LOX			83,326	229,868		
LH			11,904	32,838		
Total			95,230	262,706		
Water required on Mars				295,544		
Refurb Mass	5,444		5,444			
Lunar Polar Base						
Polar Ice Mine Refurb Mass	1,091	1,091			1091	
Mars Base						
Base Refurb Mass	50,942		50,942	50,942		
Propellant Plant Refurb Mass	3,705		3,705	3,705		
Phobos LOX Plant						
Propellant Plant Refurb Mass	3,237		3,237			
Astrotels (2)						
Crew Consumables	100,324					100,324
Xenon Propellant	5,560					5,560
Refurb Mass	14,690					14,690
Mars Spaceport (1)						
Refurb Mass	*14,690*		*14,690*			
Phobos LOX Storage Refurb Mass	257		257			
Station-keeping Xenon propellant Mass						
Earth Spaceport (1)						
Refurb Mass		14,690				
LOX/LH Production Plant Refurb mass	80	80				
Station-keeping Xenon propellant Mass						
Lunar Water Tanker (1)						
Refurb Mass						
Astrotel Cargo Freighter						
Xenon Propellant	55,497					
Refurb Mass	2,291					
Mars Cargo Freighter						
Xenon Propellant	138,958					
Refurb Mass	5,299					

These cargo use profiles have been summarized as to where the cargo are needed throughout the transportation architecture for the 15 years of operation and displayed in the following table (all masses are in kg).

Table 4-3 Total Cargo Requirements

Location------------->	At LEO	At Earth Spaceport	At Mars Spaceport	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Total Cargo Requirements						
LOX		102,935	702,938	229,868		
LH		14,705	100,420	32,838		
Water		132,346		295,544		
Xenon	200,016					5,560
Refurb Mass	148,244	39,120	78,276	54,647	1,091	14,690
Augmentation Tanks	88,047		88,047			
Crew Consumables	100,324					100,324
Communications Satellites						
Total Mass Required	536,630	289,106	969,680	612,897	1,091	120,574

Finally, this cargo has been divided into the various delivery systems, which is displayed in the next chart. These propellant and cargo use profiles provide the derived requirements for the transport of cargo and propellant throughout the architecture and the requirements on the ISRU production rates.

Table 4-4 Cargo Delivery System Manifest

Location------------->	At LEO	At Earth Spaceport	At Mars Spaceport	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Delivery Systems						
SEP Freighter Delivered Cargo		39,120	266,742			120,574
Cargo Freighter Each Trip		5,589	38,106			17,225
Lunar Water Tanker Total		132,346			1,091	
Number of Trips		7				
Lunar Water Tanker Each Trip		18,907			156	
Phobos LOX Tanker Total			702,938			
Number of Trips			30			
Phobos LOX Tanker Each Trip			23,431			
Mars Shuttle				54,647		
Number of Trips				7		
Mars Shuttle Each Trip				7,807		

4.4.8 Cargo Freighters

Two types of cargo transporters are planned, an Astrotel Cargo Freighter and a Mars Cargo Freighter. These vehicles deliver cargo from LEO to Astrotels and Spaceports. See Section 4.4.7 for the integrated cargo requirements. Cargo Freighters use xenon ion propulsion systems to spiral out of Earth orbit, shape the interplanetary trajectory to rendezvous with Astrotels or spiral into Mars orbit to Phobos. The Astrotel Cargo Freighter delivers a standard pressurized cargo bay module to the Astrotel. The cargo bay approach facilitates crew unloading.

The Freighter solar arrays consist of multiple sets of identical Astrotel solar arrays (80 kW panels). The propulsion system shares high degree of technology heritage with the Astrotel IPS. The following charts describe the design parameters for these important vehicle systems, which will be a subject of more design during Phase II.

Table 4-5 Astrotel Cargo Freighter Sizing

Mass of Cargo each trip to Astrotel	**17,225**	kg
Initial mass of Cargo Freighter in LEO	28,632	kg
Propellant Mass	7,928	kg
Final Mass of Freighter	3,479	kg
P_o	286.3	kW
Power/Propulsion Mass (M_{ps})	2291	kg

Table 4-6 Mars Cargo Freighter Sizing

Mass of Cargo each trip to Mars Spaceport	**38,106**	kg
Initial mass of Cargo Freighter in LEO	66,237	kg
Propellant Mass	19,851	kg
Final Mass of Freighter	8,280	kg
P_o	662.4	kW
Power/Propulsion Mass (M_{ps})	5,299	kg

4.4.9 In Situ Resource Utilization Systems

The following sections summarize the *in situ* resource systems for the Moon, the Earth Spaceport, Mars surface, Phobos and the Mars Spaceport. Displayed below are the solar power option. Duty cycles for all surface systems are assumed to be near 100% for the nuclear options.

4.4.9.1 Lunar Ice Mine

The automated Lunar Ice Mine is assumed to operate only during periods of sufficient solar illumination on its surface solar array, thus 34% of the time. During the night batteries or non-regenerative fuel cells provide keep-alive power. Ice is mined and melted into water and then transported to the Earth Spaceport for production of LOX/LH propellants.

Table 4-7 Lunar Ice Mine Mass and Power

Resource System Element		Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Water Required			37,040	kg					
Mining and Excavation (0.16-g)		Mine regolith (kg/hr)	1235	kg/hr	34%	0.20	247	0.0133	16.46
Soil Hauler (0.16-g)		Haul soil to reactor and slag away							
Reactor/Condensor		Heats soil from -200oC to 50°C (kg H2O/hr)	12.35	kg/hr	34%	100.00	1,235	17	209.89
Water Storage		Stores extracted water (.33 x annual water production)	12.35	kg/hr	100%	0.02	247		
Distribution		Distributes liquids and gases to where they are processed	12.35			16.00	198		
Electrolyzer		Produces propellant for launching water to L1 (kg/hr)	5.37	kg/hr	34%	20.00	107	6.5	34.92
LH liquefaction		Provides LH for transfer vehicle (kg H2/hr)	0.67	kg/hr	34%	16.50	11	20	13.43
Lox liquefaction		Provides Lox for transfer vehicle (kg O2/hr)	4.70	kg/hr	34%	6.50	31	0.5	2.35
Lox, LH storage		Assumes immediate transfer to L1						Total Power-->	277.06
Solar Array - XX kW		Produces electrical power during Lunar day (kW)			34%	4.00	1,108		
Power Mgmt & Dist		Power voltage/freq control and distribution					111		
						Total mass---->	3,294		

4.4.9.2 *Earth Spaceport LOX/LH Production and Depot*

The Earth Spaceport is an ideal location for the electrolysis of lunar water into oxygen and hydrogen, liquefaction into LOX/LH and finally storage of these propellants. Nearly continuous power is available using high efficiency solar arrays. The Earth Spaceport ISRU facility is expected to operate at a 90% duty cycle.

Table 4-8 Earth Spaceport LOX/LH Production and Depot Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Water Delivered to L1		18,907	kg					
Water Storage	Stores Lunar water after delivery to L1 (3 mo. supply)	4,727	kg	100%	0.01	47		
Electrolysis Reactor	Converts water to oxygen and hydrogen (kg/hr)	1.97	kg/hr	91%	20.00	36	6.5	12.8
Distribution	Distributes liquids and gases to where they are processed	50	kg	91%	1.00	46		
LH liquefaction	Liquefies H2	0.22	kg/hr	91%	16.50	3	20	4.4
LoX liquefaction	Liquefies Ox	1.75	kg/hr	91%	6.50	10	0.5	0.9
LH Storage	H2 gas storage at X°C (kg) - 3 mo. supply, rest in Taxi	480	kg	100%	0.15	72	0.0016	0.8
LOx Storage	Cryogenic gas storage at -183oC - 3 mo. supply, rest in Taxi	3837	kg	100%	0.07	269	0.001	3.8
							Total Power-->	22.7
Solar Array - XX kW	Produces electrical power 100% of time (kW)			100%	4.00	91		
Power Mgmt & Dist	Power voltage/freq control and distribution					9		
					Total Mass--->	583		

4.4.9.3 *Mars Dune Water Mine*

Water is recovered from the regolith of Mars. Dune fields could be an excellent location for an excavation site. Once water is recovered from the soil it is electrolyzed into oxygen and hydrogen, liquefied into LOX/LH and finally stored within the Mars Shuttle tanks. The operation on Mars is fully automatic and operates at a 34% duty cycle along with the normal day/night periods.

Table 4-9 Mars Dune Water Mine Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Water Required		19,703	kg/yr					
Dune Collection (0.38-g)	Excavates likely water bearing soil (kg/hr)	657	kg/hr	34%	0.27	177	0.00991	6.5
Soil Hauler (0.38-g)	Transports soil to reactor and slag away							
Reactor/Condensor	Heats soil to 500°C to remove water (kg/hr)	6.57	kg/hr	34%	205	1,346	35	229.9
Water Storage	Stores water before electrolysis (kg)	1147	kg	100%	0.04	46		
Electrolysis Reactor	Converts water to oxygen and hydrogen (kg/hr)	6.57	kg/hr	34%	20.00	131	6.5	42.7
Distribution	Distributes liquids and gases to where they are processed	6.57	kg/hr		3.70	24		
LH Liquifaction	Liquifies hydrogen (kg/hr)	0.82	kg/hr	34%	16.50	14	20	16.4
LOX Liquifaction	Liquifies oxygen (kg/hr)	5.75	kg/hr	34%	6.65	38	0.5	2.9
LOX Storage	Crygenic gas storage at -183°C (25% of annual production, res	5517	kg	100%	0.07	386	0.5	1.15
LH Storage	Crygenic gas storage at Y°C (25% of annual production, rest o	788	kg	100%	0.15	118	20.0	6.57
							Total Power-->	306.1
Solar Array	Produces electrical power 33% of time			34%	9.24	2,829		
Power Mgmt & Dist	Power voltage/freq control and distribution					141		
					Total Mass--->	5,251		

4.4.9.4 *Phobos LOX Plant*

Phobos regolith is excavated and processed to recover oxygen. The oxygen is then liquefied and transported to the Mars Spaceport where it can be stored in Taxi propellant tanks or propellant augmentation tanks (PATs). The plant on Phobos is fully automatic and operates at a 34% duty cycle consistent with the day/night periods.

Table 4-10 Phobos LOX Plant Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Oxygen Required kg/yr		46,863	kg/yr					
Excavation (zero-g)	Excavates Phobos regolith (kg/yr)	137,831	kg/yr					
Excavation (zero-g)	Excavates Phobos regolith (kg/hr)	45.9	kg/hr	34%	1.00	46	0.06	2.8
Soil Hauler (zero-g)	Transports soil to reactor and slag away							
Carbothermal Reactor	Reactions at 1600 °C to generate oxygen (kgO2/hr)	15.73	kg/hr	34%	96.00	1,510	16	251.7
Distribution	Distributes oxygen to where it is liquified	50	kg		1.00	50		
Liquifaction	Liquifies oxygen (kg/hr)	15.73	kg/hr	34%	6.50	102	0.5	7.9
LOX Storage	Assumes immediate transfer to Marsport							
						Total Power-->		262.4
Solar Array - XX kW	Produces electrical power 33% of time (kW)			34%	9.24	2,425		
Power Mgmt & Dist	Power voltage/freq control and distribution					121		
					Total Mass--->	4,255		

4.4.9.5 Mars Spaceport Propellant Depot

Phobos LOX and Earth LH are stored at the Mars Spaceport, in Taxi PATs and Mars Shuttle tanks. The operation at the Mars Spaceport is nearly continuous, except for occasional eclipses of the Sun by Mars.

Table 4-11 Mars Spaceport Propellant Depot Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Phobos LOX Storage*	Crygenic gas storage at -183oC - (25% ann. prod., rest in Taxi tanks)	11,716	kg	100%	0.07	820	0.00018	2.1
Earth LH Storage*	Crygenic Earth LH storage at Y°C - (25% ann. Req't, rest in Taxi tanks)	1,674	kg	100%	0.15	251	0.0084	14.1
							Total Power-->	16.2
Solar Array - XX kW at ~1.5 AU	Produces electrical power 100% of time				9.24	149		
Power Mgmt & Dist	Power voltage/freq control and distribution					7		
					Total Mass--->	1,228		

4.5 Stopover Cycler Architecture

This section describes the Stopover Cycler architecture.

4.5.1 Overview

The Stopover Cycler architecture uses the Stopover Cycler orbits for transport between Earth and Mars. These orbits are direct transfers with stops at both planets. Stopover Cycler orbits are further described in Sections 4.3 and 5.3.2. Flight duration is limited to 6 months or 180 days, although for the maximum propellant loading the trips could be considerable shorter for some opportunities. In this architecture the Interplanetary Taxi and planetary Spaceports are eliminated because the Astrotel itself is captured into a lunar orbit radius upon arriving at Earth or Phobos orbit radius upon arriving at Mars. Once in these transportation node orbits the Astrotel provides all the services to crews that the Spaceport provides in the Aldrin Low-thrust architecture. Propellant depots will be still be situated in these nodes in order to provide propellant processing and storage facilities. Because there are no continuous cycling Astrotels, the requirement for their remote resupply is eliminated along with the Astrotel Cargo Freighter. Since there is still a requirement to transport materiel to Mars, the Mars Cargo Freighter is still in operation. Chemical (LOX/LH) propulsion systems are assumed for the Astrotel orbit capture since high thrust maneuvers are required in order to keep flight duration low. Nominally these propellants are produced at ISRU sites in a similar fashion to the Aldrin Low-thrust architecture, except that considerably more propellants must be generated.

4.5.2 Integrated Propellant and Cargo Use Profiles

In this section we summarize the overall propellant and cargo masses that move through nodes of the Stopover Cycler transportation architecture over a 15-year period. Vehicle propellant, crew consumables and RRU mass requirements over 15 years have been summarized and collected in MAMA at their eventual transportation node in the following tables. All masses are kilograms.

Table 4-12 Cargo Mass Summary for Stopover Architecture

Location------------->	At LEO	At Earth Depot	At Mars Depot	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Users						
Mars Shuttle (1)						
Propellant						
LOX			83,428	230,306		
LH		11,918	11,918	32,901		
Total			95,346	263,207		
Water required on Mars				296,108		
Refurb Mass	4,403		4,403			
Lunar Polar Base						
Polar Ice Mine Refurb Mass	1,724	1,724			1724	
Mars Base						
Base Refurb Mass	50,942		50,942	50,942		
Propellant Plant Refurb Mass	3,712		3,712	3,712		
Phobos LOX Plant						
Propellant Plant Refurb Mass	6,449		6,449			
Astrotels (2)						
LOX		1,084,319	1,316,883			
LH		180,720	219,480			
Crew Consumables	112,084	56,042	56,042			
Xenon Propellant	-					-
Refurb Mass	74,640	37,320	37,320			
Mars Fuel Depot						
Refurb Mass						
Phobos LOX Storage Refurb Mass	561		561			
Station-keeping Xenon propellant Mass						
Earth Fuel Depot						
Refurb Mass						
LOX/LH Production Plant Refurb mass	104	104				
Station-keeping Xenon propellant Mass						
Lunar Water Tanker (1)						
Refurb Mass						
Astrotel Cargo Freighter						
Xenon Propellant	-					
Refurb Mass	-					
Mars Cargo Freighter						
Xenon Propellant	903,891					
Refurb Mass	34,468					

These cargo use profiles have been summarized as to where the cargo are needed throughout the transportation architecture for the 15 years of operation and displayed in the following table (all masses are in kg).

Table 4-13 Total Cargo Requirements for Stopover Architecture

Location------------->	At LEO	At Earth Depot	At Mars Depot	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Total Cargo Requirements						
LOX		1,084,319	1,400,311	230,306		
LH	231,399	192,638	231,399	32,901		
Water		1,436,576		296,108		
Xenon	903,891					
Refurb Mass	177,003	39,148	103,387	54,654	1,724	
Augmentation Tanks						
Crew Consumables	-					-
Communications Satellites						
Total Mass Required	1,312,293	2,752,681	1,735,097	613,969	1,724	

Finally, this cargo has been divided into the various delivery systems, which is displayed in the next table. These propellant and cargo use profiles provide the derived requirements for the transport of cargo and propellant throughout the architecture and the requirements on the ISRU production rates.

Table 4-14 Stopover Architecture Cargo Delivery System Manifest

Location------------->	At LEO	At Earth Depot	At Mars Depot	At Surface of Mars	At Surface of Moon	At Astrotels (2 vehicles)
Delivery Systems						
SEP Freighter Delivered Cargo		39,148	1,735,097			
Cargo Freighter Each Trip		5,593	247,871			
Lunar Water Tanker Total		1,436,576			1,724	
Number of tanker trips		58				
Lunar Water Tanker Each Trip		24,769			246	
Phobos LOX Tanker Total			1,400,311			
Number of tanker trips			30			
Phobos LOX Tanker Each Trip			46,677			
Mars Shuttle				54,654		
Number of Trips				7		
Mars Shuttle Each Trip				7,808		

4.5.3 Cargo Freighters

In the Stopover Cycler architecture only a Mars Cargo Freighter is required since the Astrotels are always resupplied at a transportation node. This vehicle delivers cargo from LEO to Mars. See Section 4.4.8 for the integrated cargo requirements. As before, the Mars Cargo Freighter uses xenon ion propulsion systems to spiral out of Earth orbit, shape the interplanetary trajectory to rendezvous with Astrotels or spiral into Mars orbit to Phobos. The cargo bay approach facilitates crew unloading.

The propulsion system of the Mars Cargo Freighter shares a high degree of technology heritage with the Astrotel IPS although the power level is considerably more.

Table 4-15 Stopover Mars Cargo Freighter Sizing

Mass of Cargo each trip to Mars Spaceport	**247,871**	kg
Initial mass of Cargo Freighter in LEO	430,855	kg
Propellant Mass	129,127	kg
Final Mass of Freighter	53,857	kg
P_o	4,308.6	kW
Power/Propulsion Mass (M_{ps})	34,468	kg

4.5.4 In Situ Resource Utilization Systems

The following sections summarize the *in situ* resource systems for the Moon, the Earth Propellant Depot, Mars surface, Phobos and the Mars Propellant Depot for the Stopover Architecture. The In situ Resource Utilization Systems (ISRU) are identical to the Aldrin Low-thrust Cycler architecture however they are scaled for the different propellant production requirements of the Stopover architecture. Displayed below are the solar power options. As with the Aldrin Low-thrust architecture, duty cycles for surface systems are assumed to be near 100% for the nuclear options.

4.5.4.1 Lunar Ice Mine

The automated Lunar Ice Mine

Table 4-16 Stopover Lunar Ice Mine Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Water Required		46,863	kg					
Mining and Excavation (0.16-g)	Mine regolith (kg/hr)	1562	kg/hr	34%	0.20	312	0.0133	20.83
Soil Hauler (0.16-g)	Haul soil to reactor and slag away							
Reactor/Condensor	Heats soil from -200oC to 50°C (kg H2O/hr)	15.62	kg/hr	34%	100.00	1,562	17	265.56
Water Storage	Stores extracted water (.33 x annual water production)	15.62	kg/hr	100%	0.02	312		
Distribution	Distributes liquids and gases to where they are processed	15.62			16.00	250		
Electrolyzer	Produces propellant for launching water to L1 (kg/hr)	25.31	kg/hr	34%	20.00	506	6.5	164.53
LH liquefaction	Provides LH for transfer vehicle (kg H2/hr)	3.16	kg/hr	34%	16.50	52	20	63.28
Lox liquefaction	Provides Lox for transfer vehicle (kg O2/hr)	22.15	kg/hr	34%	6.50	144	0.5	11.07
Lox, LH storage	Assumes immediate transfer to L1						Total Power-->	525.28
Solar Array - XX kW	Produces electrical power during Lunar day (kW)			34%	4.00	2,101		
Power Mgmt & Dist	Power voltage/freq control and distribution					210		
					Total Mass----->	5,451		

4.5.4.2 Earth LOX/LH Propellant Production and Storage Depot

The Lunar Orbit radius is an ideal location for the propellant depot where lunar water is electrolyzed into oxygen and hydrogen, LOX/LH is liquefied and finally these propellants are stored. Nearly continuous power is available using high efficiency solar arrays. The Earth Spaceport ISRU facility is expected to operate at a 90% duty cycle.

Table 4-17 Stopover Earth LOX/LH Production and Depot Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Water Delivered to L1		24,769	kg					
Water Storage	Stores Lunar water after delivery to L1 (3 mo. supply)	6,192	kg	100%	0.01	62		
Electrolysis Reactor	Converts water to oxygen and hydrogen (kg/hr)	2.58	kg/hr	91%	20.00	47	6.5	16.8
Distribution	Distributes liquids and gases to where they are processed	50	kg	91%	1.00	46		
LH liquefaction	Liquefies H2	0.29	kg/hr	91%	16.50	4	20	5.7
LoX liquefaction	Liquefies Ox	2.30	kg/hr	91%	6.50	14	0.5	1.1
LH Storage	H2 gas storage at X°C (kg) - 3 mo. supply, rest in Taxi	628	kg	100%	0.15	94	0.0016	1.0
LOx Storage	Cryogenic gas storage at -183oC - 3 mo. supply, rest in Taxi	5027	kg	100%	0.07	352	0.001	5.0
							Total Power-->	29.7
Solar Array - XX kW	Produces electrical power 100% of time (kW)			100%	4.00	119		
Power Mgmt & Dist	Power voltage/freq control and distribution					12		
					Total Mass--->	749		

4.5.4.3 Mars Dune Water Mine

Water is recovered from the regolith of Mars. Dune fields could be an excellent location for an excavation site. Once water is recovered from the soil it is electrolyzed into oxygen and hydrogen, liquefied into LOX/LH and finally stored within the Mars Shuttle tanks. The operation on Mars is fully automatic and operates at a 34% duty cycle along with the normal day/night periods.

Table 4-18 Stopover Dune Water Mine Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Water Required		19,741	kg/yr					
Dune Collection (0.38-g)	Excavates likely water bearing soil (kg/hr)	658	kg/hr	34%	0.27	178	0.00991	6.5
Soil Hauler (0.38-g)	Transports soil to reactor and slag away							
Reactor/Condensor	Heats soil to 500°C to remove water (kg/hr)	6.58	kg/hr	34%	205	1,349	35	230.3
Water Storage	Stores water before electrolysis (kg)	1147	kg	100%	0.04	46		
Electrolysis Reactor	Converts water to oxygen and hydrogen (kg/hr)	6.58	kg/hr	34%	20.00	132	6.5	42.8
Distribution	Distributes liquids and gases to where they are processed	6.58	kg/hr		3.70	24		
LH Liquifaction	Liquifies hydrogen (kg/hr)	0.82	kg/hr	34%	16.50	14	20	16.5
LOX Liquifaction	Liquifies oxygen (kg/hr)	5.76	kg/hr	34%	6.65	38	0.5	2.9
LOX Storage	Crygenic gas storage at -183°C (25% of annual production, rest on Mars Shuttle)	5527	kg	100%	0.07	387	0.5	1.15
LH Storage	Crygenic gas storage at Y°C (25% of annual production, rest on Mars Shuttle)	790	kg	100%	0.15	118	20.0	6.58
							Total Power-->	306.7
Solar Array	Produces electrical power 33% of time			34%	9.24	2,834		
Power Mgmt & Dist	Power voltage/freq control and distribution					142		
					Total Mass--->	5,261		

4.5.4.4 Phobos LOX Plant

Phobos regolith is excavated and processed to recover oxygen. The oxygen is then liquefied and transported to the Mars Propellant Depot where it can be stored in Taxi propellant tanks or propellant augmentation tanks (PATs). The plant on Phobos is fully automatic and operates at a 34% duty cycle consistent with the day/night periods.

Table 4-19 Stopover Phobos LOX Plant Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Total Oxygen Required kg/yr		93,354	kg/yr					
Excavation (zero-g)	Excavates Phobos regolith (kg/yr)	274,571	kg/yr					
Excavation (zero-g)	Excavates Phobos regolith (kg/hr)	91.5	kg/hr	34%	1.00	92	0.06	5.5
Soil Hauler (zero-g)	Transports soil to reactor and slag away							
Carbothermal Reactor	Reactions at 1600 °C to generate oxygen (kgO2/hr)	31.34	kg/hr	34%	96.00	3,009	16	501.5
Distribution	Distributes oxygen to where it is liquified	50.00	kg		1.00	50		
Liquifaction	Liquifies oxygen (kg/hr)	31.34	kg/hr	34%	6.50	204	0.5	15.7
LOX Storage	Assumes immediate transfer to Marsport							
							Total Power-->	522.7
Solar Array - XX kW	Produces electrical power 33% of time (kW)			34%	9.24	4,831		
Power Mgmt & Dist	Power voltage/freq control and distribution					242		
					Total Mass--->	8,427		

4.5.4.5 Mars Propellant Depot

Phobos LOX and Earth LH are stored at the Mars Propellant Depot that is located near Phobos in a Phobos radius orbit, in Taxi PATs and Mars Shuttle tanks. The operation at the Mars Spaceport is nearly continuous, except for occasional eclipses of the Sun by Mars.

Table 4-20 Stopover Mars Propellant Depot Mass and Power

Resource System Element	Purpose	Rate, kg/hr or Mass, kg	Units	Duty Cycle, %	Mass Factor, kg per rate or per stored kg	Total Mass, kg	Specific Power, kW per kg/hr produced	Total Power, kW
Phobos LOX Storage*	Crygenic gas storage at -183oC - (25% ann. prod., rest in Taxi tanks)	23,339	kg	100%	0.07	1,634	0.00018	4.2
Earth LH Storage*	Crygenic Earth LH storage at Y°C - (25% ann. Req't, rest in Taxi tanks)	3,857	kg	100%	0.15	578	0.0084	32.4
							Total Power-->	36.6
Solar Array - XX kW at ~1.5 AU	Produces electrical power 100% of time				9.24	338		
Power Mgmt & Dist	Power voltage/freq control and distribution					17		
					Total Mass--->	2,567		

4.6 Aldrin Low-thrust Cycler and Stopover Cycler Architecture Propellant Mass Comparison

In Figure 4-8 we compare the propellant mass requirements of the Aldrin Low-thrust and Stopover architectures as a function of the location the propellants are needed. Note that the vertical scale is logarithmic. At Earth the propellant requirements for the Stopover architecture are an order of magnitude higher than that required for the Aldrin Low-thrust architecture while at Mars the difference is only a factor of two.

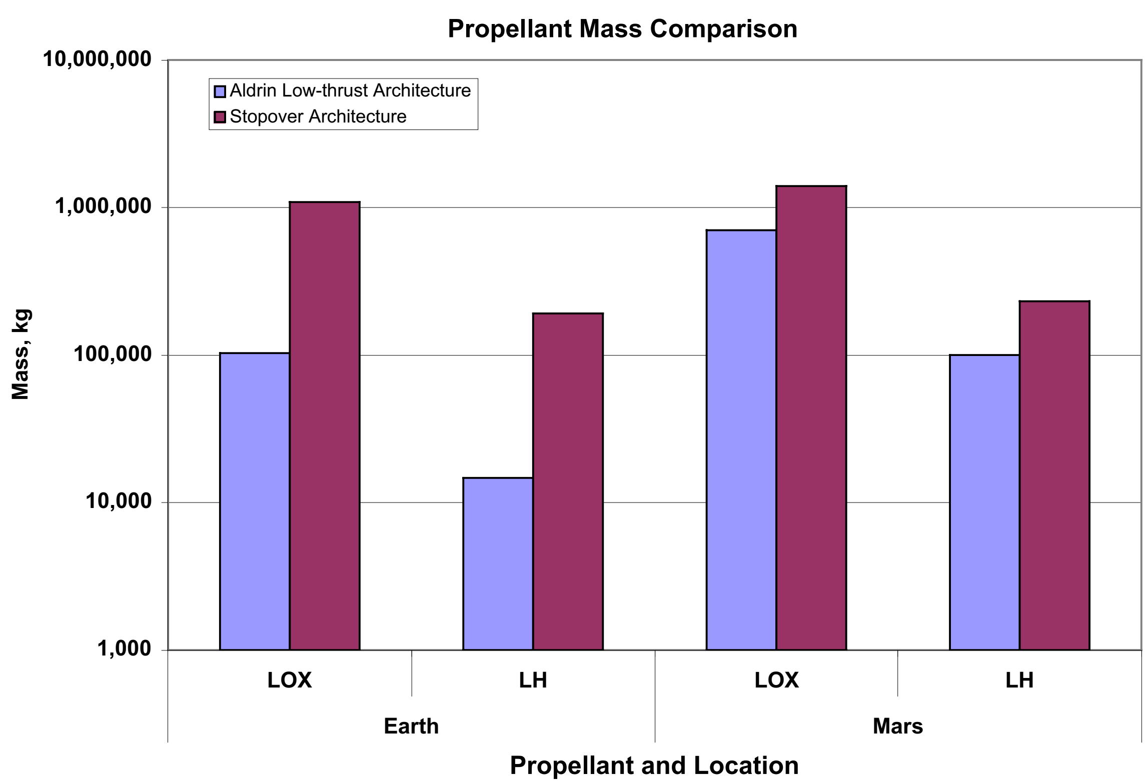

Figure 4-8 Propellant Mass and Location Requirements Comparison

5 Orbit Analysis Studies

5.1 Introduction

Phase II of this study has concentrated on continued development of the transportation architecture using the Aldrin Low-thrust Cyclers. In particular, a critical element is the hyperbolic rendezvous needed to transport a crew from some Earth or Mars orbit to the Astrotel. A 3-burn computer program has been written which minimizes the delta-V required for this transfer. One conclusion from this study is the benefit of placing the Spaceport in a lunar type orbit that enables the Taxi to do this transfer efficiently, and also be of use for operations in deep space performing other functions.

A second major effort is further optimization of the Aldrin Low-thrust Cyclers, either by modifying the basic ballistic trajectory, or by providing a considerable low thrust capability for the Astrotel. This has proved to be more difficult than anticipated, primarily because it is not clear how to alter the current cycler, and also the difficulty of performing a full trajectory optimization using low thrust.

The third effort is more detailed analyses of other possible cyclers, not necessarily to replace the Aldrin Low-thrust Cyclers, but to be able to compare their strengths and weaknesses with the Aldrin Low-thrust Cyclers choice. Two of these alternates, the 78 month semi-cycler, and the new concept which we call Stopover Cyclers have been studied in some detail, with the end purpose being a useful comparison of the three modes of transportation.

5.2 Aldrin Low-thrust Cycler Orbits

Aldrin Low-thrust cyclers are based on the ballistic Aldrin cycler orbits first developed by Aldrin and others [E. E. Aldrin, Cyclic trajectory concepts. SAIC presentation to the Interplanetary Rapid Transit Study Meeting, Jet Propulsion Laboratory (1985) and A. L. Friedlander, J. C. Niehoff, D. V. Byrnes and J. M. Longuski, Circulating transportation orbits between Earth and Mars. AIAA Paper No. 86-2009-CP, AIAAIAAS Astrodynamics Conference, Williamsburg, Va (1986)]

5.2.1 Ballistic Aldrin Cyclers

The Aldrin Cycler orbits have a period that is approximately equal to the Earth-Mars synodic period (26 months) and, when the line of apsides is rotated by gravity assist methods (average of about 51.4° each orbit), will enable Earth-to-Mars and Mars-to-Earth transfers every 26 months. Aldrin Cycler orbits come in two types, an Up Escalator and a Down Escalator orbit. The Up Escalator has the fast transfer occurring on the Earth to Mars leg while the Down Escalator is just the reverse. Figure 5-1 illustrates both orbit transfer geometries. When two Astrotels are used, an Aldrin Cycler provides relatively short transit times (~5 months) and regular transit opportunities. However, the planetary encounters occur at high relative velocities and typically, impose harsher requirements on the Taxi craft than other cyclers. Also the Aldrin Cycler requires a substantial orbit correction on 3 out of 7 orbits to maintain the proper orbit orientation. These corrections are required because of limitations of flyby altitude during the gravity assist. In the case shown, a 200-km flyby altitude constraint has been imposed. Mathematically correct, but impractical subsurface flybys eliminate the need for corrections.

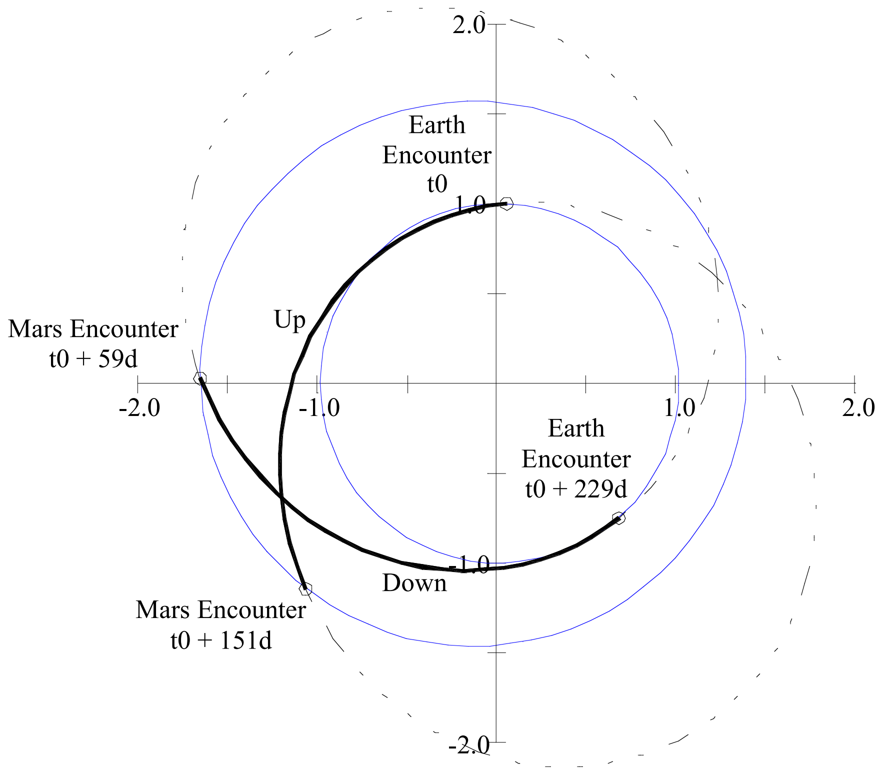

Figure 5-1 Aldrin Up/Down Cyclers

5.2.2 Low-thrust Analysis

Figure 5-2 illustrates an example low-thrust trajectory showing the low-thrust phase (bold black line) for an example Aldrin Up-cycler orbit. The low-thrust orbit corrections significantly reduce Astrotel propellant mass requirements, and this ripples through the entire architecture. Figure 5-3 illustrates the 15-year propagation of the outbound Aldrin Cycler.

The low-thrust trajectory analysis of SEP performance of 15 years of Up and Down Cyclers was generated in Phase I. The Astrotel SEP system carries out all propulsion maneuvers of the vehicle including the three large delta-Vs three orbits out of seven. The following tables describe the SEP performance requirements for two cases: a) near-optimum power for maximum payload and b) near-minimum power. Key assumptions for the analysis include 1) specific mass of the combined power and propulsion system is 8 kg/kW, 2) 15% tankage and reserve mass and 3) 75 mt initial vehicle mass. For 8 kg/kW specific mass, the power and propulsion system is only 1200 kg for a 150 kW_e system design. Propellant mass is about 3 mt for an ion engine operating at 5000 s specific impulse. The total propellant mass requirement results in a requirement for

Xenon delivery to the Astrotel of only about 430-kg average each orbit. Propulsion on time is consistent with projections of ion propulsion technology.

Table 5-1 SEP Performance Requirements for Aldrin Up (Outbound) Cycler Over 15 years

SEP Performance Requirements for Correcting the Aldrin Outbound Cycler Over 15 years

Assume: Power/Propulsion System Specific Mass = 8 kg/kWe
Propellant Tankage & Reserve Fraction = 15%
Initial Mass = 75,000 kg

Po (kWe @ 1AU)	Isp (sec)	Mps† (kg)	ΣMp (kg)	0.15*ΣMp (kg)	Mpayload (kg)	Propulsion On-Time 3 Corrections (days)		
* Near-Optimal Power for Maximum Payload								
150	2000	1200	5813	872	67,115	74	219	222
150	3000	1200	4044	607	69,149	91	273	279
150	4000	1200	3184	477	70,139	113	345	351
150	**5000**	**1200**	**2767**	**415**	**70,618**	**135**	**430**	**438**
* Near-Minimum Power								
60	2000	480	8847	1327	64,346	190	628	649
75	3000	600	6149	922	67,329	183	624	649
98	4000	784	4589	688	68,939	192	589	640
120	5000	960	3506	526	70,008	190	576	605

Table 5-2 SEP Performance Requirements for Aldrin Down (Inbound) Cycler Over 15 years

SEP Performance Requirements for Correcting the Aldrin Inbound Cycler Over 15 years

Assume: Power/Propulsion System Specific Mass = 8 kg/kWe
Propellant Tankage & Reserve Fraction = 15%
Initial Mass = 75,000 kg

Po (kWe @ 1AU)	Isp (sec)	Mps (kg)	ΣMp (kg)	0.15*ΣMp (kg)	Mpayload (kg)	Propulsion On-Time 3 Corrections (days)		
* Near-Optimal Power for Maximum Payload								
150	2000	1200	5645	847	67,308	16	216	276
150	3000	1200	4001	600	69,199	20	269	352
150	4000	1200	3329	499	69,972	25	335	460
150	**5000**	**1200**	**3007**	**451**	**70,342**	**30**	**410**	**552**
* Near-Minimum Power								
70	2000	560	8153	1223	65,064	37	484	637
85	3000	680	5633	845	67,842	38	495	648
109	4000	872	4200	630	69,298	38	475	640
120	5000	1030	3338	501	70,131	36	461	635

As a result of these analyses it was decided to select the 150 kW, 5000 s I_{sp} case for the baseline Astrotel SEP system. The baseline case is denoted by boldface type in the table above. Figure 5-2 shows orbit geometry for the low thrust maneuvers for the second correction of the Up Cycler orbit. It can be seen that the low thrust orbit correction takes a substantial portion of the orbit, 430 days, as opposed to the impulsive correction mode.

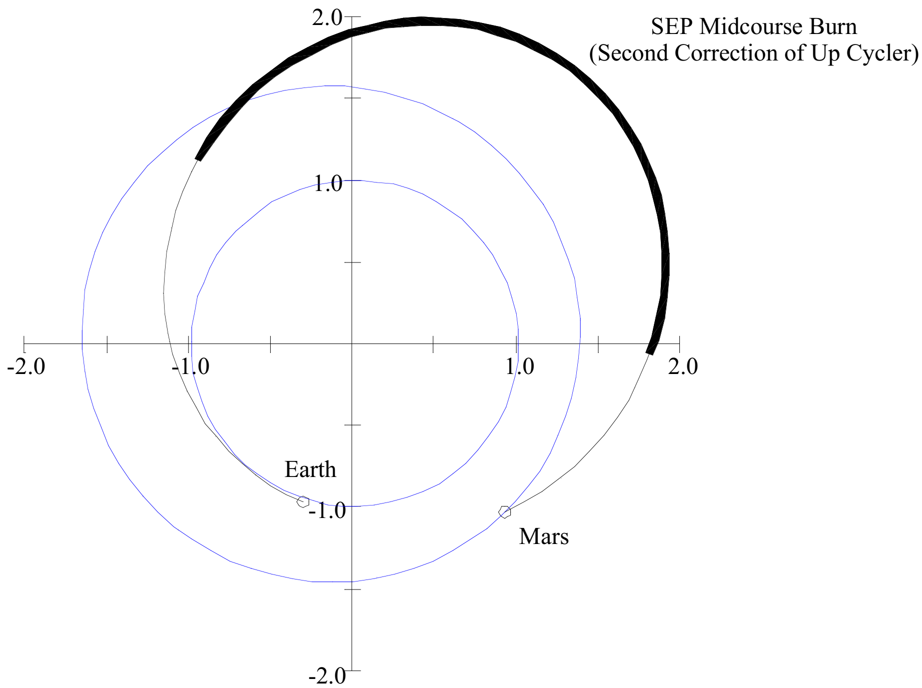

Figure 5-2 Example Low-thrust Phase on Aldrin Up-cycler Orbit

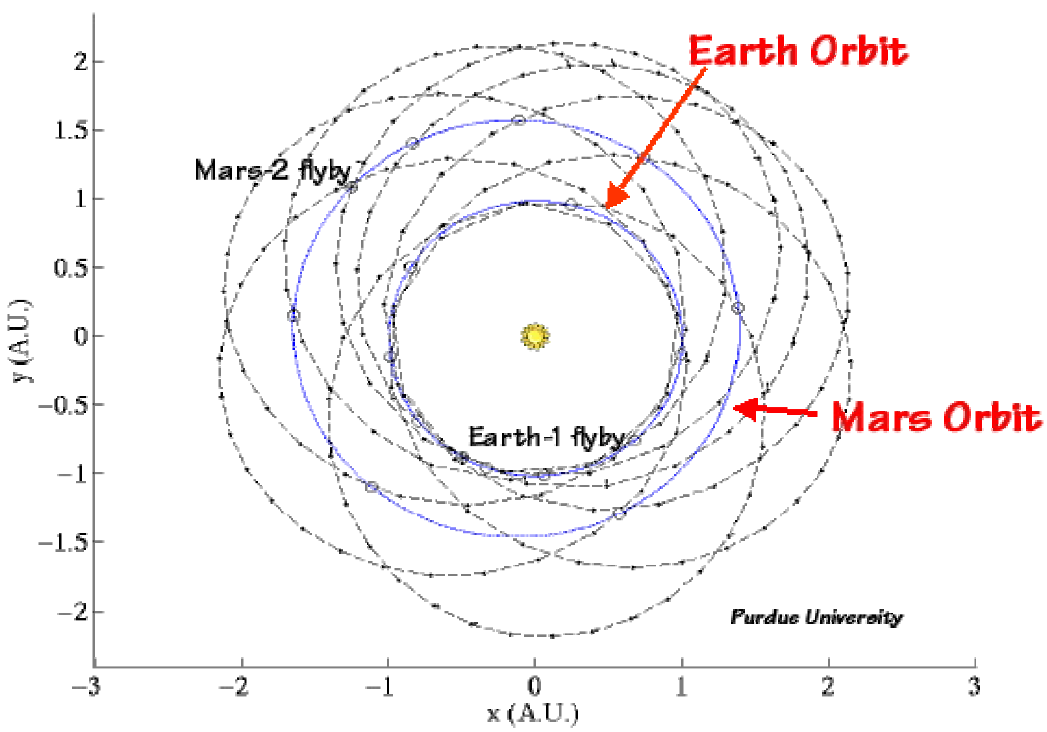

Figure 5-3 Aldrin Low-thrust Down-cycler 15-year Sequence

5.3 Transportation Orbit Options

The Aldrin Cycler is unique in that a single non-stop heliocentric orbit with a period of about 26 months (the Earth-Mars synodic period) is capable of short flight times to and from Mars. With proper phasing, one orbit (or vehicle) can provide Earth-to-Mars transfers, while a second provides Mars-to-Earth transfers, each of 5 to 6 month duration, and for every 2-year launch opportunity. Planetary gravity assist, primarily by Earth, is used to rotate the line of apsides about 50 deg during each revolution to maintain the encounters. The eccentricity and inclination of the Mars orbit, however, introduces some undesirable results. One is the need for large heliocentric maneuvers on certain revolutions to maintain the cycler orbit, and the other is a significant increase in the Mars flyby velocity, also on certain revolutions. These high velocities result in high Taxi propellant requirements, higher Mars entry velocities, and more difficult rendezvous with the Astrotel.

A major effort in Phase II of the study was a search for alternative orbit options. Two have been chosen as most promising. These are the 78-month semi-cycler, and the use of direct transfers, which we refer to as Stopover cyclers. These are discussed in the following sections.

5.3.1 Semi-cyclers

The semi-cycler trajectories connecting Earth and Mars, as the name implies, do not freely travel indefinitely about the sun with close encounters of the two planets. Instead, they begin and end at one of the planets with one or more encounters of the other without delta-V augmentation. An example, given in Figure 5-4, begins at Mars, make 5 flybys of Earth (in 4 years) with the fifth flyby being a gravity assist sending it back to Mars, where the trajectory ends with Mars orbit insertion.

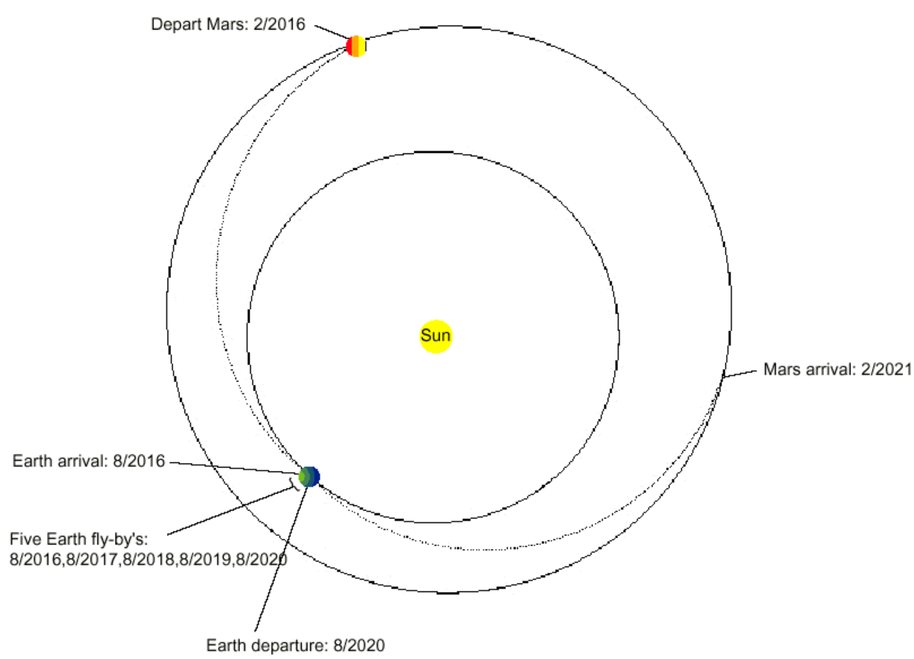

Figure 5-4 Example Semi-Cycler Showing Earth and Mars at Departure and Arrival Times (R. H. Bishop, et al, Earth-Mars Transportation Opportunities, Paper AAS 00-255)

In this case, the total flight time is 5 years, assuming about 6 months each for Mars-to-Earth at the beginning, and Earth-to-Mars at the end of the semi-cycler. The vehicle then remains at Mars about 18 months before the next Mars departure for the semi-cycler. In this period of 78 months, one vehicle would have missed 2 each of the opportunities for Mars-to-Earth and Earth-to-Mars transfer. Since the Earth and Mars opportunities appear about every 2 years, three vehicles are needed to take advantage of all opportunities.

5.3.2 Stopovers Cyclers

We considered direct orbit transfers that require propulsive maneuvers at both planets. These trajectories we call Stopover Cyclers since the Astrotel is refueled and continues to shuttle between Earth and Mars. Advantages of Stopover Cyclers are (1) lower departure and arrival velocities for given flight times, (2) flexible launch and arrival dates, (3) elimination of the hyperbolic rendezvous, (4) close vicinity of the Astrotels to the planet for replenishment and refurbishment, and (5) alternate mission uses for the Astrotels while in orbit about each planet, waiting for the next opportunity to return. For the heliocentric transfer, propellant usage for station departure and high orbit capture is minimized for each given flight time, and requirements for a one month launch period is computed. Given this parametric data, one possible launch strategy, of many, is presented to illustrate specific launch and arrival dates, and associated parameters. Two Astrotels are needed to take advantage of all launch opportunities, with each waiting at alternate planets for almost 18 months. This wait time may be used to perform items 4 and 5 above, with orbit alterations being performed with low-thrust. We assume that the Stopover Cycler Astrotel uses chemical propulsion systems (LOX/LH) instead of IPS.

The set of trajectories available for stopovers is quite large. Possible departure and arrival dates in either direction with reasonable propellant requirements can be weeks long during each 2-year launch opportunity. The subset of these trajectories of interest here would be minimum propellant usage for given flight times. These have been computed and are shown in Figure 5-5 and Figure 5-6, assuming launch from and capture into 4-day period orbits at Earth and Mars. (Since generating these parametric data we have selected capture into lunar and Phobos radius orbits in order to facilitate ISRU propellant transport to the Astrotel.) The initial capture maneuvers are made at 200–km altitude for greatest efficiency. These would represent loose-capture orbits, and the near-planet maneuvers required to meld these into the overall architecture will be studied.

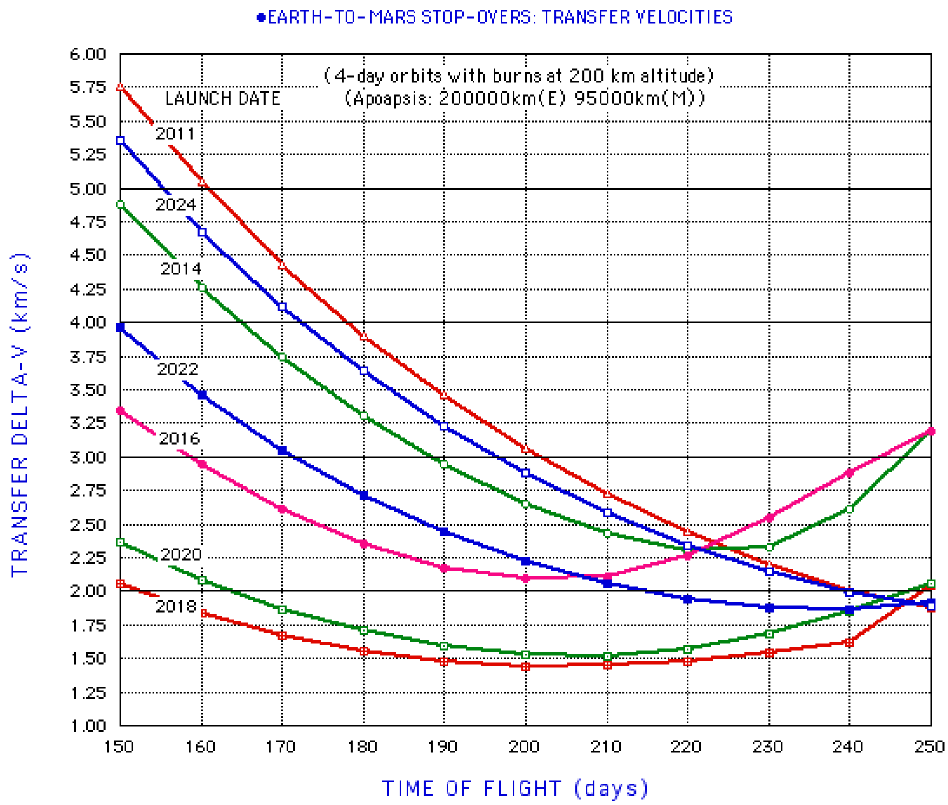

Figure 5-5 Delta-V versus Flighttime for Earth-to-Mars Stopover Trajectory Opportunities

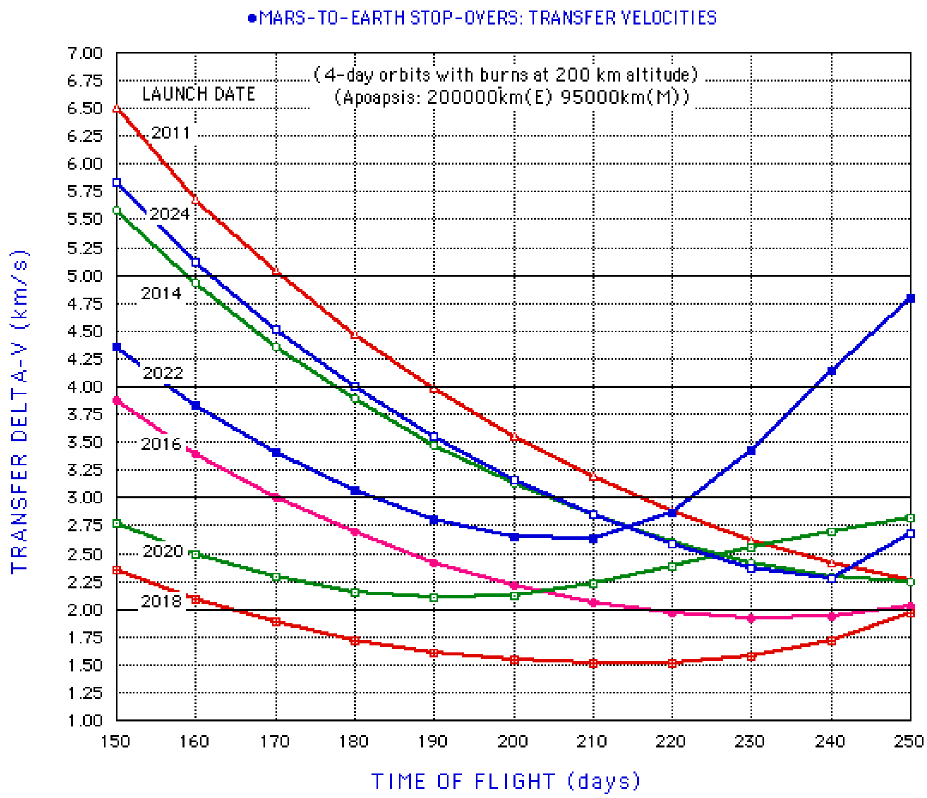

Figure 5-6 Delta-V versus Flighttime for Mars-to-Earth Stopover Trajectory Opportunities

Looking at these figures, one characteristic becomes immediately apparent. The flight time resulting in the minimum delta-V for one 15-year time period of 7 cycles (2011 to 2026) is about 220 days, or just over 7 months. For Earth to Mars, the maximum delta-V is 2500 m/s, and for Mars to Earth it is 3000 m/s. For these requirements, the flight time will sometimes be shorter. If the flight time is restricted to 5 months, these requirements jump to 5750 m/s and 6500 m/s. respectively, which is more than double. These are worse case numbers, and if propellant averages are taken, the 7-month option would be closer to 2000 m/s for Earth to Mars transfers, and 2300 m/s for Mars to Earth. The 5-month propellant averages are 3700 m/s and 4200 m/s, respectively. The penalty for the shorter flight time is about 85% above the longer, 7-month flight time. For the study of the Stopover Cyclers architecture, we selected a fixed flight duration of 6 months or 180 days, though, in operation shorter or longer flight times may be chosen. The figures below illustrate typical flight trajectories using opportunities between 2011-2024.

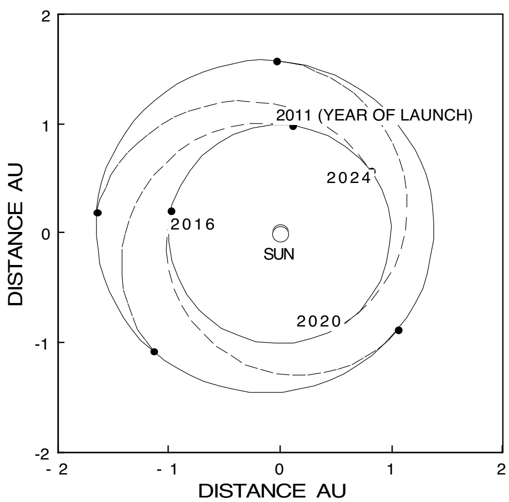

Figure 5-7 Astrotel 1 Earth-to-Mars Transfers

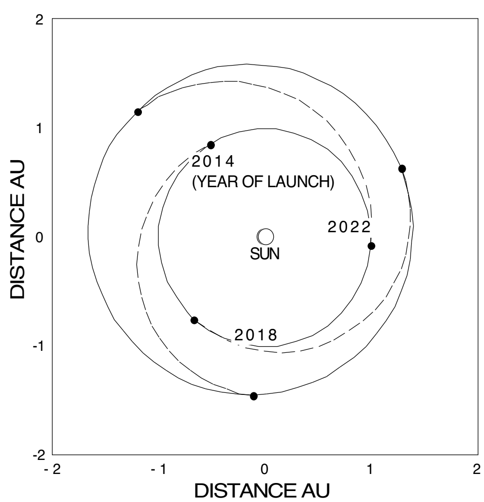

Figure 5-8 Astrotel 2 Earth-to-Mars Transfers

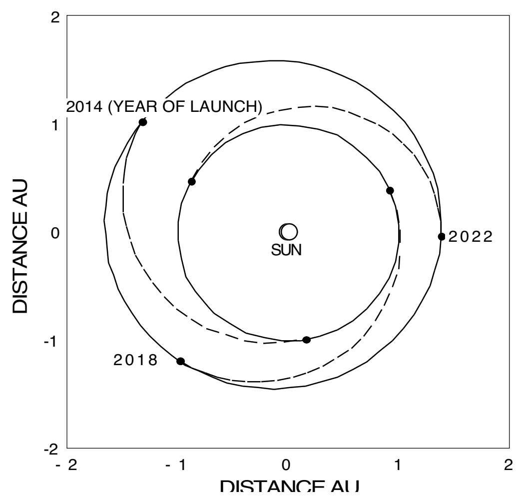

Figure 5-9 Astrotel 1 Mars-to-Earth Transfers

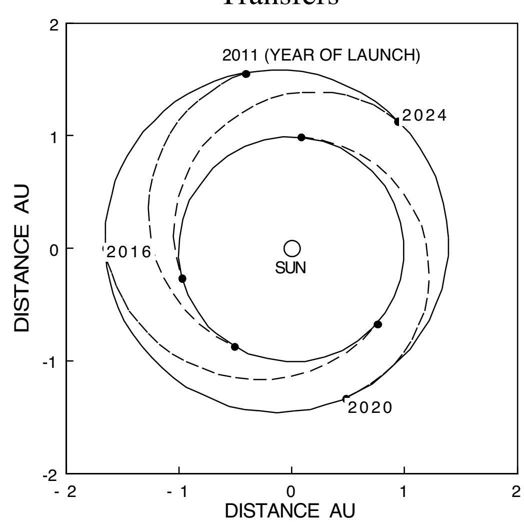

Figure 5-10 Astrotel 2 Mars-to-Earth Transfers

Stopover trajectory data and propellant requirements are summarized in the following two tables. The delta-V assumptions for these tables are consistent with capture at Earth into a lunar radius orbit and at Mars into a Phobos radius orbit. These data include all plane change delta-Vs departing Earth or Mars. Typical data is displayed in Table 5-3 and Table 5-4. This particular chart shows trajectory parameters, including total delta-V [DVT], for arrival and destination orbits at Earth and Mars being the lunar orbit and Phobos orbit, respectively. Astrotel propellant requirements have been generated for the Stopover Cycler option using the Stopover version on MAMA. These requirements, displayed in Table 5-5 and Table 5-6, are significantly larger than the Taxi propellant requirements, however we have eliminated two vehicle types, namely the Taxi and Spaceport, so the total Life-Cycle-Cost could be lower and thus this architecture is more desirable. However, the increase in propellant requirements will have a significant impact on ISRU sizing and costs.

Table 5-3 Earth-to-Mars Stopover Trajectory Data

OPTIMUM EARTH TO MARS STOP-OVER BALLISTIC TRANSFERS (MIN DV-TOTAL) Destination **PHOBOS**

LDATE (DAY)	ADATE (DAY)	TFL (DAY)	C3 (KM2/S2)	DV-TMI (KM/S)	DLA (DEG)	RLA (DEG)	VHP (KM/S)	DV-MOI-H (KM/S)	DV-MOI-P	DV-MOI-D	DAP (DEG)	RAP (DEG)	DVT (KM/S)	PHA (DEG)
25-Dec-07	22-Jun-08	180	28.222	2.12400	22.245	129.461	5.443	2.518	3.517	3.247	-19.137	281.315	5.64068	151.439
30-Jan-10	29-Jul-10	180	21.800	1.86000	1.635	159.261	4.995	2.190	3.185	2.915	-9.731	310.722	5.04512	140.077
18-Mar-12	14-Sep-12	180	14.855	1.60100	-30.182	200.946	4.020	1.530	2.523	2.253	1.494	337.938	4.12394	116.886
19-May-14	15-Nov-14	180	8.861	1.34400	-35.896	308.599	3.090	0.983	1.978	1.708	9.809	29.797	3.32247	101.984
29-Jul-16	25-Jan-17	180	14.421	1.55500	16.761	12.512	2.924	0.896	1.889	1.619	3.048	122.152	3.44448	127.901
27-Sep-18	26-Mar-19	180	25.426	2.02200	32.924	48.271	3.889	1.448	2.444	2.174	-13.170	188.844	4.46570	149.971
6-Nov-20	5-May-21	180	30.533	2.22500	35.168	84.890	4.950	2.158	3.159	2.889	-21.307	232.864	5.38401	156.271

Table 5-4 Mars-to-Earth Stopover Trajectory Data

OPTIMUM MARS TO EARTH STOP-OVER BALLISTIC TRANSFERS (MIN DV-TOTAL) Destination **MOON**

LDATE (DAY)	ADATE (DAY)	TFL (DAY)	C3 (KM2/S2)	DV-TEI (KM/S)	DLA (DEG)	RLA (DEG)	VHP (KM/S)	DV-EOI-H (KM/S)	DV-EOI-LOR	DAP (DEG)	RAP (DEG)	DVT (KM/S)	PHA (DEG)
13-Nov-11	11-May-12	180	26.801	3.842	20.390	51.636	5.887	1.647	2.383	-2.662	20.283	6.225	-34.423
23-Dec-13	21-Jun-14	180	22.586	3.846	30.882	88.806	5.362	1.408	2.145	3.944	45.573	5.991	-46.673
3-Feb-16	1-Aug-16	180	12.078	3.022	28.234	145.455	4.789	1.168	1.905	13.320	78.345	4.927	-51.356
9-Apr-18	6-Oct-18	180	7.432	2.022	-4.946	221.850	3.416	0.689	1.426	23.957	127.526	3.448	-68.819
20-Jun-20	17-Dec-20	180	11.969	3.123	-31.467	307.060	3.245	0.640	1.376	12.639	222.715	4.499	-54.918
15-Aug-22	11-Feb-23	180	18.409	3.223	-19.600	348.839	4.195	0.943	1.680	-9.704	292.624	4.903	-31.343
22-Sep-24	21-Mar-25	180	23.393	3.269	-1.266	14.707	5.452	1.448	2.184	-11.166	338.360	5.453	-24.857

Table 5-5 Earth-to-Mars Stopover Propellant Requirements

Earth to Mars			Orbit Destination	**PHOBOS**	
Launch	Arrive	Total DV, km/s	Mi, kg	Mp, kg	
25-Dec-07	22-Jun-08	5.641	382,851	273,207	
30-Jan-10	29-Jul-10	5.045	335,500	225,857	
18-Mar-12	14-Sep-12	4.124	273,532	163,888	
19-May-14	15-Nov-14	3.322	229,007	119,363	
29-Jul-16	25-Jan-17	3.444	235,285	125,642	
27-Sep-18	26-Mar-19	4.466	295,060	185,416	
6-Nov-20	5-May-21	5.384	361,676	252,032	
			Total Propellant	1,345,406	Required to be delivered to Astrotel in its Earth orb
				192,201	LH from Moon water
				1,153,205	LOX from Moon water

Table 5-6 Mars-to-Earth Stopover Propellant Requirements

Mars to Earth			Orbit Destination	**MOON**	
Launch	Arrive	Total DV, km/s	Mi, kg	Mp, kg	
13-Nov-11	11-May-12	6.225	435,828	326,185	
23-Dec-13	21-Jun-14	5.991	413,722	304,079	
3-Feb-16	1-Aug-16	4.927	326,804	217,161	
9-Apr-18	6-Oct-18	3.448	235,464	125,821	
20-Jun-20	17-Dec-20	4.499	297,272	187,629	
15-Aug-22	11-Feb-23	4.903	325,117	215,473	
22-Sep-24	21-Mar-25	5.453	367,264	257,621	
			Total Propellant	1,633,968	Required to be delivered to Astrotel in its Mars orbi
				233,424	LH from Earth
				1,400,544	LOX from Phobos

We developed a technique to realign the insertion elliptic orbit of the Astrotel, at Earth or Mars, so that about 18 months after arrival, its new orientation will allow a single in-plane periapsis burn for departure, to Mars or Earth. This is possible with a 5-burn process, or less depending on the two orbit similarities. Each burn modifies the orbit in the following way, using Earth here as an example: (1) perigee raise, (2) plane change, (3) apogee raise, (4) perigee drop, and (5) finally, the escape burn at perigee on the date of departure.

5.4 Hyperbolic Rendezvous

Perhaps one of the major concerns in the use of classic flyby cyclers for human transportation to and from Mars is the need for hyperbolic rendezvous. In Earth orbit, for example, one can take time to catch up with, and rendezvous with, another vehicle. This is not the case for the cycler. The primary restriction here is that the rendezvous must take place within about 7 days from the time of departure from the Spaceport. With flyby speeds of the Astrotel at about 6 km/s for Earth and 7 to 12 km/s for Mars, a large maneuver, preferably at low altitude, must be performed at a fairly precise time. Figure 5-11 shows the Spaceport orbit, the Astrotel flyby and three Taxi rendezvous trajectory options. We have selected the 3-burn option because it requires low delta-V and a short flighttime. The 3-burn option consists of a Taxi departure maneuver, ΔV_1, to drop-down to low periapsis altitude to do the escape maneuver, ΔV_2, and finally several days later, the rendezvous maneuver, ΔV_3. Careful preparation and planning is required for this process. The 3-burn option provides considerable flexibility in departure time. The timing and location of the initial departure ΔV_1 can be adjusted by as much as a day by using Type I (<180°) or Type II

(>180°) transfers to periapsis where ΔV_2 occurs. The primary requirement is to be at the proper position at the proper time for the ΔV_2.

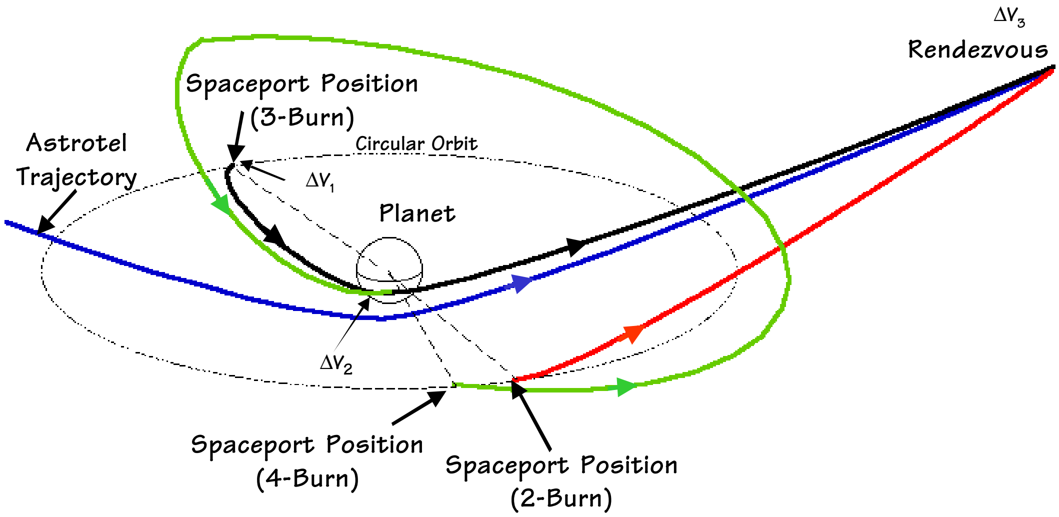

Figure 5-11 Hyperbolic Rendezvous Options in 3D

Figure 5-12 illustrates an example departure trajectory from Lunar orbit radius (LOR) in three dimensions as viewed from normal to the Moon's orbit and from a perspective view.

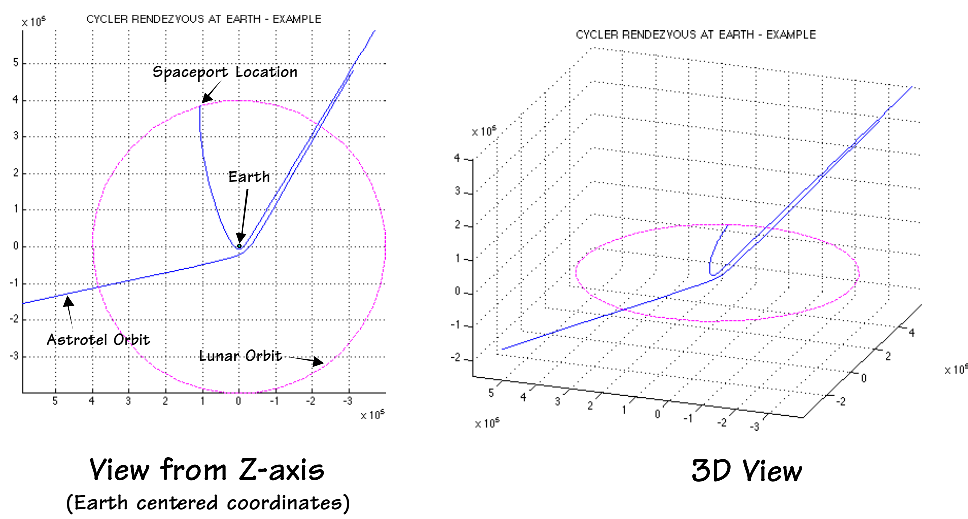

Figure 5-12 Example Hyperbolic Rendezvous from Lunar Orbit Radius in 3D

A computer program has been written to perform the three required maneuvers for hyperbolic rendezvous. Figure 5-13 illustrates a 2D version of this chart that was used in formulation of a computer program. This program has been used to generate all hyperbolic rendezvous requirements for the reference Aldrin cycler. It provides 3-dimensional solutions, and can be driven by an optimization program. It also can be easily modified to provide 2- or 4-burn solutions, where the first would be used to do a direct escape to rendezvous and the second would add a burn by flying to some distance from the planet to perform a plane change. The latter is forbidden here because of the flight time restriction. Fortunately, large plane changes are not required for the Aldrin cycler, so the 3-burn is appropriate. Because of this, closed form equations representing 2-dimensional solutions were derived and used for trajectory inputs to the MAMA program that models the transportation architecture.

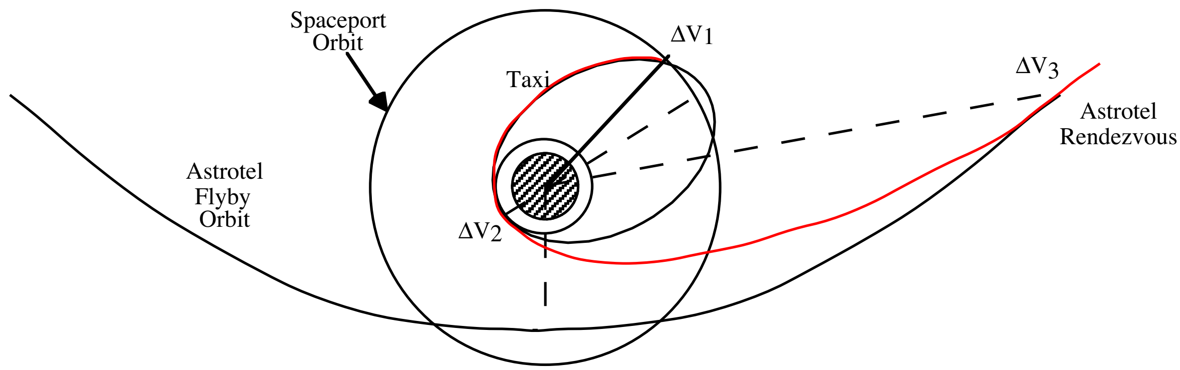

Figure 5-13 Taxi Trajectory in 2D from Spaceport to Hyperbolic Rendezvous with the Astrotel

5.5 Spaceport Location

Having to solve the hyperbolic rendezvous problem provided insight into the desired location of the Spaceport. As discussed above, plane changes are best made far from the planet where the orbital velocities are low. Velocity changes are best made close to the planet, in the gravity well. In LEO, the orbital velocity is almost 8 km/s, whereas at lunar distance, the velocity is about 1 km/s, which is a better place to make a required plane change. The L_1 libration point, which has the Moon's period, is closer to the Earth and was chosen for the Spaceport location in Phase I. The advantage is a lower orbital velocity than the Moon, and therefore a lower required plane change delta-V. The disadvantage of L_1, however, is that it is tied to the Moon and has the same period of about 28 days. For this reason, it could be in the wrong position in its orbit at the time of the first maneuver for the hyperbolic rendezvous. For the L_1 location, the Spaceport could be almost a month away from its required position for Taxi departure. This position mismatch can be mitigated by either high delta-V, which negatively impacts mission performance, or very long phasing orbits, which require excessive crew time in the Taxi.

This implies that an *active* Spaceport is needed, that can position itself into the proper hyperbolic departure location, and be there at the required time. This positioning is accomplished by changing the period of the Spaceport, which is assumed to be at about the distance of the Moon, to cause drifting to the required orbit over a period of months, and then reverting back to its original period. The velocity is proportional to drift time or about 1 m/s for 1° in longitude over a period of a month.

For Mars, the situation is different. The Spaceport, because of its other functions, particularly to aid in the in-situ production of material at Phobos has been situated in an orbit close to Phobos. Its period is about 7.5 hr, which is short enough to accommodate most launch phasing requirements.

5.6 Cycler Orbit Sequences

The sequence of key events for the three basic architecture options was generated. The following figure displays a version of the comparison of key orbit events for the Aldrin Cycler, Semi-cycler and Stopover trajectory options. This chart shows the approximate solar range of the Astrotel in its travel to and from Earth and Mars. The horizontal axis is time in years from 2012 to 2027. The vertical axis is offset solar radius. Each set of cyclic orbits is offset by 2 AU in order to facilitate displaying and comparing the relative timing. For example, the top graph displays the Aldrin Down Cycler (ADC). The numbers at the points of planetary flybys refer to crews being transferred. Approximate crew tours of duty are shown.

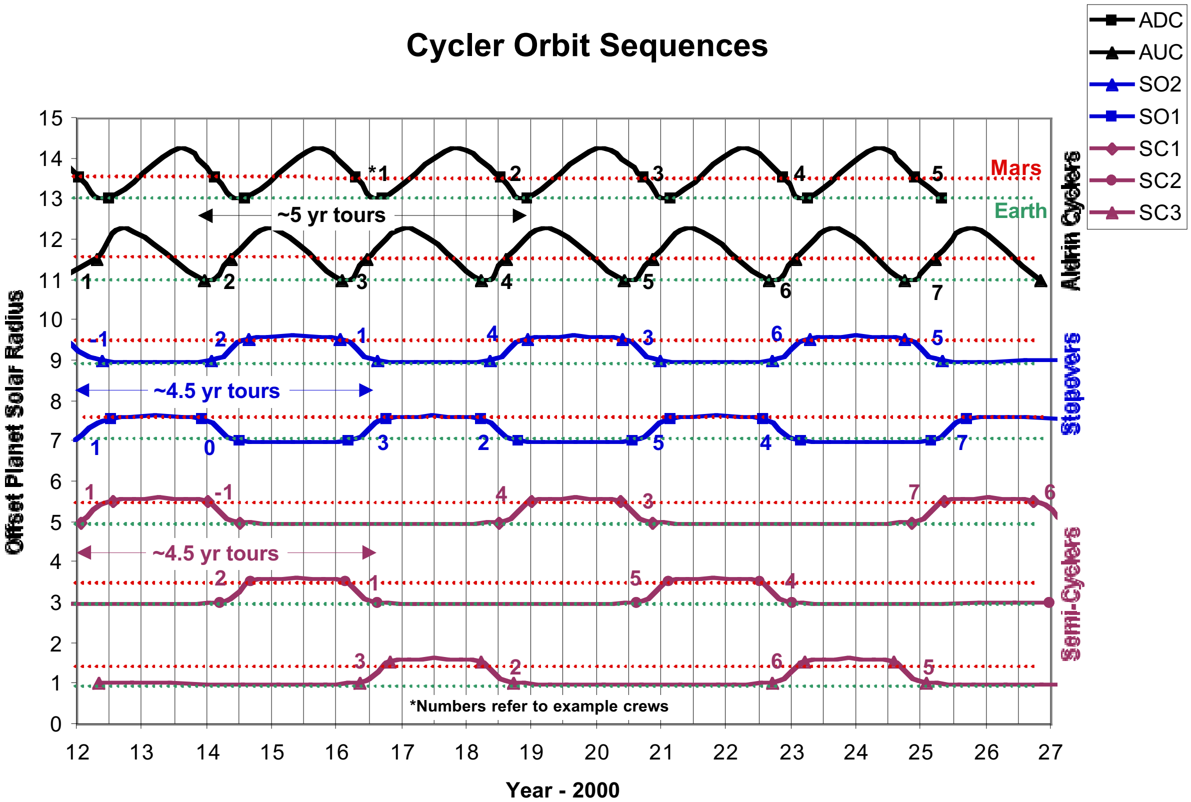

Figure 5-14 Cycler Orbit Sequences

5.7 Low-thrust Cargo Transport Orbit Sequences and Timing

A practical option is to transport cargo to Mars and continuously cycling Astrotels via reusable Cargo Freighters. An example Astrotel Cargo Freighter trajectory was analyzed. The Earth launch date is October 11, 2023 with return to Earth on May 14, 2027. This analysis resulted in a flight time (including planetocentric spirals of 966 days) to the Astrotel, only 335 days back to LEO for total round-trip flight time from LEO of 1311 days. Total propulsion on time is 608 days and there is an Earth turnaround time of 304 days before departure to the Astrotel again. For this analysis, total assumed payload mass delivered in 15 years was set at about 88 mt. Scalable mass fractions at the optimal power-to-initial mass ratio of 10 W/kg at 1 AU were 0.6016 for the payload-to-initial mass ratio and 0.2769 for the propellant-to-initial mass ratio. The following table describes the initial, payload, propellant and propulsion/power system masses and the solar array size as a function of number of sorties to the Astrotel in 15 years given the assumed cargo to be transported.

Table 5-7 Astrotel Cargo Freighter Parameters vs Number of Sorties in 15 years

# of Sorties	P_o (kW$_e$ @ 1AU)	M_o (kg)	M_{ps} (kg)	ΣM_p (kg)	0.15* ΣM_p (kg)	M_{cargo} (kg)
7	211	21,104	1,688	5,843	877	12,696
6	246	24,621	1,968	6,818	1,023	14,812
5	295	29,545	2,360	8,183	1,228	17,774
4	369	36,932	2,952	10,228	1,534	22,218
3	492	49,240	3,936	13,636	2,045	29,623

An example Mars Cargo Freighter orbit was analyzed. For cargo transport to the Mars Spaceport analysis the Earth launch date was January 5, 2011 with Earth return date of October 26, 2014. This analysis resulted in a flight time to Mars of 753 days (including a 263 day Earth escape spiral and 90 Mars capture spiral). A stay time of 178 days at the Mars Spaceport days was assumed followed by flight time of 459 days back to LEO (including a 22 day Mars escape spiral and 37 day Earth capture spiral). Total round-trip flight time from LEO is 1390 days. Total propulsion on time is 733 days and there is an Earth turnaround time of 140 days before departure to the Mars Spaceport again. For this analysis, total derived payload mass delivered in each opportunity was between 6-17 mt. Scalable mass fractions at the optimal power-to-initial mass ratio of 10 W/kg at 1 AU were 0.5753 for the payload-to-initial mass ratio and 0.2997 for the propellant-to-initial mass ratio. The following table describes the initial, propellant and propulsion/power system masses and the solar array size as a function of cargo mass to the Mars Spaceport delivered in each opportunity. Note this table only goes to a 30-mt payload level, but the results are directly scalable to higher payload mass values. In addition, these numbers assume all structure and spacecraft bus avionics are included in M_{ps}.

Table 5-8 Mars Cargo Freighter Parameters as a Function of Cargo Mass

P_o (kW$_e$ @ 1AU)	M_o (kg)	M_{ps} (kg)	ΣM_p (kg)	0.15*ΣM_p (kg)	M_{cargo} (kg)
100	10,000	800	2,997	450	5,753
150	15,000	1,200	4,496	674	8,630
200	20,000	1,600	5,994	899	11,507
250	25,000	2,000	7,492	1,124	14,384
300	30,000	2,400	8,990	1,349	17,261

The nominal plan is to have two Astrotel Cargo Freighters for the Astrotels (one for each Astrotel), each delivering cargo every other opportunity and two to four Cargo Freighters for the Mars Spaceport (one or two leaving every opportunity). This plan ensures a cargo delivery every Mars opportunity and every other opportunity for each Astrotel thus potentially building in some reliability and margin.

The tables below summarizes the a full 15 years of operations of the Mars and Astrotel Cargo Freighters from Earth to their respective targets including stay time at the target prior to return to Earth. The assumptions for this global analysis was somewhat different than the analysis above resulting is somewhat difference timing. Note that there are two vehicles in operation at any one time in order to ensure full resupply to the Mars elements of the architecture, hence Vehicle 1 operates ever other opportunity. Two dates are shown, namely the Julian date (above) and the calendar date (below). IP refers to the interplanetary phases. Refurbishment time is shown, which corresponds to the stay time at Earth prior to the next trip to Mars or the Astrotel. Table 5-9 summarizes trajectory-related information for the Astrotel Cargo Freighter. Table 5-10 summarizes similar information for the Mars Cargo Freighter.

Table 5-9 Sequence for the Astrotel Cargo Freighter

Astrotel Resupply Freighter Mission Timelines

Outbound

Opp	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	Stay Days	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	LEO Arrival	Refurb Time
				+10	+271								+132	+163					
2011 1	2455515 11/14/10	265	2455780 8/6/11	2455790 8/16/11	2456051 5/3/12	700	2456480 7/6/13	0	10	2456490 7/16/13	0	2456490 7/16/13	2456622 11/25/13	2456653 12/26/13	300	2456790 5/12/14	35	2456825 6/16/14	237
2015 3	2457062 2/8/15	263	2457325 10/29/15	2457335 11/8/15	2457596 7/26/16	700	2458025 9/28/17	0	10	2458035 10/8/17	0	2458035 10/8/17	2458167 2/17/18	2458198 3/20/18	300	2458335 8/4/18	35	2458370 9/8/18	280
2020 5	2458650 6/15/19	263	2458913 3/4/20	2458923 3/14/20	2459184 11/30/20	700	2459613 2/2/22	0	10	2459623 2/12/22	0	2459623 2/12/22	2459755 6/24/22	2459786 7/25/22	300	2459923 12/9/22	35	2459958 1/13/23	271
2024 7	2460229 10/11/23	266	2460495 7/3/24	2460505 7/13/24	2460766 3/31/25	700	2461195 6/3/26	0	10	2461205 6/13/26	0	2461205 6/13/26	2461337 10/23/26	2461368 11/23/26	300	2461505 4/9/27	35	2461540 5/14/27	203
2028 2	2461743 12/3/27	264	2462007 8/23/28	2462017 9/2/28	2462278 5/21/29	700	2462707 7/24/30	0	10	2462717 8/3/30	0	2462717 8/3/30	2462849 12/13/30	2462880 1/13/31	300	2463017 5/30/31	35	2463052 7/4/31	298
2033 4	2463350 4/27/32	263	2463613 1/15/33	2463623 1/25/33	2463884 10/13/33	700	2464313 12/16/34	0	10	2464323 12/26/34	0	2464323 12/26/34	2464455 5/7/35	2464486 6/7/35	300	2464623 10/22/35	35	2464658 11/26/35	258
2037 6	2464916 8/10/36	265	2465181 5/2/37	2465191 5/12/37	2465452 1/28/38	700	2465881 4/2/39	0	10	2465891 4/12/39	0	2465891 4/12/39	2466023 8/22/39	2466054 9/22/39	300	2466191 2/6/40	35	2466226 3/12/40	223

Mo (mt) =	33.5	Po (kWe) =	335.5	Mp min (mt) =	9.1	Mp max (mt) =	9.499	Alpha (kg/kWe) =	8

Inbound

Opp	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	Stay Days	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	LEO Arrival	Refurb Time
				+36	+303.1	+667	+700								+161.8	+194.6					
2010 1	2454947 4/25/09	263	2455210 1/13/10	2455246 2/18/10	2455513 11/12/10	2455877 11/11/11	2455910 12/14/11	700	2455910 12/14/11	0	10	2455920 12/24/11	0	2455920 12/24/11	2456082 6/3/12	2456115 7/6/12	300	2456220 10/19/12	35	2456255 11/23/12	254
2014 3	2456509 8/4/13	264	2456773 4/25/14	2456809 5/31/14	2457076 2/22/15	2457440 2/21/16	2457473 3/25/16	700	2457473 3/25/16	0	10	2457483 4/4/16	0	2457483 4/4/16	2457645 9/13/16	2457678 10/16/16	300	2457783 1/29/17	35	2457818 3/5/17	293
2018 5	2458111 12/23/17	264	2458375 9/13/18	2458411 10/19/18	2458678 7/13/19	2459042 7/11/20	2459075 8/13/20	700	2459075 8/13/20	0	10	2459085 8/23/20	0	2459085 8/23/20	2459247 2/1/21	2459280 3/6/21	300	2459385 6/19/21	35	2459420 7/24/21	238
2022 7	2459658 3/19/22	263	2459921 12/7/22	2459957 1/12/23	2460224 10/6/23	2460588 10/4/24	2460621 11/6/24	700	2460621 11/6/24	0	10	2460631 11/16/24	0	2460631 11/16/24	2460793 4/27/25	2460826 5/30/25	300	2460931 9/12/25	35	2460966 10/17/25	206
2027 2	2461172 5/11/26	263	2461435 1/29/27	2461471 3/6/27	2461738 11/28/27	2462102 11/26/28	2462135 12/29/28	700	2462135 12/29/28	0	10	2462145 1/8/29	0	2462145 1/8/29	2462307 6/19/29	2462340 7/22/29	300	2462445 11/4/29	35	2462480 12/9/29	275
2031 4	2462755 9/10/30	266	2463021 6/3/31	2463057 7/9/31	2463324 4/1/32	2463688 3/31/33	2463721 5/3/33	700	2463721 5/3/33	0	10	2463731 5/13/33	0	2463731 5/13/33	2463893 10/22/33	2463926 11/24/33	300	2464031 3/9/34	35	2464066 4/13/34	294
2035 6	2464360 2/1/35	265	2464625 10/24/35	2464661 11/29/35	2464928 8/22/36	2465292 8/21/37	2465325 9/23/37	700	2465325 9/23/37	0	10	2465335 10/3/37	0	2465335 10/3/37	2465496 3/13/38	2465529 4/15/38	300	2465635 7/30/38	35	2465670 9/3/38	187

Table 5-10 Sequence for the Mars Cargo Freighter

Mars Resupply Freighter Mission Timelines

Vehicle 1

Opp	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	Stay	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	LEO Arrival	Refurb Time
				+0	+169	+276	+400								+0	+30	+394	+400					
2011	2455642	189.8	2455832	2455832	2456001	2456108	2456232	400	2456232	48.4	233	2456512	0.7	2456513	2456513	2456543	2456907	2456913	400	2456913	16.5	2456930	249
	3/21/11		9/27/11	9/27/11	3/14/12	6/29/12	10/31/12		10/31/12			8/7/13		8/8/13	8/8/13	9/7/13	9/6/14	9/12/14		9/12/14		9/29/14	
2016	2457179	184.6	2457363	2457363	2457532	2457639	2457763	400	2457763	48.6	274	2458086	0.8	2458087	2458087	2458117	2458481	2458487	400	2458487	16.5	2458503	285.8
	6/5/15		12/6/15	12/6/15	5/23/16	9/7/16	1/9/17		1/9/17			11/28/17		11/29/17	11/29/17	12/29/17	12/28/18	1/3/19		1/3/19		1/19/19	
2020	2458789	184.6	2458974	2458974	2459143	2459250	2459374	400	2459374	48.4	265	2459687	0.7	2459688	2459688	2459718	2460082	2460088	400	2460088	16.5	2460104	236.9
	11/1/19		5/4/20	5/4/20	10/20/20	2/4/21	6/8/21		6/8/21			4/17/22		4/18/22	4/18/22	5/18/22	5/17/23	5/23/23		5/23/23		6/8/23	
2024	2460341	195.2	2460537	2460537	2460706	2460813	2460937	400	2460937	48.4	229	2461214	0.7	2461215	2461215	2461245	2461609	2461615	400	2461615	16.5	2461631	237.1
	1/31/24		8/14/24	8/14/24	1/30/25	5/17/25	9/18/25		9/18/25			6/22/26		6/23/26	6/23/26	7/23/26	7/22/27	7/28/27		7/28/27		8/13/27	
2029	2461869	185.6	2462054	2462054	2462223	2462330	2462454	400	2462454	48.5	247	2462750	0.7	2462750	2462750	2462780	2463144	2463150	400	2463150	16.5	2463167	275.3
	4/7/28		10/9/28	10/9/28	3/27/29	7/12/29	11/13/29		11/13/29			9/5/30		9/5/30	9/5/30	10/5/30	10/4/31	10/10/31		10/10/31		10/27/31	
2033	2463442	185.5	2463628	2463628	2463797	2463904	2464028	400	2464028	48.5	285	2464362	0.7	2464362	2464362	2464392	2464756	2464762	400	2464762	16.5	2464779	258.7
	7/28/32		1/30/33	1/30/33	7/18/33	11/2/33	3/6/34		3/6/34			2/3/35		2/3/35	2/3/35	3/5/35	3/3/36	3/9/36		3/9/36		3/26/36	
2037	2465037	188.2	2465226	2465226	2465395	2465502	2465626	400	2465626	48.3	241	2465915	0.7	2465915	2465915	2465945	2466309	2466315	400	2466315	16.5	2466332	231.8
	12/9/36		6/16/37	6/16/37	12/2/37	3/19/38	7/21/38		7/21/38			5/6/39		5/6/39	5/6/39	6/5/39	6/3/40	6/9/40		6/9/40		6/26/40	

Mo (mt) = 59.66 Po (kWe) = 600 Mp min (mt) = 12.6 Mp max (mt) = 15.527 Alpha (kg/kWe) = 8

Vehicle 2

Opp	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	Stay	Begin Spiral	Spiral Days	Start IP	Thruster Activity On	Thruster Activity Off	Thruster Activity On	Thruster Activity Off	IP Days	Begin Spiral	Spiral Days	LEO Arrival	Refurb Time
				+0	+169	+276	+400								+0	+30	+394	+400					
2014	2456408	185.6	2456593	2456593	2456762	2456869	2456993	400	2456993	48.5	247	2457289	0.7	2457289	2457289	2457319	2457683	2457689	400	2457689	16.5	2457706	275.3
	4/25/13		10/27/13	10/27/13	4/14/14	7/30/14	12/1/14		12/1/14			9/23/15		9/23/15	9/23/15	10/23/15	10/21/16	10/27/16		10/27/16		11/13/16	
2018	2457981	185.5	2458167	2458167	2458336	2458443	2458567	400	2458567	48.5	285	2458901	0.7	2458901	2458901	2458931	2459295	2459301	400	2459301	16.5	2459318	258.7
	8/15/17		2/17/18	2/17/18	8/5/18	11/20/18	3/24/19		3/24/19			2/21/20		2/21/20	2/21/20	3/22/20	3/21/21	3/27/21		3/27/21		4/13/21	
2022	2459576	188.2	2459765	2459765	2459934	2460041	2460165	400	2460165	48.3	241	2460454	0.7	2460454	2460454	2460484	2460848	2460854	400	2460854	16.5	2460871	231.8
	12/27/21		7/4/22	7/4/22	12/20/22	4/6/23	8/8/23		8/8/23			5/23/24		5/23/24	5/23/24	6/22/24	6/21/25	6/27/25		6/27/25		7/14/25	
2026	2461103	189.8	2461293	2461293	2461462	2461569	2461693	400	2461693	48.4	233	2461973	0.7	2461974	2461974	2462004	2462368	2462374	400	2462374	16.5	2462391	249
	3/3/26		9/9/26	9/9/26	2/25/27	6/12/27	10/14/27		10/14/27			7/20/28		7/21/28	7/21/28	8/20/28	8/19/29	8/25/29		8/25/29		9/11/29	
2031	2462640	184.6	2462824	2462824	2462993	2463100	2463224	400	2463224	48.6	274	2463547	0.8	2463548	2463548	2463578	2463942	2463948	400	2463948	16.5	2463964	285.8
	5/18/30		11/18/30	11/18/30	5/6/31	8/21/31	12/23/31		12/23/31			11/10/32		11/11/32	11/11/32	12/11/32	12/10/33	12/16/33		12/16/33		1/1/34	
2035	2464250	184.6	2464435	2464435	2464604	2464711	2464835	400	2464835	48.4	265	2465148	0.7	2465149	2465149	2465179	2465543	2465549	400	2465549	16.5	2465565	237
	10/14/34		4/17/35	4/17/35	10/3/35	1/18/36	5/21/36		5/21/36			3/30/37		3/31/37	3/31/37	4/30/37	4/29/38	5/5/38		5/5/38		5/21/38	
2039	2465802	195.2	2465998	2465998	2466167	2466274	2466398	400	2466398	48.4	229	2466675	0.7	2466676	2466676	2466706	2467070	2467076	400	2467076	16.5	2467092	237
	1/13/39		7/28/39	7/28/39	1/13/40	4/29/40	8/31/40		8/31/40			6/4/41		6/5/41	6/5/41	7/5/41	7/4/42	7/10/42		7/10/42		7/26/42	

5.8 Applications of Astrotel Orbit Concepts

Applications of Astrotel orbit concepts occur for high Earth orbit (HEO) operations, high Mars orbit (HMO) operations, in design techniques using combinations of hyperbolic rendezvous and low-thrust propulsion for HEDS and robotic exploration missions. HEO could be a departure point to deep space for missions to Mars and asteroids, Lagrange points and for lunar exploration. In addition, there are servicing missions for near-Earth science observatories, e.g. SIRTF. For HMO Operations, one can enter regions of near-Mars space where perturbations can dramatically change satellite motion and/or use low-thrust propulsion to direct perturbations to useful function. Some examples include adding or subtracting orbit energy to return to the Mars surface or to escape for Earth return, performing orbit modification to achieve a desired future position and velocity, e.g. for rendezvous, or for helping to stabilize distant locations for communication and observation spacecraft. New trajectory design techniques using combinations of hyperbolic rendezvous and low-thrust propulsion can be applied to transfer geosynchronous satellites to high "junk yard" orbits and LEO/lunar cargo transfer. Hyperbolic rendezvous and low-thrust concepts can be applied to for robotic exploration missions including Mars sample return, University micro-scout Mars explorers, and asteroid missions.

6 Aero-assist Studies

6.1 Introduction

This section summarizes the aero-assist studies carried out during Phase I and discusses the new work done in Phase II to date.

6.2 Phase I Study Summary

In the Phase I Astrotel Study, aerocapture analysis was performed of the Taxi at Earth and Mars. Though Earth aerocapture was found to drive aeroshell thermal design, Earth aerocapture was found to require lower g-loads than at Mars aerocapture. A Taxi vehicle design was selected that had a raked elliptical cone shaped aeroshell with a maximum L/D of 0.63. This vehicle was sufficient to provide the required centripetal force at the initial horizontal flight part of an aerocapture for all Mars aerocapture opportunities. It was found, however, that at the highest entry speed (12.5 km/s), the Taxi vehicle was required to be at a low altitude to achieve the appropriate level of lift-generated centripetal force in order to remain in the atmosphere. At this level of aero-cruise the total aerodynamic g-load (lift and drag) was found to be about 7 Earth gee, which was assumed excessive for the crew. A number of strategies were analyzed in order to reduce the crew g-loads for these highest entry speed opportunities. These strategies included 1) descending into the atmosphere at a shallow angle and using drag to reduce velocity before reaching aero-cruise altitude, 2) a steeper entry angle to bleed off more speed prior to aero-cruise, 3) the use of a ballute device at entry, 4) the reduction of velocity by propulsive means before entry, 5) propulsive thrusting in the velocity direction (which is very counter-intuitive) during the initial aero-cruise phase, and 6) allowing only experienced and exceptionally fit crews fly those opportunities. Studies showed that propulsive thrusting at about 1 gee in the general velocity direction could reduce this peak initial g-load below 5 gees.

The atmospheric entry of the Mars Shuttle was also analyzed and it was determined that the design originally proposed by the NCOS study was robust enough, given current technology, to allow direct entry into the Martian atmosphere. Direct entry eliminates a modest delta-V required to first circularize the vehicle prior to entry targeting and thus reduces Phobos propellant production requirements and its storage at the Mars Spaceport.

6.3 Taxi Aerocapture Studies

Studies and simulations were carried out to understand the aerocapture maneuver at Mars and at Earth. Figure 6-1 displays a schematic of the Taxi aerocapture maneuver at Mars.

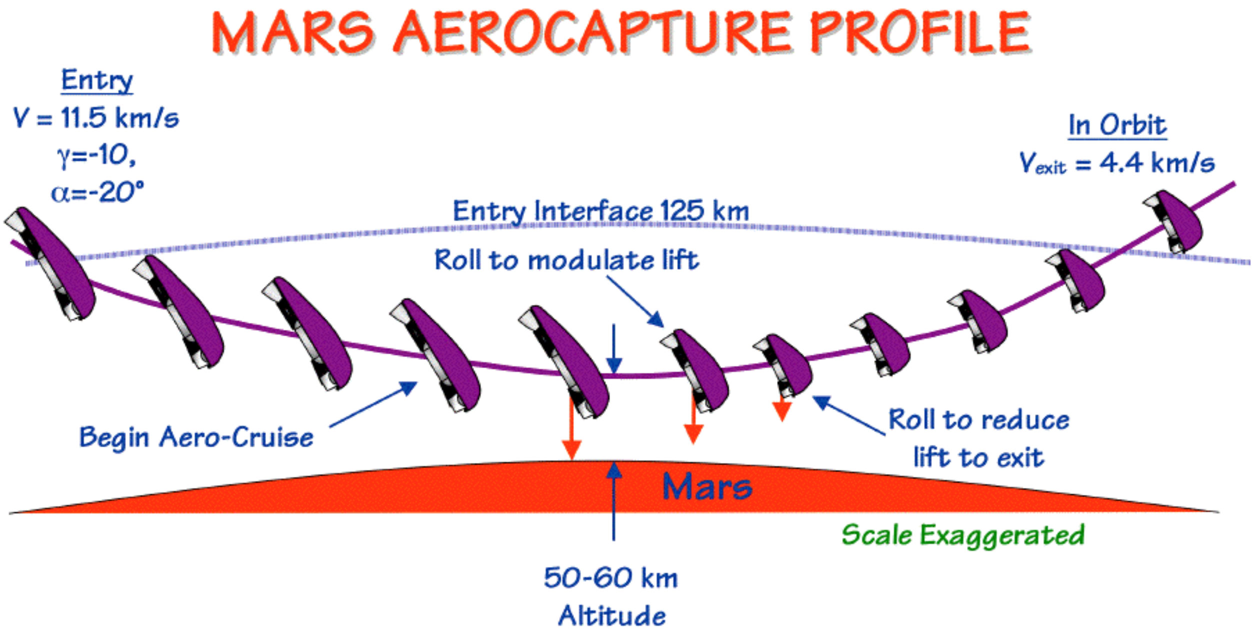

Figure 6-1 Schematic of Aerocapture Maneuver at Mars

6.3.1 Mars Aerocapture Analysis

In Phase II we have determined that level flight for Mars aerocapture will use roll control of the Taxi at fixed angle of attack (AOA). The Mars requirements dictate that the AOA will be set for maximum L/D. The conventional guidance process calls for an initial period of quasi-equilibrium glide to evaluate the error in entry angle, if any, and apparent variation of the atmospheric density and scale height from the values expected. Generally one corrective roll angle position will be used, with a roll reversal if needed to reduce the cross-track deviation to an acceptable level.

Several runs were made of our integrating simulation software in an attempt to develop and understand the technique for generation of aerocapture profiles in order to be able to automate it via the computer model. The main control variable is the Bank Angle that rotates the lift vector to be able to modulate the level of lift relative to the vertical.

The following figures display several parameters from a Taxi aerocapture trajectory simulation (with manual guidance) that results in a velocity that would take the Taxi to the radius of Phobos orbit. The Taxi was assumed to have an entry mass of 22,380 kg and a diameter of 12 m. For this profile the Angle of Attack (AOA) is fixed at –20 degrees for a lift coefficient of 0.60 and a drag coefficient of 0.95. The AOA is the difference between where the axis of symmetry of the aeroshell is pointed relative to the direction of the air flowing past the aeroshell. At zero AOA there is no lift. For particular values of AOA, the lift varies with drag. Angle of Bank (AOB) is defined as the roll angle about the wind or ram direction. Convention is that when AOB is 180° the lift is pointed positive toward the planet. Initially the AOB is set at 180° or full lift down towards Mars in an attempt to offset the centrifugal forces trying to throw the vehicle out of the atmosphere. After reaching the approximate altitude for aero-cruise, the AOB is adjusted to 90° putting all the lift in the horizontal direction, essentially zeroing out the vertical component of lift. After about 30 s, the AOB is moved to about 130° creating enough vertical component of lift toward Mars to offset centrifugal forces. The AOB is adjusted as needed throughout aero-cruise until the required budget of velocity is lost (at ~250 s) where upon the AOB rolls to zero to align the lift vector away from Mars so that the Taxi swiftly leaves the atmosphere on its way to Phobos.

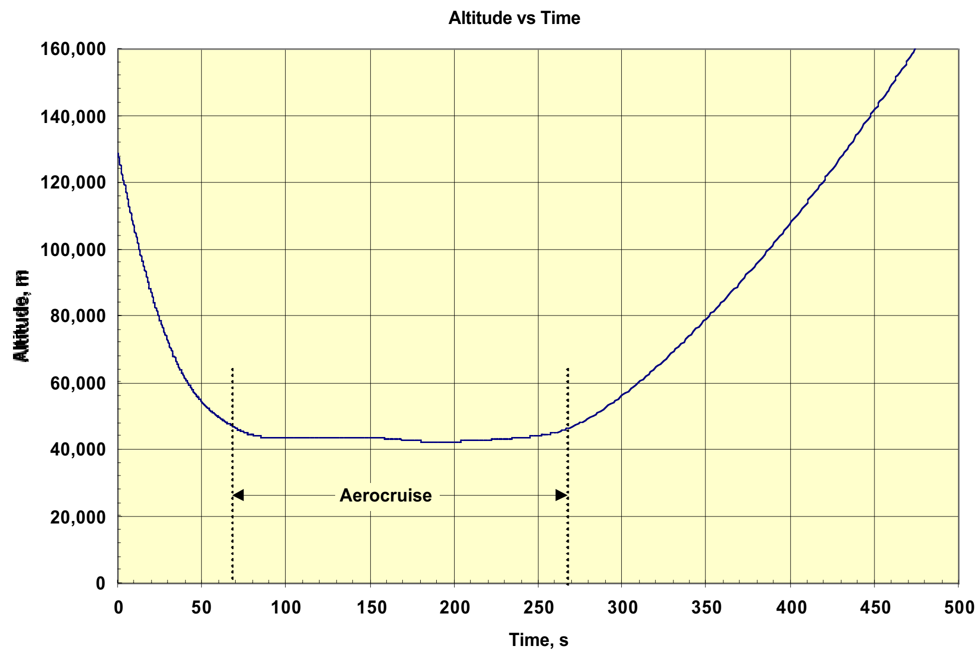

Figure 6-2 Taxi Mars Aerocapture Altitude Profile

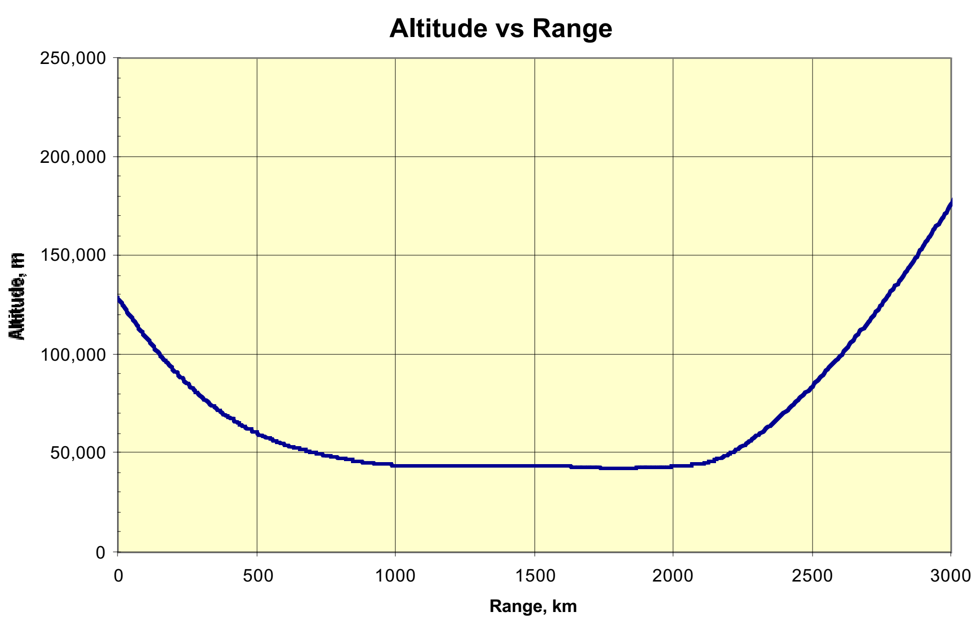

Figure 6-3 Taxi Mars Aerocapture Altitude vs. Range Profile

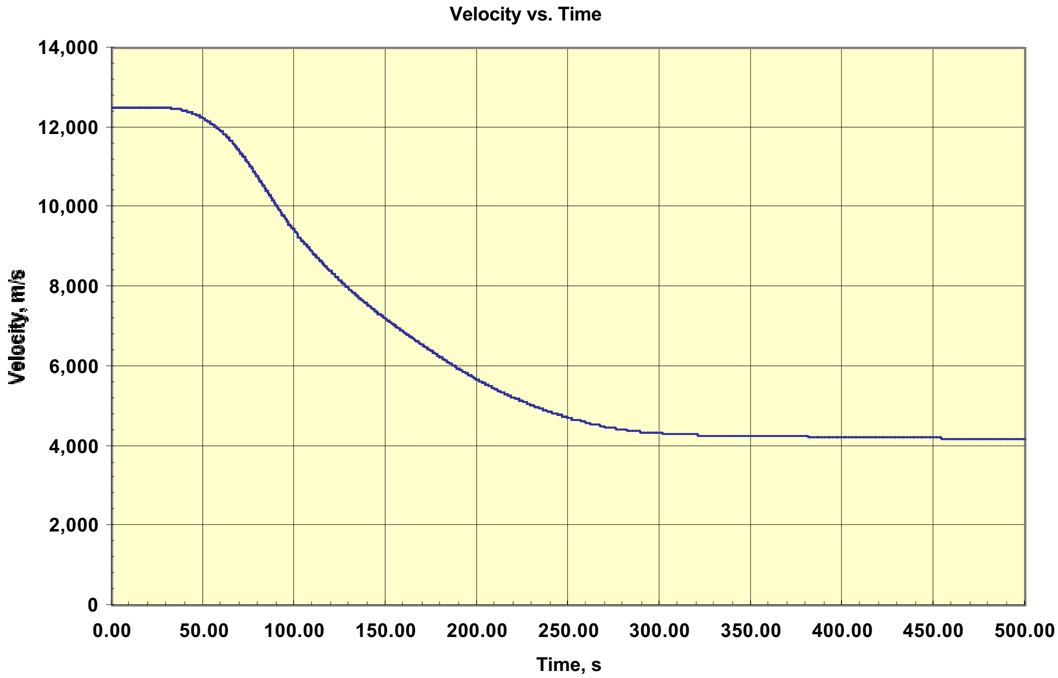

Figure 6-4 Taxi Mars Aerocapture Velocity Profile

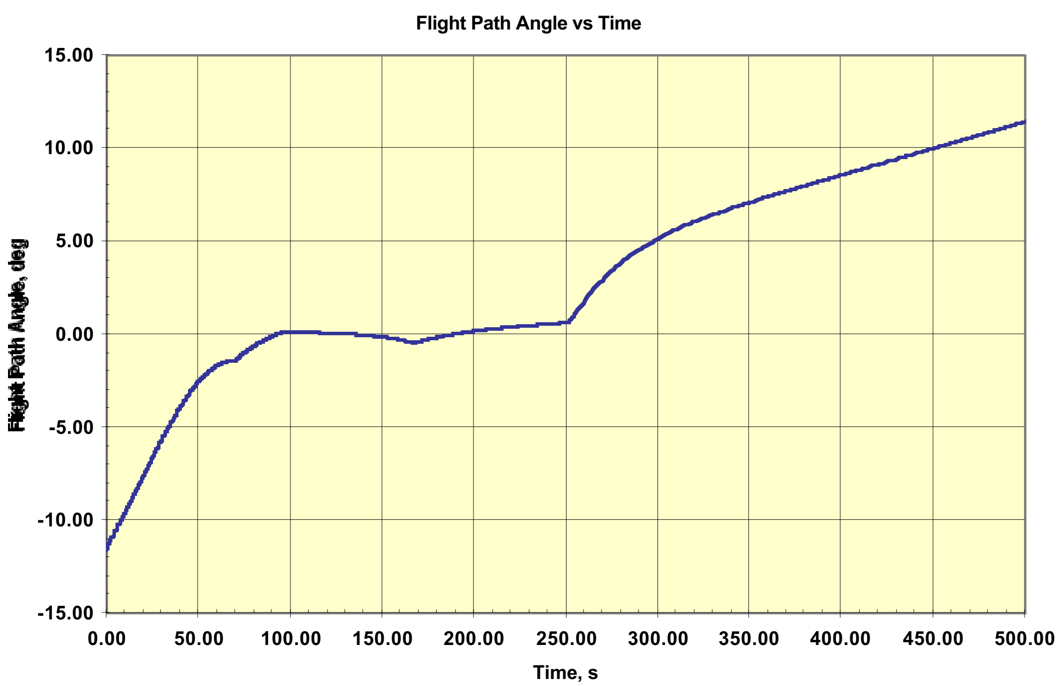

Figure 6-5 Taxi Mars Aerocapture Flight Path Angle Profile

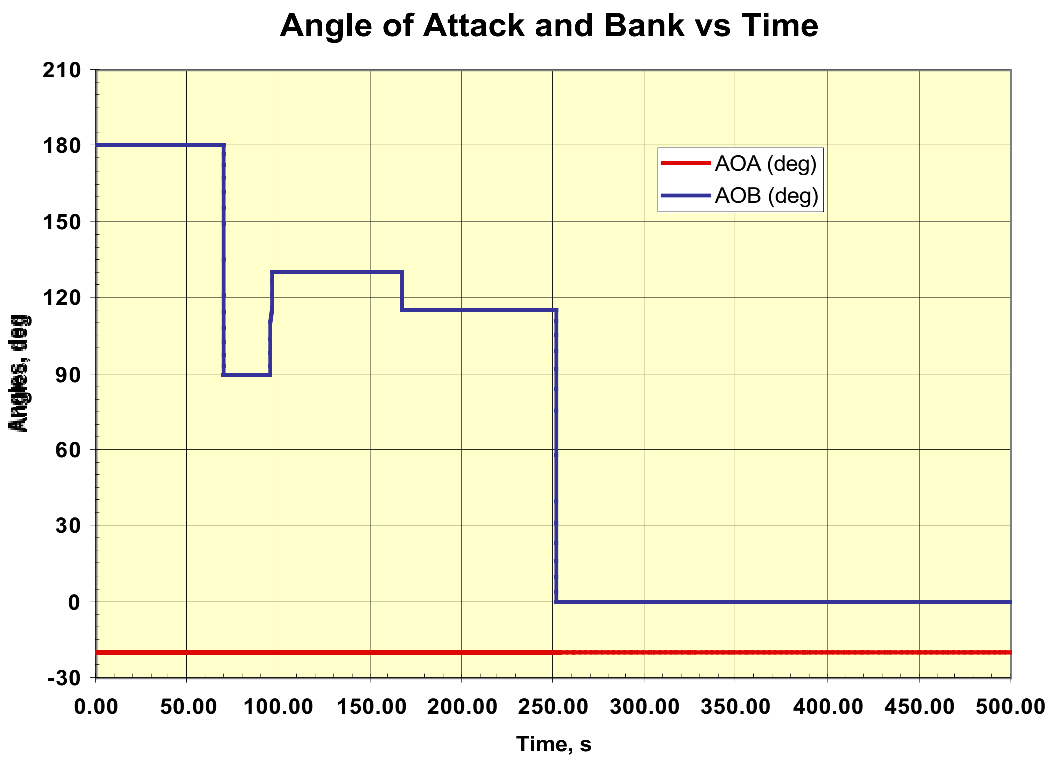

Figure 6-6 Taxi Mars Aerocapture Angle of Attack and Bank Angle Profiles

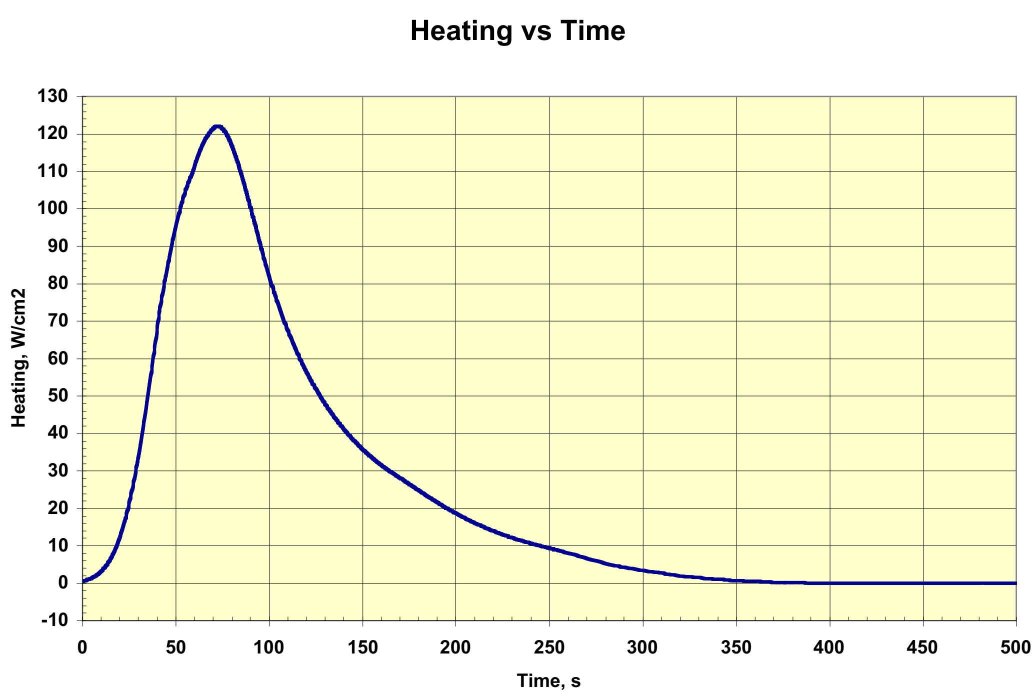

Figure 6-7 Heat Load vs. Time

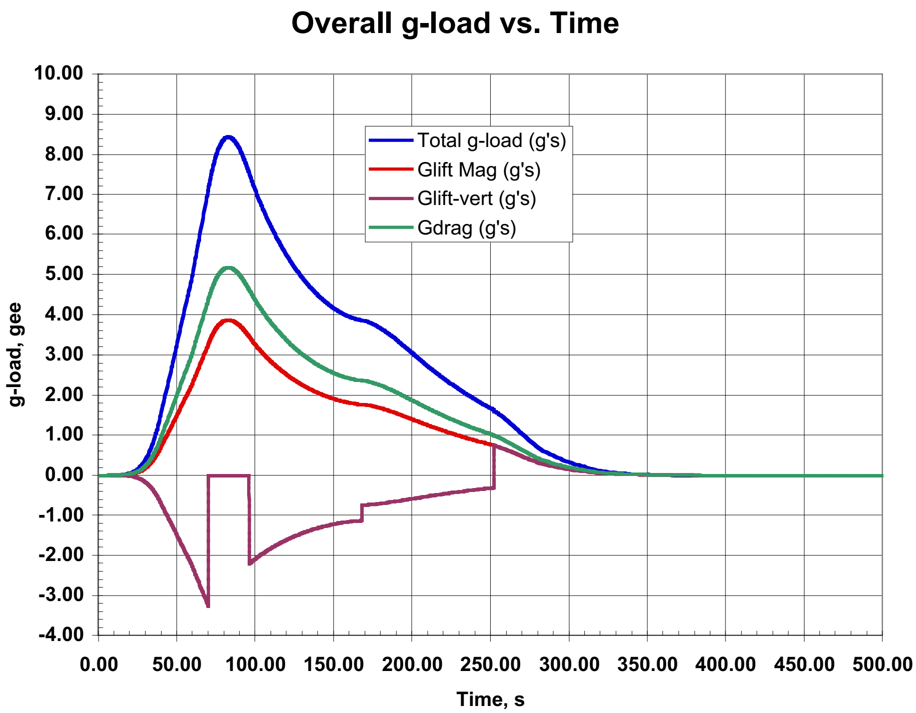

Figure 6-8 Various components of g-load vs. Time

As can be seen in Figure 6-8, overall g-load exceeds the maximum of 5. As discussed earlier, Phase I studies indicated that for the highest entry speed of 12.5 km/s, the Taxi needs to incorporate thrust maneuvers during the aerocapture in order to keep g-loads below 5. In addition the vehicle attitude needs to be trimmed to ensure the maximum lift/drag ratio of 0.63 and it needs to enter the atmosphere initially with full lift down. This orientation of the lift vector will avoid a major attitude change in the short time period before it reaches the level flight altitude, estimated to be about 50 km. As the measured onboard aero deceleration (lift and drag) rises to approach the g-load limit level adopted as 5, the engine is burned at a level of thrust of about 1.5 gee at angle about 45° to the vertical in a direction to reduce the aero drag. When the measured g-load falls below 5 the engine thrust is reduced and soon brought to zero. Thereafter, control of level flight is achieved by rolling the vehicle at fixed AOA to maintain level flight. As the speed reduces the roll angle moves from downward (180°) towards 90° (horizontal lift). At a speed of about 4.5 km/s the taxi applies full lift up (roll angle 0°) and exits the atmosphere on its way up to the orbit radius of Phobos.

A brief analysis was done to assess the possibility of reducing the entry speed at Mars by use of a ballute in the initial low-density portion of the atmospheric pass. It was found, however, that the entry path length, the short entry time, and the limit of 5 g-load effectively limit the delta-V that can be achieved by a ballute to about 1000-1500 m/s for any reasonable size and mass of ballute system.

The next set of charts displays aerocapture parameters for a lower entry speed for which the maximum g-load is 5.

6.3.2 Earth Aerocapture Analysis

At Earth, the delta-V to be lost in aerocapture is relatively small (the entry speed is about 11.5 km/s and the desired exit speed to go to an apoapsis at the orbit radius of the Moon is about 10.8 km/s). Preliminary studies indicate that the full lift of the taxi is not required, because there will be an entry corridor of several degrees (in the range about 4° to 8°). It may be appropriate to maintain the AOA at the Mars value (for maximum L/D) but to enter with a substantial upward lift, e.g., a roll angle of 45°, and adjust roll angle as the onboard measurements determine the actual entry angle. A target altitude for level flight at Earth is about 80 km. At the entry speed of 11.5 km/s, a downward lift acceleration of about 1 gee added to gravity will maintain level flight. Since the period of aero-cruise in the atmosphere is much shorter at Earth than at Mars, the Taxi will simply execute the conventional guidance process that is a quasi-equilibrium glide at initial entry and a computed roll angle for the exit portion of the atmospheric pass.

6.4 Mars Shuttle Entry Analysis

In Phase II the Mars Shuttle vehicle has been studied, assuming a general Viking/Pathfinder shape that has a peak lift-to-drag (L/D) ratio of about ±0.3. The advantages of this shape are significant space and volume for crew, fuels and cargoes and the existence of significant technological heritage. The Mars Shuttle deploys a large surface for entry that folds up against the vehicle for ascent to provide a smaller cross-section and reduced drag.

6.4.1 Entry Orbit

Entry profiles were evaluated for atmospheric entry from a 500-km circular orbit and also for direct entry from Phobos orbit radius (POR) for which the entry speeds are about 3.6 km/s and 4.2 km/s, respectively. We analyzed the expected accuracy of a de-orbit from POR (about 500 m/s) and determined that the resultant entry angle accuracy would be about 0.2 deg, an acceptable value. Figure 6-9 displays the entry angle as a function of the vacuum periapsis altitude. Direct de-orbit, as opposed to going into an intermediate orbit, results in a savings of about 500 m/s. Figure 6-10 shows this de-orbit delta-V at POR as a function of vacuum periapsis altitude.

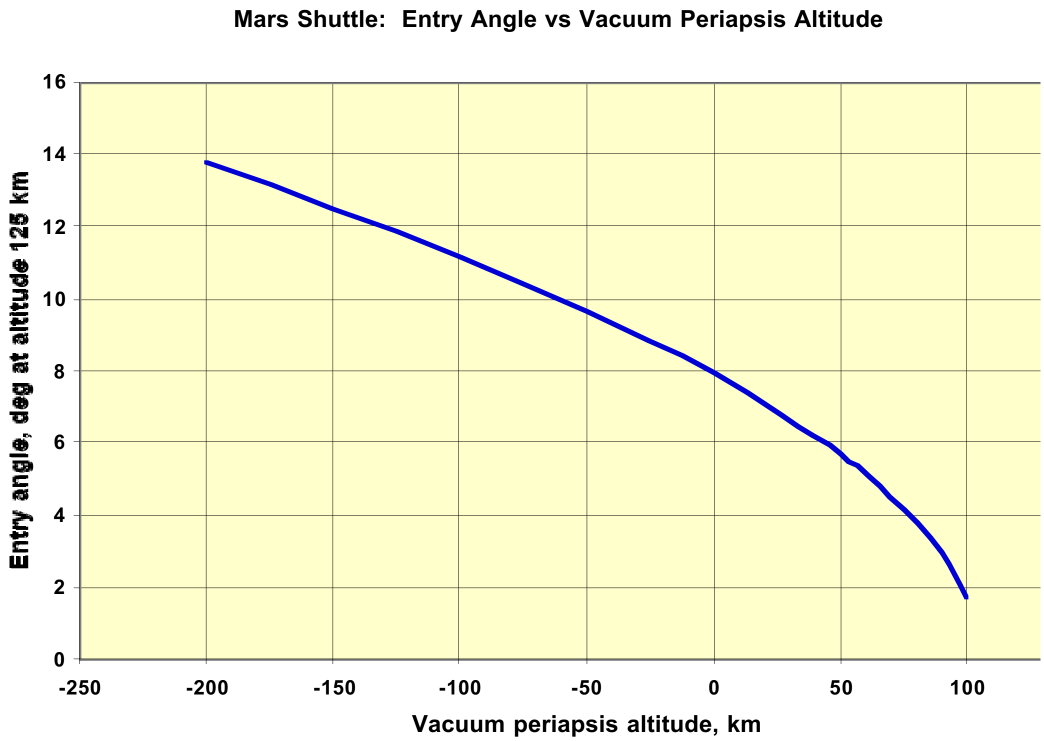

Figure 6-9 Entry Angle vs. Vacuum Periapsis Altitude for Entry from Phobos Orbit Radius

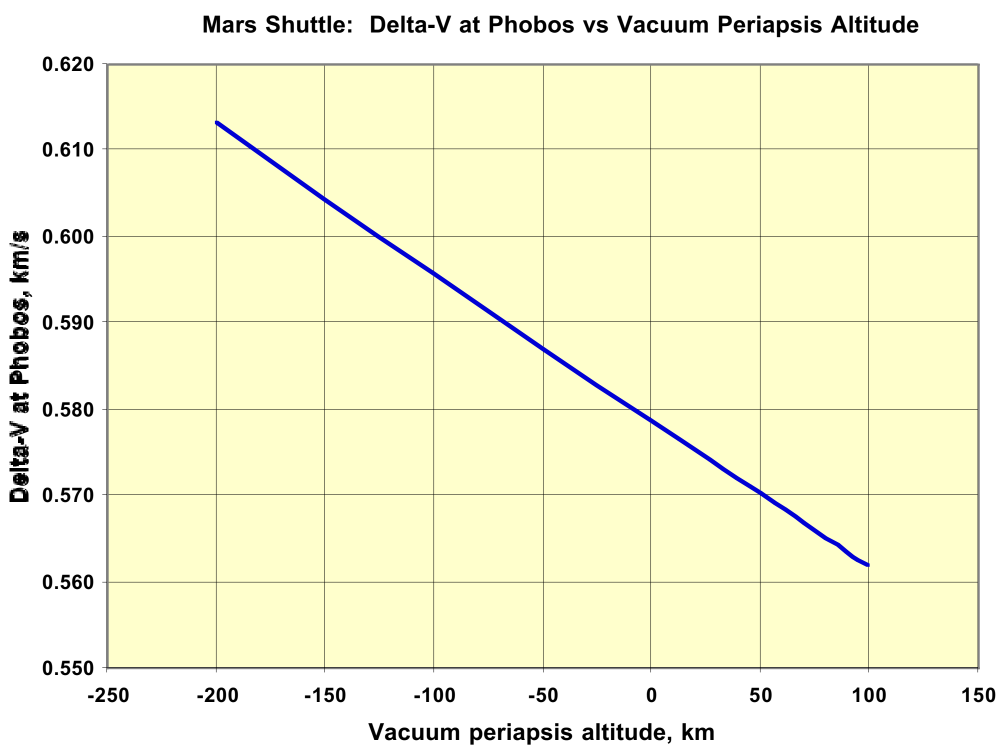

Figure 6-10 Delta-V at Phobos Orbit Radius vs. Vacuum Periapsis Altitude

6.4.2 Lift-to-Drag Analysis

Mars Shuttle trajectories were evaluated with constant L/D values of ±0.3 and 0 at entry angles between 7 and 11 deg. The L/D = -0.3 cases generally give excessively high peak g-loads during entry, which means that the Mars Shuttle will likely employ a fixed L/D somewhere in the range 0 to +0.3. The following figures display a number of entry parameters as a function of L/D and entry angle, from entry to the point where the speed has dropped to 500 m/s, including altitude at Mach 2; maximum aerodynamic g-load; range; and peak reference stagnation heating rate (for a 1-m diameter reference sphere).

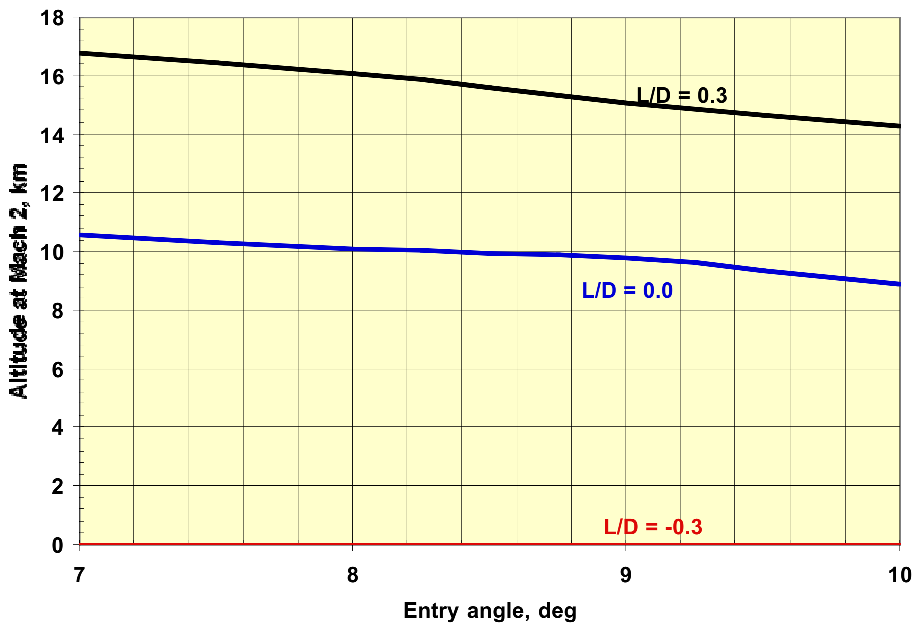

Figure 6-11 Altitude at Mach 2 as a Function of Entry Angle

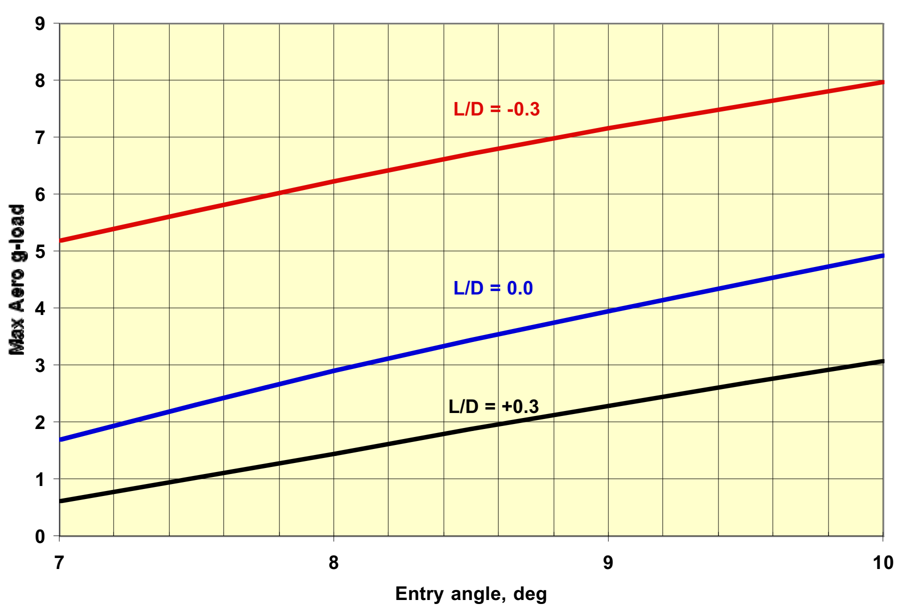

Figure 6-12 Maximum Aerodynamic g-load as a Function of Entry Angle

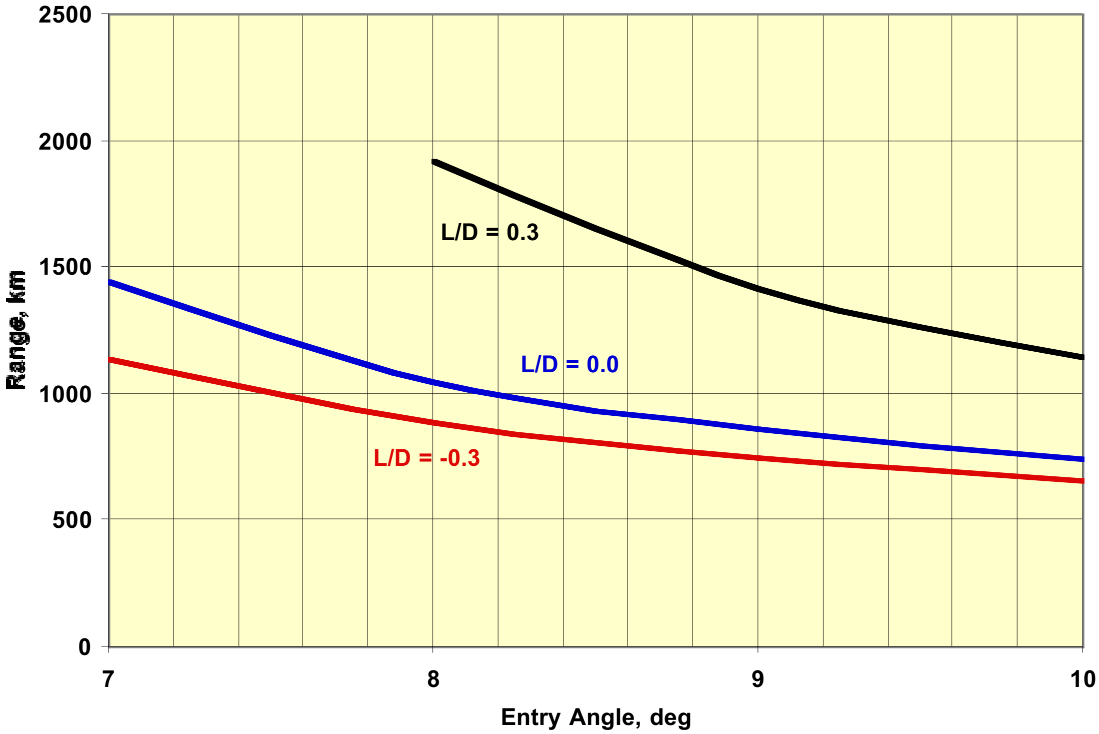

Figure 6-13 Range as a Function of Entry Angle

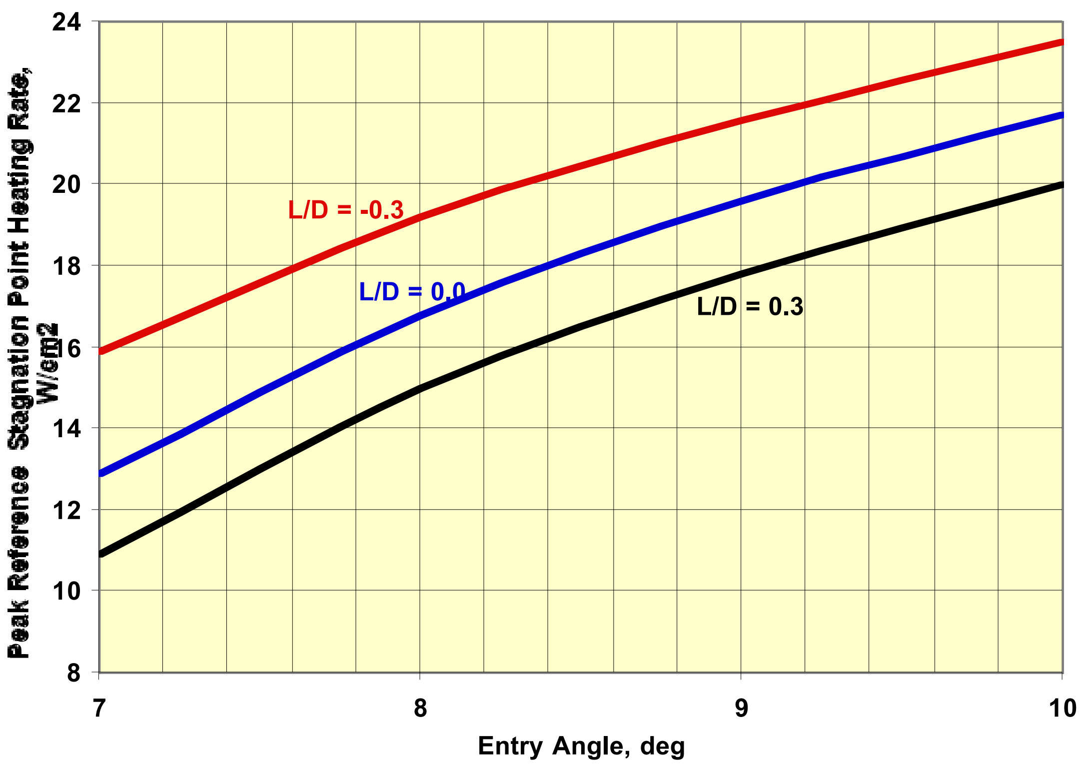

Figure 6-14 Peak Reference Stagnation Point Heating as a Function of Entry Angle

6.4.3 Entry Profile

A typical entry profile is displayed in the following figures for L/D = 0, ballistic coefficient of 100 kg/m^2 and nominal entry angle of 8° at an entry interface altitude 125 km. Entry trajectories for a ballistic coefficient in the range B = 30 to 120 kg/m^2 (B = m/C_dA, where m is the mass, C_d is the drag coefficient, and A is the cross sectional area) were examined. A nominal B = 100 kg/m^2 (the current design is at B = 70 kg/m^2) was chosen for evaluation of the peak entry heating rate and calculating the required delta-V for landing. The peak-heating rate for a vehicle with nose radius 9 m (about the maximum for this case) is about 3 W/cm^2. This peak heating value can be radiated at a surface temperature of about 900 K, at an emissivity of 0.8, or by use of an ablating low-temperature thermal protection system (TPS). Targeting to a surface point will be achieved by using the customary roll angle control modulation to adjust the vertical component of lift. The range variation attainable is considerably more than is needed to compensate for entry angle errors and expected atmospheric density variations.

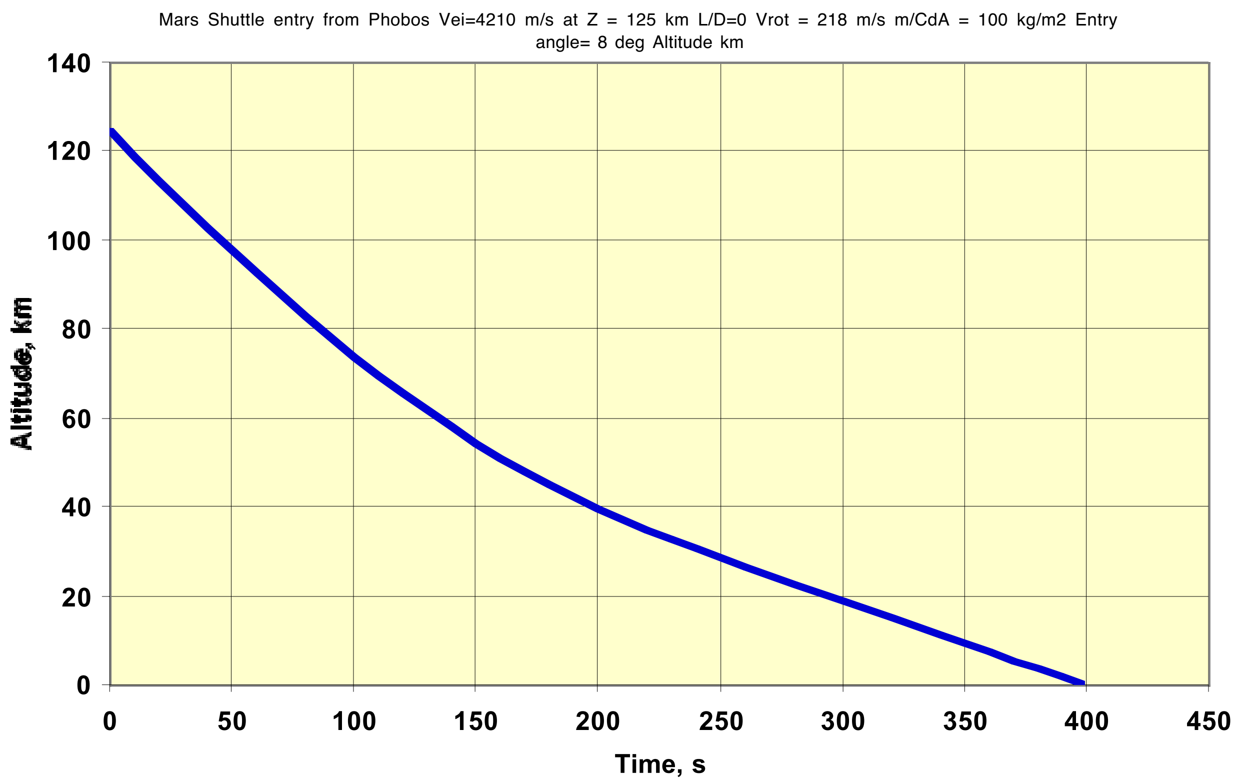

Figure 6-15 Mars Shuttle Entry: Altitude vs. Time

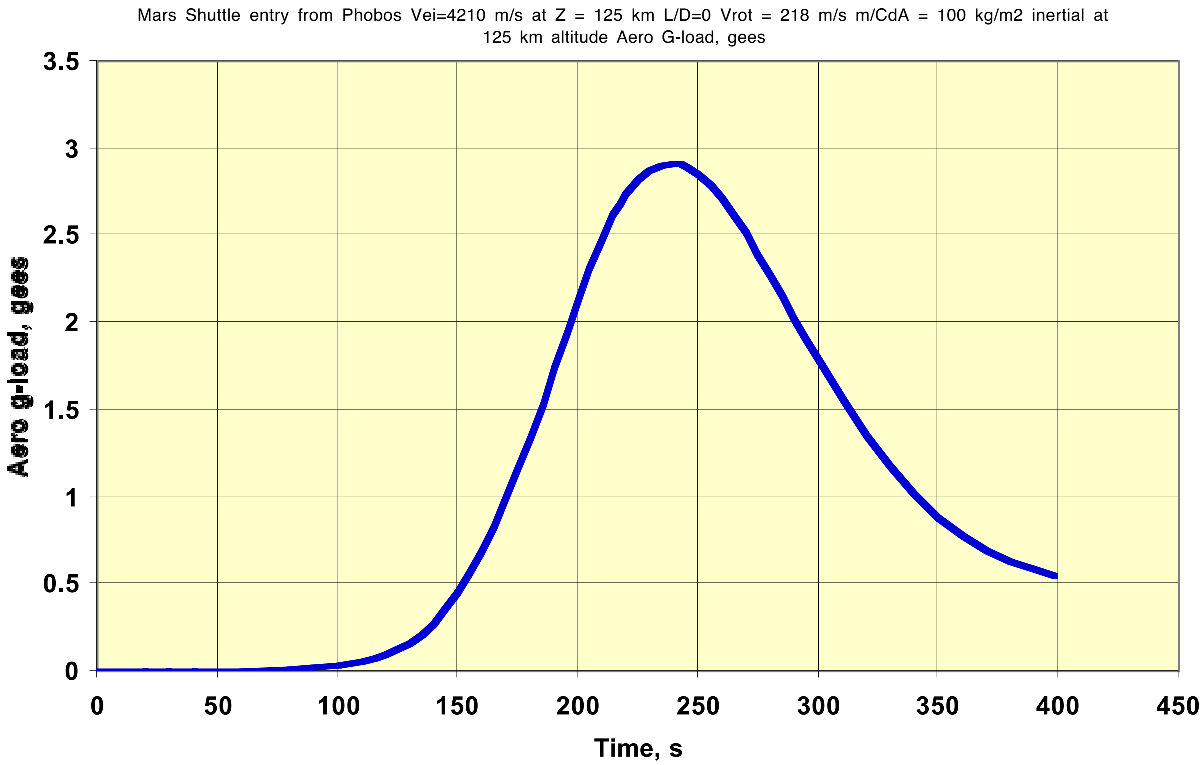

Figure 6-16 Mars Shuttle Entry: Aerodynamic g-load vs. Time

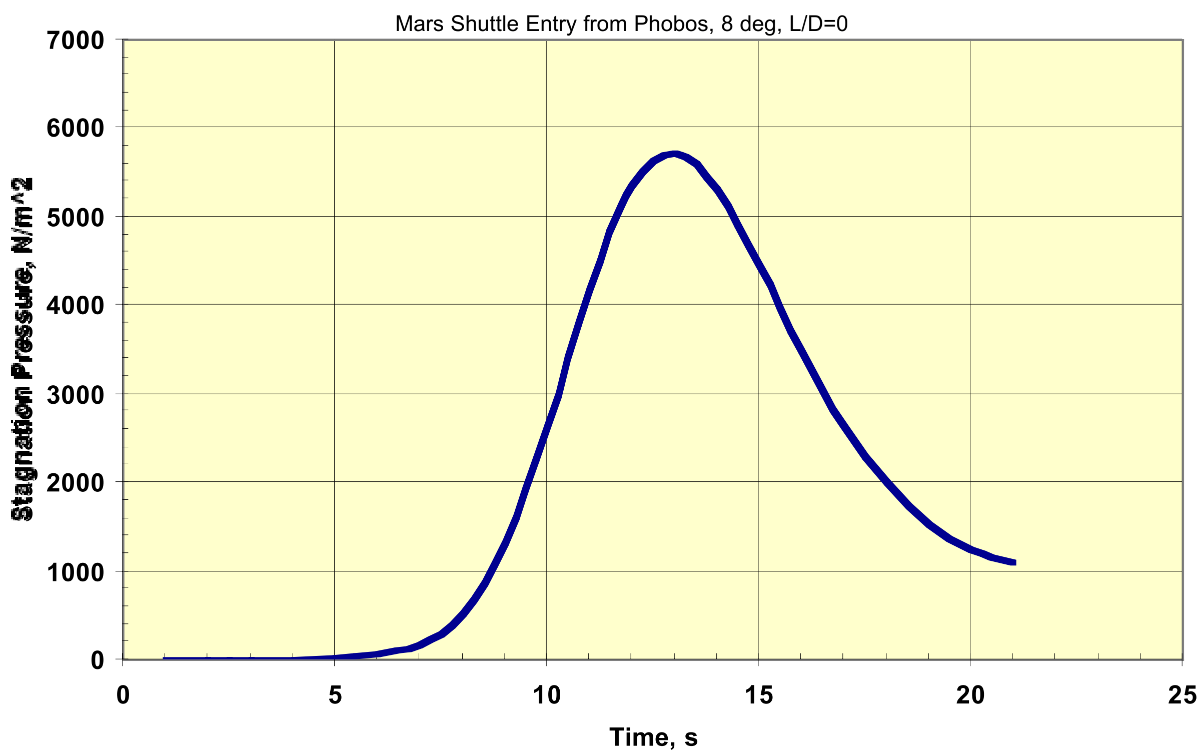

Figure 6-17 Mars Shuttle Entry: Stagnation Pressure vs. Time

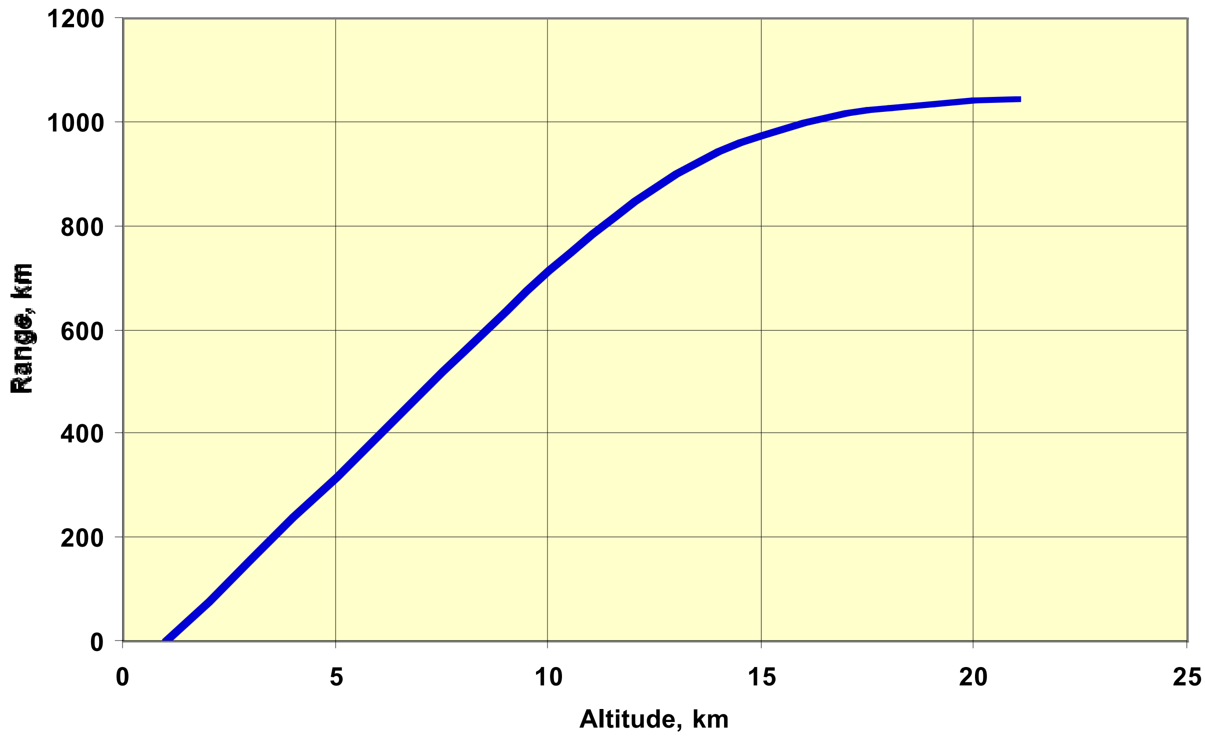

Figure 6-18 Mars Shuttle Entry: Range vs. Altitude

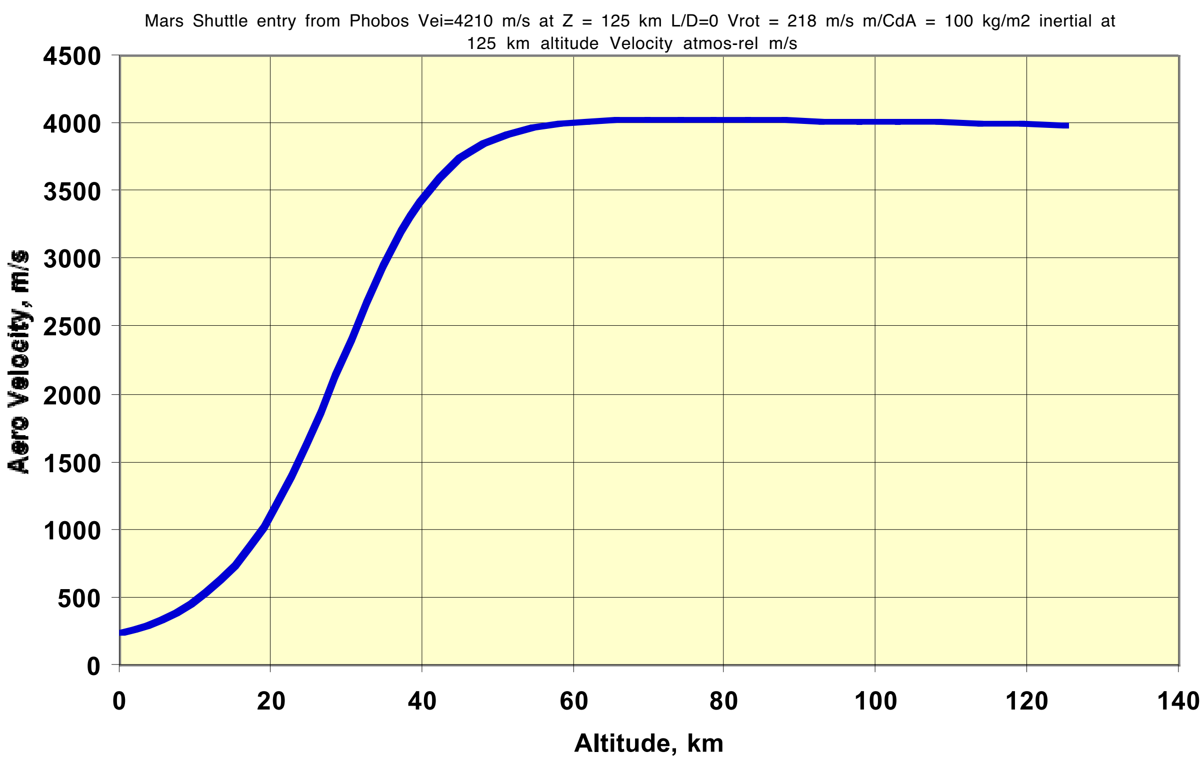

Figure 6-19 Mars Shuttle Entry: Velocity vs. Altitude

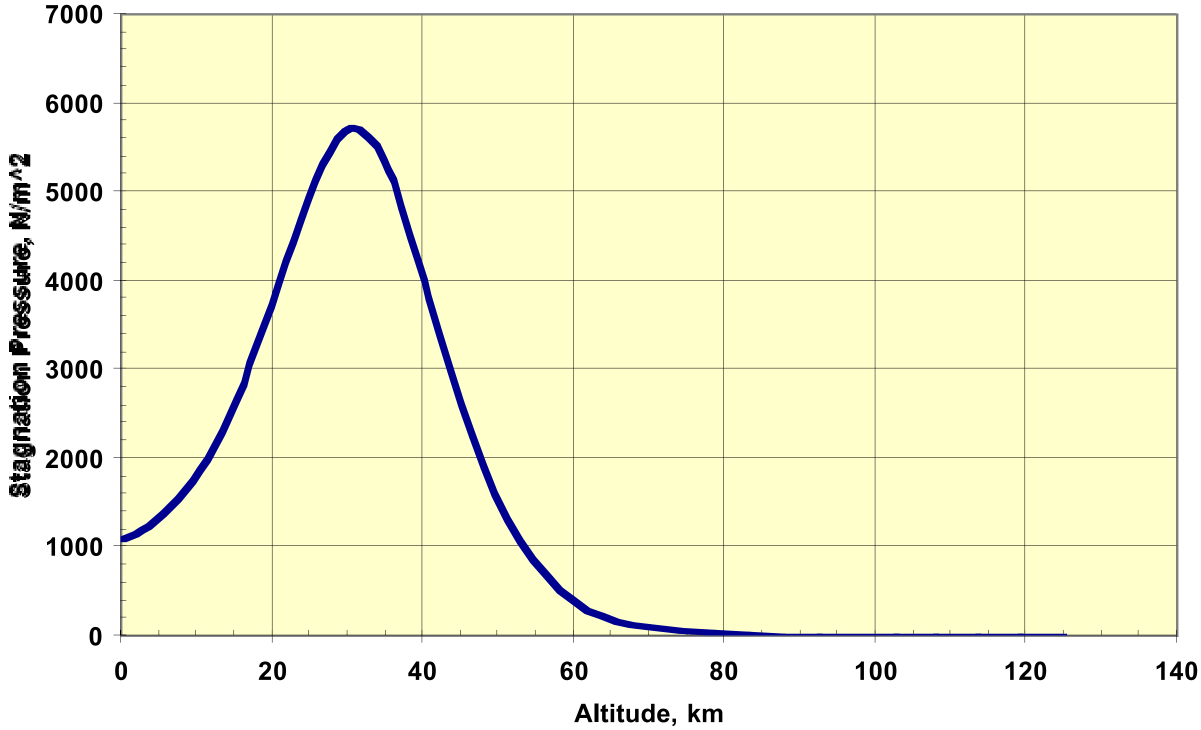

Figure 6-20 Mars Shuttle Entry: Stagnation Pressure vs. Altitude

6.4.4 Landing Deceleration Analysis

Landing deceleration was analyzed assuming a L/D = 0, ballistic coefficient of 100 kg/m^2 and nominal entry angle of 8° at an entry interface altitude 125 km. For this analysis, a decelerating engine burn thrust of a steady 1.5 Earth gees was applied starting at different altitudes from 10 to 6 km in the descent phase. The required delta-V decreased as the burn began at lower altitudes. For the 6-km burn start case, the vehicle essentially was decelerated to zero velocity in about 2 km, which consumed about 300 m/s of propellant. In the following figures, the speed and altitude are normalized such that at zero altitude or time from landing the speed is zero. Figure 6-21 shows the delta-V to decelerate with 1.5 gee, beginning at altitudes of 10, 8 and 6 km versus the entry angle. Figure 6-22 shows altitude as a function of time for a landing deceleration of 1.5 gee beginning at altitude 6 km. Figure 6-23 shows the atmosphere relative speed as a function of altitude for a deceleration of 1.5 gee beginning at altitude 6 km. Figure 6-24 shows the decelerating g-load as a function of time from landing for 1.5 gee beginning at altitude 6 km. It is probable that in reality the latter part of the decelerating burn and the burns to hover and fly to the target and to counteract wind would be combined and would use a variable thrust level. In any event, an allocation of about 300 m/s for deceleration is considerably less than the 1000 m/s assumed in Phase I. These results mean that the number of "RL60-class" rocket engines needs to be 3 in order to have a reserve engine in the event of a one-engine-out situation during descent from orbit to the surface of Mars.

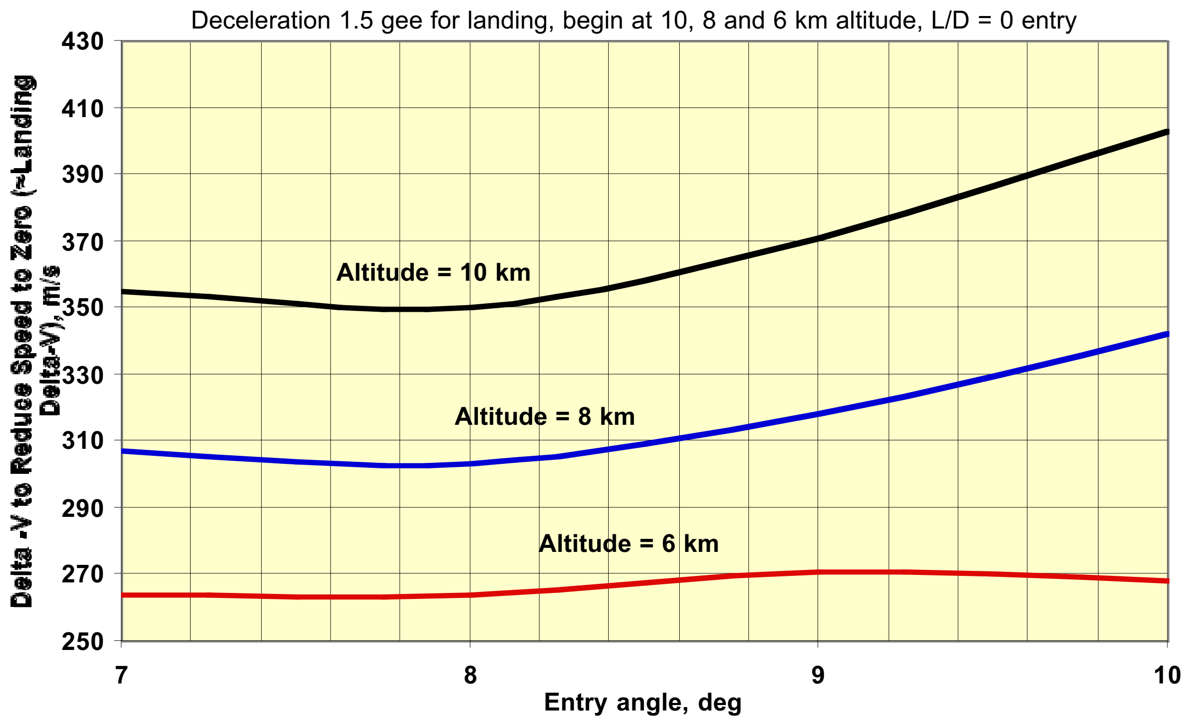

Figure 6-21 Landing Delta-V vs. Entry Angle and Burn Start Altitude

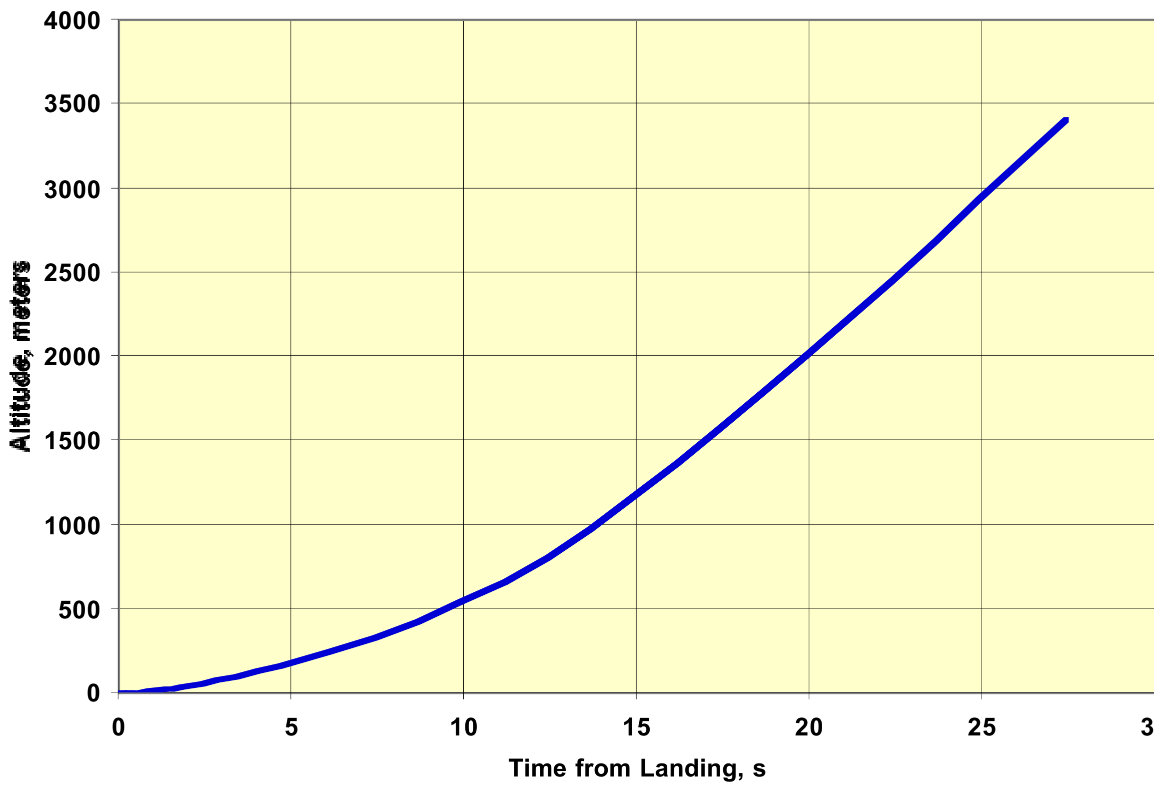

Figure 6-22 Altitude vs. Time for Entry Angle 8°, Deceleration of 1.5 gee, Beginning at 6 km.

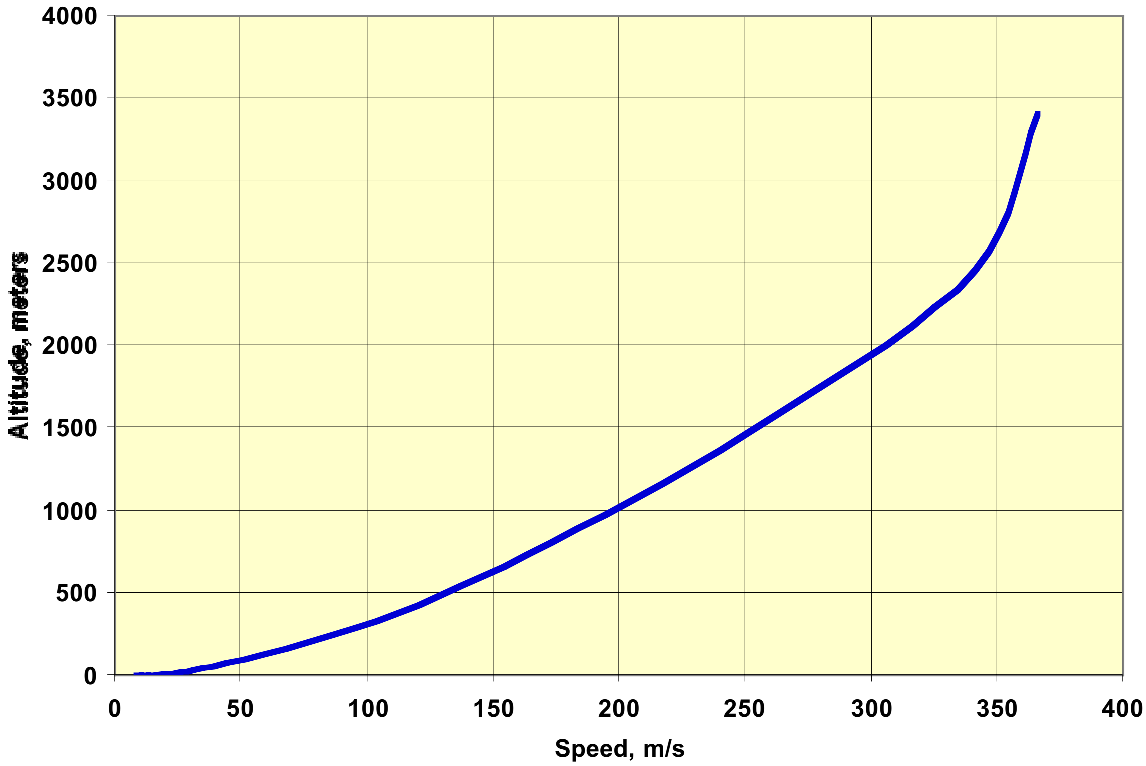

Figure 6-23 Altitude vs. Atmosphere Relative Speed for Deceleration of 1.5 gee, Beginning at 6 km, Entry Angle 8

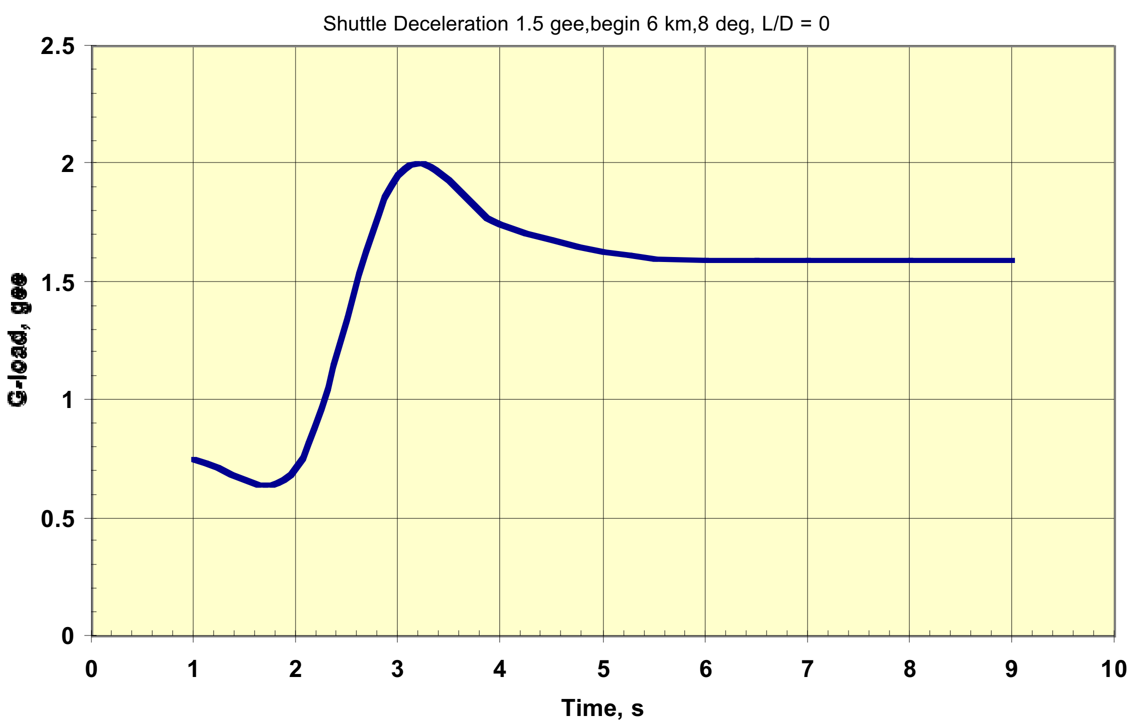

Figure 6-24 Deceleration g-load vs. Time from Landing for 1.5 Gee, 8° Entry Angle, Beginning at Altitude 6 km

6.5 Mars Shuttle Ascent Analysis

The Mars Shuttle ascent problem was first simulated with a COSPAR mean atmosphere model and a nominal vehicle with subsonic drag coefficient C_d of 0.7. The simulation software provides for 3 burns, with thrust along the V-vector, and gravity turning. The first two burns were coalesced, since there is no staging, and the burns stop when it is computed that the vehicle apoapsis will reach a predetermined altitude, e.g. 125 km. The 3rd burn is brought on at some point and ceases when the vehicle reaches the desired apoapsis. The variable parameter that has most effect on the trajectory is the ascent path angle early in the ascent. If the path is too steep, the vehicle fights gravity for too long, and the burn stops with the speed low and the path steep. If the path is too shallow the vehicle fights aerodynamic drag for too long and fails to ascend to orbit. A useful metric in evaluating the performance of a particular ascent profile is to evaluate the difference between the ideal velocity associated with the mass expulsed in reaching a circular orbit (calculated using the rocket equation or $m_f = m_i e^{-\Delta V/gIsp}$) and the actual velocity attained in orbit. To gain insight into the dominant parameters this difference in velocity was analyzed as a function of the thrust/mass ratio at launch, timing and length of coast periods and the parameter m/C_dA where m is the vehicle mass, C_d is the drag coefficient and A is the area. This velocity difference or "loss" is caused by the (1) effects of the atmosphere (drag loss), (2) misalignment between the velocity and thrust vectors during the burns (steering loss) and (3) burns not being instantaneous impulses and thus occurring over a range of planet radii (gravity loss). Gravity losses result from a burn occurring over time and a wide variation in planet radius since a velocity change is much more effective (increases orbit energy more) if executed as close to the center of the planet as possible.

In the analysis carried out, a RL60-class rocket engine was assumed and a number of ascent variables explored. These variables included initial thrust acceleration in the range about 0.8 to 2.0 Earth gee, Ballistic Coefficient, m/C_dA (m = mass, kg, C_d is average drag coefficient, and A is the cross-sectional area, m^2) from 60 to 1200 kg/m^2, and initial flight path angle from 88° to 80°. For each initial flight path angle the acceleration was varied until the total delta-V to reach a circular orbit at the radius of Phobos was determined, and the minimum Delta-V was selected. Optimum delta-V was calculated as a function of ballistic coefficient, m/C_dA. Graphs of Mach number and acceleration were also plotted versus time and altitude and shown below.

A delta-V of about 5.15 km/s results from a C_d of 0.9 (hemispherical nose cap), which could be reduced about 50 to 150 m/s if a sharper nose could be used, a ballistic coefficient of 784 kg/m^2 assuming a 10-m diameter vehicle at launch with the aeroshell stowed, and a 55,423 kg launch mass. The Mach number plot indicates that the vehicle quickly goes supersonic, so that the governing C_d is for supersonic and hypersonic flight. The following chart displays delta-V versus the ballistic coefficient of the vehicle. As with the descent analysis, these results mean that the number of RL60-class" rocket engines needs to be 3 in order to have a reserve engine in the event of a one-engine-out situation during launch and ascent from the surface of Mars. The mass ratio from launch to final orbit at Phobos altitude was computed with instantaneous burns to raise the apoapsis to that of Phobos and then to circularize. First results indicate a final mass/initial mass ratio of about 0.33.

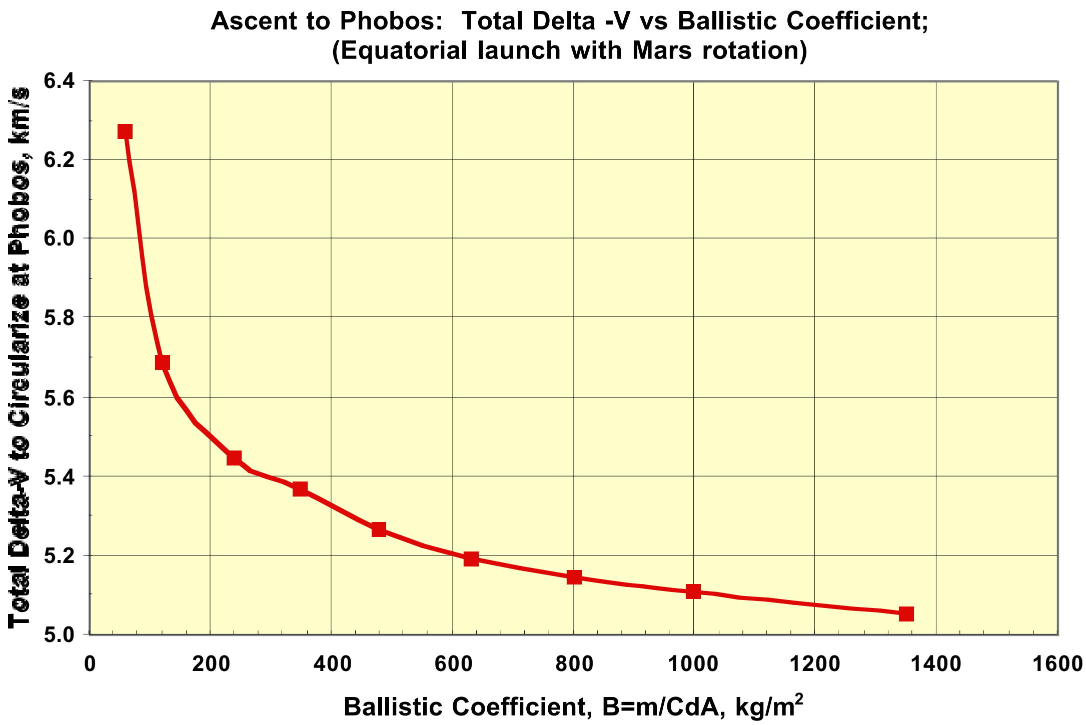

Figure 6-25 Delta-V versus Ballistic Coefficient

The next figure displays the Mars Shuttle altitude versus time for the launch to Phobos.

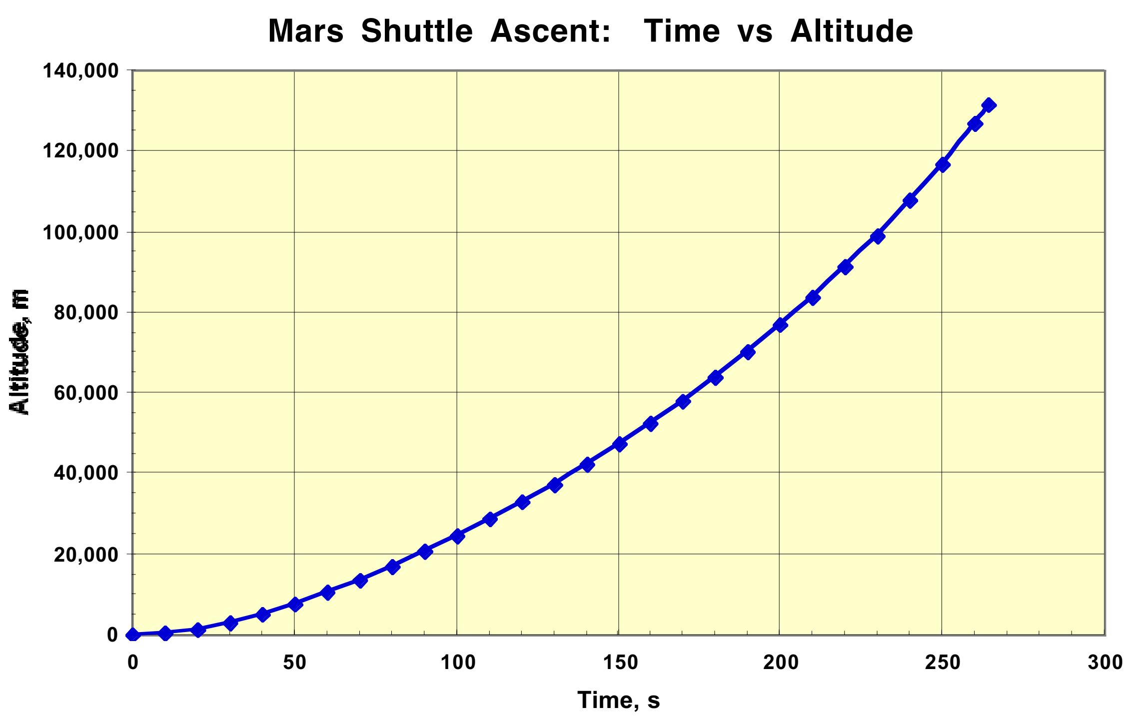

Figure 6-26 Ascent Time versus Altitude

The next charts show the Mars Shuttle velocity as a function of time.

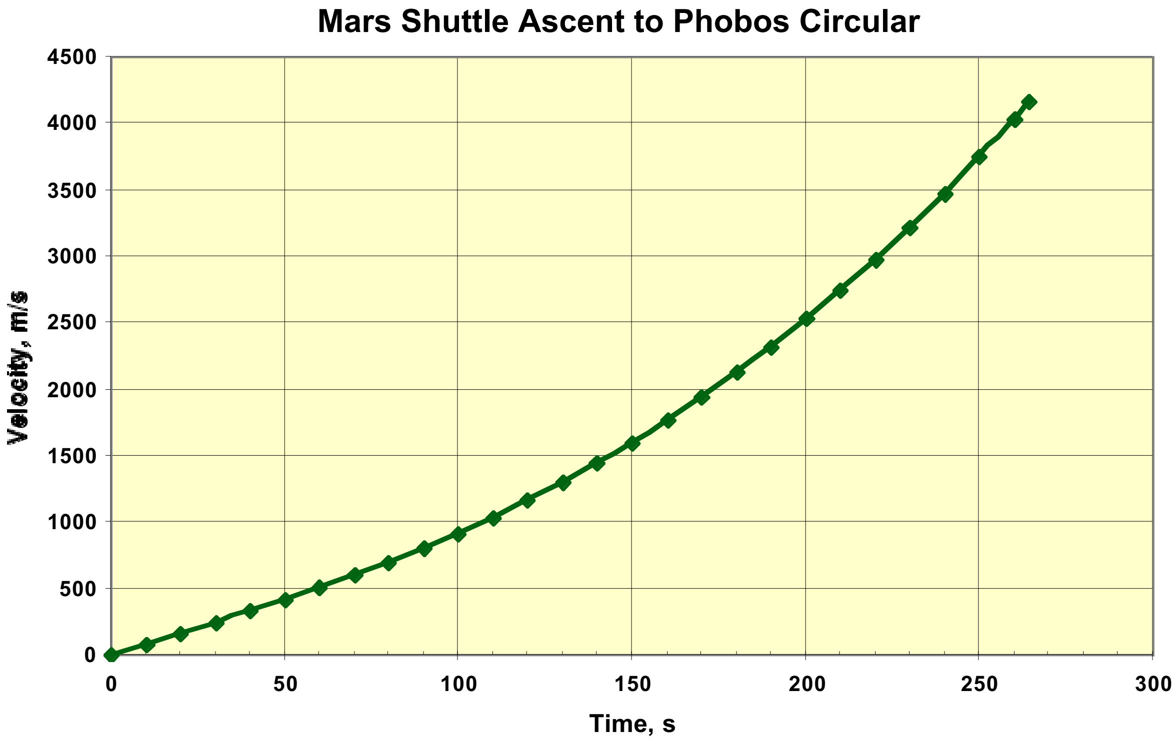

Figure 6-27 Ascent Velocity versus Time

The g-load as a function of time for the launch profile is shown in the next chart. Note the peak g-load is at the end of the burn.

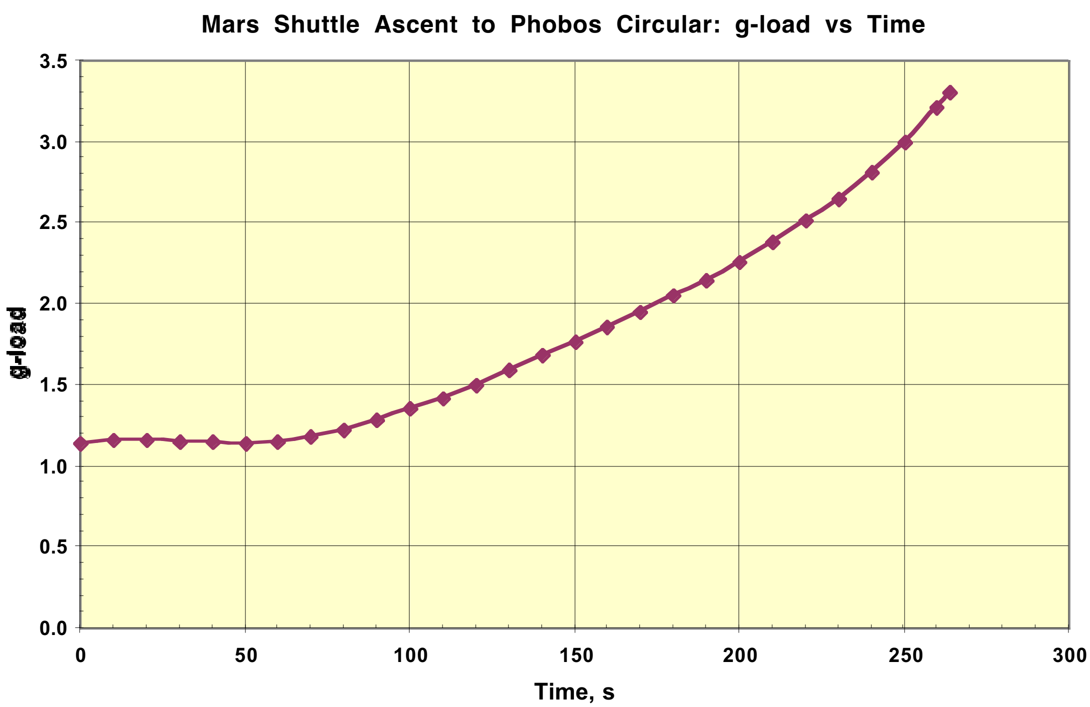

Figure 6-28 Ascent g-load versus Time

The next chart displays the flight path angle as a function of time throughout the ascent phase.

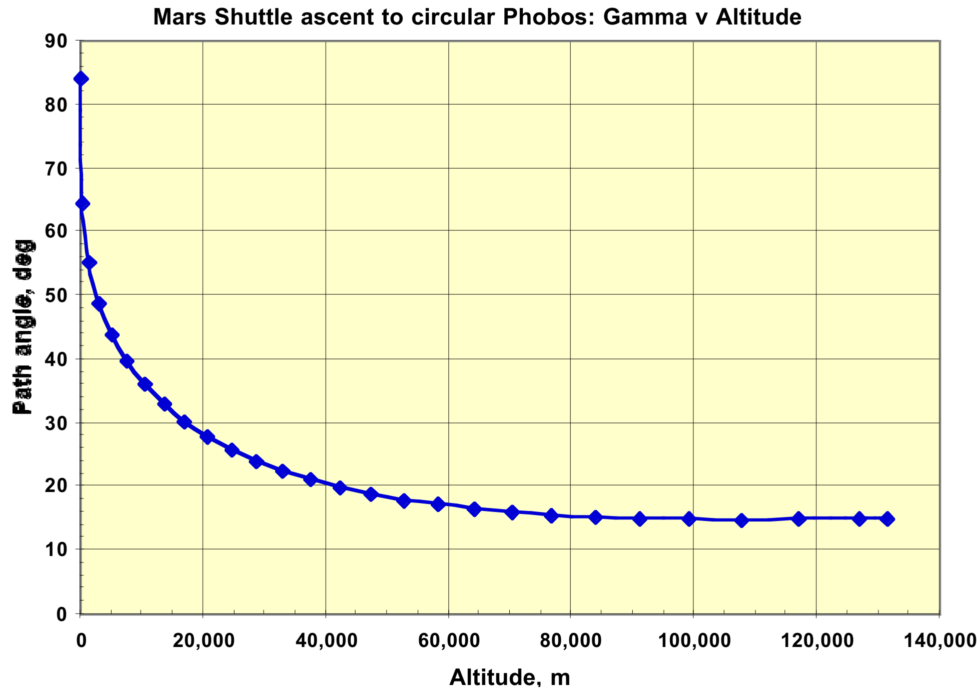

Figure 6-29 Ascent Flight Path Angle versus Time

The following figure charts the dynamic pressure as a function of time. Note that maximum dynamic pressure (Max Q) is about 70 s after launch.

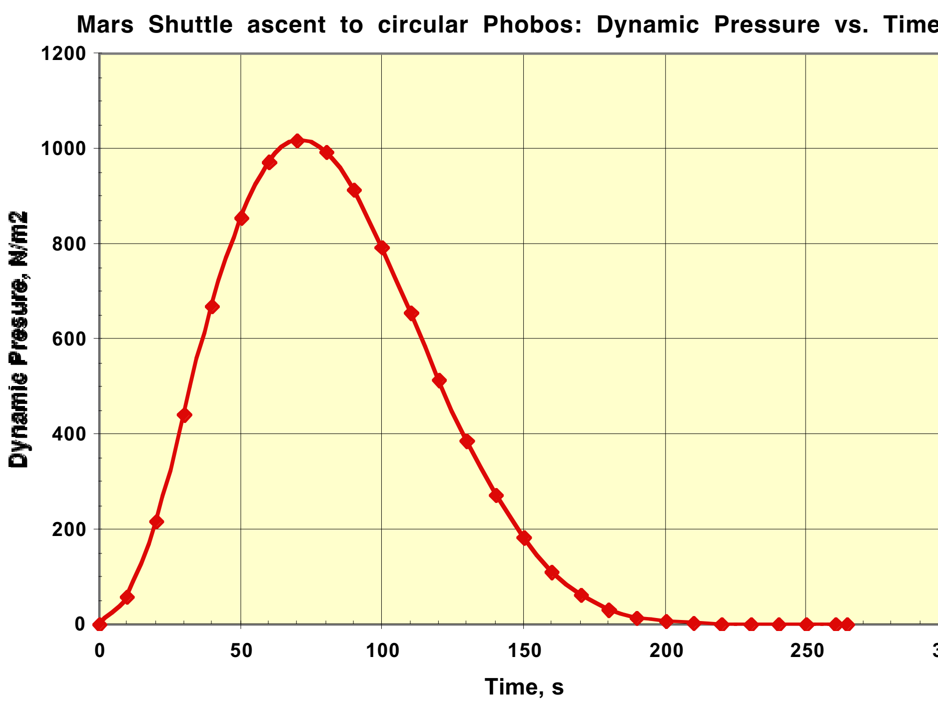

Figure 6-30 Ascent Dynamic Pressure versus Time

Finally the next chart displays the Mach number versus time.

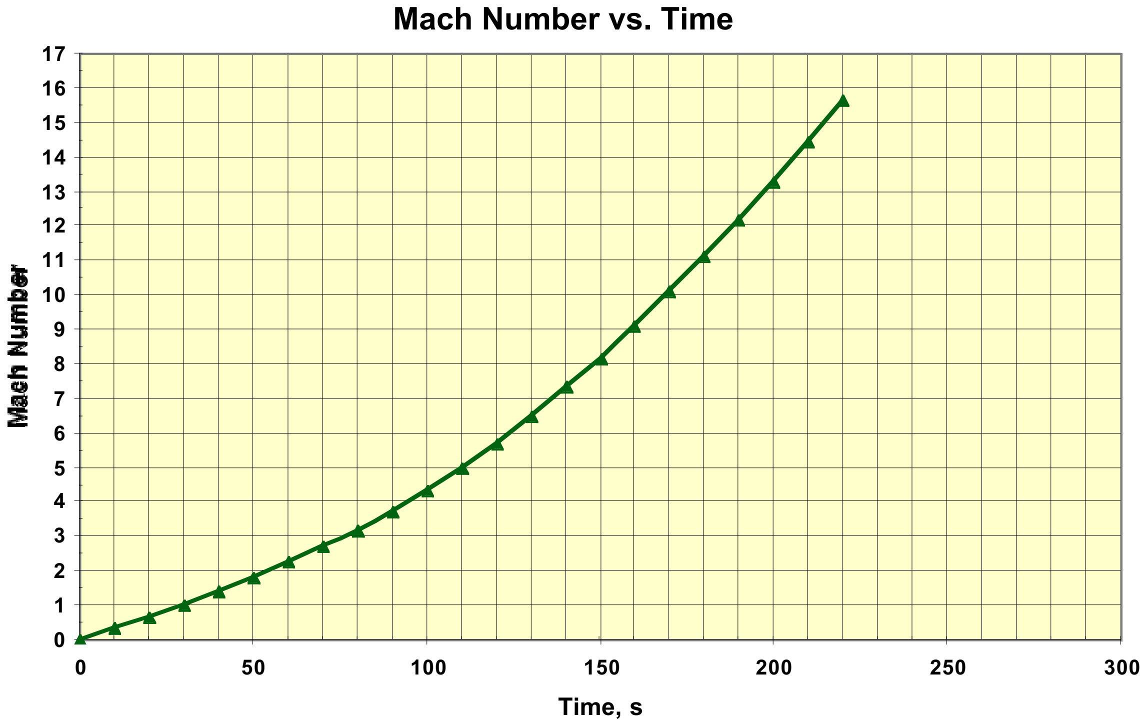

Figure 6-31 Ascent Mach Number versus Time

7 Planetary Resource Utilization Systems Studies

The GAC Astrotel transportation architecture between Earth and Mars requires in-situ resource utilization (ISRU) system to provide propellant cost effectively. The Colorado School of Mines (CSM), under the Extraterrestrial In-Situ Resource Systems Concepts Development Project supported the Astrotel architecture by developing a conceptual design for the ISRU support system. The design proposes extraction of oxygen or water from regolith excavated from on the surfaces of the Moon, Mars, and its moon Phobos. Further information about the ISRU conceptual design is in the appended report from CSM. Note that the numbers in bracket [XX] refer to references in the CSM Final Report in Appendix 3 of Phase II Interim Report.

7.1 In Situ Resource Utilization (ISRU) Conceptual Designs

The CSM project's limited resources were applied primarily to the Phobos ISRU conceptual design. Conceptual designs from other projects underway at CSM for Lunar and Martian ISRU systems were integrated with the Phobos design to build the overall ISRU system design for Astrotel. Note that there are small inconsistencies between the requirements for the ISRU systems as shown in this section and the current derived MAMA ISRU requirements developed during Phase II.

7.1.1 Mars ISRU Conceptual Design

Excavation and extraction systems for Mars must be designed to withstand the low-pressure atmosphere (560 Pa), the low gravity (3.73 m/s^2) and the highly varying temperature variations (140 K to 300 K). A bucket wheel excavator (BWE) was designed for excavation of Martian regolith by Muff [35]. This design provided the baseline from which the Phobos BWE was analyzed. The Muff [35] BWE is sized to excavate 50 kg/hr of regolith and has a mass of 50 kg. Its dimensions are 0.58-m wide, 1.5-m long, and 0.5-m tall.

Simulation and analysis of the conceptual design made it evident that excavator geometry can be changed to decrease excavation forces. Decreasing excavation forces meant a decrease in system mass and power consumption. This led to an optimization of the BWE design to minimize the excavation forces. Optimization of the conceptual design reduced the computed torque applied to the bucket-wheel by 72% (5.39 Nm to 1.49 Nm) and the computed power consumed while removing regolith by 52% (0.0723 W to 0.0345 W).

Based on the results from the optimization, a prototype excavator has been designed, and fabrication of the excavator has begun. Once built, the prototype will be instrumented and excavation control algorithms developed for laboratory scale tests. Tests will produce actual excavation forces for model verification.

7.1.2 Phobos ISRU Conceptual Design

Phobos ISRU conceptual design work consisted of studies and development of a Phobos Regolith Excavator, the development of options for Phobos oxygen and water extraction and propellant production, and Phobos Regolith carbothermal reduction.

7.1.2.1 Phobos Regolith Excavator

It is difficult to design an excavation and extraction system for Phobos since it lacks an atmosphere, the gravity is extremely low (0.005 m/s^2), and the temperature variations are extremely high (150 K to 270 K). A trade study was done on proven terrestrial excavation alternatives to determine which would be most effective when modified for the Phobos environment which requires obstacle avoidance, rock sorting, continuous excavation duty cycle, excavator flexibility, and low mass excavation. The bucket-wheel excavator system (BWE) scored the highest. CSM Graduate Research Assistant Tim Muff built a model of an extraterrestrial BWE for Mars in the motion-solving software package, Visual NASTRAN Desktop. Lee Johnson modified and used it for the Phobos excavation modeling. The model, shown in Figure 7-1, has the ability to maneuver the boom and bucket-wheel around obstacles, while screening out stones from entering the storage bin using a grating over the auger that transports material to the bin. The BWE provides a continuous excavation duty cycle, since it can excavate materials and simultaneously transport the materials to storage without stopping excavation. Excavation forces for the BWE system are primarily horizontal and provided by the mass of the entire excavator instead of only the bucket mass, shown in Figure 7-2, allowing an excavator to work in extremely low gravity without exterior anchoring, provided the mass (material and/or excavator) provides ample traction for excavation and forward movement.

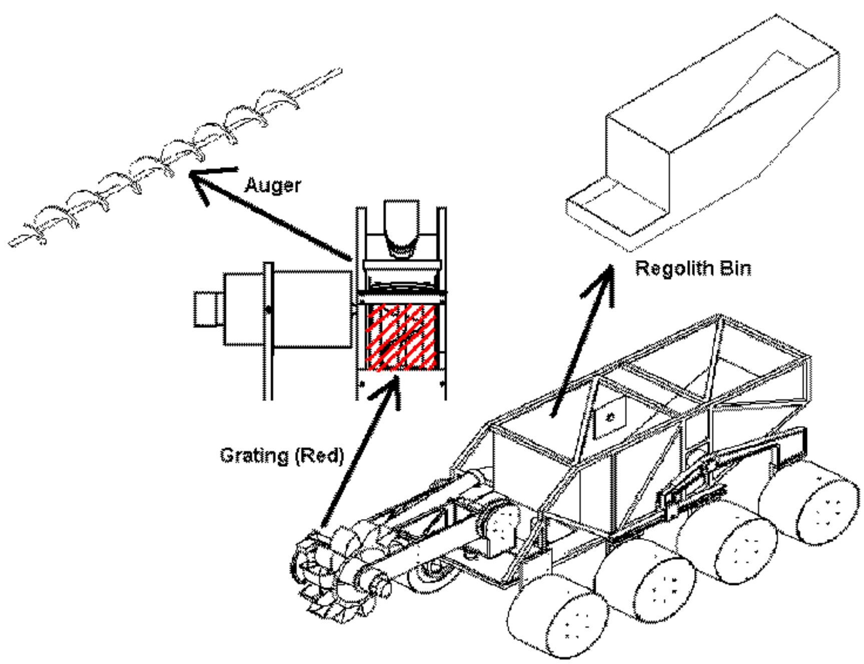

Figure 7-1 Prototype Mars Bucket-Wheel Excavator Design [35]

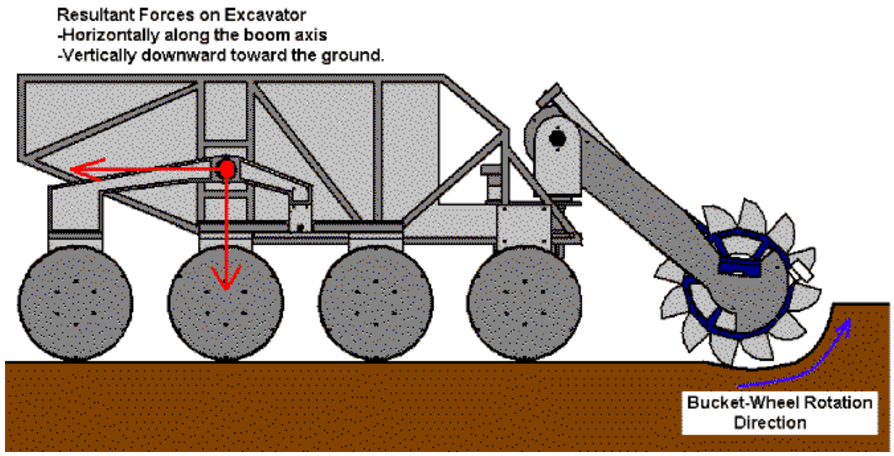

Figure 7-2: BWE Excavation Reaction Forces.

Muff's model for Mars was modified for Phobos by lowering gravity and adding a proper basic wheel penetration, traction, and rolling resistance model, per Bekker [38], which took into account both wheel penetration into the regolith, and shearing of the ground. The traction limits for the regolith were compared against a set of excavation forces developed by varying excavation parameters (such as production rate and bucket digging depth). The excavation parameters that produced forces below the available traction limits, which would cause no wheel slip, were used for the final excavator analysis trials.

The micro-gravity environment of Phobos poses a limitation on the bucket-wheel's ability to deliver regolith into the auger. If bucket-wheel rotation was too high, regolith was thrown from the buckets, or did not discharge. Two models were developed to find the limiting speeds for the bucket-wheel rotation and the boom-sweeping rate, shown in Figure 7-3 and Figure 7-4 when the buckets contained 1-mm diameter regolith particles. The limiting speed for correct discharge was found to be 6 °/s. The bucket-wheel rotational velocity limited the production rate and digging depths for the excavator.

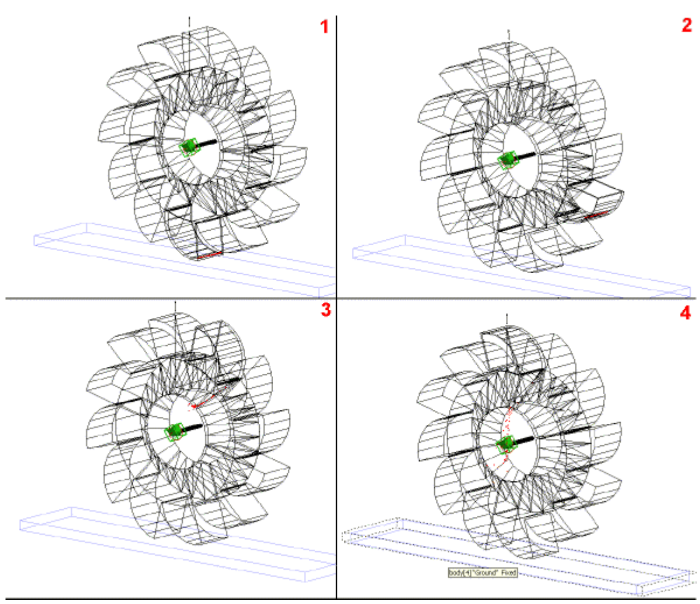

Figure 7-3: Bucket-Wheel & Regolith Particles Model.

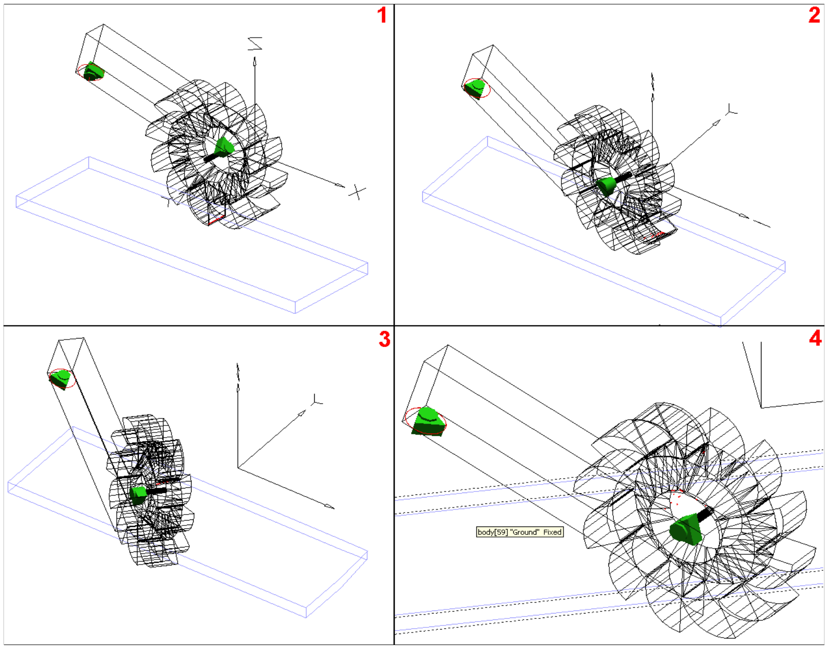

Figure 7-4 Bucket-Wheel/Boom & Regolith Particles Model.

Multiple BWE simulations calculated torque, instantaneous power, average power over one boom sweep, and average power over one hour of excavation from the set of production rates and digging depths that allowed the excavator to advance without wheel slippage. The model added rolling resistance and integrated to calculate the instantaneous power, average power over the excavation period, and specific power (power per mass of material excavated). The scenario with the highest production rate and lowest specific power respectively yielded the best set of excavation parameters. The optimal parameters and associated power requirements are listed in the following table.

Table 7-1 BWE Model Characteristics for Phobos Excavation Conceptual Design.

Characteristic	Value
Production Rate (Q)	20 kg/hr
Forward Depth of Cut (FDOC)	0.04 m
Depth of Cut (Z)	0.02 m
Required Traction	19.3 N
Available Traction	19.8 N
Average Total Power Excavation (1 Hour)	6.29 W
Average Total Power Movement (1 Hour)	0.11 W

Future additions to the NASTRAN BWE model could include:

1) Integration of known regolith composition and physical properties from future studies of or exploration missions at Phobos.
2) Enhanced soil interaction algorithm to account for soil shearing (FEA), compaction, and excavator wheel flotation.
3) Integration of work, power, and optimization calculations into NASTRAN Desktop.

4) Enhanced materials handling model of material movement and power consumption to compare ballistic, conveyor, and pneumatic systems.
5) An excavation area model that would optimize the travel path of the excavator given power, time, and other constraints.
6) A scaling model that would optimize the BWE physical dimensions to specific excavation environment and constraining criteria.

7.1.2.2 Trade Studies for Phobos Oxygen and Water Extraction and Propellant Production

The Astrotel Phase I baseline assumed that only oxygen would be available from Phobos' regolith and that the regolith would be processed on Phobos. Several spreadsheets were constructed, based on the baseline Phobos oxygen production system run during the Phase I study, to assess the:

1) implications if Phobos actually has 10% extractable water in the regolith; and
2) benefit of mining regolith on Phobos and transferring it to a spaceport for processing.

The calculations for the second question were carried out for two different spaceport locations with different ΔV requirements (100 m/s and 700 m/s) for transport from Phobos to the spaceport. These preliminary estimates can be improved when the final location of the Mars spaceport is determined. The calculations assumed that the spacecraft mass is 10% of the mass of the payload that it carries, which should be sufficient as the impulse required for transferring propellant to the spaceport is small.

The general conclusions are:

1) The production of water on Phobos requires less total mass than the production of oxygen. Although the abundance of extractable water is about 1/3 of that assumed for oxygen in the baseline, the reactors are more efficient at lower operating temperatures and less power is required for extraction. The Phobos water scenario assumes that water is produced on Phobos and transferred to the spaceport for electrolysis and liquefaction. It doesn't make sense to electrolyze water and liquefy the gases at Phobos, as water is easier to transport to the spaceport than the cryogens and power is more available at the spaceport.
2) Transporting regolith from Phobos to a spaceport for processing was studied to determine whether the gain of efficiency of the processing and power systems in free space would offset the need for more hardware and propellant use in carrying large amounts of regolith back and forth. Production at the spaceport can be effective at small ΔV's, but becomes unreasonable at ΔV's greater than a few hundred m/s. It was assumed that the regolith would have to be returned to Phobos after extracting the propellant, so propellant has to be provided for both transfers. The spacecraft is not large, but at 700 m/s ΔV, the amount of propellant used to carry the regolith back and forth became nearly half of the mass of the propellant that could be produced from the regolith load.

In the original baseline calculation, the mass of the spacecraft transporting oxygen to the spaceport was not included. The baseline was modified to include the mass of the tanker. As something must be transported from Phobos to the spaceport in all scenarios, this spacecraft must be included for a complete comparison of how much mass would have to be delivered to the Mars system.

7.1.2.3 Phobos Regolith Carbothermal Reduction

The true composition of Phobos is unknown, however from its low albedo and density it has been considered to be of a carbonaceous chondrite composition. Most carbonaceous chondrites contain bound water, but the observed reflectance spectrum of Phobos by the Russian Phobos spacecraft found no water absorption line.

Table 7-2 lists the composition of carbonaceous chondrite. Oxygen might be produced by heating the regolith, assuming that elemental carbon is available for the reduction of iron oxide and silicon dioxide. Several problems exist with this concept:

1) The amount of carbon that can be obtained by carbothermal reduction is directly related to the carbon content. If all the carbon is converted to CO, the mass of oxygen that can be obtained is 1.3 times the mass of carbon. To obtain the 34% of regolith mass to oxygen conversion assumed by the original Astrotel concept, a carbon recovery and recycling system is required.
2) Carbonaceous chondrite materials contain high amounts of sulfur which is an efficient fouler of catalysts that are used to convert CO to O_2. A means of sulfur removal is required.
3) The composition of the gases evolved from pyrolysis of carbonaceous chondrite material will be complex and will require a means of adjustment to simplify the reactor system.

Table 7-2 Chemical Composition of Carbonaceous Chondrite

Component	%
SiO_2	28.23
MgO	19.03
FeO	12.22
C	4.43
FeS	20.67
MeO*	15.42

*Metal oxides, not reduced by carbothermal reduction, considered inert.

As a part of the research into the propellant production aspect of the Astrotel concept, three groups of CSM chemical engineering students addressed the question of how oxygen may be extracted from dehydrated carbonaceous chondrite material, taking into account the issues listed above. Their findings are:

1) The first team calculated the equilibrium gas composition to be expected when carbonaceous chondrite material is heated to different temperatures. The C/H ratios in which the Phobos regolith is heated can be adjusted to drive the released sulfur into the CS_2 form. As long as the weight ratio of carbon to sulfur in the regolith is greater than 0.2, in a reactor that recycles carbon and hydrogen, carbon replenishment is not required. However, to reduce unwanted reactions in the effluent stream a method must be found to reduce its temperature below that of the gas reactor.

2) The second team independently addressed the problem associated with the removal of sulfur (however they did not consider the possibility of CS_2 removal). They considered three approaches:
   A) Removing sulfur as molten FeS at temperatures around 1200 °C.
   B) Using a modified Claus process [Gary, J.H. (1994) Petroleum Refining: Technology and Economics. Marcel Dekker, New York] where H_2S and SO_2 are reacted to produce water and elemental sulfur (a feasible process but complicated).
   C) Using an absorption agent to remove H_2S (extremely complex and not studied in depth).
3) The third team considered the process of taking a gas stream from the pyrolysis/sulfur removal steps and producing oxygen. This process was based on the assumption that the incoming gas stream is a mixture of CH_4 and CO. The resultant gas processing system was sized to produce 69mt/yr of O_2.

From the work done, it is concluded that the production of oxygen by carbothermal reduction of the regolith on Phobos is feasible. Although not studied in detail, the separation of CS_2 is considered to be the most promising approach. Preliminary mass and power estimates have been provided for the Astrotel MAMA model. However, several key issues require additional study:

1) Integrating carbothermal reduction reactor and gas processing systems.
2) Adapting chemical engineering processes (such as flash tanks and distillation columns) to zero gravity.
3) Designing a sulfur removal process involving the separation of CS_2.

Designing the gas processing system to define the mechanisms for controlling C/H in the sulfur removal process.

7.1.3 Lunar ISRU Conceptual Design

The discovery that Lunar Polar Regions might contain water-ice has increased the likelihood of water extraction from surface regolith. A senior capstone design course team designed an integrated excavation/extraction system capable of excavating 350,000 kg/yr of polar regolith, yielding 3,500 kg/yr of water if the regolith contains 1% ice by weight. An integrated system could excavate the soil, heat it, remove the water, and then reject the leftover soil. The excavation/extraction system was constrained to a mass of 200 kg, including the power source, a size of 1 m by 0.5 m by 1 m, capable of operating in temperature ranges at the Lunar South Pole (25 K to 80 K). This would require excavating 50 kg of regolith per hour, assuming a continuous duty cycle with forward motion of 2 m/hour, cutting depth of 5 cm, and cutting width of 0.25 m. The students designed a continuous scraper excavation system and a thermal extraction system. Design of a prototype laboratory system is underway with construction and testing expected over the next year.

The excavator, shown in Figure 7-5, contains a cylinder with spikes that break up the regolith ahead of the scraper. The Mars/Phobos BWE design could also be used on the Moon and is an alternative approach that will be considered. The excavated regolith is conveyed to the extractor.

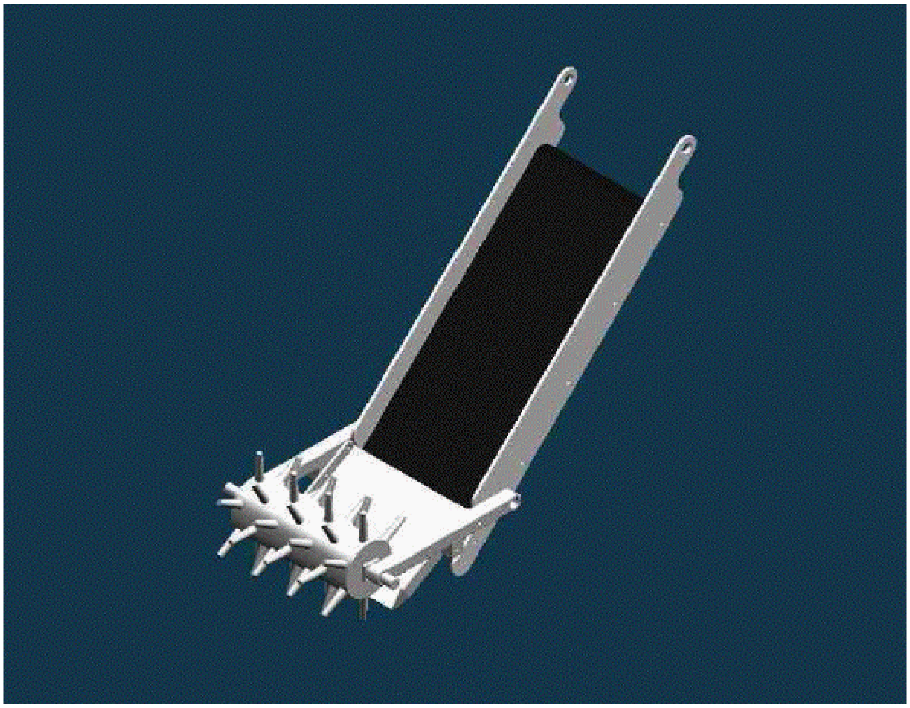

Figure 7-5 Lunar Ice Excavator

The extractor, shown in Figure 7-6, sublimates ice from the regolith and stores the collected water, in an acceptable phase, for delivery to a central storage system.

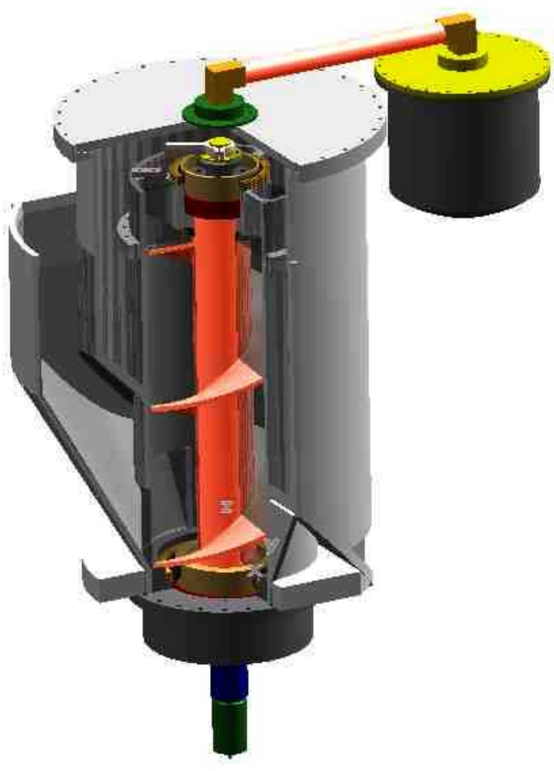

Figure 7-6: Lunar Ice Excavator Extractor.

The total mass of the system is 45.2 kg, not including the power supply. The system holds more than 44 kg (11.1 kg in the hopper, 11.1 kg in the auger core [max], and 22.4 kilograms in the outer bin [max]) of regolith at steady state. The Extract, Collect, and Store Unit (ECSU) processes nearly its own mass in regolith in each hour of operation. The system holds up to 1.7 kg of water ice in this configuration. Table 7-3 summarizes the power required. The amount of power required is consistent with the use of a Sterling isotope dynamic power system.

Table 7-3 Lunar Ice Excavator Power Requirement Summary

Component	Power Requirements (W)
Extractor	242
Excavator	62
Total	**304**

7.2 ISRU Propellant Production and Storage

The Astrotel architecture features the production and cryogenic storage of propellants produced from in-situ resources as well as designs for storage and delivery systems at the propellant production sites and at the Earth and Mars spaceports. The in-situ produced liquid hydrogen and oxygen propellants would eliminate the need and cost of transporting propellant from Earth to Mars, the moon or the Earth Spaceport.

The ISRU propellant depot models were developed from requirements given by the Phase I MAMA. The designed cryogenic propellant production and storage facilities have five major components:

1) Storage Tanks
2) Electrolysis
3) Dryers & Radiators
4) Liquefiers & Radiators
5) Solar Cells

7.2.1 Mars Spaceport Propellant Depot

Table 7-4 summarizes the requirements for the Mars Spaceport propellant depot, which is in an orbit of Mars at the radius of Phobos , according to the Phase I version of MAMA.

Table 7-4 Mars Spaceport (O2 Liquefaction and LOX/LH Storage)

Oxygen $_{(Gas)}$ Generation From Reactor	23.2 kg/hr
Oxygen Liquefaction Rate	23.2 kg/hr
Phobos LOX Storage (25% annual production, rest in Taxi)	17,272 kg
Earth LH Storage (25% annual production, rest in Taxi)	2,467 kg

The Mars Spaceport depot conceptual design assumed:

1) 3-month storage of propellant on facility, rest in Taxi [41].
2) Facility does not supply power to Taxi when docked; Taxi has its own power source.

3) Facility keeps liquid hydrogen from Earth liquefied, and takes in oxygen gas from Phobos' surface and liquefies & stores it. Liquefaction of oxygen in orbit allows for a possible 100% duty cycle. (No Dryers necessary)
4) Oxygen gas delivered daily to facility, which operates at 100% duty cycle [41].
5) Linear proportioning of capacities, masses, sizes, power, and capabilities from other systems using similar equipment [1, 41].

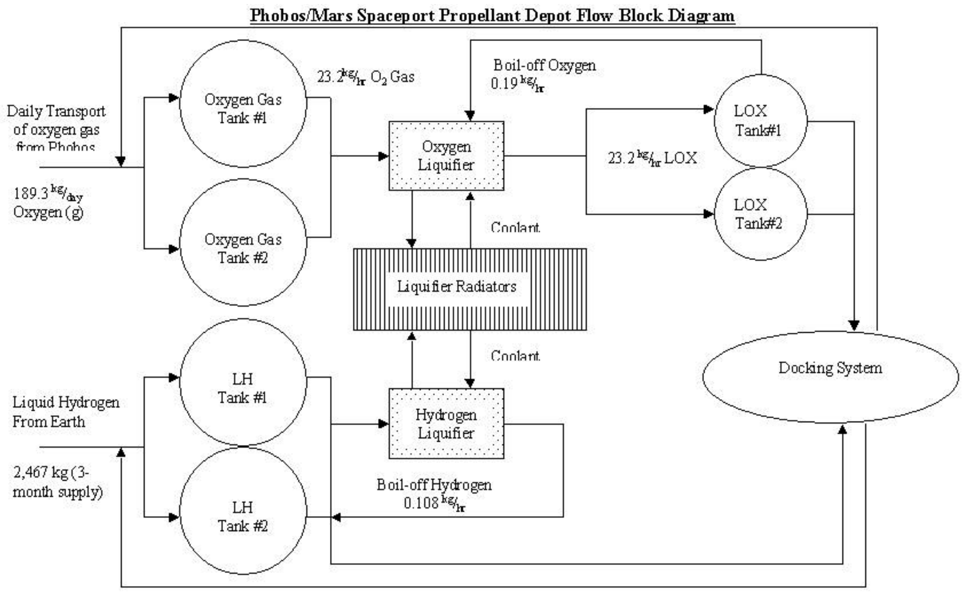

Figure 7-7: Mars Spaceport Propellant Depot Flow Block Diagram.

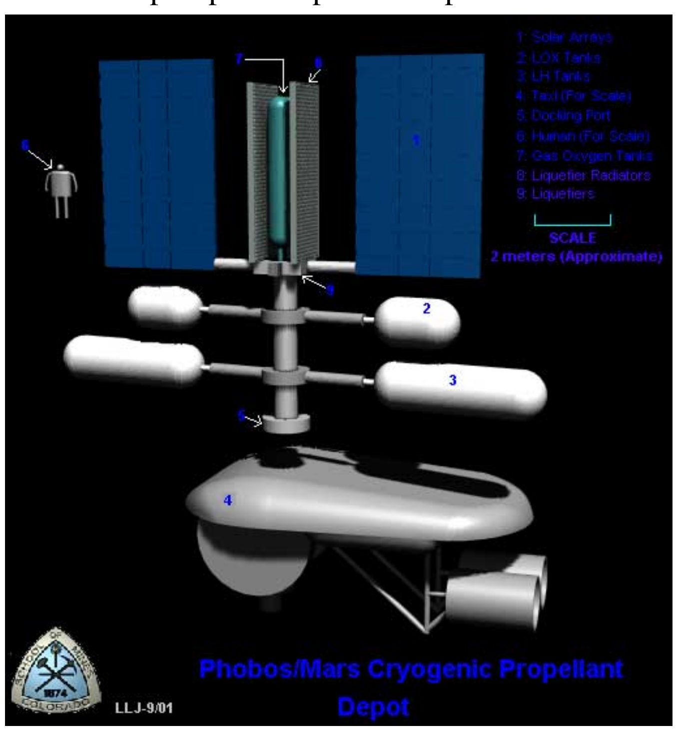

Figure 7-8: Mars Spaceport Propellant Depot Rendered 3D Solid Model.

7.2.2 Mars Base Propellant Depot

Table 7-5 summarizes the requirements for the surface propellant depot at the Mars Base, according to the Phase I MAMA. Figure 7-9 shows the Mars Base propellant flow block diagram.

Table 7-5 Mars Base (LOX/LH Production/Storage, H2O Storage):

Total Water Storage (From reactor, holding for electrolysis)	1,147 kg
Electrolysis Rate	8.07 kg/hr
LH Liquefaction Rate	1.01 kg/hr
LOX Liquefaction Rate	7.06 kg/hr
LH Storage (25% annual production, rest at Mars Spaceport)	30,593 kg
LOX Storage (25% annual production, rest at Mars Spaceport)	3,824 kg

The Mars Base depot conceptual design assumed:

1) 4 month water storage on facility (1.147 mt of liquid water) [41].
2) All rates used from reference 41.
3) 33% annual production and storage of LOX and LH @ Mars base stored on site, the rest is stored in the Mars Shuttle tanks (Storage and boil-off rates are: Hydrogen 3.3% per month, and Oxygen 0.8% per month [43].

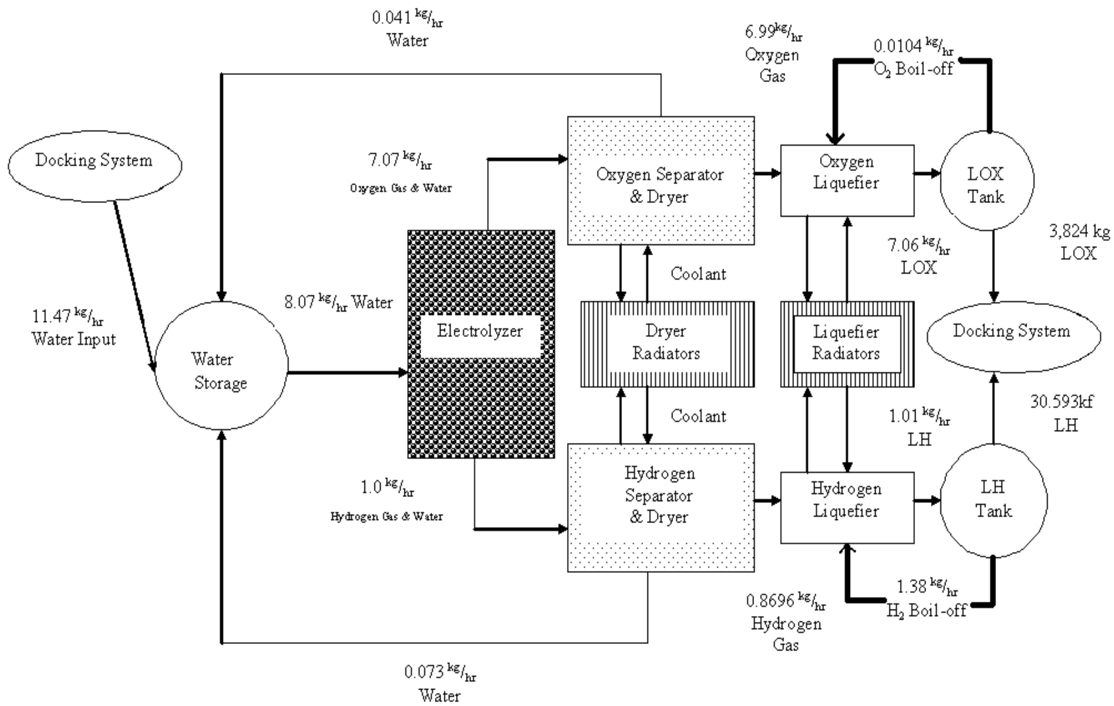

Figure 7-9: Mars Base Propellant Depot Flow Block Diagram.

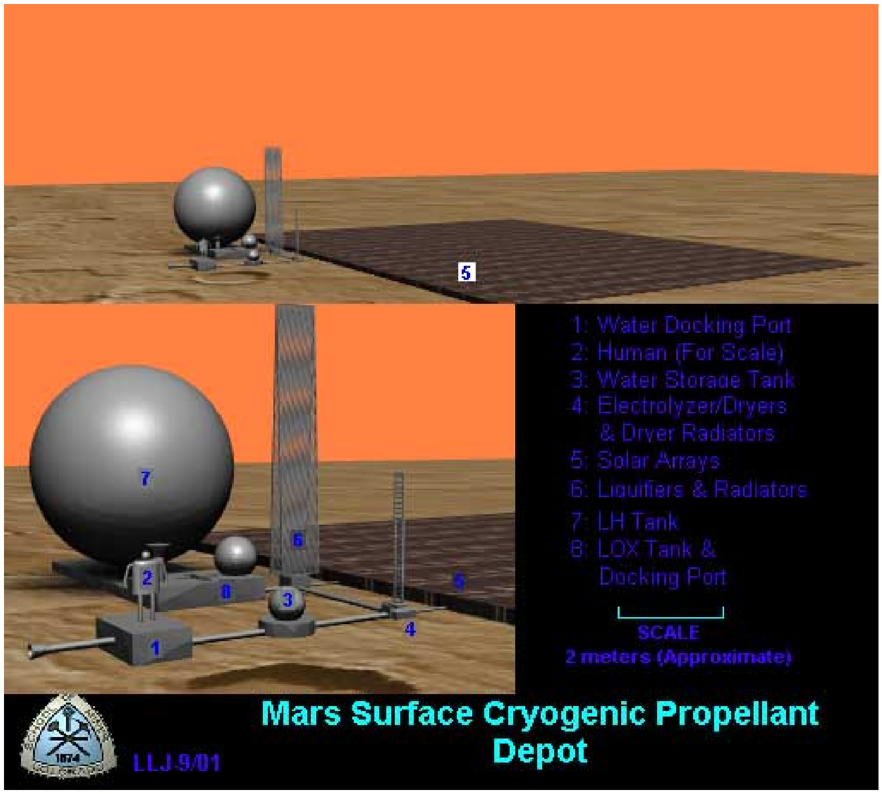

Figure 7-10 Mars Base Propellant Depot Rendered 3D Solid Model.

7.2.3 Lunar Polar Propellant Depot

Figure 7-11 summarizes the requirements for the surface propellant depot on the Moon, according to the Phase I MAMA. Figure 7-12 shows the Lunar Polar Propellant Depot propellant flow block diagram.

Figure 7-11 Lunar Polar (LOX/LH Production/Storage, H_2O Storage):

Total Water Required	30,473 kg
Water Storage (0.33 annual water production)	10.16 kg/hr
Electrolyzer (Produces propellant for launching water to L1)	4.5 kg/hr
LH Liquefaction (Produces LH for transfer vehicle)	0.56 kg/hr
LOX Liquefaction (Produces LOX for transfer vehicle)	3.94 kg/hr
LOX Daily Storage	32.15 kg/day
LH Daily Storage	4.57 kg/day

The lunar surface depot conceptual design assumed:

1) 4-month water storage on facility (10.06 mt of liquid water) [1].
2) All rates used from reference 41.
3) Daily transfer of LOX and LH to L1 space facility, cryogenic LOX and LH tanks sized accordingly. (Boil-off negligible during short storage time)

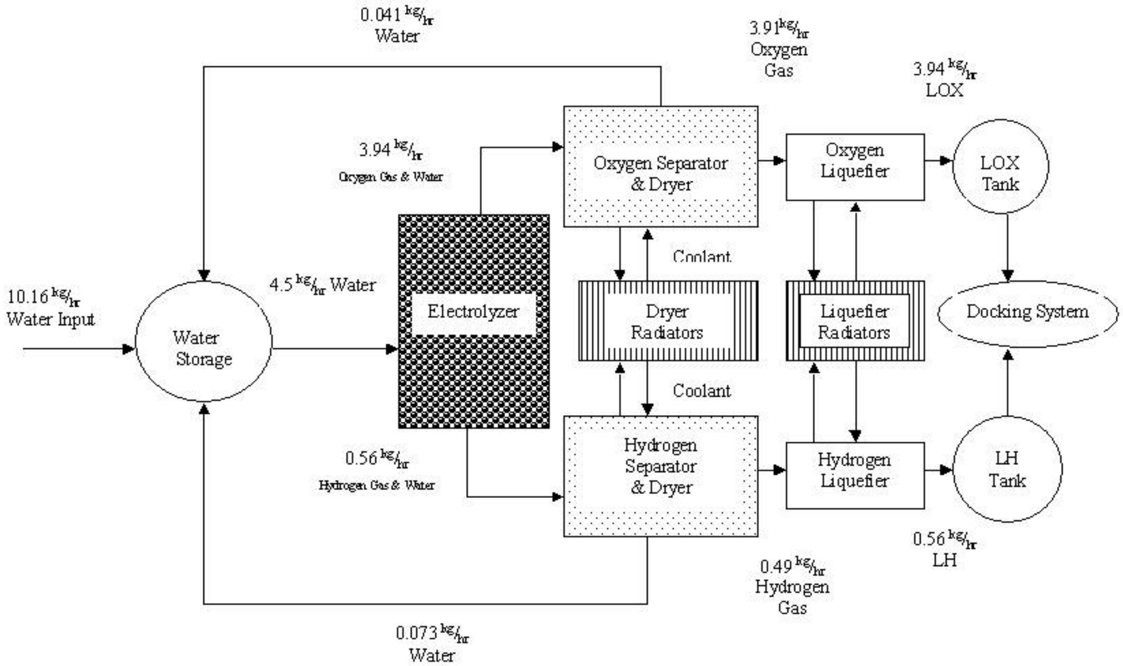

Figure 7-12 Lunar Polar Propellant Depot Flow Block Diagram.

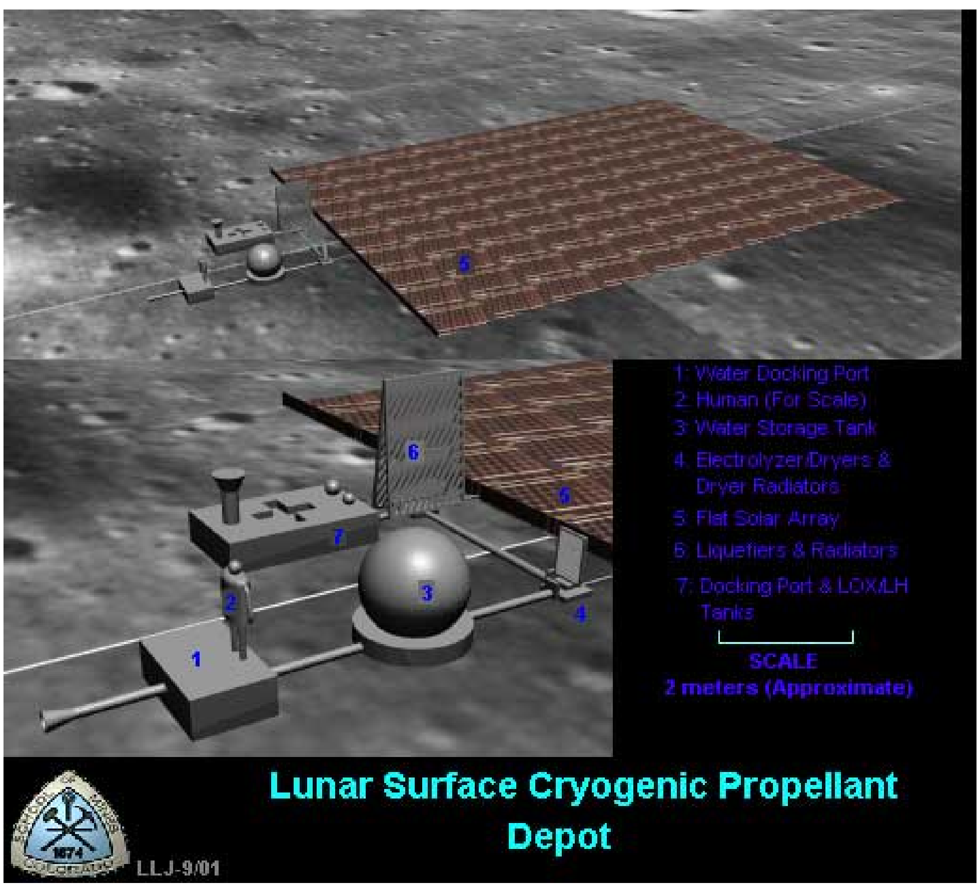

Figure 7-13 Lunar Polar Propellant Depot Rendered 3D Solid Model.

7.2.4 Earth Spaceport Propellant Depot

Table 7-6 summarizes the requirements for the orbital propellant depot at lunar orbit radius according to the Phase I MAMA. Note, that in the current Phase II plan the Earth Spaceport is in an orbit at lunar radius instead of at L1. Figure 7-14 shows the Earth Spaceport Propellant Depot propellant flow block diagram.

Table 7-6 Earth Spaceport (LOX/LH Production/Storage, H2O Storage):

Total Water Storage (3 month lunar water supply)	3,818 kg
Electrolysis Rate	1.59 kg/hr
LH Liquefaction Rate	0.18 kg/hr
LOX Liquefaction Rate	1.42 kg/hr
LH Storage (3 month storage, rest in Taxi)	387 kg
LOX Storage (3 month storage, rest in Taxi)	3,099 kg

The following assumptions were made during the Earth Spaceport depot conceptual design:

1) 3 month water storage on facility, rest in Taxi (3.818 mt of liquid water) [41].
2) All rates used from reference 41.
3) Assume 100% duty cycle on solar arrays.

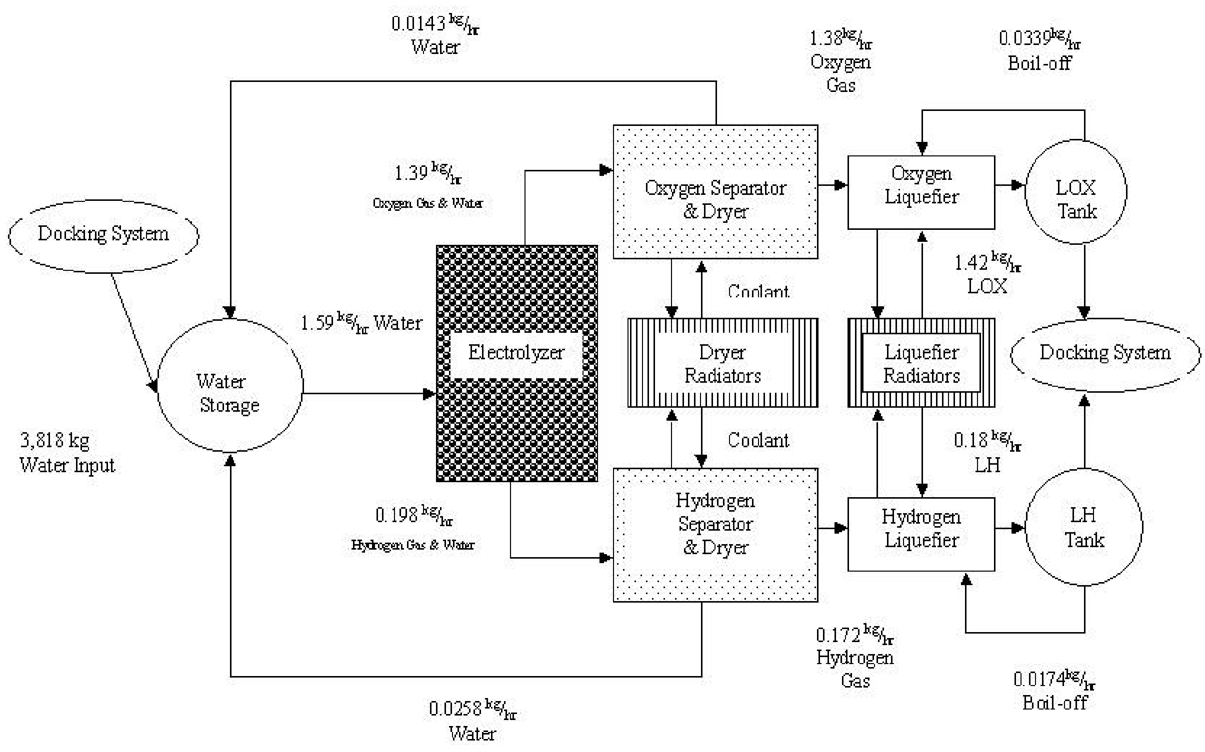

Figure 7-14 Earth Spaceport Propellant Depot Flow Block Diagram.

Figure 7-15 Earth Spaceport Propellant Depot Rendered 3D Solid Model.

7.3 Recommendations for Future Robotic Exploration for ISRU

As stated previously, little is known about the surface of Phobos and several conflicting theories exist about the physical and chemical composition of Phobos regolith. We recommend science studies about Phobos in the future to measure:

1) Surface regolith and underlying base material properties, including size composition, compaction, friction angle, composition by depth, thermal, etc.

2) Location of surface and subsurface anomalies, such as mineral concentrations, caverns, and gravitational attraction.

3) The surface environment including temperature, wind, dust, radiation, and micro-meteor impact

8 Flight Systems Development

8.1 Introduction

In this section we describe common technologies that are used in all architectures, the common system elements used in several vehicles, and the detailed flight systems designs.

8.2 Common Transportation Architecture Technologies

Several common technologies exist that are used throughout the architecture, namely, propulsion, power and radiation protection technology.

8.2.1 Propulsion Systems

8.2.1.1 Chemical (LOX/LH) Propulsion

Many options exist for the design basis for the LOX/LH engines used in the Mars Transportation architecture. For the purpose of the Phase I study, however, a 7:1 mixture ratio LOX/LH, Pratt & Whitney (P&W) RL-series propulsion system has been assumed. The RL10 engine is an Expander Power Cycle engine, meaning that the Hydrogen is circulated through the nozzle to pre-heat it and provide high-pressure gas power for the fuel and oxidizer turbo-pumps. For typical LOX/LH engines the mixture ratio is 5 or 6:1. Today's Space Shuttle main engines are set at a 6:1 mixture ratio while the P&W RL10, used on Centaur, actually varies its mixture ratio between 5:1 and 6:1 in flight to ensure near simultaneous fuel and oxidizer depletion. The implications of the 7:1 mixture ratio are 1) reduced the hydrogen volume required and thus reduced vehicle size and 2) somewhat reduced thrust level. Nominal parameters of the RL10B-2 engine at 5-6:1 mixture ratios are a thrust of 110.1 kN (24,750 lbs_f), I_{sp} of 464 s, propellant burn rate of 24.4 kg/s and a mass of about 277 kg. Thrust and burn rate of this engine operating at a 7:1 mixture ratio is estimated at 66.7 kN (15,000 lbs_f) and 14.8 kg/s. A nice feature of this engine is its extendable nozzle, which enables it to fit in a volume about half of its deployed state (its length goes from about 4.1 m to 2.1 m long). Figure 8-1 is a drawing of the RL10B-2 engine with nozzle retracted.

A higher thrust engine is desirable for the Mars transportation architecture. P&W is now developing an advanced RL-series engine called an RL60, having 60,000 lbs. of thrust, roughly three times higher than the RL10. In addition, the current RL60 design driver is missions to GEO that require 2 engine restarts for a total of only 3 burns and a total engine life of only 700 s. Maximum engine gimbaling capability of ±5° is expected. This engine has an I_{sp} of 459 s at a mixture ratio of 7:1. For the purpose of the Mars transportation architecture, a higher thrust engine than the existing RL10 is desired in order to reduce finite burn and gravity losses due to the large, long delta-V burns at Mars. The RL60 engine is about the same form factor as the RL10, the increase performance coming from running the chamber pressure about three times higher.

Considerable engine technology development may be required to enable several hours of reliable operation in remote locations, however, the Space Shuttle main engines were originally designed to operate for about 8 hours over about 55 starts without major overhaul. The specific wear issues relate generally to rotating machinery items such as bearings, seals, etc. For a multi-year mission requirement, there are additional considerations that need to be addressed (e.g. slow flowing of static seals, slow degradation of material properties specifically used in the engine due to factors such as radiation).

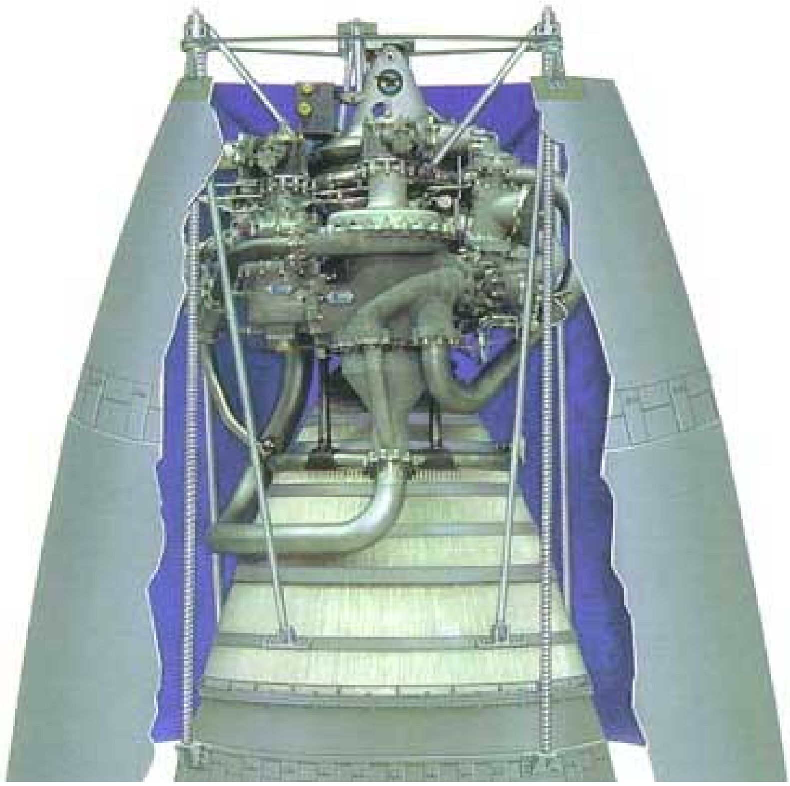

Figure 8-1 Pratt & Whitney RL10B-2 Engine

For the purpose of this study we have assumed a LOX/LH engine based on the new RL60 engine but with a de-rating to account for the higher mixture ratio, the projected longer burn, and longer life requirements. The assumed rocket engine for the architecture has the following characteristics:

Table 8-1 LOX/LH Engine Characteristics

Parameter	Value
Mixture Ratio	7:1
Thrust	266.9 kN (60,000 lbs_f)
Specific Impulse	460 s
Propellant Burn Rate	59.2 kg/s
Engine Mass	500 kg

8.2.1.2 Solar-Powered Ion Propulsion

After more than 40 years of development by NASA, solar-powered ion propulsion systems (IPS) are now operational. JPL's Deep Space 1 (DS1), which used IPS, launched in October of 1998. DS1's IPS consisted of a throttling, single 30-cm diameter, 2.5 kW input Xenon ion thruster operating at an exhaust velocity of about 30.4 km/s (I_{sp} of 3100 s) capable of thrust levels from about 21-92 mN. The following figure illustrates the key components of the DS1 IPS.

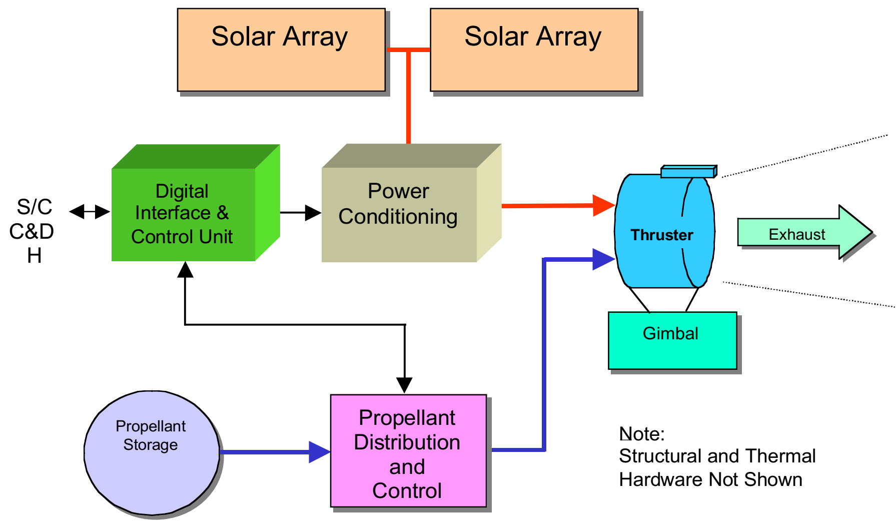

Figure 8-2 Simplified Block Diagram of an IPS (courtesy J. Brophy, JPL)

An ion propulsion system (IPS) converts solar generated electrical energy to momentum of positively charged molecules or ions (since the 1980s noble gases, like Xenon, have replaced Mercury for ion thrusters). This conversion is accomplished by first ionizing suitable atoms by electron bombardment and then accelerating them in the desired direction by using two electrically charged grids. The magnitude of the applied voltage and the charge-to-mass ratio of the ions determines the exhaust velocity. The momentum of these ions reacts against a spacecraft propelling it in the opposite direction. An IPS can be extremely efficient if sufficient solar power is available and a long time is allowed for making velocity changes. The reason for this efficiency can be seen in the "rocket equation." The rocket equation is, $\Delta V=V_e*\ln(m_i/m_f)$, where V_e is the exhaust velocity of the thruster and m_i and m_f are the initial and final mass of the spacecraft. In the rocket equation the delta-V is directly proportional to the exhaust velocity of the rocket engine. If the initial spacecraft mass is only 10% greater than the final mass, (meaning $m_i=1.1m_f$), the delta-V capability of the DS1 IPS is nearly 3 km/s. Given the same final mass but a LOX/LH engine with a V_e of 4.5 km/s, more than 90% of the final mass is required to achieve the same delta-V ($m_i=1.9m_f$). It is this efficiency that makes an IPS very attractive for making the occasional course corrections required of the Astrotels, providing station-keeping forces for the Spaceports and providing the primary motive force behind the interplanetary cargo freighters delivering consumables, propellants and refurbishment hardware to Spaceports and Astrotels.

8.2.1.2.1 Status and Plans

The DS1 IPS is obviously a resounding success but considerable technology advance is still needed by the Mars transportation systems under study. Fortunately, there is ongoing NASA research into IPS technology that is moving in the proper direction. The following figure illustrates one possible evolution of IPS technology development.

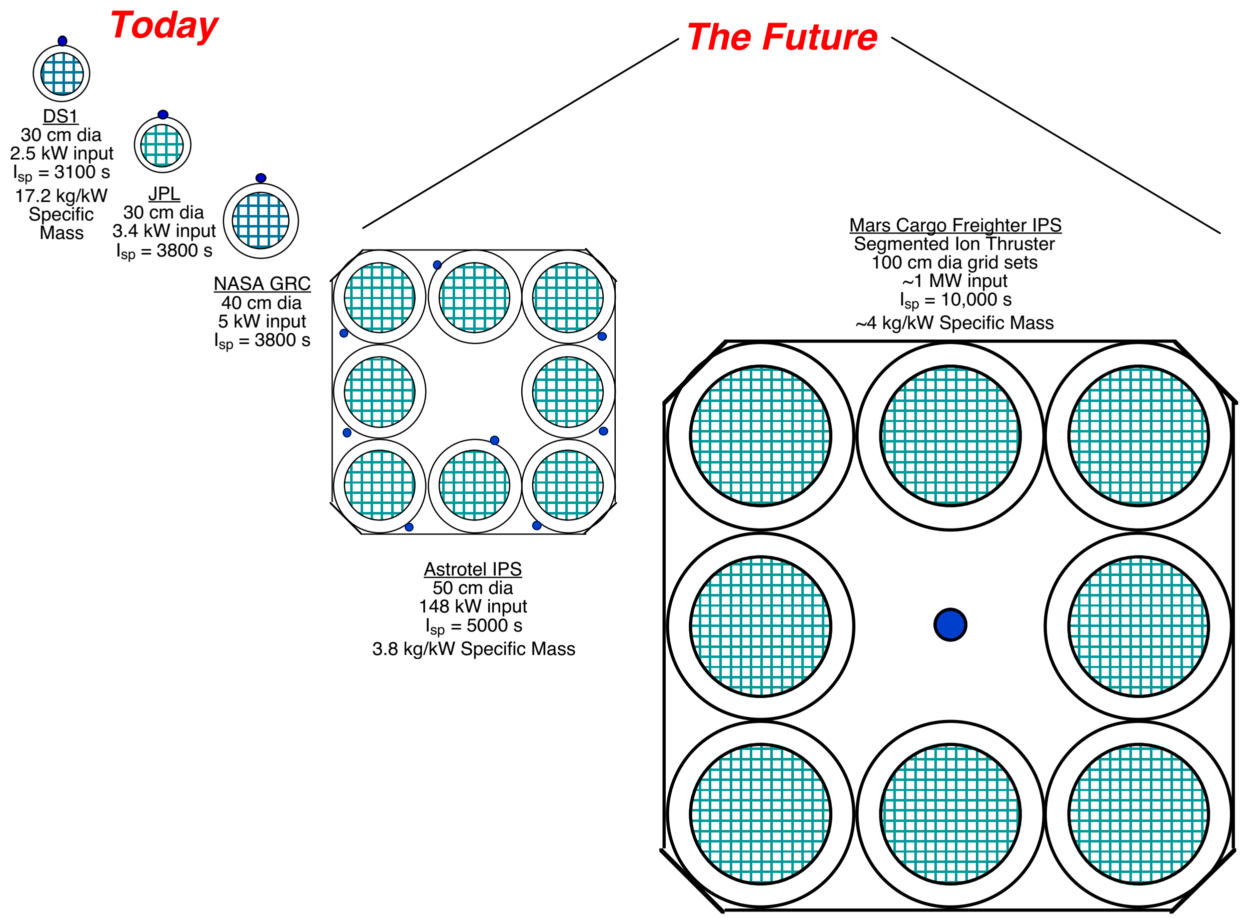

Figure 8-3 Example Evolution of IPS Technology

JPL has worked on a modest upgrade of the DS1 thruster which increases the input power to the Power Processing Unit (PPU) by about 35% to 3.4 kW at the same time increasing the specific impulse by about 23% to 3800 s. This thruster design is slated for several possible missions including a comet nucleus sample return. NASA Glenn Research Center is working on a 40-cm diameter thruster that also would operate at a specific impulse of 3800 s.

Conventional single engine designs are limited as higher powers are processed and exhaust velocities are increased because of the difficulty in making the accelerator system (thruster span-to-gap ratios, accelerating voltage constraints, and current handling capability). In order to maintain constant power density across a thruster the grid separation must remain constant. Current thrusters have an engine-span-to-grid-gap ratio of about 500 (a 30-cm diameter engine could have its high voltage grids separated by a gap of only 0.6 mm depending on voltage across). As the desire to process more power grows, the engine diameter grows. Assuming practical limits to the electric field between the grids the span-to-gap ratio can eventually grow beyond the state-of-the-art.

Because of the need to process much larger levels of power over larger area thrusters and because of the practical limits to span-to-gap ratios, multiple grid sets are attractive. Multiple grid sets, along with their smaller individual ion source components, electrically connect several grid sets together so that they simulate a larger diameter thruster yet still retaining the desired span-to-gap ratio for each engine segment. Additionally, only one neutralizer is required for the multiple grid set and the smaller individual ion source chambers reduce complexity and other plasma problems. Multiple grid sets per engine can significantly increase the beam current per engine. Such Segmented Ion Thruster (SIT) designs include individual propellant ionization chambers for each engine segment and only one neutralizer for the set of two or more thrusters. Multiple aperture grid ion propulsion is assumed for the Astrotel and Cargo Freighter propulsion systems because of the advantages discussed above.

8.2.1.2.2 Astrotel IPS

There are several system options for the Astrotel IPS depending on technology advance including the upgrade of DS1 engine technology or designing a new 8-set SIT.

8.2.1.2.2.1 Upgrade DS1 Engine Technology

Upgrade of DS1 single engine technology from a 30-cm diameter, 2.5 kW, 3100 s specific impulse, and 19.2 kg/kW specific mass system to a 50-cm diameter, 17.2 kW input, 5000 s specific impulse, and 3.8 kg/kW specific mass system. This appears to be a modest improvement in technology, especially since 30-cm diameter engines have already been run at 20 kW input power about 15 years ago, though not with the lifetime capability required for the Astrotel system. Given the requirements of the Astrotel IPS, J. Brophy at JPL generated a model of this engine. The model inputs and resultant performance are presented in the following tables.

Table 8-2 Astrotel Ion Engine Performance Input Data

ION ENGINE PERFORMANCE INPUT DATA	
Parameter	**Value**
Engine Type	RING CUSP
Propellant (AMU)	131.3
Beam Diameter (cm)	50
Specific Impuse (s)	5000
Max. Span-to-Gap Ratio	500
Minimum Grid Gap (mm)	0.6
Max. E-Field (V/mm)	2600
Max. Disch. Current (A)	100
Max. Beam Current (A)	6.75
Maximum R-Ratio	0.9
Minimum R-Ratio	0.55
Perveance Coef. Xe x10E9	2.48
Perveance Exponent	1.5
Screen Grid Tranparency	0.75
Beam Flatness Parameter	0.6
Divergence Thrust Loss	0.98
Double Ion Ratio	0.1
Discharge Voltage (V)	28
Disch. Chmbr Prop. Eff.	0.92
Discharge Loss (eV/ion)	180
Keeper Current (A)	0
Keeper Voltage (V)	4
Coupling Voltage (V)	15
Neut. Keeper Current (A)	2
Neut. Keeper Voltage (V)	15
Neut. Flow Fraction	0.05

Table 8-3 Model-Estimated Performance of Astrotel Ion Engine

CALCULATED ION ENGINE PERFOMANCE	
Parameter	**Value**
Thrust (N)	0.515
Engine Input Power (kW)	17.23
Total Engine Efficiency	0.733
Thrust-to-Power Ratio (mM/kW)	29.91
Beam Voltage (V)	2353
Total Voltage (V)	2615
Net-to-Total Voltage Ratio	0.9
Beam Current (A)	6.75
Discharge Current (A)	43.38
Grid Gap (mm)	1.006
Actual Span-to-Gap Ratio	497.2
Screen Hole Diameter (mm)	3.35
Effective Acceleration Length (mm)	1.95
Maximum Beam Current Density (mA/cm2)	5.73
Average Beam Current Density (mA/cm2)	3.44
Double Ion Thrust Loss Factor	0.97
Total Propellant Efficiency	0.87
Total Propellant Flow Rate (g/s)	0.01052

The overall ion propulsion system mass breakdown based on the upgrade of the DS1 IPS is shown in the following table.

Table 8-4 Astrotel IPS Mass Breakdown

Conventional Approach with Redundancy -- XFS				
Item	**QTY**	**Unit Mass, kg**	**Total Mass, kg**	**Comments**
Engine	8	18.40	147.2	
PPU	8	36.40	291	
DCIU	2	3.00	6	1 required plus one spare
Regulator	2	0.45	0.90	
Service Valve - HP	1	0.01	0.01	
Service Valve - LP	17	0.01	0.17	
Pressure Transducer	2	0.25	0.50	
Latch Valve - HP	2	0.10	0.20	8.71
Latch Valve - LP	8	0.10	0.80	
Filter	1	0.13	0.13	
Var. Reg. With Flow Meter	24	0.15	3.60	
Tubing	1	2.00	2.00	
Fittings	1	0.40	0.40	
Gimbal	8	5.52	44.16	One for each thruster
Misc. Thermal (5% of dry mass)	1	24.86	24.86	
Cabling (5% of dry mass)	1	24.86	24.86	
Structure (4% of dry mass)	1	21.88	21.88	
Total			***569***	

The summary assumptions and description for this option for the Astrotel IPS is shown in the following table.

Table 8-5 Summary Assumptions and Description for Astrotel IPS

Assumptions		
1 150 kW System Input Power		
2 2800 kg total propellant processed		
3 Xenon propellant		
4 Gridded ion engines		
5 Specific Impulse = 5000 s		
PROPULSION MODULE SUMMARY		Notes
Number of Engines	8	
Number of Operating Engines	8	
IPS Thrust (N)	4.12	
IPS Input Power (kW)	148.2	
Engine Input Power (kW)	17.23	
Engine Thrust (N)	0.515	
Engine Unit Mass (kg)	18.4	
Gimbal Mass (kg)	5.5	
Propulsion Module Cabling Mass (kg)	24.9	
Xenon Feed System (kg)	8.71	
Xenon Tank Mass (kg)	51.3	Assumes a tank sized for 1027 kg of xenon
Propulsion Module Structure (kg)	21.88	
Propulsion Module Dry Mass (kg)	**272**	Not including PPUs
Propulsion Module Specific Mass (kg/kW)	1.8	
POWER PROCESSING SUBSYSTEM SUMMARY		
Number of PPUs	8	
PPU Specific Mass (kg/kW)	1.96	
PPU Unit Mass (kg)	36.4	
PPU Efficiency	0.93	
Radiator Area per PPU (m2)	3.1	
Total PPU Mass (kg)	**291**	
Total PPU Radiator Area (m2)	24.8	Radiator mass not included
Number of DCIUs	2	
DCIU Unit Mass (kg)	3	
Total IPS Mass (kg)	***569***	
Total IPS Specific Mass (kg/kW)	***3.84***	Note the total NSTAR IPS specific mass is 19.2 kg/kW

The Xenon tank was sized for about one third the Xenon required over a 15-year cycle of operations. Note that the system specific mass is 3.8 kg/kW, a factor of over 4-times improvement over the current DS1 IPS. Such an improvement is projected to be possible in the 2010 timeframe.

8.2.1.2.2.2 Alternative Astrotel IPS

Instead of having eight individual engines, an eight-set segmented ion thruster (SIT) could be employed. This approach might offer some simplification and hardware part reduction since there would only be one neutralizer and one high voltage power supply.

If the 100-cm diameter grid set operating at 125 kW input could be developed, a throttled 2 grid-set SIT could be employed for the Astrotel IPS and an eight grid-set SIT could be used for the Mars Cargo Freighter.

8.2.1.2.3 Cargo Freighter IPS

For the low thrust trajectory analysis and the system definition of the Astrotel and Mars Cargo Freighter vehicles we assumed a propulsion specific mass of 4 kg/kW, consistent with the estimated performance in 2010. In addition, we assume a power system specific mass of 4 kg/kW, for a total power and propulsion system specific mass of about 8 kg/kW. The Astrotel and Mars Cargo Freighter input power requirements are expected to be much larger than the Astrotel requirements between 300-900 kW. This large size will likely require SIT configurations. For the purpose of the low thrust trajectory analysis, a specific impulse of only 5000 s was assumed. If a specific impulse of 10,000 s becomes a reality for a SIT system of this size, significant improvement in performance will occur.

8.2.1.3 References

1. Popp, M., Bullock, J. R., Santiago, J. R., Development Status of the Pratt & Whitney RL60 Upper Stage Engine, AIAA-2002-3587, presented at 36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Indianapolis, IN, July 7-10, 2002.
2. Brophy, J., "Near-Term, 100 kW-Class Ion Engines", AIAA Paper # 91-3566, AIAA/NASA/OAI Conference on Advanced SEI Technologies, Cleveland, OH, September 1991.
3. Brophy, J., "Ion Propulsion System Design for the Comet Nucleus Sample Return Mission", AIAA Paper # 2000-3414, 36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Huntsville, AL, July 2000.
4. Polk, J., et. al., "In-Flight Performance of the NSTAR Ion Propulsion System on the Deep Space One Mission", IEEE paper, 2000.

8.2.2 Power Systems

Two power conversion system choices were examined, namely nuclear and photovoltaic arrays. Factors considered in power generation technology selection are life cycle cost, specific power (power generated divided by generation and storage mass), modularity, and safety (space operations and manufacturing). Because solar photovoltaic power generation appeared very attractive due to the projected very low cost and mass, it was selected for the baseline architecture. This baseline selection was tested in Phase II with respect to life-cycle cost and the results are displayed in Section 9. Energy storage options are also discussed.

8.2.2.1 Solar Photovoltaic Power Generation

There are two very different photovoltaic powered missions involved in this report – deep space and planetary or satellite surface operation. The near term technology applicable to both these missions will be discussed first and then projected to technologies likely to be available in 2010. For the projection of solar array technology to 2010, cells are expected to have improved efficiency and structures and optical systems should become lighter.

8.2.2.1.1 Deep Space Solar Arrays

Deep space missions are those which take the spacecraft out of Earth orbit to orbits between Earth and the Moon, the L1 point and past Mars. Since Mars is not too distant from the sun, all of these missions can be handled with essentially the same power generation system technology. The size of power generation systems for the Mars transportation architecture ranges from about 12 kW to nearly 1 MW.

8.2.2.1.1.1 Mechanical and Optical Technologies

The near term technology was reviewed with regard to demonstrated performance at least at the solar array module level, cost and possibility of improvement. Two aspects were studied including lightweight deployable and concentrator arrays.

Lightweight photovoltaic energy conversion systems, or solar arrays, have been used on most space missions. However, very large solar arrays (100 kW) are not common so there is little experience applicable to this study. The analysis presented in this section is based upon projection of past development onto present demonstration technologies. Lightweight, deployable, photovoltaic arrays have been demonstrated for space missions ["Advanced Photovoltaic Solar Array Design," TRW Report No. 46810-6004-UT-00, 3 November 1986] and are being incorporated into a number of programs. Dependent upon the technology selected, large (in excess of 2kW), high performance arrays cost about $850/W. Using current costs, a 160 kW array, as suggested for the Astrotel, could cost 136 million dollars.

A solar cell concentrator approach, which should reduce array cost, uses fewer of the expensive elements, i.e. the solar cells. A 15 times concentrator array uses roughly one-twelfth the number of solar cells. Concentrator arrays have now been space qualified [P. A. Jones et al., “The SCARLET Light Concentrating Solar Array,” 25th IEEE-PVSC, 1996] on the DS-1 spacecraft. The SCARLET array has achieved over 200 W/m^2 areal power density and 45 W/kg specific power. Figure 8-4 illustrates the SCARLET array in flight.

A combination of lightweight array and concentrator technology is in the demonstration phase [M. J. O’Neill, “The Stretched Lens Ultralight Concentrator Array,” 28th IEEE-PVSC, 2000]. The stretched lens array (SLA) has been incorporated into a deployable, flexible-blanket planar space array concept called Aurora by AEC-ABLE. Figure 8-5 illustrates a stretched lens Array Prototype. Figure 8-6 illustrates a deployment concept for the SLA.

Aurora components have already demonstrated a cell efficiency of 30% and a lens efficiency of 92%. Operational efficiency at beginning of life (BOL) is expected to be 22% or about 300 W/m^2 areal power density. This corresponds to a near term expectation of 170 W/kg BOL specific power at the deployed wing level. The combination of SCARLET and SLA technology potentially provides a 50% increase in areal power density and almost a 300% increase in specific power.

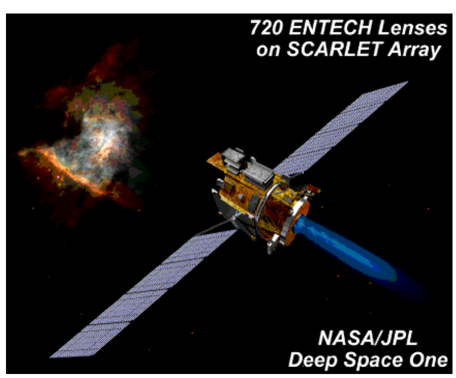

Figure 8-4 Picture of SCARLET Array on DS1 Spacecraft (Courtesy ENTECH)

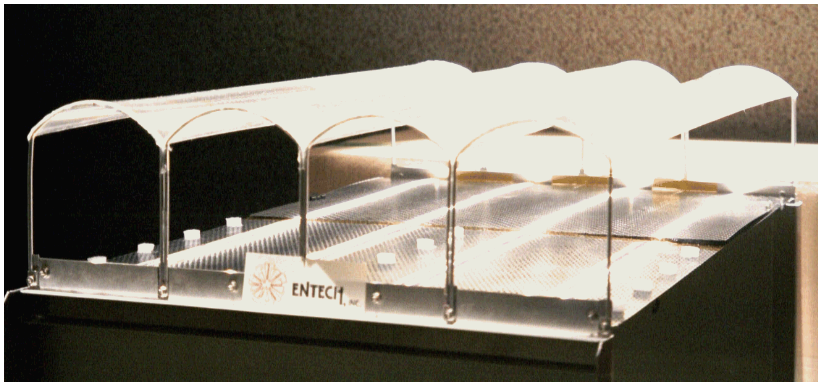

Figure 8-5 Stretched Lens Array Module Prototype (Courtesy ENTECH)

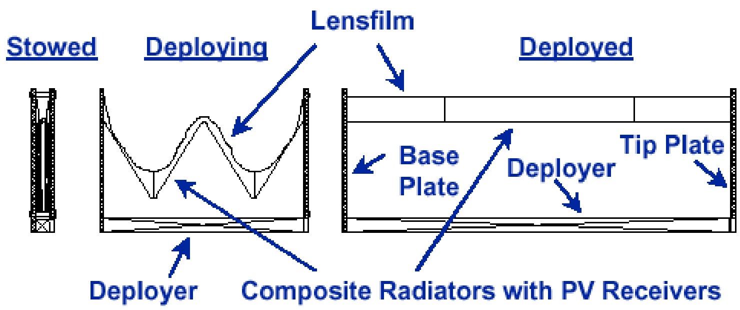

Figure 8-6 SLA Deployment Concept (Courtesy ENTECH)

Table 8-6 Aurora Array Performance

Beginning of Life (BOL) Performance Parameters	
Average Cell Efficiency at 8 Suns and Room Temperature (Demonstrated)	30%
Average Cell Efficiency at 80C (GEO Operational Temperature)	26%
Lens Efficiency (Demonstrated)	92%
Cell-to-Panel Packing Factor	95%
Wiring/Mismatch Factor	95%
Operational Array Efficiency (Product of Last Four Values)	**22%**
Areal Power (W/sq.m.)	**296**
Areal Mass (kg/sq.m.)	**1.74**
Specific Power (W/kg)	**170**

8.2.2.1.2 Solar Cell Technology

JX Crystals projects 32-35% cell efficiency in 10 years. This is a conservative estimate since they have already achieved 30% with non-optimized cells [L. Fraas et al., "30% Efficient InGaP/GaAs/GaSb Cell-Interconnected-Circuits for Line-Focus Concentrator Arrays," 28th IEE-PVSC, 2000]. The following figure shows a typical stacked cell set. The top picture is a completed InGaP/GaAs/GaSb circuit, the middle is a circuit with GaSb IR cells and the bottom is the substrate with metal traces. These overlay each other to form the integrated cell.

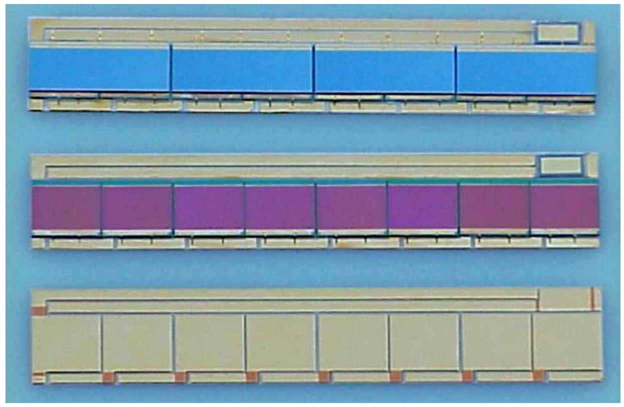

Figure 8-7 Example of a Stacked Cell Set (Courtesy JX Crystals)

Mechanically stacked cells have been assumed due to the present uncertainty over the feasibility of 4 junction photovoltaic cells [P. Iles, "Future of Photovoltaics for Space Applications," Progress in PV Research and Apps. 8, 39-51, 2000]. Other selection criteria were the lower projected cost of the mechanically stacked cells and the ability to electrically connect the stacked cells to take advantage of the larger currents produced by the bottom cells.

8.2.2.1.3 Integrated Solar Array Design and Costs

Discussions with AEC-ABLE, ENTECH and JX Crystals indicate that there is a high likelihood of significant mass reduction to achieve 600 W/m^2 and 340 W/kg in 10 years. For the purposes of this study we have derated these numbers to 450 W/m^2 and 250 W/kg. An issue to be discussed is the cost of arrays. Two elements of array cost are the cost of the cell itself and the cost of the mechanical/optical systems.

Cost of a mechanically stacked cell can be assumed to be slightly higher than the cost of a triple junction cell. For simplicity we will assume a factor of 1.2 times the cost of a triple junction cell or about $300/W in year 2000 dollars. The mechanically stacked cell consists of an epitaxially formed double junction (InGaP on GaAs) cell stacked on top of a diffused single junction gallium antimonide (GaSb) cell. The double junction cell is slightly less expensive than a triple junction cell and the diffused GaSb cell is much less expensive than an epitaxial cell. Each cell has two wiring connections. The upper cells are wired in parallel since they are high voltage, low current cells. The lower cells are wired in series since they are low voltage, high current cells. There will be a different number of each type of cell in order to match voltages. At the end of each voltage balanced module the two wire strings can be connected. This module then is a natural size for application of a stretched lens 15-times optical concentrator and attendant thermal radiator. Due to optical losses and packing considerations, the final concentrator array can be assumed to achieve about a 12-times reduction in required cell aperture area. This gives a 12-times factor for reduction of cell costs along with attendant stringing costs. The reduction in stringing costs is improved since it can be done with rugged automated wire bonding machines rather than fussy cell bonding machines. The final cost savings on cells would be only a factor of 10 since the stacked cells are more expensive. The expense of bonding all of the strings onto a module in a series of large areas is reduced to that of mechanical assembly and wiring the separate modules together. There may be a cost savings here but it is hard to quantify at this time. Cost of the stretched lens concentrator and thermal radiator is less than that of an equivalent area of solar cells. The total area of array required is reduced by a factor of 25/30 which is the ratio of the operating efficiencies at GEO of a triple junction cell to that of the stacked cell. The net result of all of these changes is to produce a final cost per watt of an Aurora type space array of about $700/W. Additional savings might be realized from the large size of the array but this can not be easily quantified especially 10 years into the future.

8.2.2.1.4 Planetary Surface Solar Arrays

Surface operations requiring solar array power are contemplated on the Moon, Mars, and Phobos. These locations can be serviced with essentially the same technology – at least as a first approximation. The types of solar arrays that can be placed on a planetary surface include a) a rigid structural space or terrestrial solar array modules, which can be oriented at a fixed angle or pointed in one or two axes to track the sun or b) a low-cost, lightweight, flexible, thin-film solar array that can be laid out on the surface over large areas. We have selected the lightweight, low-cost approach.

An additional consideration for surface array operation on natural satellites or planets might be the inclusion of a electrostatic or mechanical robotic dust removal system or "Dustbot."

8.2.2.1.4.1 Near Term Technology

Planetary (Mars) and satellite (Moon and Phobos) surfaces provide a fixed surface for mounting a photovoltaic array and thus an opportunity to reduce mass by eliminating most of the structure. Due to planet rotation, the solar angle on the array changes continuously and is almost never normal to the surface of the cells. Thus the size of the array must account for the varying solar angles. In addition, operations (including ISRU), duty cycles and capacities must be designed to respond to the varying solar energy input. At the Lunar poles the Sun is almost always at the horizon. A suitable location for a large surface mounted array must be found that will allow sufficient solar illumination. The Lunar South Pole has a mountainous region where an array could be positioned at an angle of 40° to the horizontal. Significant oblique solar illumination will require several times the array area as opposed to when the sun is normal to the array.

A United Solar press release on their web site [http://www.ovonic.com/unitedsolar/uninews] claims that their thin-film triple-junction amorphous silicon (see following figure) modules on the MIR space station have a specific power greater than 500 W/kg (2 kg/kW).

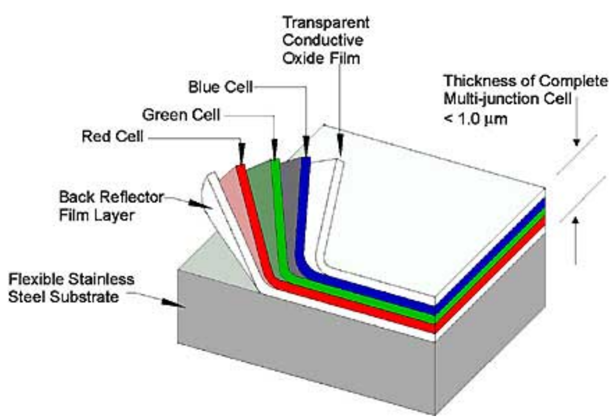

Figure 8-8 Triple Junction a-Si Cell

The United Solar modules were a special run with the cells deposited on a thin (between 0.5 and 1 mil thick) stainless steel substrate rather than the standard 5mil substrate. Neither areal power or conversion efficiency is noted in the press release. If the stainless steel substrate were 0.8 mil thick and no other mass is assumed then the conversion efficiency can be calculated at 5.9%. This low efficiency imposes a large supporting structure mass penalty on a spacecraft but not on a fixed ground installation.

Iowa Thin Film Technologies (ITFT) is producing single-junction amorphous silicon modules on a 5-mil thick polymer (Kapton) substrate. The ITFT modules operate at 5% efficiency and have been made on special order on 2-mil thick Kapton. These special modules had a specific power of about 650 W/kg (1.54 kg/kW).

8.2.2.1.5 2010 Technology

Thin film photovoltaic cell systems of amorphous silicon, cadmium telluride, copper indium diselenide (CIS), and copper indium gallium diselenide (CIGS) are all being developed for application on flexible substrates like thin polymer films. One near-term technology – triple junction amorphous silicon solar cells - has achieved 10% efficiency for long lifetime and their future is very promising. When compared with conventional crystalline solar cells, thin film solar cells have several advantages for planet surface operations application. Advantages include physical flexibility afforded by their thin-film construction, ability to be fabricated into monolithic cell strings and good resistance to radiation.

NASA has studied advanced thin-film, flexible solar array systems for Lunar and Mars surface applications utilizing amorphous silicon solar cells for flexible solar array blankets [Colozza, A. J., *Design and Optimization of a Self-deploying PV Tent Array*, NASA CR 187119, June 1991]. The areal density of these planetary solar array systems was stated to be about 20 g/m^2. This represents an array using a polymer substrate less than 1-mil (25 um) in thickness and has little or no allowance for wiring and probably no allowance for any deployment or attachment hardware. A more realistic estimate would be about 4 times higher mass or 80 g/m^2 for a 2-mil substrate, wiring and minimal hardware. CIGS technology is very likely to produce AM0 (air mass zero – space) cells on polymer substrates with about 14% conversion efficiency in 10 years. Using the 80 g/m^2 and 14% figures gives a calculated specific power of 2400 W/kg (0.42 kg/kW). This value needs to be derated to account for packing factor and operating temperature. Using estimates of 0.9 for packing factor and 0.9 for temperature derating yields a more reasonable specific power of 1920 W/kg (or about 0.52 kg/kW).

A design value of 1 kg/kW is appropriate for this study, which includes stakes, tie downs, robotic dusters and other necessary surface deployment and maintenance hardware. A much more conservative specific power of 250 W/kg (4 kg/kW) has been assumed in the calculations of surface power mass requirements.

8.2.2.2 Nuclear Power Generation

Two types of nuclear reactor systems were evaluated for the Astrotel architectures, namely, surface systems, for use at the Mars Base and at resource utilization sites, and space systems to provide power for the Astrotel itself and the cargo vehicles that utilize electric propulsion systems. Figure 8-9 displays the mass and cost of space reactors as a function of power level.

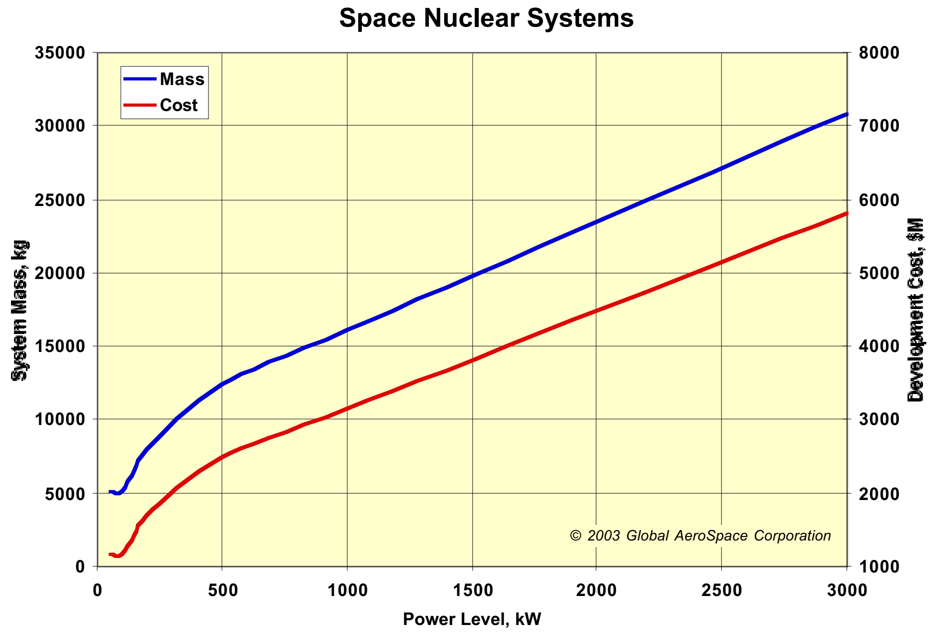

Figure 8-9 Space Nuclear Reactor System Mass and Cost as a Function of Power Level

The primary reference for these data comes from a paper entitled "Liquid Metal Cooled Reactor for Space Power" by Weitzberg presented at Space Technology & Applications International Forum (STAIF) 2003. The scaling is not reliable for power levels below about 100 kW. Figure 8-10 displays similar information for surface reactors.

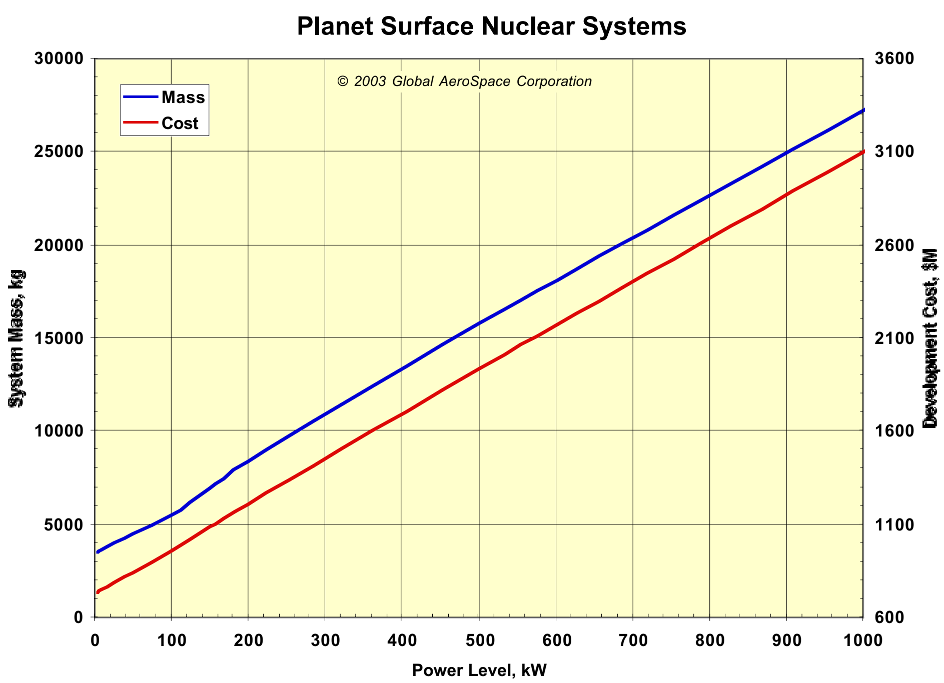

Figure 8-10 Surface Nuclear Reactor System Mass and Cost as a Function of Power Level

The surface scaling comes primarily from a paper entitled "Design Concept for a Nuclear Reactor-Powered Mars Rover" by Elliot, Lipinski, and Poston also presented at STAIF 2003.

8.2.2.3 Energy Storage

Energy storage requirements are in two categories, namely the need for power for small, non-solar-powered spacecraft like Taxis and power storage for solar-powered space and surface systems during nighttime or eclipse operation. Since photovoltaic energy conversion requires sunlight, the "fixed" base solar-powered space (Astrotels and Spaceports) and surface (resources plants) systems must have an energy storage system to handle nighttime or eclipse power needs. In most surface system cases, limiting major operations to the daytime hours minimizes nighttime energy requirements. Nighttime requirements are generally keep-alive power in order to keep electronics from failing at cold temperatures. In the case of power for small spacecraft like Taxis, small, high energy density energy storage systems are needed. In addition, they must be either rechargeable or refillable at a transportation node.

8.2.2.3.1 Energy Storage Options

The types of energy storage that can produce electricity directly are fuel cells, ultra capacitors, and batteries. For larger systems flywheels sometime can also become competitive in energy density with some types of batteries. Flywheel storage has not been widely used and has a serious limitation with friction losses when the storage period is long term. High capacity capacitors still do not have an adequate power density for space use. This reduces the energy storage options to fuel cells and secondary batteries.

8.2.2.3.1.1 Fuel Cells

Fuel cells using hydrogen and oxygen are used in the Space Shuttle and are a proven technology. Standard fuel cell energy conversion systems require pressurized tanks for both fuel and oxidizer. Estimated specific power for a fuel cell is 700-1000 W-hr/kg [Nesmith, W., DOE HQ, Personal communications, October 1997]. New technology options include a regenerative fuel cell that contains both a fuel cell and an electrolysis capability in the same unit. Regenerative fuel cells for energy storage have been investigated for application to solar-powered aircraft at energy densities about 400 W-hr/kg including tanks though only about 250 W-hr/kg has currently been demonstrated. For a non-regenerative system, oxygen and hydrogen for fuel cell operation can be generated and stored at a surface based facility to refuel the cells periodically. Space energy storage might benefit from the use of regenerative fuel cells where the hydrogen and oxygen are reacted to form water when power is needed. Later when surplus power is available the water can be broken down by electrolysis into hydrogen and oxygen.

8.2.2.3.1.2 Secondary Batteries

The types of secondary batteries that should be considered are nickel-hydrogen, nickel-metal hydride, lithium-ion, and lithium-ion polymer. These presently have the approximate energy densities (W-hr/kg) shown (cell size and packaging dependant):

Table 8-7 Battery energy densities, W-hr/kg

Nickel-hydrogen	60
Nickel-metal hydride	80
Lithium-ion	100
Lithium-ion polymer	130

Nickel-hydrogen has been included because of its continuing outstanding performance in space. Information is available that shows performances of 60,000 cycles at 60% depth-of discharge (DOD). However, the nickel-hydrogen battery basic energy density really has not improved in some time. Some improvements have been obtained in smaller batteries with common (2 cells per pressure vessels) and single pressure vessels (all cells in one). Nickel-metal hydride promises the volume of nickel-cadmium with an energy density exceeding nickel-hydrogen. Larger cell sizes are becoming available. Lithium-ion batteries are being made in larger sizes and they may even exceed the 100 W-h/kg given. Battery charging has been somewhat critical with successful methods using bypass circuits across each cell to prevent overcharge on a cell level. A concern is that high cycle life has not really been demonstrated. Additional data shows a loss of capacity with cycling that seems to level out after about 500 cycles. Lithium-ion polymer is really a lithium-ion battery with a different kind of separator and packaging. The comments on lithium-ion charging and capacity loss also apply to the lithium-ion polymer battery.

At this time, the lithium-ion polymer appears to be the best candidate for space Astrotel use because of projected improvements in energy density and the low cycle life requirement. For Astrotel space missions the number of cycles is generally low since batteries would essentially be standby power or used during eclipses by a planet or natural satellite. Table 8-8 summarizes lithium-ion polymer cell characteristics.

Table 8-8 Typical Lithium-Ion Polymer Cell Characteristics

Energy density:	125-140 Wh/kg
Operating/storage temperature range	-20°C to + 60°C
Charge conditions	C/2 max. to 4.2V max. (0 to 40°C)
Discharge voltage	4.0 to 3.25 V (3.7typ @ 25°C)
Typical discharge rate	(C/2) [C – cell capacity]

For the 2010 technology horizon assumed, available cells of at least 25 Ah and energy density of at least 200 W-h/kg should be available.

8.2.2.3.2 Energy Storage Requirements and Assumptions

The following table describes the energy storage requirements and assumptions for various systems of the Mars transportation architecture.

Table 8-9 Mars Transportation System Energy Storage

System	Power kW	Duration days	Energy kWhr	Storage Media	Energy Density kWhr/kg	Mass kg
Taxi	10	10	2400	NRFC LOX/LH	0.7-1.0	2400-3430
Mars Shuttle	10	2	480	NRFC LOX/LH	0.7-1.0	480-690
Astrotel	10-30	0.33	80-240	Li Ion Polymer	0.2	400-1200
Surface (Mars)	1.0	0.75	180	Li Ion Polymer	0.2	900
Surface (Mars)	1.0	0.75	180	NRFC LOX/LH	0.7-1.0	180-260

NRFC – non-regenerative fuel cells

8.2.3 Radiation Protection

Interplanetary radiation includes high energy Galactic Cosmic Radiation (GCR), which is very difficult to protect against, and solar flare particle events (SPEs), which consist mostly of high-energy protons. GCR is continuous while SPEs only occur during major solar storms. SPEs can last from minutes to hours. It is necessary to provide protection against SPEs because this radiation can be quite harmful and can cause death for unprotected humans. A major SPE, had it occurred with Apollo astronauts on the moon, would likely have killed them. The effect of GCR is expected to result in a small increased risk of cancer over the crew times usually considered. The following figure describes one model for the shielding required of various materials in order to reduce cell damage (transformation) by a particular amount for a one year exposure (in NRC report on Radiation Hazards to Crews of Interplanetary Missions, 1996).

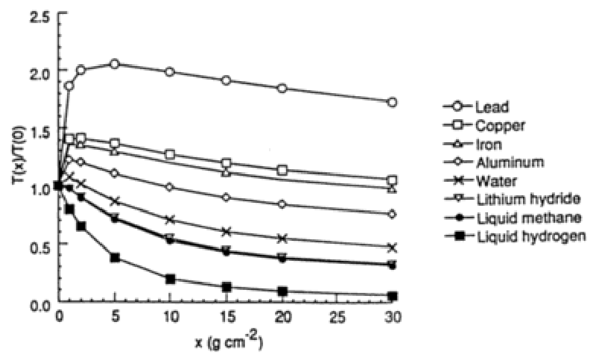

Figure 8-11 Required Shielding of Various Materials

The effect of high Z materials for shielding against radiation is counter-intuitive. Lead shielding actually increases the dose of damaging radiation as compared to no shielding at all. This increase is due to more particles being generated as the result of collisions in the shielding material. Very thick layers of lead are required for any protection at all. In discussions with SAIC personnel at JSC, current reference mission planning assumes the use of available water and water-bearing food stuffs in the transfer vehicle cargo for protection. In the future, there may also be consideration of an onion skin approach to shielding materials, which could be added over time. Substantially shielding of crew sleeping quarters, where crews will spend a significant amount of their time, can significantly reduce the overall GCR dose. Liquid hydrogen is a very good shielding material for SPEs, a 30-cm thickness reducing the cell damage by an order of magnitude below unprotected cells. Of course water is easier to store at room temperature, but it only reduces cell damage by 50% with the same thickness of shield. It is clear that an SPE storm shelter of some kind will be required on the Astrotel because the protection of the entire vehicle will likely be prohibitively massive. The best way to protect the crew against GCR dose is to limit flight times in interplanetary transit to the shortest practical values.

8.3 Common Flight System Elements

There are two flight systems elements that are utilized in multiple flight systems. The crew module is used in both the Interplanetary Taxi vehicle and the Mars Shuttle with a few variations. The Habitability Module, based on the NASA developed TransHab, is used in both the Astrotel vehicle and the Spaceport designs.

8.3.1 Crew Module Development

In Phase II we reviewed the Apollo scaling for the crew module that was used in Phase I for both the Taxi and Mars Shuttle vehicles. As a result of this analysis a redesign of the crew module was carried out.

8.3.1.1 Apollo Scaling

Estimate of the mass of the crew module, based on scaling up the Apollo Command Module from 3 to 10 people, is about 8 mt excluding radiation shielding. The scaling up from Apollo includes a reduction in mass for a number of electronics elements (radios, computers, controls) and modest advancements on structures. We scaled up the volume of the vehicle to account for the additional crew. The mass estimate of this module based on scaling up the NCOS Mars Shuttle vehicle is about 5 mt, a 3 mt difference. This is a large difference that forced us to redesign the crew module in detail before going further in defining more of the Mars Shuttle or Taxi system details. The following table summarizes the crew module scaling from Apollo.

Table 8-10 Crew Module Scaling from Apollo

System Element	Apollo	2002 Technology w/o propulsion	Crew Module 3->10 Crew	Crew Module w/o Aeroshell
		smaller, lighter elect & radios	Expand cone, retain same m3 per crew	Eliminate Apollo aeroshell mass
Apollo Command Module				
Structure	1,567	1,000	2,571	2,571
Aeroshell	848	490	1,110	-
RCS	400	400	400	400
Recovery Equip	245	-	-	-
Nav	505	100	100	100
Telem	200	100	100	100
Elect	700	100	100	100
Comm	100	50	50	50
Crew Accom	550	400	1,333	1,333
Crew Mass	216	245	818	818
Misc	200	200	200	200
ECLS	200	200	667	667
Propellans	75	75	150	150
	5,806	3,360	7,599	6,489
Aeroshell to module mass ratio	0.17	0.17	0.17	-
Apollo Service Module				
Structure	1,910	1,500		
Electrical	1,200	1,000	2,000	2,000
Propulsion	3,000			
Propellants	18,413			
	24,523	2,500	2,000	2,000
Total Mass	**30,329**	**5,860**	**9,599**	**8,489**
Crew	3	3	10	10
Height, m	3.5	3.5	4.8	4.8
Radius, m	2.0	2.0	2.7	2.7
h/r ratio	1.8	1.8	1.8	1.8
Volume, m^3	13.9	13.9	35.8	35.8
Crew volume, m^3/crew	2.06	2.67	2.06	2.06
Hab Volume, m^3	6.2	8.0	20.6	20.6

8.3.1.2 Crew Module Redesign

As a result of the Apollo scaling difficulties, we redesigned in detail the crew module that is used in both the Taxi and Mars Shuttle vehicles. Detail estimates of structure, crew accommodation, radiation shielding and life support masses were developed. The basic cylindrical design is retained, however, the cylinder is shortened and volume within the module identified as to its use, i.e. crew or equipment. The crew module is cylindrical in shape and surrounded by about 3-cm thick polyethylene radiation shield to protect the crew during major solar particle events. We also consolidated the rotating "***g-hammocks***" to one side of the cylinder to provide more contiguous volume for equipment. Average crew volume is still 2.1 m^3 including the g-hammock itself at 1.65 m^3 plus community volume in the center of the cylinder. At least two crew members will have additional Taxi control and monitoring equipment adapted to their g-hammock. Crew hammocks rotate in order to accommodate the varying acceleration vector during aerocapture maneuvers and very different acceleration vectors during staged propulsive rocket burns. Maximum g-load is nominally 5. We also moved the airlock entirely out of the main cylinder. A separate Utility Module attaches to one end of the Crew Module and contains life support and power subsystems. The utility Module is analogous to the Service Module of Apollo, but without propulsion elements.

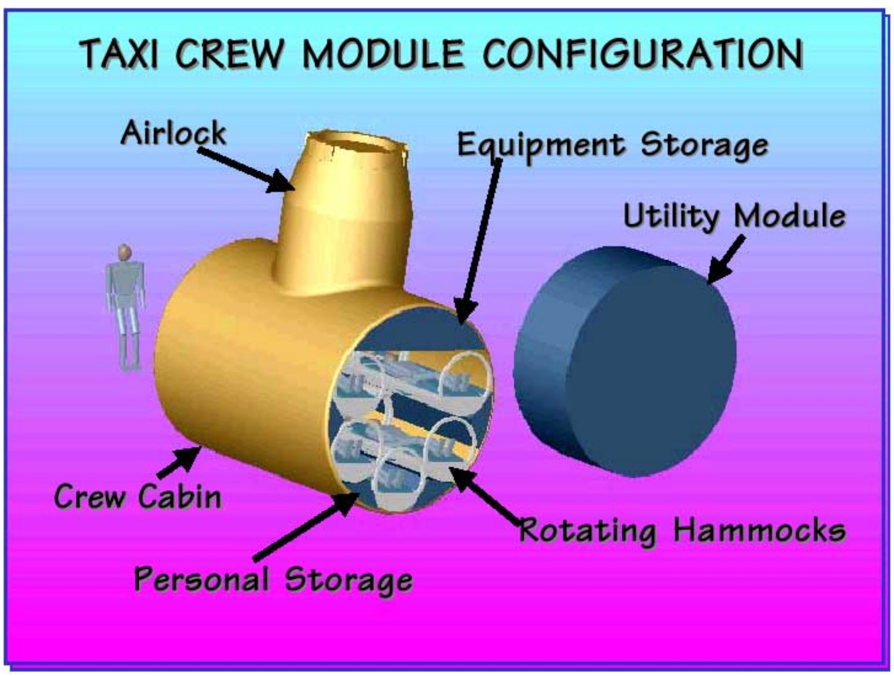

Figure 8-12 Common Crew Module Design

Table 8-11 displays the current mass estimate for both the Mars Shuttle and Taxi Crew Module and it includes estimates for refurbishment mass. The major differences between the two crew modules are the radiation shield, more environmental control and life support systems (ECLSS) mass and additional fuel cells in the Taxi Crew Module, since flights of up to 7 days are planned, and the second airlock in the Mars Shuttle Crew Module to allow easy crew egress once on the Martian surface. All other systems are identical.

Table 8-11 Common Crew Module Mass Estimates

System Elements	Mars Shuttle Crew Module	Mars Shuttle Refurb Mass in 15 years, %	Mars Shuttle Refurb Mass, kg	Taxi Crew Module	Taxi Refurb Mass in 15 years, %	Taxi Refurb Mass, kg
Crew Cabin						
Structure	1431	5%	72	1431	5%	72
Airlock plus Tunnel	810	5%	41	0	5%	0
Insulation, 30mm	188	100%	188	188	100%	188
Nav	100	100%	100	100	100%	100
Telem	100	100%	100	100	100%	100
Elect	100	30%	30	100	30%	30
Comm	50	100%	50	50	100%	50
Crew Accom	694	30%	208	694	30%	208
Crew Mass	818	0%	0	818	0%	0
Misc	200	50%	100	200	50%	100
Radiation Shield	0	0%	0	1830	0%	0
Subtotal	4490		888	5510		847
Utility (Service) Module						
ECLSS	121	100%	121	497	100%	497
Electrical	171	75%	129	1200	75%	900
Subtotal	292		250	1697		1397
Total Mass	**4783**		**1137**	**7207**		**2244**

8.3.1.3 Life Support

The ECLSS mass estimates come from the following assumptions shown in Table 8-12. The assumptions are a 10-person crew and a 7- and 1-day trip times for the Taxi and the Mars Shuttle respectively.

Table 8-12 ECLSS Mass Estimates

Atmosphere Management	Type	Mass one canister, kg/4p-d	10p (min 3 canister-d)	Taxi	Mars Shuttle
CO2 removal	LiOH	7	21	147	21

Water Management (wash, flush, urine)	Mass, kg/p		Taxi	Mars Shuttle
Multifiltration (drinking, wash)	10		100	100
VCD (Urine, flush)	25		250	0

Total Required ECLSS Mass	Taxi	Mars Shuttle
	497	**121**

Notes: p=people
d=day

The Taxi radiation shield assumes a thickness of 3 cm of polyethylene covering the entire outer shell. The radiation shield is 1830 kg. Because the exact timing of the Mars Shuttle trips can be allowed to vary and because the trip times are measured in hours there is no radiation shield mass allocation.

8.3.2 Habitability Module

The TransHab Module is an inflatable home in space that was initially developed for the International Space Station (ISS). We assume a modified TransHab Module is used for the Astrotel crew habitation. In addition, it has been considered as a habitability module for future human missions to Mars and as a possible hotel for tourists to visit in Earth orbit. As currently designed the TransHab is a home to a crew of 6 astronauts on board the space station, which includes sleeping compartments, food preparation and eating facilities, windows, exercise gym and food storage areas. The concept for TransHab originated in 1997 at NASA JSC during studies of future Mars missions. Development has continued and has included vacuum chamber testing in late 1998. See http://spaceflight.nasa.gov/station/assembly/elements/transhab for more information.

8.3.2.1 TransHab Design

The TransHab is an inflatable structure so as to allow it to be launched in a smaller stowed volume and inflated once on orbit. This approach enables a much larger volume per crew than otherwise would be available from rigid structures that can be launched into space. The 30-cm thick shell is composed of several layers of differing materials to provide the maximum protection to the crew from orbital debris and meteoroid impacts. The multi-layer meteoroid and orbital debris (MOD) design facilitates particle break up before penetration of the envelope. The shell consists of several sheets of meteoroid and orbital debris shielding, a Kevlar restraint layer and several layers of redundant pressure-retaining bladders. The length of the TransHab is 11 m including pressurized air-lock and equipment tunnel and its diameter after inflation is 8.2 m. The total enclosed volume is about 340 m^3. At launch, as designed for the ISS, it is 13.2 mt. The central core of the TransHab is a lightweight structure made from carbon-fiber composite materials. This structure provides the base for the three floors and several compartments. A central tunnel provides access between the three floors of crew space and the pressurized air-lock to the docking port. Extendable floors are unfolded and erected after shell inflation. In the ISS design, an integral water storage volume surrounds the crew sleeping quarters to provide protection from high-energy charged particles. A pressurized docking cone is also included on one end of the ISS design.

Figure 8-13 illustrates the ISS TransHab with crew. At the very top of this figure is the pressurized tunnel. Next down is Level 3 which is the Crew Health Care floor that contains exercise equipment, exercise area and soft material stowage space. Level 2, next below, is the crew quarters and environmental controls and life support equipment areas. The crew quarters are individual spaces surrounding the central access tunnel. Surrounding the crew quarters is an integral water tank that provides additional protection to the crew against high-energy particle impacts while they are sleeping. Level 1 is where the galley and wardroom are located, which includes food storage, preparation, eating and clean up facilities. At the very bottom of the

TransHab, as seen in this view, is the unpressurized "tunnel" where the inflation systems and tanks are contained.

Figure 8-13 TransHab Design Cutaway (courtesy NASA/JSC)

8.3.2.2 Adaptations for Astrotel and Spaceport Application

Several adaptations will be required for use of the TransHab design for the Astrotel and Spaceport concepts. Some of the more obvious modifications will likely include:

- Expansion of crew quarters from 6 to 10. This may require reducing the currently available space or carving out new space from the health or galley areas.

- Expanding the radiation protection volume to include most of the volume occupied by the crew. This could include an internal bladder inside the current inside dimension of the TransHab or it could include filling one or more of the current outer shell bladders or MOD volumes with protective material such as water or polyethylene.

- Replacement of the unpressurized "tunnel" with another docking port for attaching pressurized cargo bays. Once the system is inflated, this equipment is not necessary.

Inclusion of a command and control area, perhaps replacing storage areas, to provide space for electronics and interplanetary communications systems.

8.4 Astrotel Concept

The Mars transportation system architecture concept uses small, highly autonomous space ships, we have dubbed *Astrotels*, for transporting humans to and from Earth and Mars on cyclic or near-resonant orbits between these planets. Human flight time each way is reasonably short, between 5 and 6 months.

8.4.1 Astrotel Design Description

Key elements of these ships are that they are highly autonomous and transport only human and other high value cargo, use highly efficient solar electric propulsion, and not require artificial gravity. The 70 mt mass includes a habitability module for a crew of ten. The size and volume of this system would provide a crew volume of about 6 times that available to today's Space Shuttle crew (7). The astronaut living space is a three-story structure patterned after the *TransHab* modules which have been studied by NASA (see Section 8.3.2). Figure 8-14 is a schematic of one concept for an Astrotel that is approaching Mars. The two smaller modules between the TransHab and the solar array are cargo bays. The Astrotel Cargo Freighter autonomously delivers all cargo to the Astrotel contained within a standard cargo bay. These are pressurized modules to facilitate crew unloading of consumables and RRU hardware. Once emptied the cargo bay could be discarded or used to provide added crew volume.

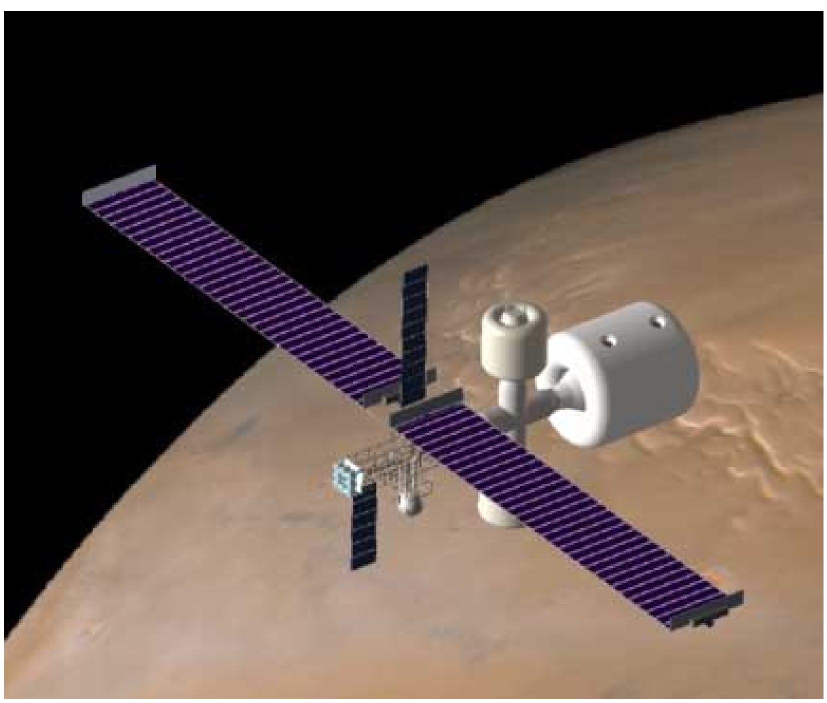

Figure 8-14 Computer Design of one Concept for an Astrotel

8.4.2 Astrotel IPS Module

During three orbits out of seven in 15 years, the Astrotel orbit will need to undergo modification by use of its ion propulsion system. The low thrust analysis presented earlier optimized the maneuvers by timing them for minimum propellant usage, not necessarily near the optimum impulsive delta-V at the orbit aphelion. In addition, since propellant mass has a cost associated with it, the focus is on the maximum payload cases, in particular the 150 kW, 5000 s I_{sp} case. This case was best achieved by an ion propulsion system. The required propellant supply averages only about 400 kg per cycle to meet the 2767-kg propellant required for 15 years of corrections. Xenon has the advantages of being inexpensive, easy to store, and having considerable ion propulsion experience. A 644 kg, Xenon propellant, ion propulsion system (IPS) thruster system is included in the Astrotel design. Xenon ion thrusters are situated at one end of the Astrotel in order to facilitate pointing the thrust vector toward the center of gravity of the system. The ion propulsion system is based on the scaling presented in Section 8.2.1.2. The following figure shows the details of the propulsion system including the eight 50-cm engines, radiators, and the xenon propellant tank. Not shown, but located behind the thrusters is the power processing electronics for the ion propulsion system. The thermal radiator assembly for the IPS is shown below the thruster assembly and oriented 90° to the direction to the Sun. The 3 m^3 volume xenon tank is sized to contain all the propellant required for the 15 years cycle.

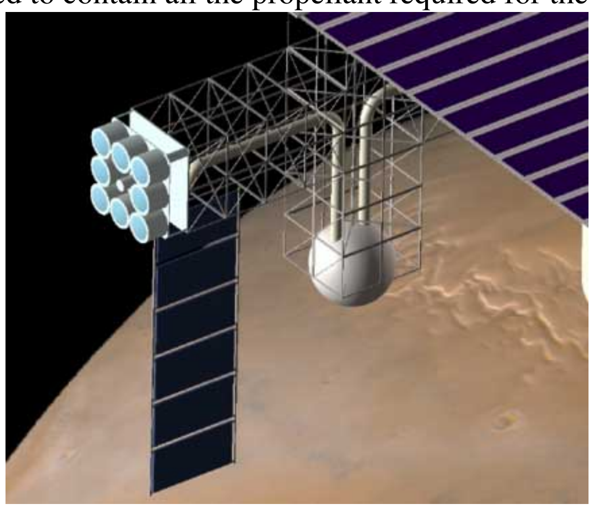

Figure 8-15 Astrotel Propulsion Module

8.4.3 Astrotel Solar Array

As discussed earlier, various options exist for solar array cell selection. At this point we have specified the use of multi-component, mechanically stacked solar cells in a concentrator solar array configuration at a specific power of about 250 W/kg (or ~4 kg/kW). We assumed a 10 kW requirement in addition to the propulsion power requirement for a total array size of 160 kW. During crew habitation the ion propulsion system will usually be off allowing significant reserve power for crew activities. It is estimated that such a solar power generation system will mass about 785 kg and require about 384 m^2 of area, which could be accommodated in two separate deployable array panels each 6-m wide by 32-m long. The solar arrays can be articulated ±90° along their long axis in order to track the sun while orienting the propulsion system thrust vector optimally.

8.4.4 Astrotel Mass Summary

The total estimated Astrotel mass is about 70 mt, which includes about 28 mt of hab module, propulsion/power systems, and cargo storage plus about 32 mt for reserve, radiation shielding and an escape pod. The mass breakdown, in Table 8-13, will be further developed in the proposed Phase II effort as the radiation shielding requirements and the need and requirements for an escape pod are evaluated.

The Astrotel sizing was evaluated based on the revised JSC Mars reference mission systems designs. It is based on the JSC Reference Mission Version 3.0 (JSC Adv. Dev. Off. Report #EX13-98-036, June 1998) Earth Return Vehicle sizing. The JSC mass numbers were combined with the 160 kW power and IPS SEP subsystem. The JSC Mars reference mission consumables numbers for the Earth return vehicle were used though scaled up for a 12-member crew (the baseline crew size is 10, so there is some reserve included). Consumables are 2 kg/person/day for Physical Chemical Life Support and 2.2 kg/person/day for crew accommodation (food, etc.). This is not a totally closed life support system (LSS). The next table is the mass summary for the current Astrotel vehicle.

Table 8-13 Astrotel Equipment and Mass Summary, kg

Subsystem or Item		Dry Mass	Consumables	Subtotal Mass	15-year Consumables Mass	15-year Refurbish-ment Mass
Physical/Chemical Life Support		2,778	3,200	5,978	22,400	1,389
Crew Accomodation		5,000	3,520	8,520	24,640	2,500
Structure		5,500		5,500		
EVA Equipment and Consumables		1,183	446	1,629	3,122	1,183
Communications and Information		320		320		320
Thermal Control		550		550		275
Power		785		785		785
Solar Array	640					
Internal Electrical Power Distribution	100					
Energy Storage	45					
Propulsion		643		643		643
Thrusters	147					
Power Processing Units	291					
Radiators	24					
Propellant Management	59					
Gimbals	44					
Cabling, structure, thermal, DCIU	78					
Attitude Control		500		500		250
Radiation Shielding		9,254		9,254		
Escape Pod and Reserve		22,000		22,000		
Crew		1,000		1,000		
Utility Module Base		5,000				
Permanent Cargo Bay		3,000				
Spares		2,100		2,100		
Total Mass		**59,613**	**7,166**	**66,779**	**50,162**	**7,345**

8.4.5 Stopover Cycler Architecture Astrotel

The Astrotel for the Stopover Cycler architecture has several differences with the Aldrin Astrotel. These differences are summarized below:

- More structural mass to support g-loads during high-thrust escape and capture manuevers
- Less power since IPS replaced by chemical propulsion, and
- More consumables due to 180 day flight duration

The following table summarizes the Stopover Astrotel mass breakdown. The dry Stopover Astrotel is about 35 mt heavier than the baseline Astrotel.

Table 8-14 Stopover Astrotel Mass Summary, kg

Subsystem or Item	Dry Mass	Consumables	Subtotal Mass
Physical/Chemical Life Support	2,778	3,600	6,378
Crew Accomodation	5,000	3,960	8,960
Structure	11,000		11,000
EVA Equipment and Consumables	1,183	446	1,629
Communications and Information	320		320
Thermal Control	550		550
Power	265		265
Propulsion	31,138		31,138
Attitude Control	500		500
Radiation Shielding	9,254		9,254
Escape Pod and Reserve	22,000		22,000
Crew	1,000		1,000
Utility Module Base	5,000		
Permanent Cargo Bay	3,000		
Spares	2,100		2,100
Total Mass	**95,088**	**8,006**	**103,094**

8.5 Interplanetary Taxi Concept

Taxis provide transportation between Spaceports and Astrotels in the Aldrin Low-thrust Cycler architecture. In order to minimize propulsive energy use, Taxis use advanced aeroassist technologies for planetary orbit capture. Aerocapture takes maximum advantage of planetary atmospheric drag to slow the vehicle on its approach from planetary space.

8.5.1 Key Sizing Assumptions

The sizing of the Taxi vehicle has been carried out given the following key assumptions:

- Minimal radiation protection (equivalent to ~3 cm polyethylene surrounding crew module) for the crew is provided since transfer times to/from the Astrotels could be <7 days

- No cargo is transported to the Astrotel by the Taxi except crew
- ~10-15% of the entry mass is aeroshell
- LOX/LH propulsion system at Isp of 460 s and thrust of 60,000 lbs./engine
- Fuel cell energy storage, no solar array power source
- Propellant tank augmentation (expendable drop tanks and in some cases additional engines) are required at Mars

8.5.2 Taxi Design Description

The nominal Taxi system aeroshell design is an elliptical raked cone (see Section 6.3). Taxis utilize LOX/LH propulsion to escape planets and place them and their crew onto hyperbolic rendezvous trajectories with the interplanetary orbiting Astrotels. Figure 8-16 depicts an early two-engines version of the Taxi departing the Earth Spaceport at with the Moon in the background. This figure illustrates the crew module, propellant tanks, rocket engines (in their deployed position), and the aeroshell. Propellant capacity of the basic Taxi vehicle is 17.3 mt. Rendezvous time to Astrotels is measured in days in order to reduce the duration of crew time in the expected cramped quarters. Crew volume is comparable to what the Apollo astronauts had on their flights to the Moon and back.

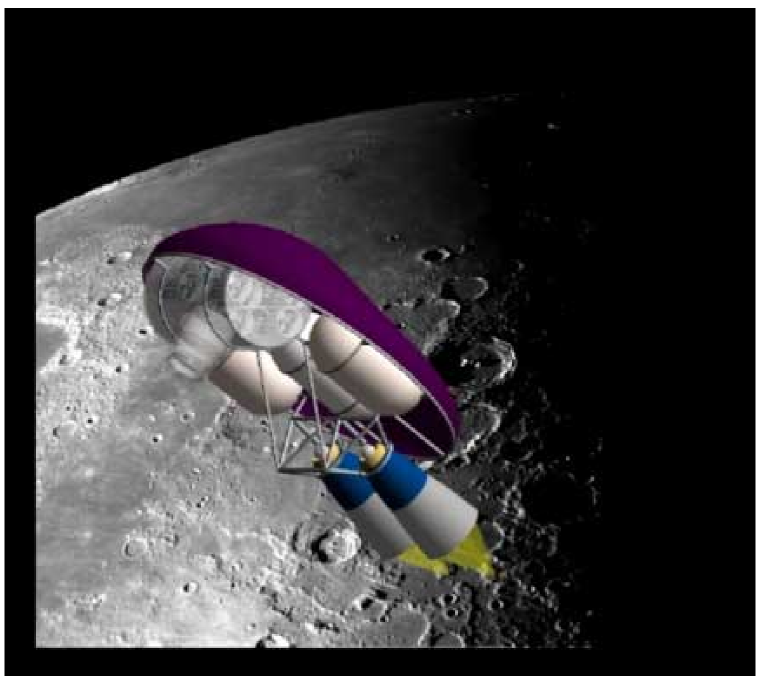

Figure 8-16 Taxi Departing L-1

The following figure is a scale drawing of an early version of the Taxi as it undergoes aerocapture at Mars. The view is as seen from 50 km above Valles Marineris during aero-cruise. Note the rocket engines are in their stowed position. During this time the tanks are almost empty, containing only the propellant necessary to rendezvous with the Mars Spaceport after aerocapture. The crew module is shown in *see-through* mode so one can observe the crew g-seats, which rotate in order to accommodate the varying g-load direction and the quite different thrust direction during propulsive maneuvers.

Figure 8-17 Taxi During Mars Aerocapture

Figure 8-18 illustrates the early two-engine Taxi vehicle docked at an Astrotel.

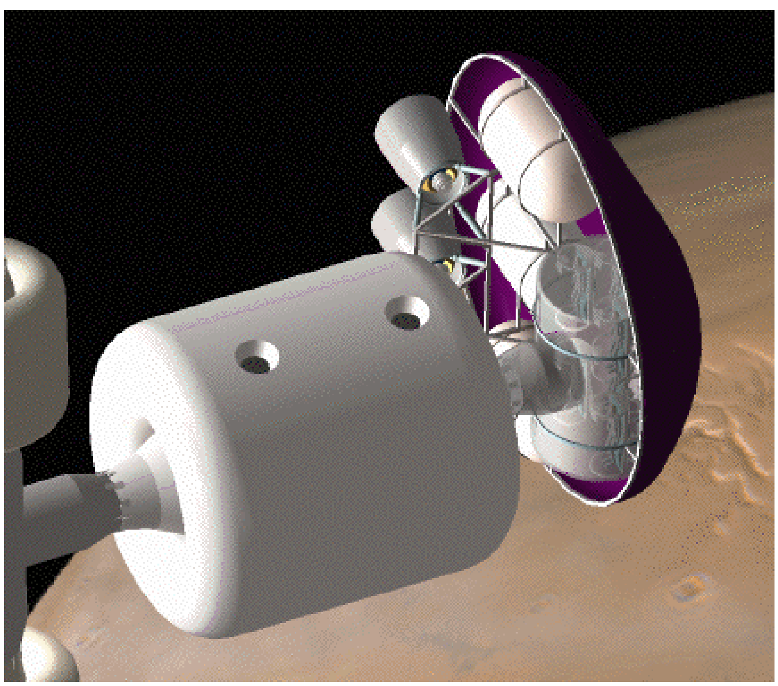

Figure 8-18 Taxi Docked at Astrotel

8.5.3 Taxi Propellant Requirements

The departure delta-Vs at Mars are significantly larger than at Earth due to the higher V-infinity of the inbound (down) cycler orbit as it passes Mars (see Section 5.2). For these larger delta-Vs, the Taxi escape maneuver must occur in stages to be most efficient. Two stages are required for 3 of 7 opportunities, where the delta-V is less than 6.7 km/s. Staging is accomplished with the addition of up to 6.6 mt of expendable propellant augmentation tanks (PATs). Tanks are added to the foreword section of the Taxi in line with the thrust direction of the three RL60-class engines. For the two stage opportunities up to 44 mt of additional propellant must be used. Three stages are required for the other 4 opportunities, where the delta–V can reach up to 10.5 km/s. Staging is accomplished by adding up to 27.3 mt of tanks foreword of the tanks just discussed. These additional tanks hold the 138 mt of propellant in the first stage.

Propellant requirements for the Taxi vehicles have been estimated from the Taxi system design and the delta–Vs generated in Phase I and shown above. The following tables describe the 15-year propellant requirements (kg) and in the case of Mars departures, the additional tanks required for staging due to these large delta-Vs. The propellant requirements will drive the propellant production system requirements at Mars and Earth. For the purposes of this analysis the augmentation tankage was calculated as exact percentages of additional propellant (rubber tanks) as opposed to fixed increments of tank sizes. In actuality, there are fixed tank sizes and each set of tankage will be optimized depending on actual delta-V requirements of each opportunity.

Table 8-15 Up Escalator Taxi Propellant Requirements at Earth, kg

Planet	Date	ΔVin (m/sec)	ΔVout (m/sec)	Total ΔV (m/s)	Mf	Mi	Total Propellant
Earth	13-Nov-11		2,473	3,222	16,037	32,754	16,717
Mars	24-Apr-12	749					
Earth	18-Dec-13		2,490	3,302	16,037	33,345	17,308
Mars	18-May-14	812					
Earth	26-Jan-16		2,508	3,240	16,037	32,891	16,854
Mars	16-Jun-16	733					
Earth	17-Mar-18		2,454	3,175	16,037	32,416	16,380
Mars	9-Aug-18	721					
Earth	3-Jun-20		2,442	3,167	16,037	32,363	16,326
Mars	10-Nov-20	726					
Earth	22-Aug-22		2,482	3,244	16,037	32,915	16,879
Mars	24-Jan-23	762					
Earth	26-Sep-24		2,542	3,284	16,037	33,214	17,177
Mars	14-Mar-25	742					
Earth	1-Nov-26		2,538	2,538			
	Average Mars	749	**Total 15-year Propellant at Earth**				**117,641**

Table 8-16 Down Escalator Taxi Propellant Requirements at Mars, kg

Planet	Date	ΔVin, m/s	ΔVout, m/s	Total ΔV, m/s	ΔV1, m/s	ΔV2, m/s	ΔV3, m/s	Total Mp, kg	Aug Tanks, Mt1, kg	Aug Tanks, Mt2, kg	Total Aug Tanks and Expendable Engines, kg
Earth	22-May-10	928									
Mars	8-Jan-12		6,897	7,825	1,171	3,352	3,302	87,409	3,910	6,606	10,515
Earth	25-Jun-12	929				-			-		-
Mars	15-Feb-14		5,732	6,654		3,352	3,302	61,336		6,604	6,604
Earth	4-Aug-14	922				-			-	-	-
Mars	19-Apr-16		5,506	6,424		3,122	3,302	56,430		5,868	5,868
Earth	15-Sep-16	918				-			-	-	-
Mars	11-Jul-18		4,940	5,858		2,556	3,302	46,011		4,306	4,306
Earth	8-Dec-18	918				-			-		-
Mars	24-Sep-20		7,478	8,397	1,743	3,352	3,302	103,967	6,393	6,606	12,999
Earth	18-Feb-21	918				-			-		-
Mars	9-Nov-22		9,528	10,449	3,795	3,352	3,302	199,516	20,726	6,606	27,331
Earth	30-Mar-23	922				-			-		
Mars	1-Dec-24		8,719	9,648	2,994	3,352	3,302	153,458	13,817	6,606	20,423
Earth	28-Apr-25	928									

Stages	Code
2	(green)
3	(yellow)

Total 15-year Propellant at Mars, kg 708,128 **Tanks, kg 88,047**

Total 15-year Propellant at Mars, mt 708.1 **Tanks, mt 88.0**

8.5.4 Taxi Redesign (Stages and Augmentation Tanks)

Redesign of the Taxi vehicle was carried out to incorporate augmentation tanks required for the Mars departures. In Phase I and early in Phase II, tankage mass has been included in the system masses, however the actual physical design had been deferred. Up to two stages of augmented tanks have been assumed for departing Mars as shown in Table 8-16. The largest stage has even included additional engines in order to minimize finite burn and gravity losses. Guidelines for augmentation tank stage design are:

- No increase in Taxi aeroshell diameter
- Placement of engines to ensure thrust vector aligned with cg and gimbaling minimized
- Engine nozzles retracted if thrusting during aerocapture
- Structural mounts will not pierce front of aeroshell
- Long axis of tanks oriented with thrust direction to maximize attitude stability during thrusting

A design concept was selected that enables the thrust vector to be aligned through the cg of all stages of the Taxi at Mars (see Figure 8-19). The configuration on the right is the three-stage vehicle, fully-load system (243 mt with 200 mt of propellant) required for 4 out of 7 opportunities. The middle figure is the two-stage vehicle that is required for the other 3 opportunities, which masses 84 mt including 61 mt of propellant. The vehicle on the left is the last stage used at Mars and the only configuration required at Earth departure which masses 33 mt fully loaded and includes 17 mt of propellant. Maximum propellant loads of the first and second stage augmentation tanks are 44 and 138 mt respectively. The current design includes a small cryogenic refrigeration system and thermal insulation required for keeping the LOX/LH at cryogenic temperatures. In addition this design includes manifolds to direct fuel and oxidizer to the engines under g-load.

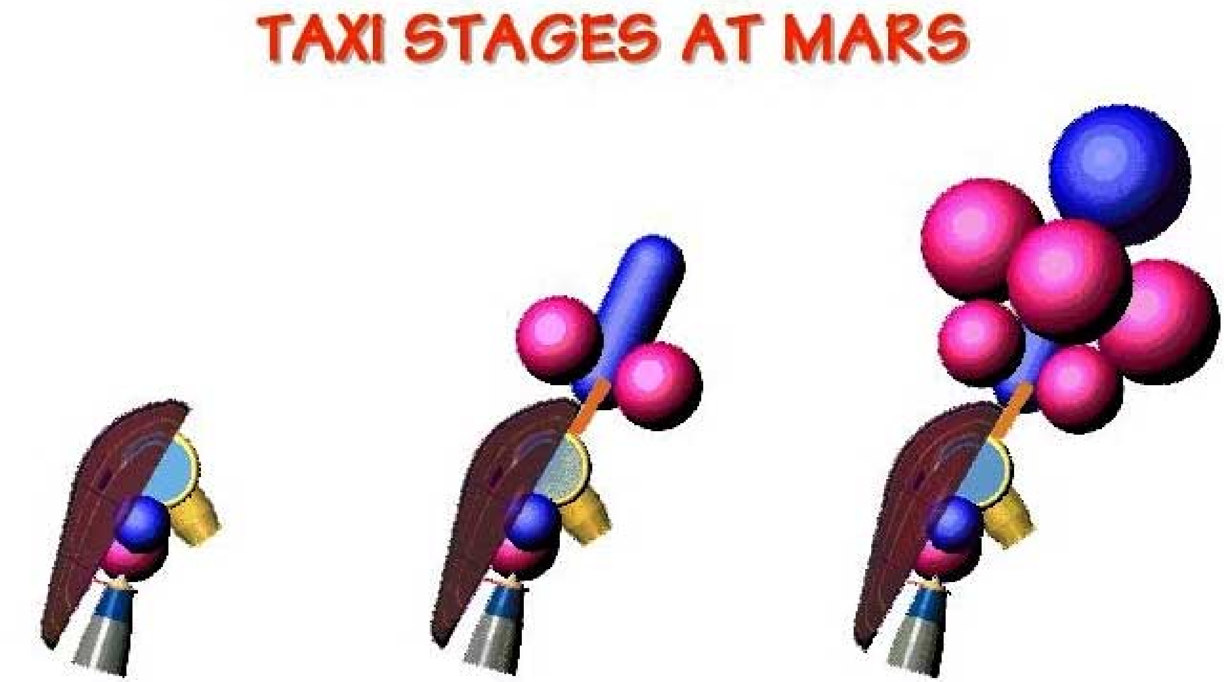

Figure 8-19 Taxi Stages for Mars Departure

The propellant tank configuration (blue = LOX and red = LH) was selected after evaluation of several solutions with the lowest mass of tankage and structure, compactness and simplicity of design in mind. The thrust vector of the engines passes through the vehicle's center of gravity requiring minimum engine gimbaling (within ±5 deg) during operation of all three stages, including 3rd stage with empty tanks. All propellant tanks except the 2nd stage LOX tank are spherical. The 2nd stage LOX tank is cylindrical with hemispherical heads and performs dual functions: besides storing LOX it is used as the main structural element supporting all 1st stage tanks and 2nd stage LH tanks and transferring resulting forces to the 3rd stage. Aluminum tank jackets includes multi-layer insulation (MLI). The 3rd stage tanks (two LOX and two LH tanks) are supported by Aeroshell structure (reinforcing ribs). The shells are reinforced to properly distribute dynamic pressure and concentrated support loads. The 2nd stage LOX tank is connected to two central Aeroshell reinforcing ribs similar to a boat keel. The attachments, besides small bending moment, are loaded with dynamic force resulting from accelerating of 1st and 2nd stages, transferred through 2nd stage LOX tank shell and reinforcing rings. Four 2nd stage LH tanks are attached to front reinforcing ring of the 2nd stage LOX tank by tension members and supported on its rear, reinforcing ring. Tension members are tangential to the respective tank shells. The 1st stage LH tanks are similarly attached to the 1st stage LOX tank and supported on the 2nd stage LOX tank front reinforcing ring. The 1st stage LOX tank, by far the heaviest component of the system, is supported from the 2nd stage LOX tank front reinforcing ring. All the tanks are internally and/or externally reinforced so the concentrated as well as dynamic pressure loads are properly distributed to the shell. Figure 8-20 through Figure 8-21 illustrate in CAD and line drawings, respectively, the propellant augmentation tanks for 1st and 2nd Taxi stages of the three-stage Taxi.

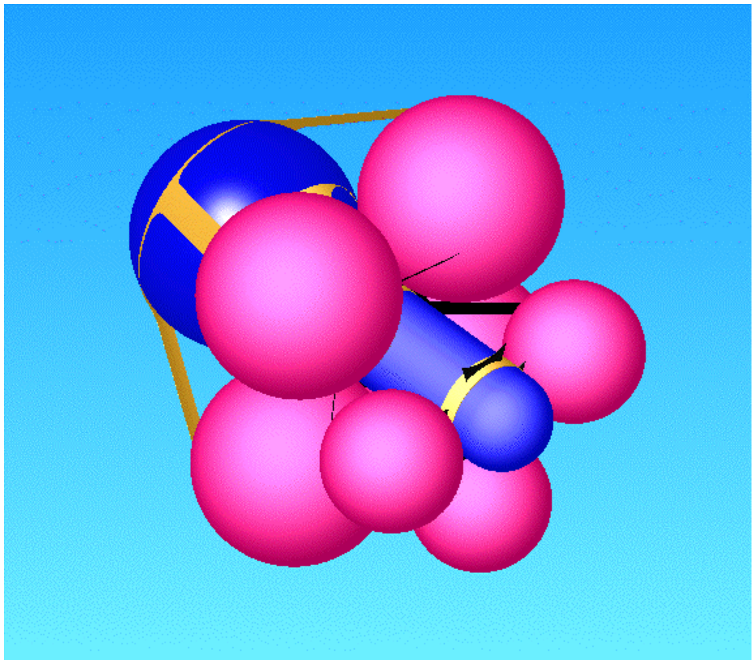

Figure 8-20 Propellant Augmentation Tanks for 1st and 2nd Taxi Stages

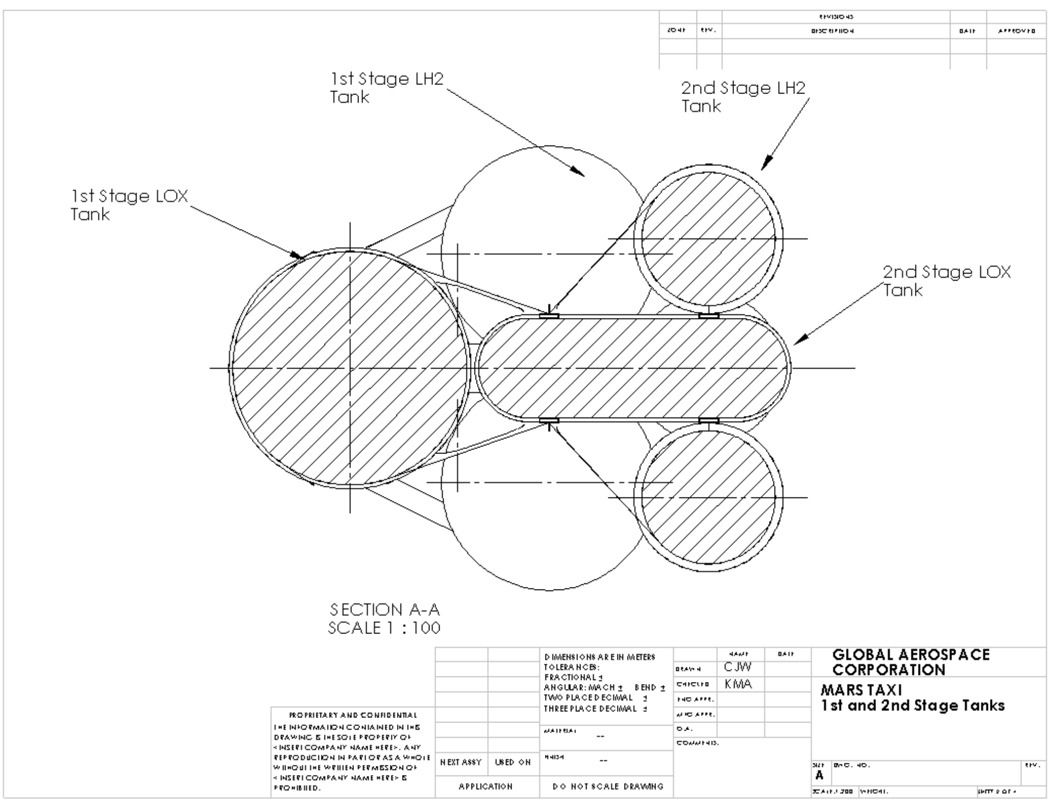

Figure 8-21 Line Drawing of Propellant Augmentation Tanks for 1st and 2nd Taxi Stages

Figure 8-22 to Figure 8-24 illustrate the full three-stage Taxi vehicle in a number of different views. Note the rocket engines are shown in their retracted state.

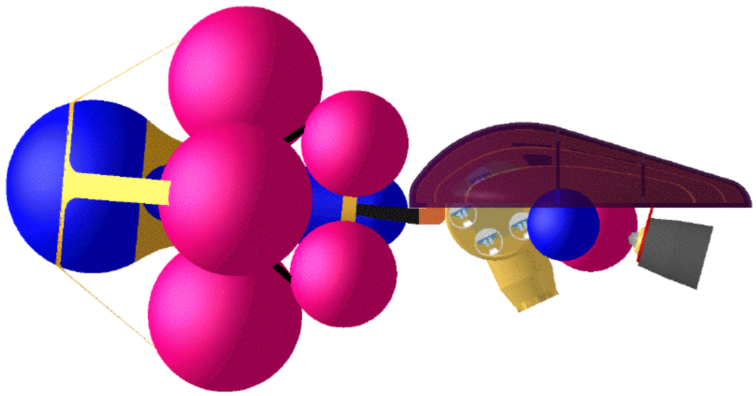

Figure 8-22 Side View of Three-stage Taxi

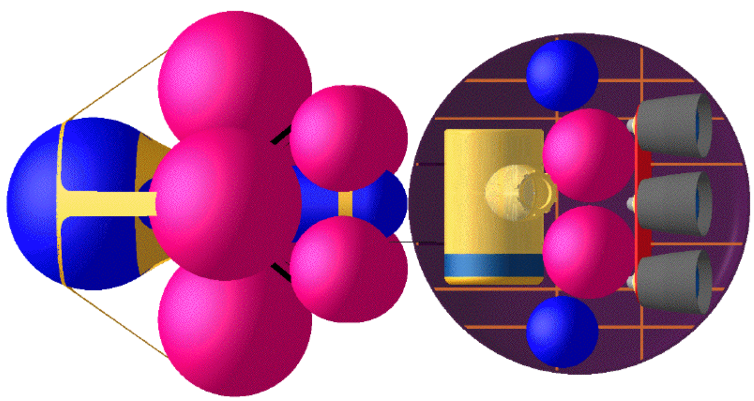

Figure 8-23 Top View of Three-stage Taxi

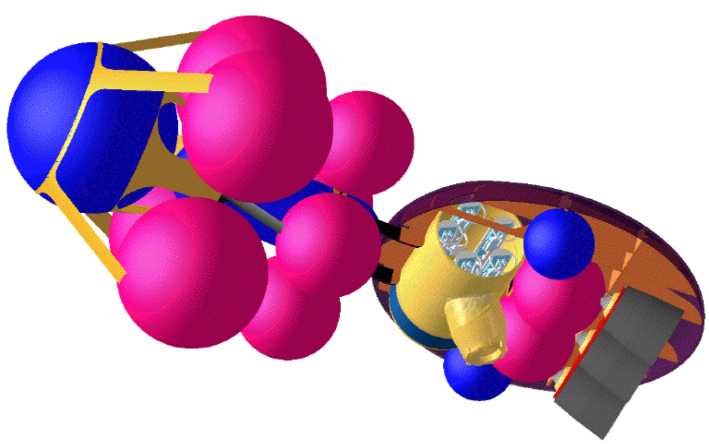

Figure 8-24 Perspective of Three-stage Taxi

The Taxi is propelled with three Pratt & Whitney RL60 engines, rated at 60,000 lbs. each. The engines may be gimbaled ±5°, both vertically and horizontally. All three engines are shown installed in a common frame but in the future individual gimbals and actuators for each engine will be designed.

8.5.5 Taxi Mission Profile

Two Taxis operate in the Mars transportation architecture. In a typical sequence a Taxi departs Earth and rendezvous 7-10 days later with the Up Astrotel to Mars for its 5-month trip to Mars. Several days before Mars arrival, the Taxi departs the Astrotel and deflects its trajectory to a Mars aerocapture. After aerocapture the Taxi, now in orbit, rendezvous with the Mars Spaceport, where it docks. This Taxi remains docked to the Mars Spaceport as the crew departs the Spaceport on the Mars Shuttle toward the Mars Base. After an average of 2.3 years, the next crew boards the Taxi and departs the Mars Spaceport to rendezvous with the Down Astrotel for its 5-month trip back to Earth. Total Taxi mission duration is an average of 2.8 years from Earth departure to return. Once back at the Earth Spaceport, major refurbishment and upgrades are planned including possible replacement of aeroshell components. Figure 8-25 illustrates the Taxi profile departing the Mars Spaceport on its way back to Earth.

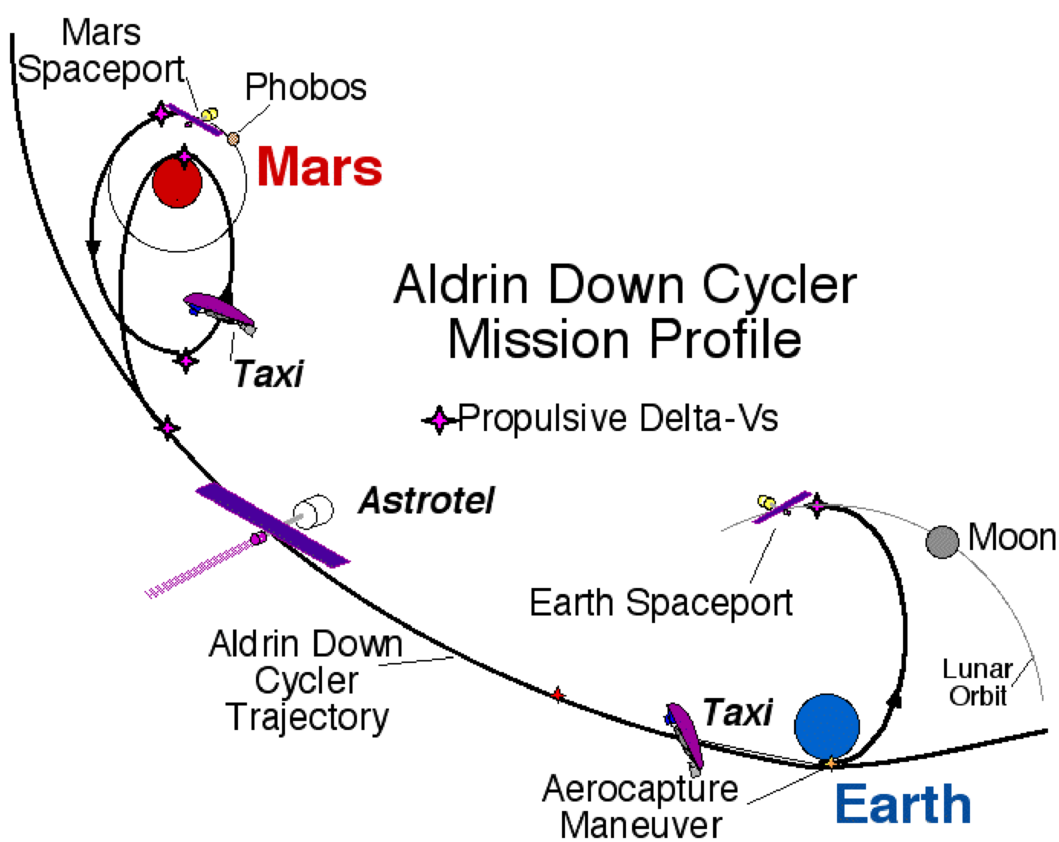

Figure 8-25 Taxi Mission Profile on Return to Earth

8.5.6 Taxi Mass Summary

The following table summarizes the mass for the basic Taxi vehicle that can depart Earth, rendezvous with the Up Astrotel, carry out the Mars aerocapture maneuver and perform propulsive maneuvers to rendezvous with the Mars Spaceport.

Table 8-17 Basic Taxi Vehicle Mass Summary, kg

Subsystem or Item		Dry Mass	Consum-ables	Single Stage Mass
Taxi Crew Module		7,207	519	7,726
Primary Structure		1,000		1,000
Propulsion		4,356		4,356
Engines	1,500			
Tankage	2,596			
Residual and Reserve	260			
Total		12,562	519	13,081
Aeroshell		2,955		2,955
Grand Total Dry Mass		**15,518**	**519**	**16,037**
Final Stage Mass				**16,037**
Propellant for each Stage				**17,307**
Grand Total Wet Mass				**33,344**
Delta-Vs, m/s				**3,302**

8.6 Mars Shuttle System Concept

This section summarizes the work in Phase I, describes the Phase II design process, discusses the entry orbit selection and mission delta-Vs, Mars Shuttle design options and concepts, detailed design of aeroshell, mass breakdown, and the overall configuration.

8.6.1 Phase I Study Summary

In Phase I the Mars Shuttle design was based on the 1985 NCOS design except it was scaled for a crew of 10 versus 4 and we assumed a direct entry trajectory from Phobos orbit radius (POR) rather than an intermediate 500-km altitude circular orbit. The NCOS Mars Shuttle design was scaled in the size of the crew module from a 4 person crew to a 10 person crew which increased the crew module mass from 2,262 kg to 4,823 kg; most of this increase occurring in primary structure, power system, life support and crew member mass. Delta-Vs were reevaluated and new propellant requirements determined. Due to the crew module mass increase the overall system mass grew due to increases in the proportional mass elements including propellant tankage, aeroshell and primary structure. The total dry mass of the Mars Shuttle as scaled in Phase I was 22,584 kg. There was no other Mars Shuttle configuration or detailed design work carried out during Phase I.

8.6.2 Design Process

The Mars Shuttle design process was iterative. An early step in this process was to determine all delta-Vs including the ascent phase burns for launch, trans-Phobos, and POR circularization and the descent phase burns for de-orbit from POR and landing. The landing burn is comprised of a component for initial deceleration, hover to target and an allocation to account for surface winds. Another early step was to develop, evaluate and compare detailed configuration design options, such as aeroshell placement, location of rocket engines, cargo storage schemes, etc. Aero-assist parameters such as entry angle, entry speed, required L/D, ballistic coefficient, crew g-loads, heating rates, down range distance, and deceleration delta-V, must be examined in order to select an optimum set for the design. A detailed configuration study then developed design concepts for aeroshell, propellant tanks, crew module, and landing gear and their packaging. Finally, the design of the aeroshell took into account heating rates, aerodynamic forces, structural loads, and packaging requirements.

8.6.3 Entry Orbit Selection and Mission Delta-Vs

Two options were studied for de-orbit and entry of the Mars Shuttle. The first option is direct entry from POR using a single delta-V at POR that targets the Mars Shuttle to the proper landing target. The second option is a three burn sequence that places the Mars Shuttle first onto a intermediate, trans-Phobos orbit with a low Mars periapsis altitude (say 500 km), then a circularization burn, followed by a final de-orbit burn to entry. This second option would be required if entry angle accuracy is insufficient for safe entry. These two options are illustrated in Figure 8-26. We have selected the direct entry option because it requires less fuel and entry angle accuracy is not expected to be an issue.

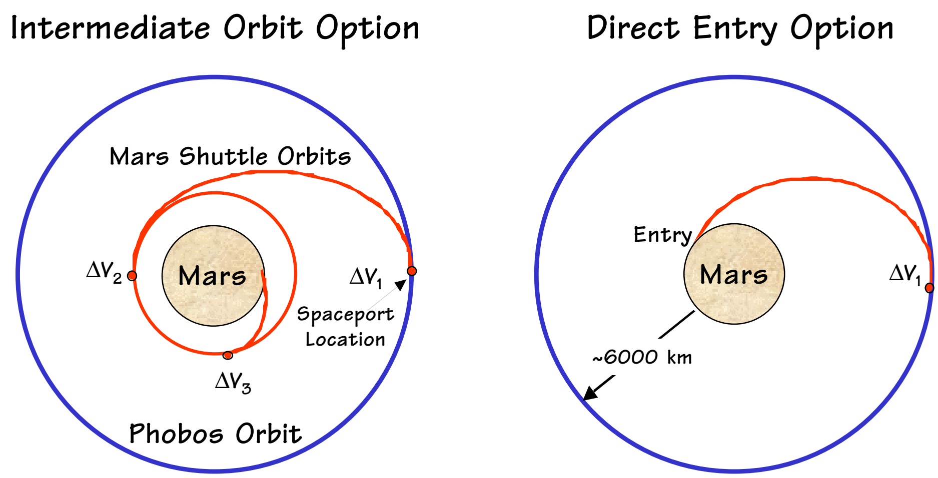

Figure 8-26 Mars Shuttle Entry Orbit Options

The delta-V comparison for return to Mars from Phobos for these two options is summarized in Table 8-18. The direct entry option saves 659 m/s in delta-V. The landing portion is assumed identical for each option. Landing consists of budgets for deceleration, hovering and surface winds. The deceleration budget is 270 m/s that corresponds to a entry interface of 125 km altitude, L/D of zero, entry angle of 8°, and burn thrust at 1.5 gees starting at 6-km altitude (see Figure 6-21). There is a 230 m/s budget for hovering, which corresponds to a 1-minute hover over a distance of 1.85 km. The possibility of surface winds to a level of about 50 m/s are covered in a further 80 m/s of delta-V. We have not yet carried out a landing site availability analysis for the direct entry option, which could force changes to the design.

Table 8-18 Delta-V Comparison: Intermediate Orbit Entry versus Direct Entry

ΔV DESCRIPTION	Intermediate Orbit ΔV, km/s	Direct Entry ΔV, km/s
DE-ORBIT		
ΔV1	0.500	0.571
ΔV2	0.627	-
ΔV3	0.103	-
Total De-orbit	1.230	0.571
LANDING		
Deceleration	0.270	0.270
Hover	0.230	0.230
Winds	0.080	0.080
Total Landing	0.580	0.580
Total De-orbit & Landing	1.810	1.151

Ascent delta-Vs are discussed in Section 6.5. Key variables are drag coefficient, C_d, of the Mars Shuttle in its launch configuration, drag cross-section area (as defined by vehicle diameter), level of thrust, timing and positions of coast periods and initial orbit requirements, if any. We assume the Mars Shuttle is launched on a direct trajectory to Phobos orbit with no intermediate orbit. The aerodynamic parameters assumed are a drag coefficient, C_d, of 0.8-0.9, a cross-section area of 78.5 m^2, an initial mass of about 55 mt, and a resulting ballistic coefficient of 800-900 kg/m^2, which yields a total calculated ascent and rendezvous delta-Vs about 5.1 km/s.

8.6.4 Mars Shuttle Design Concepts

We have always assumed a low L/D aero-vehicle design for the Mars Shuttle entry. A low L/D aeroshell would have an aeroshell shape similar to that used by the Viking and Pathfinder robotic missions but with advanced thermal protection systems. The primary reason for this shape is the ease of incorporating cargo containers, fuel tanks and an existing crew module into the design behind the aeroshell. A high L/D design would be more efficient in aeroshell size and mass, however it would be a considerable challenge to package all desired components within such an aeroshell and still keep it lightweight. The low L/D design, however, may require stowing the aeroshell prior to launch in order to minimize cross-section area and drag during ascent to reduce launch delta-V.

Figure 8-27 displays four options for configuring the aeroshell for entry and launch. The ***Deployable Aeroshell*** concept can have the aeroshell at the aft (#1) or the forward end (#2) of the vehicle. Cargo is attached to the backside of the aeroshell at entry. Advantages of the aft aeroshell design are that it provides a clear docking port at the forward end and the aeroshell attachment structure can do double duty by helping to support the engine and landing gear. Disadvantages of the aft aeroshell placement are that the engine nozzles and landing gear must penetrate the aeroshell and the aeroshell deployment. The advantages of the forward aeroshell placement are that neither engines nor landing gear penetrate the aeroshell. The disadvantages of the forward aeroshell placement are that the docking port penetrates the aeroshell, the fuel tanks and crew module experience g-loads that are 180° apart during a mission, and the aeroshell attachment structure is single function. The ***Fixed Aeroshell*** concept has a disadvantage of significant increase in launch delta-V due to the higher drag. The major advantage of this concept is that the aeroshell design can be optimized in a fixed position and thus the design is simple. Cargo is attached to the backside of the aeroshell at entry. The ***Fixed Aeroshell, Low Drag Launch*** configuration provides the same advantages of the fixed aeroshell configuration but with a shorter vehicle that can have significantly lower drag at launch, thus reducing the delta-V penalty. In addition, the fuel tanks can be larger in volume overall reducing the tankage mass. Another advantage of this concept is a lower center-of-gravity during the entry. In this concept, cargo can be fully enclosed within the aeroshell at entry. The ***Disposable Aeroshell*** concept offers significant reductions in propellant requirements during launch, which currently drive the tank size and number of engines.

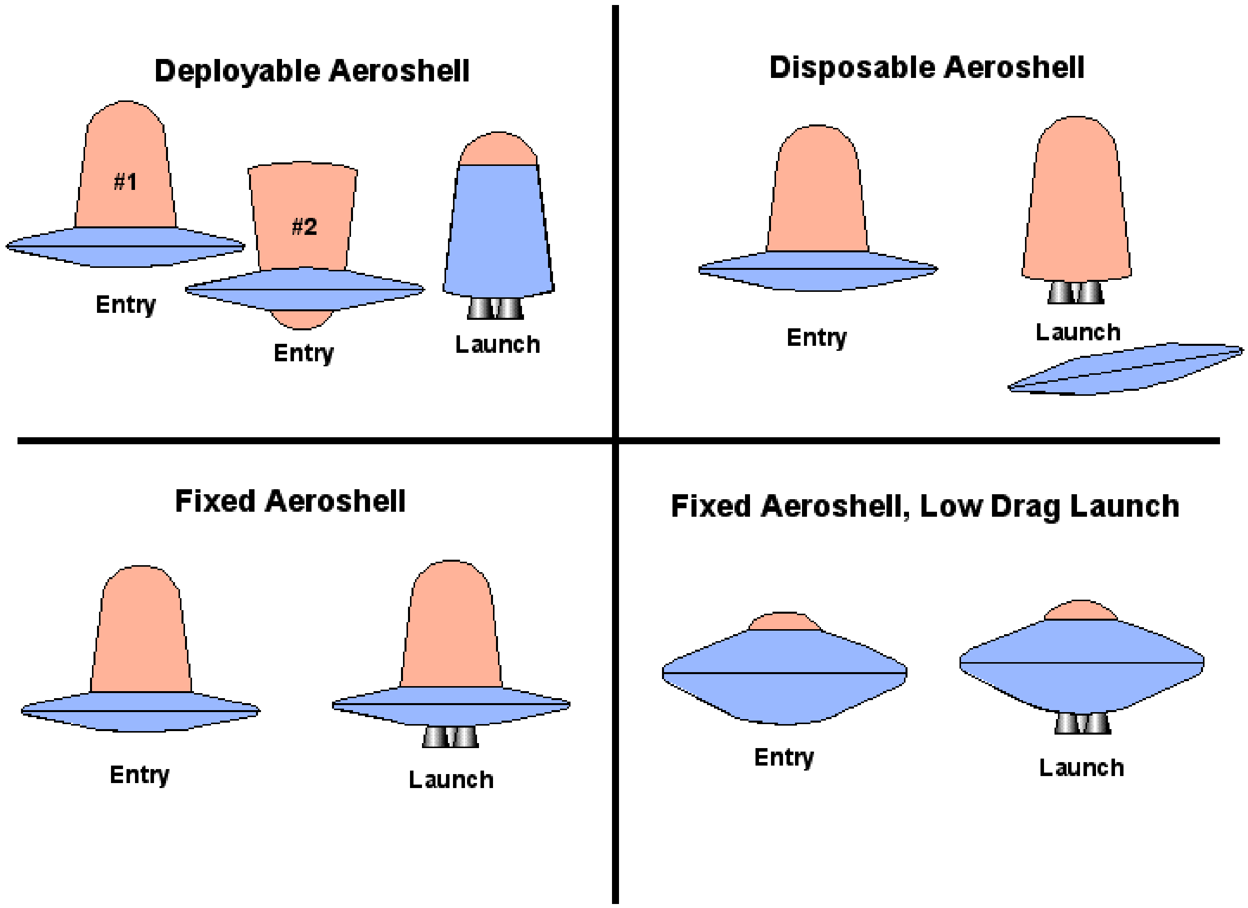

Figure 8-27 Mars Shuttle Entry and Launch Aeroshell Options

For the current Mars Shuttle design we have assumed the aft Deployable Aeroshell design (#2) and a correspondingly lower launch delta-V.

8.6.5 Mars Aeroshell Design

The Mars Shuttle aeroshell is about a 70° half-cone angle with a rounded forebody. The aeroshell is made of a lightweight, load-bearing structure in order to handle aerodynamic and engine thrust loads. The aeroshell structure must also accommodate the load of the cargo that is attached to the backside of the aeroshell. A particular design challenge for the aft deployable Mars Aeroshell is to meet the thermal and mechanical requirements and yet be stowable for launch from the Martian surface. Consideration was first given the detailed requirements for the Mars Shuttle aeroshell. The maximum aerodynamic load is calculated to be 0.038 atm or about 0.57 lb./in^2. The area of the 10-m radius aeroshell is currently estimated at 314 m^2. The maximum temperatures on the Shuttle heat shield during entry into Mars were evaluated on the basis of the vehicle radius at the nose and the shoulder. Heating rates are low as compared to Taxi aerocapture but will still require reasonably good thermal protection systems (TPS). TPS temperatures are expected between 590-755 °C from which one can choose an appropriate heat shield material. For example, Space Shuttle high-temperature reusable surface insulation (HRSI) SiO_2 ceramic tiles would be sufficient for the Mars Shuttle TPS since temperatures are expected to be well below their 1,260°C limit. Reinforced carbon-carbon may be necessary near low radius corners such as the aeroshell edge.

The conventional 70° half-cone angle, Viking-shaped aeroshell is fabricated of aluminum spars with aluminum honeycomb surfaces upon which Space Shuttle type TPS, as discussed above, is applied. The aeroshell comes in 30 segments, alternating segments folding along alternating hinge paths. The aeroshell is deployed while at the Mars Spaceport for the Mars entry. During the trip to the surface, the aeroshell also serves as a location to affix the up to 10 mt of cargo destined for the Mars surface. After landing the cargo is removed and the aeroshell segments rotated back toward the main body of the vehicle to reduce ascent drag. Because the rocket engines and landing gear penetrate the aeroshell in the aft deployable aeroshell design, deployable doors must be included for the access through the aeroshell. These doors are fixed in place until after reaching low sonic speeds, e.g. Mach 2, when they are opened, allowing the rocket engine nozzles and the landing gear to deploy. These doors can remain open until the return of the Mars Shuttle to the Mars Spaceport. The following figure illustrates an early aeroshell design concept showing the aerobrake in its deployed and stowed state.

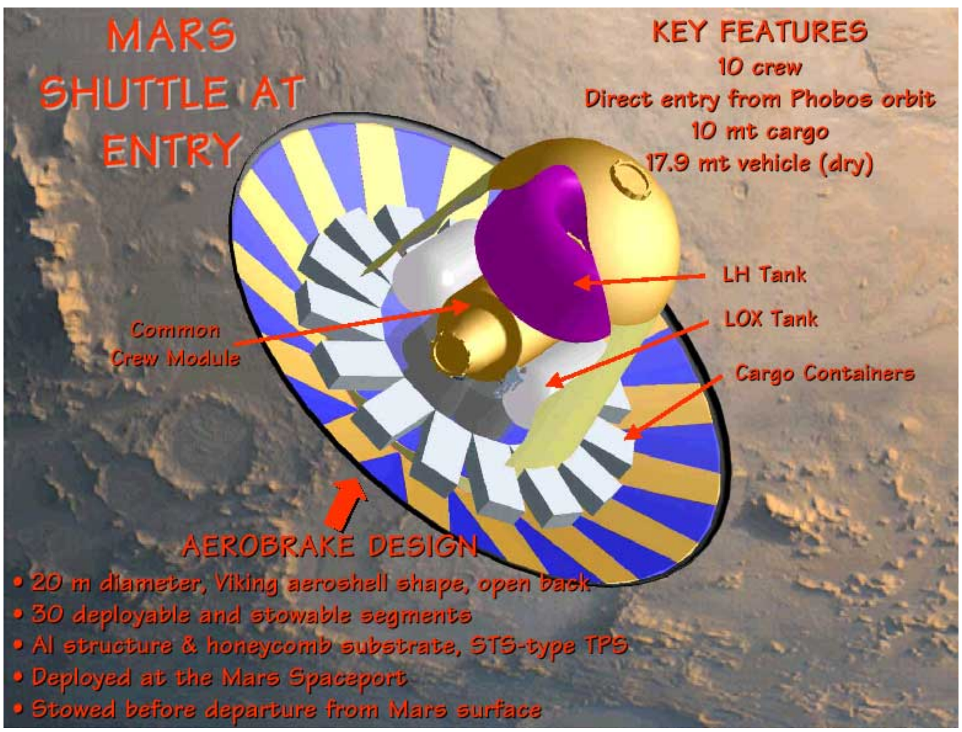

Figure 8-28 Mars Shuttle Aeroshell Design Concept

The following discussion provides the details of the mechanical design of the aeroshell and its mechanical stow/deploy scheme. Three separate sets of panels make up the aeroshell. Each set of panels is deployed/stowed in sequence. An animation has been generated in order to illustrate the stow and deploy process for Mars Shuttle appendages (aeroshell, landing gear, engine nozzles, etc.). The following figure illustrates the stowage of the latest design Mars Shuttle aeroshell in sequence from left to right.

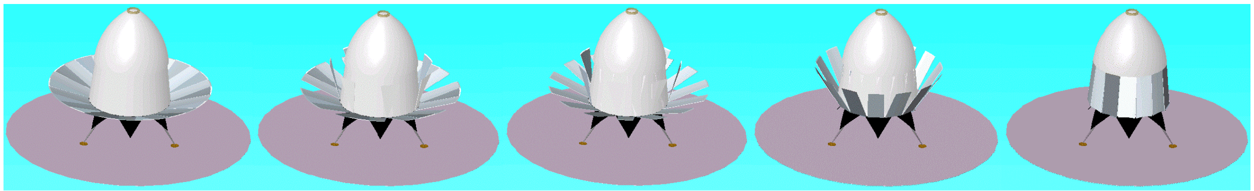

Figure 8-29 Mars Shuttle Aeroshell Stow Sequence

The next figure views the Mars Shuttle from below while the aeroshell is partially deployed. The engine doors are open and the extended nozzles and landing gear are shown.

Figure 8-30 Mars Shuttle Aeroshell Stow Sequence View from Below

8.6.6 System Mass Summary

Table 8-19 summarizes the overall mass breakdown for the Mars Shuttle system for the aft deployable aeroshell design. This table and the subsystem masses are generally the same as developed in Phase I, except that the proportional masses, like tankage, structure, aeroshell, etc., are recalculated based on the current Phase II delta-Vs. The ratios for proportional masses are summarized in Table 8-20. As we continue in the detailed design of the Mars Shuttle in Phase II, we will determine these masses by design rather than by ratios.

Table 8-19 Mars Shuttle Mass Breakdown

Subsystem	Mass, kg	Refurb Mass in 15 years, %	Refurb Mass, kg
Crew Module			
Total Crew Module	4,783	46%	2,185
Propulsion Module			
Tanks, Insulation & Plumbing	3,753	5%	188
Engines	1,500	100%	1,500
Landing Gear	333	10%	33
Aerobrake	4,000	30%	1,200
Attitude Control (dry)	200	100%	200
Attitude Control (prop)	554	0%	-
Primary Structure	2,771	5%	139
Total Propulsion Module	13,111		3,259
Total Mars Shuttle	17,894		5,444
Mars Cargo Sized	10,000	kg	

Table 8-20 Mars Shuttle Proportional Mass Ratios

Tankage Factor	10.0%	of propellant mass
Proportional Mass Factors	6.6%	
Structure	5.0%	of initial mass
Landing Gear	0.6%	of initial mass
Attitude Control	1.0%	of initial mass
Aerocapture	15.0%	of Entry mass

Table 8-21 summarizes the key system masses for the Mars Shuttle for different flight phases.

Table 8-21 Key System and Propellant Masses

ITEM	MASS, kg
Dry, Empty System	17,894
System at Launch	55,423
Fuel at Launch	37,529
System at Phobos Departure	41,498
Mars Base Cargo	10,000

Phobos Departure Fuel	4,861
System at Entry	36,637
Fuel for Landing at Entry	8,743
Landed System	27,894

8.6.7 Configuration Design

Figure 8-31 illustrates an early Phase II Mars Shuttle configuration design in ascent with the deployable aeroshell concept. This version shows a hemispherical nose cap.

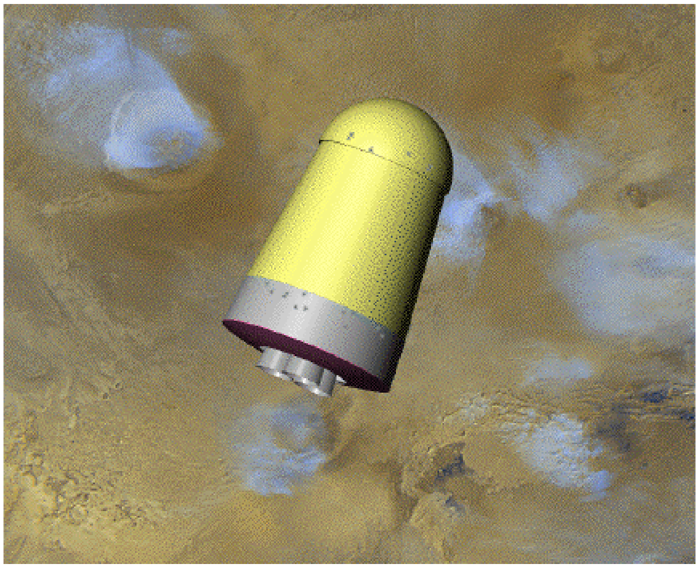

Figure 8-31 Early Mars Shuttle Design Launch Configuration

Figure 8-32 shows the current design Mars Shuttle at entry (left) and at the moment of engine burn (right). In reality, the Mars Shuttle would be oriented almost vertically at engine ignition. Note that due to lower propellant requirements than originally assumed the vehicle is shorter. In addition, we have a more aerodynamic shape of the nose to take advantage of a somewhat lower C_d.

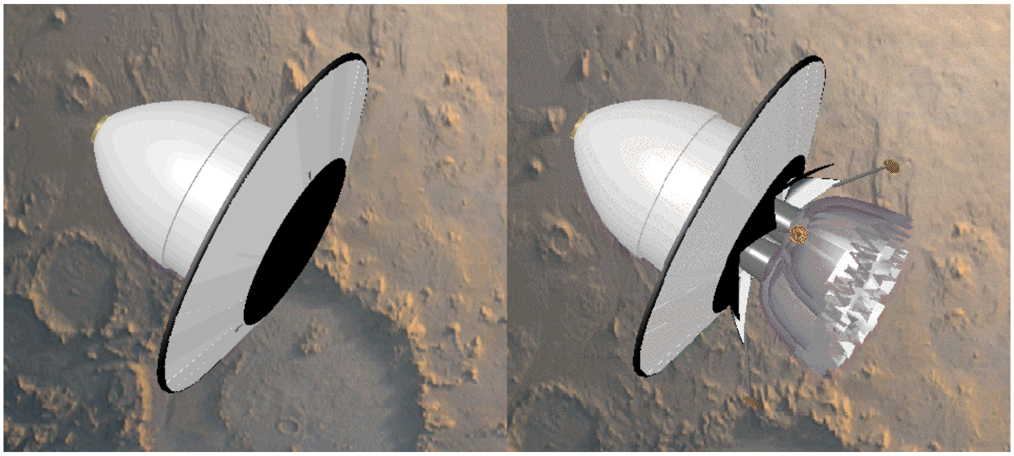

Figure 8-32 Mars Shuttle Entry Configuration

8.6.7.1 Primary Structure

The primary structure is a 12-m high space frame that has a 10-m diameter at its base or aft end and a small diameter at the forward end of the vehicle near the aerodynamically shaped nose. Within, and supported by the structure, are the LOX and LH propellant tanks, propulsion distribution and control equipment, and the crew module, which is planned to be identical to that used by the Taxi vehicle. The structure also supports the aeroshell in its deployed and stowed state. The structure near the aft end of the vehicle is strengthened in order to support the loads of internal hardware, the aeroshell and landing gear.

8.6.7.2 Internal Configuration

The components are placed inside the space frame in order to ensure the proper location of the center-of-gravity (cg) with respect to the center-of-pressure (cp) of the system. The crew module is placed at the aft end of the vehicle and access is provided to the exterior. A long tunnel extends upward toward the forward end of the vehicle where a docking port and airlock are situated to facilitate transfer of crew to and from the Mars Spaceport. A separate airlock at the level of the crew module allows the crew to depart the vehicle to the surface by use of a ladder. LOX tanks are placed at the aft end and LH tank is located above them in a torroidal tank, because LOX is much denser than LH ensuring a low cg. The LOX is split into two moderate sized tanks arrange on the periphery of the crew module. The LH tank surrounds the crew upper access tunnel.

8.6.7.3 Propulsion

Mars Shuttle propulsion consists of three 500 kg RL60-class, LOX/LH rocket engines (see Section 8.2.1.1). These engine will operate at 267 kN [60,000 lb.] thrust, a 7:1 mixture ratio, an I_{sp} of 460 s, and a propellant mass flow rate of 59.2 kg/s. Two engines are required for launch from the surface, however, three engines are required for the landing. Since redundancy is required, three engines are carried at all times. Since three engines produce more thrust than necessary, throttling may be required for both launch and landing.

9 Mission Architecture and Model Analysis (MAMA) Development

In Phase II, a concept derived in Phase I to integrate all the various contributing analyses to support system-level analysis was implemented in a tool named MAMA. This tool was used to assess various architecture design trades. The tool development and trade study results are summarized here.

9.1 Introduction

The concept of integrating all the Astrotel analysis tools was derived to support trade studies of alternative approaches for Mars Astrotel Concepts. The integrated tool developed during Phase II, called MAMA, helps identify technologies and approaches with the potential for high leverage. These technologies include, but are not limited to, innovative trajectories, in situ resource utilization, advanced power systems, and propulsion. The tool generates high-level metrics to help quickly compare results from in-depth analyses and also allows penetration into the lower level details. MAMA captures requirements for all life cycle cost elements and sensitivity analyses were performed with early prototype versions to direct development of trade studies to investigate in more detail.

MAMA provides system analysis capabilities by linking inputs/outputs between multiple lower-level models. The approach derived in Phase I, and implemented in Phase II, is shown in Figure 9-1. A separate model is created for each Element, and the Element models can refer to any of the Support Analyses models to assess design alternatives and technology utilization. MAMA also provides "standardized" system outputs based on results rolled up from the lower-level models. MAMA can display outputs for a baseline and multiple options simultaneously to facilitate trade study analyses.

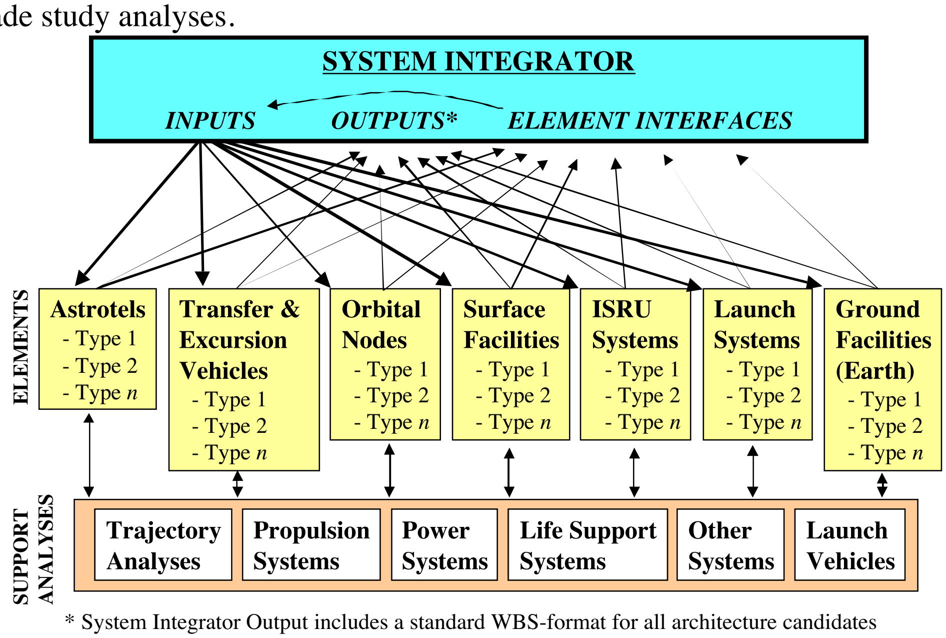

Figure 9-1 MAMA Information Flow and System Integration

9.2 Phase II Summary

The Phase II MAMA development effort enhanced and refined the concept developed in Phase I. The new version expanded on the level of integration demonstrated by MAMA Version 1 and provides a single interface to all of the lower-level Astrotel models. This section describes the Phase II MAMA approach and design, cost analysis and trade studies.

9.2.1 Approach

The idea for MAMA came up near the end of the Phase I effort. Its primary purpose was to collect the outputs of the various models in one place to facilitate cost analysis. Making changes required going into multiple separate files and a substantial amount of data entry to accurately capture the impacts to all the lower-level models. For this reason, only a single concept was explored in detail and a limited set of trades were assessed.

In Phase II, the goal was to enable MAMA to assess a wider range of architecture design options. MAMA includes database capabilities to capture inputs and selected outputs from an Astrotel concept run. Architecture designs can be created to test impacts from concept design alternatives and advanced technologies. By comparing results from these assessments, analysts get a better understanding of what the best technologies and mission architectures for a Mars Astrotel system might be. The MAMA databases save results from each run to facilitate comparisons and trade studies.

To organize the various architecture-level and element-level alternatives, multiple Trade Trees were developed and a hierarchical structure was established to collect alternative selections for a given design. Examples of MAMA Trade Trees are shown in Figure 9-2.

A sensitive and important aspect of the study is the supporting cost analysis. The cost estimating methodology used for MAMA can be characterized as a “quasi” grassroots approach, where estimates are “built up” using a detailed WBS that captures all life cycle phases, starting with required advanced technology development and going through the end of planned operations. Lower-level estimates are derived from a combination of analogies and parametric relationships, using appropriate data and adjustments tailored for each Element. Tools used to support cost analysis for various estimate elements include an SAIC Planetary Model with Price H that can estimate component-level costs and other development items, the NASA-Air Force Cost Model (NAFCOM) that includes parametrics and a historical cost database, NASA’s Space Operations Cost Model (SOCM) for MO&DA, CSOC pricing information for operations services, Earth Launch Vehicle cost data, and many other sources that are Element-unique. An example of the detail included in the WBS is shown in Table 9-1. It is important to note no matter how detailed the approach, it would be optimistic to claim even ±30-50% accuracy. The approach is better suited to estimate relative impacts from different approaches and can help quantify impacts from advanced technologies.

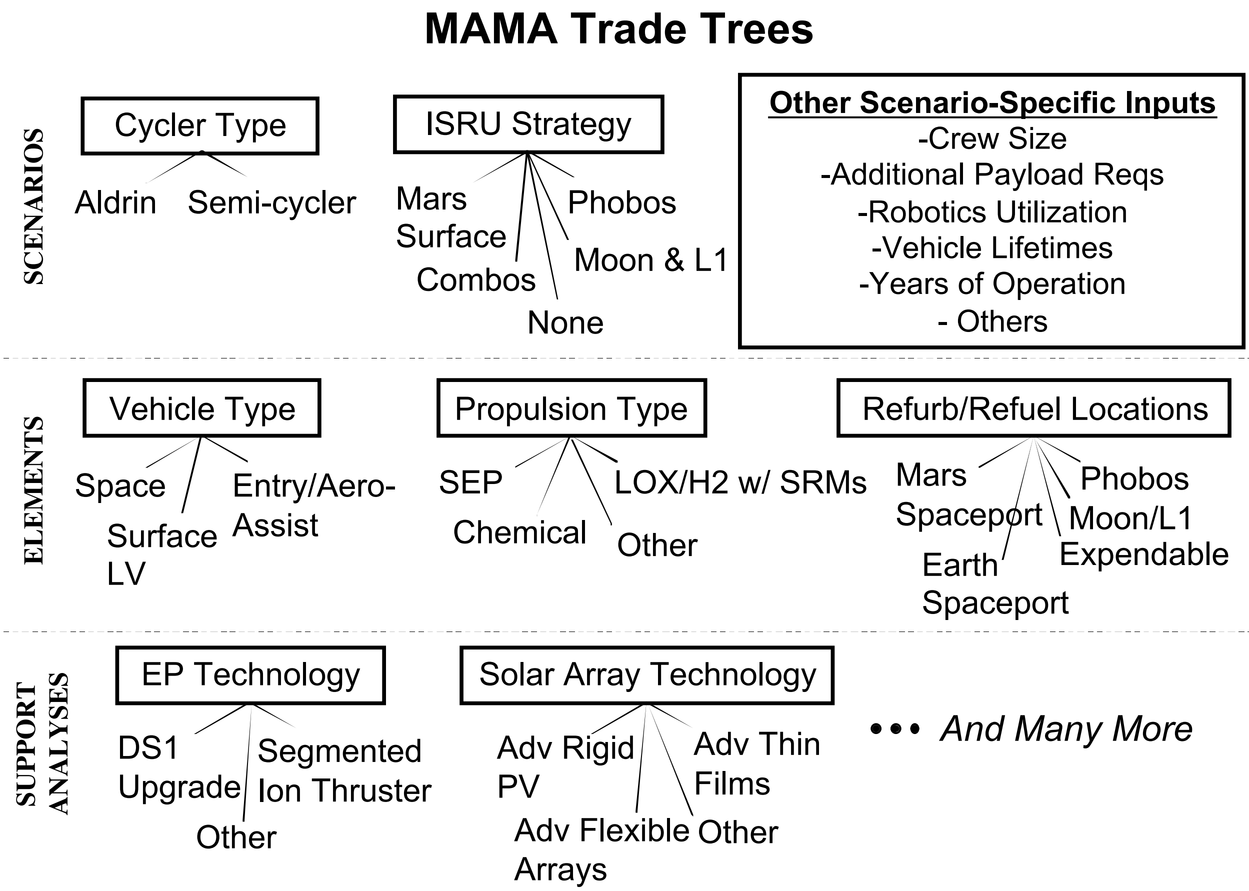

Figure 9-2 Examples of MAMA Scenario-specific inputs and Trade Trees

The MAMA model architecture concept is based on an approach successfully applied to several past studies and model development efforts. The multi-level approach is based on methods used in the SOCM tool. The idea for integrating multiple standalone detailed analysis and design tools was combined with the multi-level approach to develop a tool to assess various space solar array and power system design options (Array Design Assessment Model – ADAM). In a much less sophisticated manner, past SAIC in situ resource utilization studies have incorporated similar techniques. The primary benefit of developing these tools is to provide better insight into the potential life cycle impacts of changes to requirements, design, and technologies.

Table 9-1 Example of detail in Work Breakdown Structure (WBS)

MARS CYCLER CONCEPT LIFE CYCLE COST WORK BREAKDOWN STRUCTURE -Version 1 8/1/2000

Concept Name

Life Cycle Cost WBS Elements *	**Life Cycle Cost Phase **** **Adv Tech Dev** Adv Tech Dev	Adv Tech Testing	**Development** Flight System Design	Flight Subsys Fab	Flight System I&T	LV Ground Facility Proc	**Launch** Launch & Checkout	**Operations** Orbital Assy, I&T	Startup Ops	SS Ops incl Refurb/ Maint	Disposal
1.0 Advanced Technology Dev											
1.1 General R&D											
1.2 Facilities											
1.3 Flight Demos/Major Tests											
1.4 Sys-Unique Test Facil/HW											
1.5 Other ATD Costs											
2.0 Flight System Development											
2.1 Flight Elements											
2.1.1 Flight Element 1	*repeated for each Flight Element*										
2.1.1.1 Subsystem Components	*repeated for each Flight Element Subsystem; can be linked to a detailed mass statement/equipment list*										
2.1.1.2 Subsystem I&T	*repeated for each Flight Element Subsystem*										
2.1.1.3 System I&T	*repeated for each Flight Element*										
2.1.1.4 System Support/Mgt	*repeated for each Flight Element*										
2.2 Flight System Dev Support											
2.3 Other Flight System Costs											
2.4 Development Reserves											
3.0 Launch Services											
3.1 Launch Approval											
3.2 Launch Processing											
3.3 Launch Vehicle											
3.4 Other Launch Services Costs											
4.0 Operations											
4.1 Operations Project Mgmt											
4.2 Integrated Logistics											
4.2.1 Repairs											
4.2.2 Spares											
4.2.3 Ground Transport & Handling											
4.2.4 Inventory Management											
4.2.5 Maintenance, Ground											
4.2.6 Maintenance, Space											
4.3 Flight Operations											
4.3.1 Maintenance & Support											
4.3.2 Mission Ops Control											
4.3.3 Flight Planning											
4.3.4 Flight Ops Design/Dev											
4.4 Training Operations											
4.4.1 Flight Operations											
4.4.2 Ground Operations											
4.5 Launch Operations											
4.5.1 Element Processing											
4.5.2 GSE Maintenance & Support											
4.5.3 Launch Services											
4.6 In-Space Crew Support											
4.6.1 EVA											
4.6.2 IVA											
4.6.3 EVA/IVA Support											
4.7 Comm/Data Handling Ops											
4.7.1 Data Handling											
4.7.2 Ground-Ground Comm											
4.7.3 Tracking Network											
4.8 Operations Proj Supp Costs											
4.8.1 System Integ Mgmt & Supp											

*** Each WBS Element can include add'l detailed output categories for:**
- **Facilities**
- **GSE**
- **Materials/Hardware**
- **Consumables**
- **Other**
- **Program Support**

**** Life Cycle Cost Phases are spread across fiscal years based on a concept's life cycle schedule**

9.2.2 MAMA Design

This section will describe the general design of the MAMA tool. As described in Section 9.1, MAMA integrates many lower-level modules describing flight element and mission designs, technology characteristics, and life cycle costing. It provides a single interface to enter inputs and view/compare results. It also has a database that allows users to quickly pull up a given concept to use as a point of departure for a new design.

The MAMA Main Menu is shown in Figure 9-3. It organizes Astrotel architecture elements into four separate areas: Vehicles, Transportation Nodes, Other System Elements, and Supporting Analyses. Items included in each of these areas can be accessed using the buttons at the top of the Main Menu. The Main Menu also has a Master Comparison Table, accessed through the buttons at the bottom of the menu, which allows users to compare various aspects of the currently loaded architecture design to multiple other designs that have been stored in the MAMA database. Primary power system type can be assigned to be either nuclear reactor or solar arrays. One significant difference between nuclear reactors and solar arrays is how they scale to higher power levels. Solar arrays scale more linearly, while the reactors tend to have a somewhat large fixed mass, even for low power levels. They can have a lower specific power (W/kg) than arrays at either high power levels or great distances from the sun. Table 9-2 shows the differences between 100kWe and 150kWe nuclear reactor designs. The power system designs are tailored for either space or planetary surface applications. Table 9-3 shows the worksheet for space applications and Table 9-4 shows the worksheet for planetary surface applications. All designs include sufficient radiation protection for the crew.

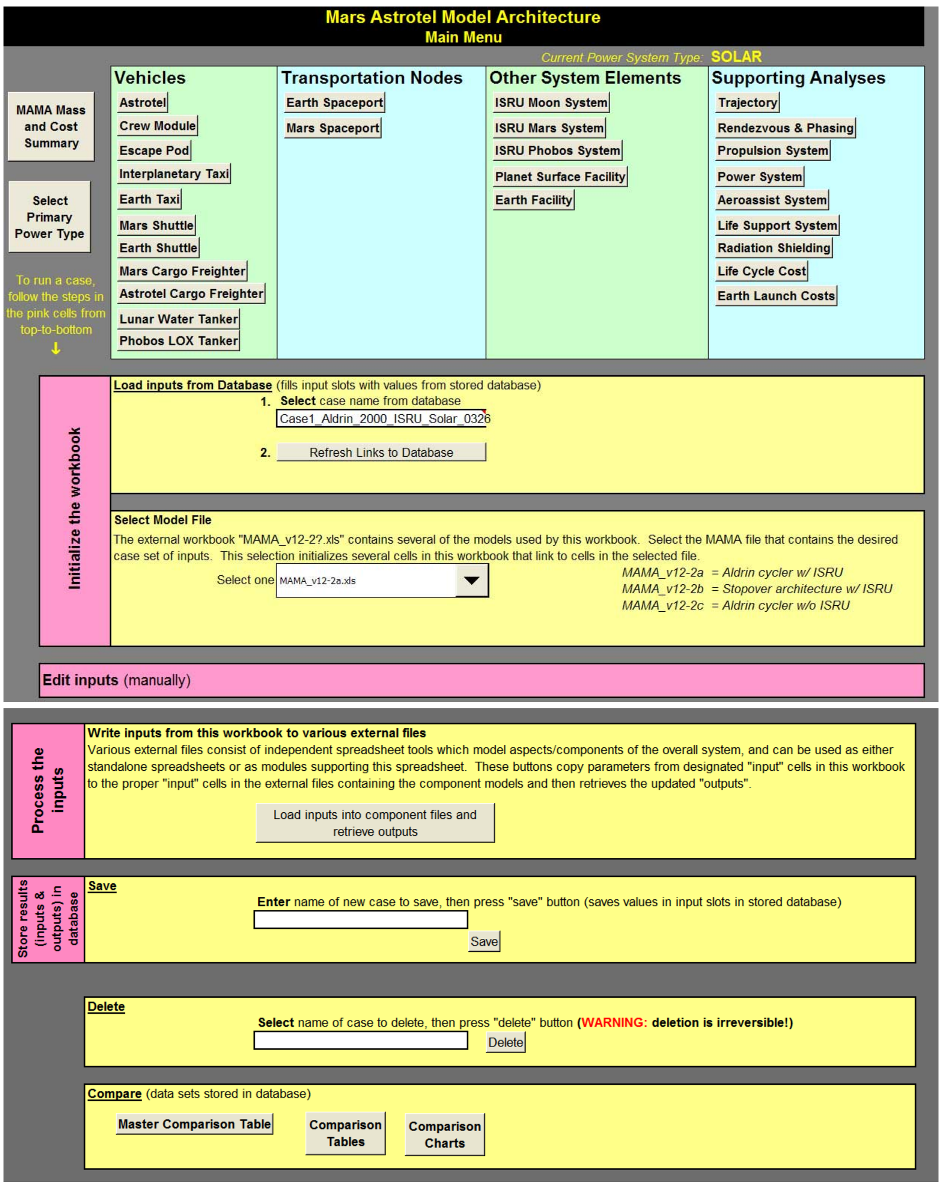

Figure 9-3 MAMA Main Menu

Table 9-2 Nuclear Reactor System Scaling

Comparison of 100 and 150 kWe systems

	100kWe	150kWe		W/kg 100kWe	W/kg 150kWe		$M 100kWe	$M 150kWe	
Nuclear subsystem	1900	2245	kg	52.6	66.8	W/kg	1000	1200	$M
Brayton Conversion	1800	2200	kg	55.6	68.2	W/kg	80	100	$M
Power-conv Heat rejection									
radiators	900	1350	kg	111.1	111.1	W/kg	50	80	$M
support struct/mech	300	500	kg	333.3	300.0	W/kg	25	35	$M
Cabling	150	225	kg	666.7	666.7	W/kg	10	15	$M
Total Nuclear Power System	5050	6520	kg	19.8	23.0	W/kg	1165	1430	$M

Table 9-3 Nuclear Reactor System Worksheet for Space Transportation Vehicles

Space Fission Reactor - Mass & Cost Worksheet

Mass & Cost estimates based on reference data:

Power Level **200** kWe (50 - 2000kWe)

Mass Estimates:			W/kg	
Nuclear subsystem	2,591		77.2	
Brayton Conversion	2,600		76.9	
Power-conv Heat rejection				
radiators	1,800		111.1	
support struct/mech	700		285.7	
Cabling	300		666.7	
Total Nuclear Power System	**7,991**	**kg**	25.0	W/kg
Cost Estimates:				
Nuclear subsystem	1,400		7.0	$/W
Brayton Conversion	120		0.6	$/W
Power-conv Heat rejection				
radiators	110		0.6	$/W
support struct/mech	45		0.2	$/W
Cabling	20		0.1	$/W
Total Nuclear Power System	**1,695**	**$M**	8.5	$/W

MASS BREAKDOWN for 100kWe SP100/Brayton DESIGN

Element Masses (kg)			
Reactor			
Support Structure			
Reactor instr & control			
Primary heat transport			STAIF ref may incl radiator
Shielding	800		=additional Shielding for manned systems (WAG)
Gamma			
Neutron			
Total Nuclear Subsystem Mass	1900	kg	
Additional Elements Required:			not incl in STAIF ref
Brayton Conversion	1800	kg	
Power-conv Heat rejection			
radiators	900	kg	
support struct/mech	300	kg	
Cabling	150	kg	
Total Nuclear Power System	5050	kg	

Source: "Liquid Metal Cooled Reactor for Space Power" by Weitzberg (STAIF 2003)

Table 9-4 Nuclear Reactor System Worksheet for Planet Surface Applications

Planetary Surface Fission Reactor - Mass & Cost Worksheet

Mass & Cost estimates based on reference data:			
Power Level	**240**	kWe	(1 - 200 kWe)
Mass Estimates:			W/kg
Nuclear subsystem	2,161		111.1
incl Reactor, PMAD, and Shield			
Brayton Conversion	2,920		82.2
Power-conv Heat rejection			
radiators	2,160		111.1
support struct/mech	450		533.6
Cabling	1,800		133.3
Total Nuclear Power System	**9,491**	**kg**	25.3 W/kg
Cost Estimates:			
Nuclear subsystem	924		7.7 $/W
Brayton Conversion	106		0.9 $/W
Power-conv Heat rejection			
radiators	128		1.1 $/W
support struct/mech	31		0.3 $/W
Cabling	120		1.0 $/W
Total Nuclear Power System	**1,309**	**$M**	10.9 $/W

MASS BREAKDOWN for 3kWe MARS SURFACE DESIGN		
Element Masses (kg)		
Reactor	545	w/ Stirling conversion system (~200kg)
Support Structure	35	
PMAD	35	
Radiator	72	20 m2 area
Shielding	1169	
Gamma	274	
Neutron	895	
Total Nuclear System Mass	1856	does not include sufficient cabling
Specific Mass	1.616	W/kg
Source: "Design Concept for a Nuclear Reactor-Powered Mars Rover" by Elliot, Lipinski, and Poston (STAIF 2003)		

9.2.3 Cost Assumptions and Analysis

Costs were estimated for all life cycle phases, including Advanced Technology Development, Development, Launch, and Operations. These phases also included sub-phases, as shown earlier in Table 9-1. Table 9-1 also shows the WBS used to track the estimates.

Because much of the cost analysis requires lower-level mass and performance data, the cost portion of the model captures this data. Component-level mass/performance data is collected from the various lower-level MAMA modules and costs are estimated by analogy to this data. MAMA's Mass and Cost Summary Main Menu provides access to this information, as well as additional tabular and graphical summary outputs. Analogies can be assigned, modified, or added using this interface.

Table 9-5 shows the analogy data points used along with the Non-Recurring and Recurring costs and masses for each. Component-level results for individual flight elements can be viewed through the MAMA Cost Main Menu (Figure 9-4) but are summed up to a single WBS item in Table 9-1.

Costing of Nuclear Reactor Systems was performed in a separate module. Estimates for these items are based on scaling relationships developed from the data shown in Figure 9-2 and the analogy points shown in Figure 9-2 and Figure 9-3. It is important to note these analogy points are only recent concepts and estimates based on cost projections and they will have greater uncertainty than the estimates for other items.

Mission operations costs are estimated using a fixed percentage of the recurring hardware development cost. This is a gross simplification, and although it provides a reasonably rough estimate, it does not accurately capture differences in operational complexity between different technology approaches. For example, since the development costs for a nuclear reactor far exceed the costs for solar arrays, estimates of operations costs for nuclear reactor options are also higher, even though maintenance and support requirements may be similar or even easier. Currently, NASA does not have an available tool that captures technology impacts on operations and there is a lack of relevant analogy data. A more refined method for estimating operations costs for these types of missions is needed.

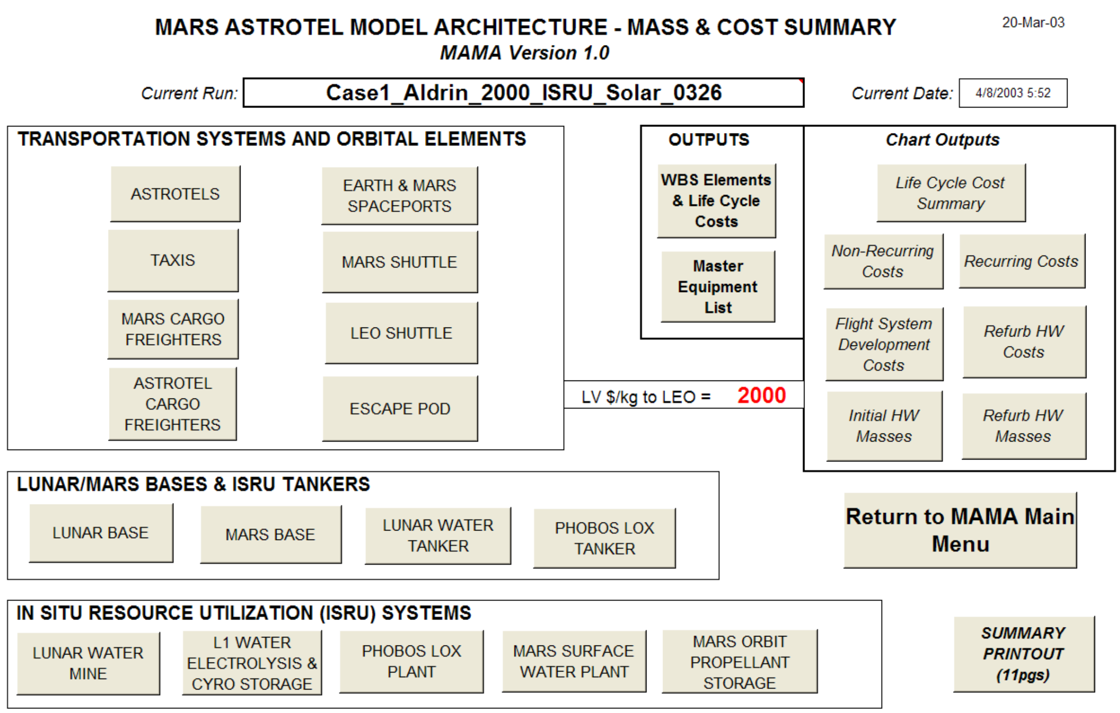

Figure 9-4 Mass and Cost Summary Main Menu

Table 9-5 Reference Data For Estimating Development Costs

MAMA Reference Info for Costing

Reference Name	Reference NRC	Reference RC	Reference Mass, kg	Total Cost	FY	$K/kg
Spacelab - Subsystem - ECLS	91.81	123.27	533.9	215.08	2001	403
Spacelab - Subsystem - Crew Accommodations	11.99	6.64	205.5	18.63	2001	91
Spacelab - Subsystem - Structure Mechanical	322.03	194.24	7,438.1	516.27	2001	69
Spacelab - Subsystem - PMAD	96.33	56.86	943.0	153.20	2001	162
Spacelab - Subsystem - Thermal	111.87	111.65	446.8	223.51	2001	500
Spacelab - Subsystem - CDH	232.97	248.61	888.1	481.58	2001	542
STS - Subsystem - ECLS	276.18	105.07	2,735.6	381.25	2001	139
STS - Subsystem - Propulsion	664.94	211.64	6,893.8	876.58	2001	127
STS - Subsystem - RCS	246.16	103.57	1,389.4	349.73	2001	252
STS - Subsystem - CDH	1,205.30	267.18	2,569.6	1,472.48	2001	573
STS - Subsystem - APU	145.60	28.52	419.1	174.12	2001	415
STS - Subsystem - PMAD	300.20	129.09	4,632.6	429.29	2001	93
STS - Subsystem - Thermal	667.95	228.15	11,294.6	896.10	2001	79
STS - Subsystem - Structure Mechanical	1,475.48	558.37	37,424.0	2,033.86	2001	54
SAIC Estimate - SEP - PPU & Thrusters	7.24	3.59	51.7	10.83	2001	209
SAIC Estimate - SEP - SA	0.93	5.13	11.3	6.06	2001	538
SAIC Estimate - ADM for MSR EEV	7.67	6.84	25.1	14.51	2001	578
SAIC Estimate - SRM MAV for MSR	16.79	8.57	263.5	25.36	2001	96
Spacelab - Overall System	2,056.63	983.09	10,455.4	3,039.72	2001	291
STS - Overall System	9,874.78	2,388.09	69,650.7	12,262.87	2001	176
SAIC Estimate - Large EP - Thrusters	10.73	70.97	428.4	81.70	2001	191
SAIC Estimate - Large EP - PPU	46.93	72.43	597.1	119.36	2001	200
SAIC Estimate - Large EP - Structure Mechanical	106.59	49.21	1606.6	155.80	2001	97
SAIC Estimate - MRO - Communications	7.32	14.97	44.7	22.29	2001	498
SAIC Estimate - MRO - CDH	14.86	12.27	44.7	27.13	2001	606
SAIC Estimate - Lander Propulsion for MSR	2.19	1.54	78.5	3.73	2001	48
SAIC Estimate - Lander ADM for MSR	18.23	20.00	78.5	38.23	2001	487
SAIC Estimate - MAV Chemical Propulsion for MSR	0.54	0.20	5.9	0.74	2001	124
SAIC Estimate - MAV SRM Propulsion for MSR	2.34	2.55	225.6	4.89	2001	22
SAIC Estimate - Tankage for Large Orbiter	37.85	28.27	1550.0	66.12	2001	43
MGS - Overall S/C	53.94	48.61	578.2	102.55	2001	177
Pathfinder - Rover	17.668	8.577	13.3	26.25	2001	1975
Solar Array at $500/W		1000				
Solar Array at $500/W and 100W/kg		50000				
Current Estimate for Astrotel	3836	2025	128,556	5861	2001	46
X-38-based CRV	430	125	10660.0	555	2001	52

9.2.4 Trade Studies

Although many trades were investigated, the final version of the MAMA tool was applied to a small set of focused trades. Options traded included Architecture Type (Aldrin or Stop-over), Launch Costs ($2K/kg or $10K/kg), ISRU (with or without), and Power Source Type (Solar Array or Nuclear Reactor). Table 9-6 summarizes the various trade studies that were performed. Detailed results for each case are provided in Table 9-7 through Table 9-30 and Figure 9-5 through Figure 9-16.

Table 9-6 MAMA Trade Studies Performed

Case	Architecture	Launch Costs	ISRU	Power
1	Aldrin	2,000	yes	Solar
2	Aldrin	10,000	yes	Solar
3	Aldrin	2,000	yes	Nuclear
4	Aldrin	10,000	yes	Nuclear
5	Aldrin	2,000	no	Solar
6	Aldrin	10,000	no	Solar
7	Aldrin	2,000	no	Nuclear
8	Aldrin	10,000	no	Nuclear
9	Stopover	2,000	yes	Solar
10	Stopover	10,000	yes	Solar
11	Stopover	2,000	yes	Nuclear
12	Stopover	10,000	yes	Nuclear

Table 9-7 Case 1: Aldrin, ISRU, Solar, $2k/kg LV; Flight Element Masses

Case1_Aldrin_2000_ISRU_Solar_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**653,037**	**451,220**
1	Astrotel	**2**	55,754	111,509	17,173
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*5,664*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	17,173
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	8,335	8,335	5,334
	Mars Cargo Freighter Xe Propellant				*139,889*
7	Astrotel Cargo Freighter	**1**	3,553	3,553	2,340
	Astrotel Cargo Freighter Xe Propellant				*56,688*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	6,917	6,917	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	278,757	278,757	50,942
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	3,074	3,074	1,031
15	L1 Water Electrolysis & Cyro Storage	**1**	584	584	80
16	Phobos LOX Plant	**1**	4,262	4,262	3,243
17	Mars Surface Water Plant	**1**	5,261	5,261	3,712
18	Mars Spaceport LOX/LH2 Storage	**1**	1,230	1,230	258
19					
TOTALS	Flight Elements, dry			**653,037**	**148,656**
	Flight Elements, wet			**0**	**302,565**

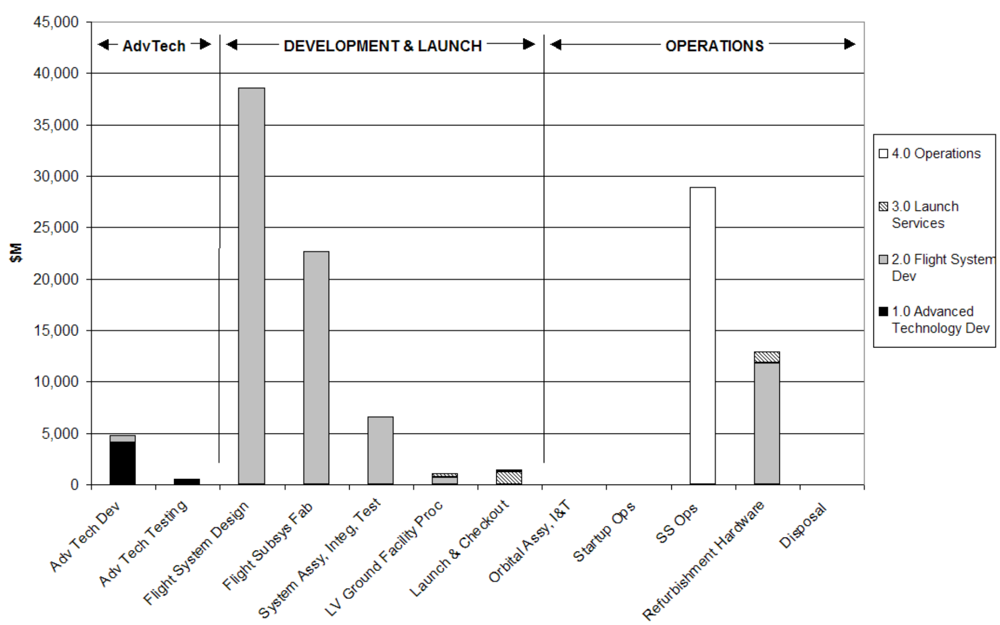

Figure 9-5 Case 1: Aldrin, ISRU, Solar, $2k/Kg LV; Summary Cost Results

Table 9-8 Case 1: Aldrin, ISRU, Solar, $2k/Kg LV; Life Cycle Costs by WBS

Case1_Aldrin_2000_ISRU_Solar_0326 — 15 = years of operation — Return

Life Cycle Cost WBS Elements *	Life Cycle Cost Phase, FY'00 $M ** Adv Tech Dev: Adv Tech Dev	Adv Tech Testing	Development: Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch: Launch & Checkout	Operations: Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	4,085	540	0	0	0	0	0	0	0	0	0	0	4,626
2.0 Flight System Development	681	0	38,601	22,663	6,564	699	0	0	0	0	11,839	0	81,048
3.0 Launch Services	0	0	0	0	0	425	1,306	0	0	0	1,048	0	2,779
4.0 Operations	0	0	0	0	0	0	161	0	0	28,936	0	0	29,096
LCC TOTAL	4,766	540	38,601	22,663	6,564	1,124	1,467	0	0	28,936	12,887	0	117,549
1.0 Advanced Technology Dev	4,085	540											4,626
1.1 General R&D	681	340											1,021
1.2 Facilities	3,404												3,404
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	681	0	38,601	22,663	6,564	699	0	0	0	0	11,839	0	81,048
2.1 Flight Elements	516	0	29,243	17,169	4,973	529	0	0	0	0	8,969	0	61,400
2.1.1 Astrotel	65	0	3,261	3,405	1,098	64	0	0	0	0	845	0	8,739
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	0	0	5,186
2.1.4 Mars Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	484	0	5,670
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	8	0	196	148	48	8	0	0	0	0	307	0	715
2.1.7 Astrotel Cargo Freighter	3	0	85	64	21	3	0	0	0	0	135	0	311
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	4	0	187	92	24	3	0	0	0	0	0	0	310
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	283	0	14,135	8,750	2,319	316	0	0	0	0	4,755	0	30,559
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	7	0	355	127	34	5	0	0	0	0	79	0	606
2.1.15 L1 Water Electr & Cryo Storage	0	0	19	13	3	0	0	0	0	0	4	0	39
2.1.16 Phobos LOX Plant	8	0	418	190	50	7	0	0	0	0	253	0	926
2.1.17 Mars Surface Water Plant	9	0	427	215	57	8	0	0	0	0	278	0	994
2.1.18 Mars Spaceport LOX/LH2 Storage	0	0	22	22	7	1	0	0	0	0	14	0	67
2.2 Flight System Dev Support	52	0	2,924	1,717	497	53	0	0	0	0	897	0	6,140
2.3 Other Flight System Costs													0
2.4 Development Reserves	113	0	6,433	3,777	1,094	116	0	0	0	0	1,973	0	13,508
3.0 Launch Services						425	1,306				1,048		2,779
3.1 Launch Approval						386					118		504
3.2 Launch Processing						39					27		66
3.3 Launch Vehicle							1,306				902		2,209
3.4 Other Launch Services Costs													0
4.0 Operations							161			28,936			29,096
4.1 Operations Project Mgmt							16			2,894			2,910
4.2 Integrated Logistics							16			2,894			2,910
4.3 Flight Operations							16			2,894			2,910
4.4 Training Operations							16			2,894			2,910
4.5 Launch Operations							16			2,894			2,910
4.6 In-Space Crew Support							16			2,894			2,910
4.7 Comm/Data Handling Ops							16			2,894			2,910
4.8 Operations Proj Supp Costs							16			2,894			2,910
4.9 Other Operations Costs							16			2,894			2,910
4.10 Operations Reserves							16			2,894			2,910
LIFE CYCLE COST TOTAL	4,766	540	38,601	22,663	6,564	1,124	1,467	0	0	28,936		0	117,549

Table 9-9 Case 2: Aldrin, ISRU, Solar, $10k/Kg LV; Flight Element Masses

Case2_Aldrin_10000_ISRU_Solar_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**653,037**	**451,220**
1	Astrotel	**2**	55,754	111,509	17,173
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*5,664*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	17,173
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	8,335	8,335	5,334
	Mars Cargo Freighter Xe Propellant				*139,889*
7	Astrotel Cargo Freighter	**1**	3,553	3,553	2,340
	Astrotel Cargo Freighter Xe Propellant				*56,688*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	6,917	6,917	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	278,757	278,757	50,942
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	3,074	3,074	1,031
15	L1 Water Electrolysis & Cyro Storage	**1**	584	584	80
16	Phobos LOX Plant	**1**	4,262	4,262	3,243
17	Mars Surface Water Plant	**1**	5,261	5,261	3,712
18	Mars Spaceport LOX/LH2 Storage	**1**	1,230	1,230	258
19					
TOTALS	Flight Elements, dry			**653,037**	**148,656**
	Flight Elements, wet			**0**	**302,565**

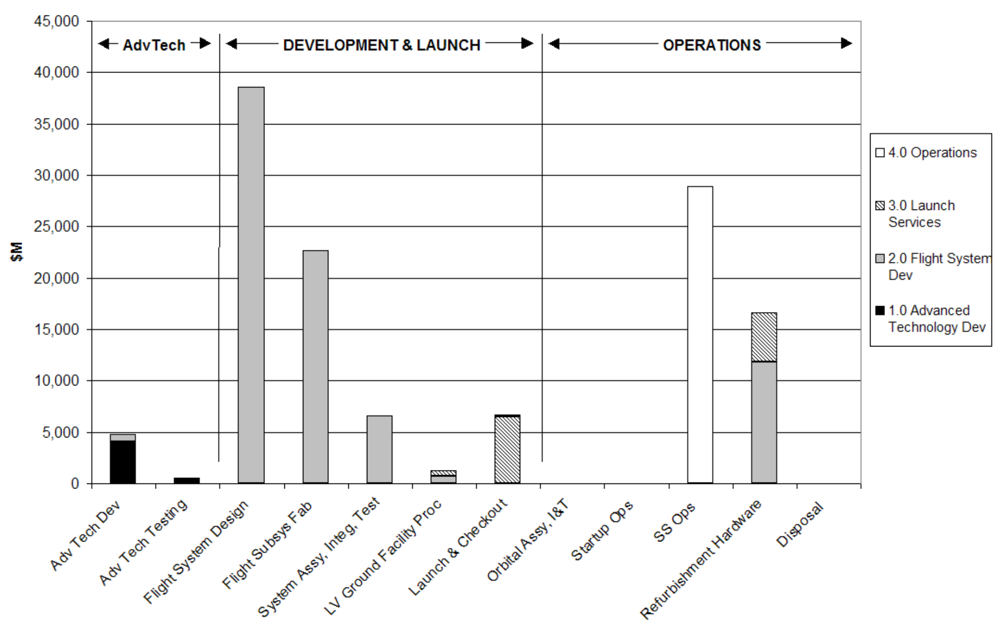

Figure 9-6 Case 2: Aldrin, ISRU, Solar, $10k/Kg LV; Summary Cost Results

Table 9-10 Case 2: Aldrin, ISRU, Solar, $10k/Kg LV; Life Cycle Costs By WBS

Case2_Aldrin_10000_ISRU_Solar_0326		15	= years of operation								Return		
	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	4,085	540	0	0	0	0	0	0	0	0	0	0	4,626
2.0 Flight System Development	681	0	38,601	22,663	6,564	699	0	0	0	0	11,839	0	81,048
3.0 Launch Services	0	0	0	0	0	582	6,530	0	0	0	4,766	0	11,878
4.0 Operations	0	0	0	0	0	0	161	0	0	28,936	0	0	29,096
LCC TOTAL	4,766	540	38,601	22,663	6,564	1,281	6,691	0	0	28,936	16,605	0	126,648
1.0 Advanced Technology Dev	4,085	540											4,626
1.1 General R&D	681	340											1,021
1.2 Facilities	3,404												3,404
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	681	0	38,601	22,663	6,564	699	0	0	0	0	11,839	0	81,048
2.1 Flight Elements	516	0	29,243	17,169	4,973	529	0	0	0	0	8,969	0	61,400
2.1.1 Astrotel	65	0	3,261	3,405	1,098	64	0	0	0	0	845	0	8,739
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	0	0	5,186
2.1.4 Mars Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	484	0	5,670
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	8	0	196	148	48	8	0	0	0	0	307	0	715
2.1.7 Astrotel Cargo Freighter	3	0	85	64	21	3	0	0	0	0	135	0	311
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	4	0	187	92	24	3	0	0	0	0	0	0	310
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	283	0	14,135	8,750	2,319	316	0	0	0	0	4,755	0	30,559
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	7	0	355	127	34	5	0	0	0	0	79	0	606
2.1.15 L1 Water Electr & Cryo Storage	0	0	19	13	3	0	0	0	0	0	4	0	39
2.1.16 Phobos LOX Plant	8	0	418	190	50	7	0	0	0	0	253	0	926
2.1.17 Mars Surface Water Plant	9	0	427	215	57	8	0	0	0	0	278	0	994
2.1.18 Mars Spaceport LOX/LH2 Storage	0	0	22	22	7	1	0	0	0	0	14	0	67
2.2 Flight System Dev Support	52	0	2,924	1,717	497	53	0	0	0	0	897	0	6,140
2.3 Other Flight System Costs													0
2.4 Development Reserves	113	0	6,433	3,777	1,094	116	0	0	0	0	1,973	0	13,508
3.0 Launch Services						582	6,530				4,766		11,878
3.1 Launch Approval						386					118		504
3.2 Launch Processing						196					135		331
3.3 Launch Vehicle							6,530				4,512		11,043
3.4 Other Launch Services Costs													0
4.0 Operations							161			28,936			29,096
4.1 Operations Project Mgmt							16			2,894			2,910
4.2 Integrated Logistics							16			2,894			2,910
4.3 Flight Operations							16			2,894			2,910
4.4 Training Operations							16			2,894			2,910
4.5 Launch Operations							16			2,894			2,910
4.6 In-Space Crew Support							16			2,894			2,910
4.7 Comm/Data Handling Ops							16			2,894			2,910
4.8 Operations Proj Supp Costs							16			2,894			2,910
4.9 Other Operations Costs							16			2,894			2,910
4.10 Operations Reserves							16			2,894			2,910
LIFE CYCLE COST TOTAL	4,766	540	38,601	22,663	6,564	1,281	6,691	0	0	28,936		0	126,648

Table 9-11 Case 3: Aldrin, ISRU, Nuclear, $2k/Kg LV; Flight Element Masses

Case3_Aldrin_2000_ISRU_Nuclear_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**998,339**	**5,667,161**
1	Astrotel	**2**	60,547	121,094	26,758
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*6,063*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	26,758
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	303,987	303,987	194,552
	Mars Cargo Freighter Xe Propellant				*5,101,877*
7	Astrotel Cargo Freighter	**1**	3,841	3,841	2,529
	Astrotel Cargo Freighter Xe Propellant				*61,283*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	6,917	6,917	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	283,191	283,191	55,796
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	7,700	7,700	7,150
15	L1 Water Electrolysis & Cyro Storage	**1**	584	584	80
16	Phobos LOX Plant	**1**	26,583	26,583	24,128
17	Mars Surface Water Plant	**1**	13,658	13,658	12,236
18	Mars Spaceport LOX/LH2 Storage	**1**	1,230	1,230	258
19					
TOTALS	Flight Elements, dry			**998,339**	**397,615**
	Flight Elements, wet			**0**	**5,269,546**

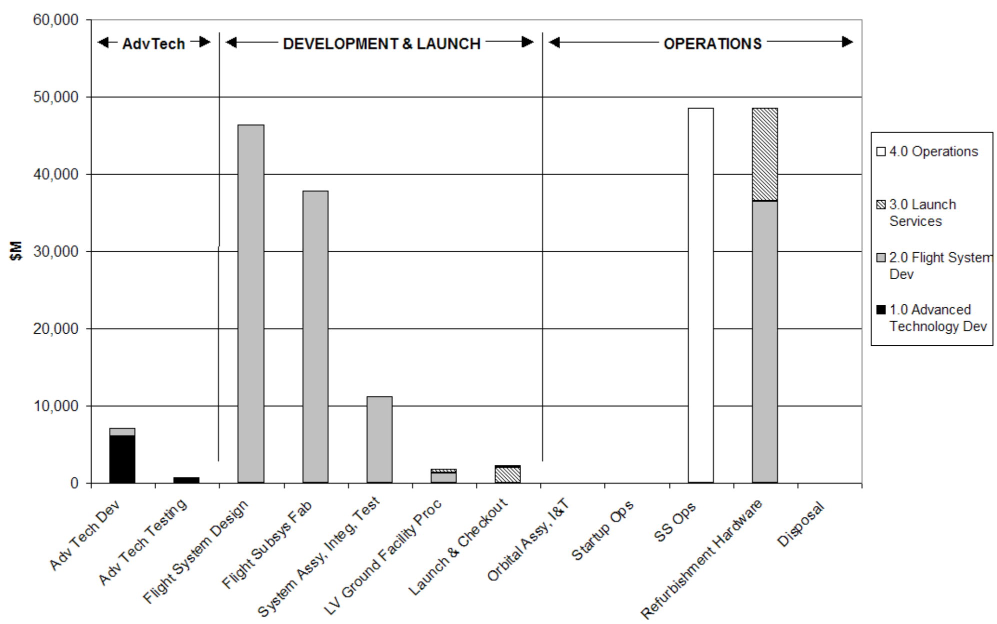

Figure 9-7 Case 3: Aldrin, ISRU, Nuclear, $2k/kg LV; Summary Cost Results

Table 9-12 Case 3: Aldrin, ISRU, Nuclear, $2k/kg LV; Life Cycle Costs By WBS

Case3_Aldrin_2000_ISRU_Nuclear_0326

	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	6,060	705	0	0	0	0	0	0	0	0	0	0	6,765
2.0 Flight System Development	1,010	0	46,397	37,880	11,169	1,290	0	0	0	0	36,495	0	134,241
3.0 Launch Services	0	0	0	0	0	524	1,997	0	0	0	12,039	0	14,560
4.0 Operations	0	0	0	0	0	0	270	0	0	48,558	0	0	48,828
LCC TOTAL	7,070	705	46,397	37,880	11,169	1,814	2,266	0	0	48,558	48,534	0	204,394
1.0 Advanced Technology Dev	6,060	705											6,765
1.1 General R&D	1,010	505											1,515
1.2 Facilities	5,050												5,050
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	1,010	0	46,397	37,880	11,169	1,290	0	0	0	0	36,495	0	134,241
2.1 Flight Elements	765	0	35,149	28,697	8,461	977	0	0	0	0	27,648	0	101,698
2.1.1 Astrotel	63	0	3,169	5,390	1,738	102	0	0	0	0	2,720	0	13,182
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	0	0	5,411
2.1.4 Mars Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	1,060	0	6,471
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	252	0	6,293	4,668	1,506	247	0	0	0	0	9,312	0	22,277
2.1.7 Astrotel Cargo Freighter	3	0	81	60	19	3	0	0	0	0	121	0	287
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	4	0	187	92	24	3	0	0	0	0	0	0	310
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	273	0	13,635	8,894	2,357	321	0	0	0	0	5,052	0	30,532
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	6	0	301	800	212	29	0	0	0	0	1,412	0	2,759
2.1.15 L1 Water Electr & Cryo Storage	0	0	19	13	3	0	0	0	0	0	4	0	39
2.1.16 Phobos LOX Plant	32	0	1,586	2,293	608	83	0	0	0	0	3,918	0	8,519
2.1.17 Mars Surface Water Plant	13	0	660	1,284	340	46	0	0	0	0	2,221	0	4,564
2.1.18 Mars Spaceport LOX/LH2 Storage	0	0	22	22	7	1	0	0	0	0	14	0	67
2.2 Flight System Dev Support	77	0	3,515	2,870	846	98	0	0	0	0	2,765	0	10,170
2.3 Other Flight System Costs													0
2.4 Development Reserves	168	0	7,733	6,313	1,861	215	0	0	0	0	6,082	0	22,374
3.0 Launch Services						524	1,997				12,039		14,560
3.1 Launch Approval						464					365		829
3.2 Launch Processing						60					340		400
3.3 Launch Vehicle							1,997				11,334		13,331
3.4 Other Launch Services Costs													0
4.0 Operations							270			48,558			48,828
4.1 Operations Project Mgmt							27			4,856			4,883
4.2 Integrated Logistics							27			4,856			4,883
4.3 Flight Operations							27			4,856			4,883
4.4 Training Operations							27			4,856			4,883
4.5 Launch Operations							27			4,856			4,883
4.6 In-Space Crew Support							27			4,856			4,883
4.7 Comm/Data Handling Ops							27			4,856			4,883
4.8 Operations Proj Supp Costs							27			4,856			4,883
4.9 Other Operations Costs							27			4,856			4,883
4.10 Operations Reserves							27			4,856			4,883
LIFE CYCLE COST TOTAL	7,070	705	46,397	37,880	11,169	1,814	2,266	0	0	48,558		0	204,394

Table 9-13 Case 4: Aldrin, ISRU, Nuclear, $10k/kg LV; Flight Element Masses

Case4_Aldrin_10000_ISRU_Nuclear_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**998,339**	**5,667,161**
1	Astrotel	**2**	60,547	121,094	26,758
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*6,063*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	26,758
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	303,987	303,987	194,552
	Mars Cargo Freighter Xe Propellant				*5,101,877*
7	Astrotel Cargo Freighter	**1**	3,841	3,841	2,529
	Astrotel Cargo Freighter Xe Propellant				*61,283*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	6,917	6,917	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	283,191	283,191	55,796
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	7,700	7,700	7,150
15	L1 Water Electrolysis & Cyro Storage	**1**	584	584	80
16	Phobos LOX Plant	**1**	26,583	26,583	24,128
17	Mars Surface Water Plant	**1**	13,658	13,658	12,236
18	Mars Spaceport LOX/LH2 Storage	**1**	1,230	1,230	258
19					
TOTALS	Flight Elements, dry			**998,339**	**397,615**
	Flight Elements, wet			**0**	**5,269,546**

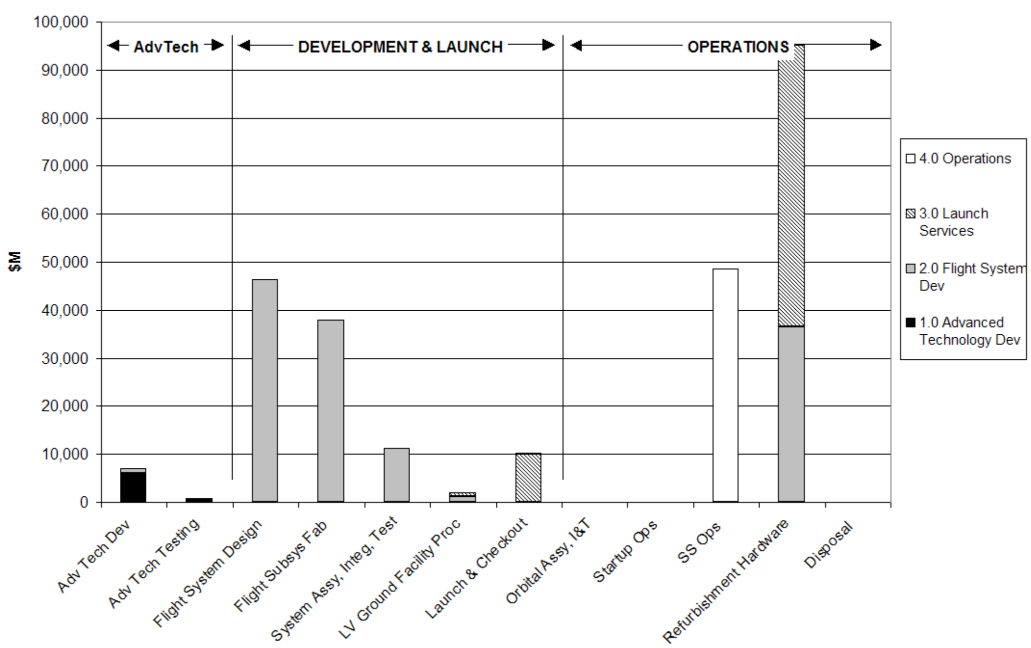

Figure 9-8 Case 4: Aldrin, ISRU, Nuclear, $10k/kg LV; Summary Cost Results

Table 9-14 Case 4: Aldrin, ISRU, Nuclear, $10k/kg LV; Life Cycle Costs By WBS

Case4_Aldrin_10000_ISRU_Nuclear_0326		15	= years of operation								Return		
	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	6,060	705	0	0	0	0	0	0	0	0	0	0	6,765
2.0 Flight System Development	1,010	0	46,397	37,880	11,169	1,290	0	0	0	0	36,495	0	134,241
3.0 Launch Services	0	0	0	0	0	763	9,983	0	0	0	58,737	0	69,484
4.0 Operations	0	0	0	0	0	0	270	0	0	48,558	0	0	48,828
LCC TOTAL	7,070	705	46,397	37,880	11,169	2,054	10,253	0	0	48,558	95,232	0	259,318
1.0 Advanced Technology Dev	6,060	705											6,765
1.1 General R&D	1,010	505											1,515
1.2 Facilities	5,050												5,050
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	1,010	0	46,397	37,880	11,169	1,290	0	0	0	0	36,495	0	134,241
2.1 Flight Elements	765	0	35,149	28,697	8,461	977	0	0	0	0	27,648	0	101,698
2.1.1 Astrotel	63	0	3,169	5,390	1,738	102	0	0	0	0	2,720	0	13,182
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	0	0	5,411
2.1.4 Mars Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	1,060	0	6,471
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	252	0	6,293	4,668	1,506	247	0	0	0	0	9,312	0	22,277
2.1.7 Astrotel Cargo Freighter	3	0	81	60	19	3	0	0	0	0	121	0	287
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	4	0	187	92	24	3	0	0	0	0	0	0	310
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	273	0	13,635	8,894	2,357	321	0	0	0	0	5,052	0	30,532
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	6	0	301	800	212	29	0	0	0	0	1,412	0	2,759
2.1.15 L1 Water Electr & Cryo Storage	0	0	19	13	3	0	0	0	0	0	4	0	39
2.1.16 Phobos LOX Plant	32	0	1,586	2,293	608	83	0	0	0	0	3,918	0	8,519
2.1.17 Mars Surface Water Plant	13	0	660	1,284	340	46	0	0	0	0	2,221	0	4,564
2.1.18 Mars Spaceport LOX/LH2 Storage	0	0	22	22	7	1	0	0	0	0	14	0	67
2.2 Flight System Dev Support	77	0	3,515	2,870	846	98	0	0	0	0	2,765	0	10,170
2.3 Other Flight System Costs													0
2.4 Development Reserves	168	0	7,733	6,313	1,861	215	0	0	0	0	6,082	0	22,374
3.0 Launch Services						763	9,983				58,737		69,484
3.1 Launch Approval						464					365		829
3.2 Launch Processing						300					1,700		2,000
3.3 Launch Vehicle							9,983				56,672		66,655
3.4 Other Launch Services Costs													0
4.0 Operations							270			48,558			48,828
4.1 Operations Project Mgmt							27			4,856			4,883
4.2 Integrated Logistics							27			4,856			4,883
4.3 Flight Operations							27			4,856			4,883
4.4 Training Operations							27			4,856			4,883
4.5 Launch Operations							27			4,856			4,883
4.6 In-Space Crew Support							27			4,856			4,883
4.7 Comm/Data Handling Ops							27			4,856			4,883
4.8 Operations Proj Supp Costs							27			4,856			4,883
4.9 Other Operations Costs							27			4,856			4,883
4.10 Operations Reserves							27			4,856			4,883
LIFE CYCLE COST TOTAL	7,070	705	46,397	37,880	11,169	2,054	10,253	0	0	48,558		0	259,318

Table 9-15 Case 5: Aldrin, No ISRU, Solar, $2k/kg LV; Flight Element Masses

Case5_Aldrin_2000_NoISRU_Solar_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**668,800**	**906,060**
1	Astrotel	**2**	55,754	111,509	17,173
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*5,664*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	17,173
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	34,767	34,767	22,251
	Mars Cargo Freighter Xe Propellant				*583,497*
7	Astrotel Cargo Freighter	**1**	3,553	3,553	2,340
	Astrotel Cargo Freighter Xe Propellant				*56,688*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	5,970	5,970	0
10	Mars Shuttle	**1**	6,307	6,307	3,105
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	278,757	278,757	50,942
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	0	0	0
15	L1 Water Electrolysis & Cyro Storage	**1**	46	46	5
16	Phobos LOX Plant	**1**	0	0	0
17	Mars Surface Water Plant	**1**	46	46	9
18	Mars Spaceport LOX/LH2 Storage	**1**	1,430	1,430	300
19					
TOTALS	Flight Elements, dry			**668,800**	**159,888**
	Flight Elements, wet			**0**	**746,172**

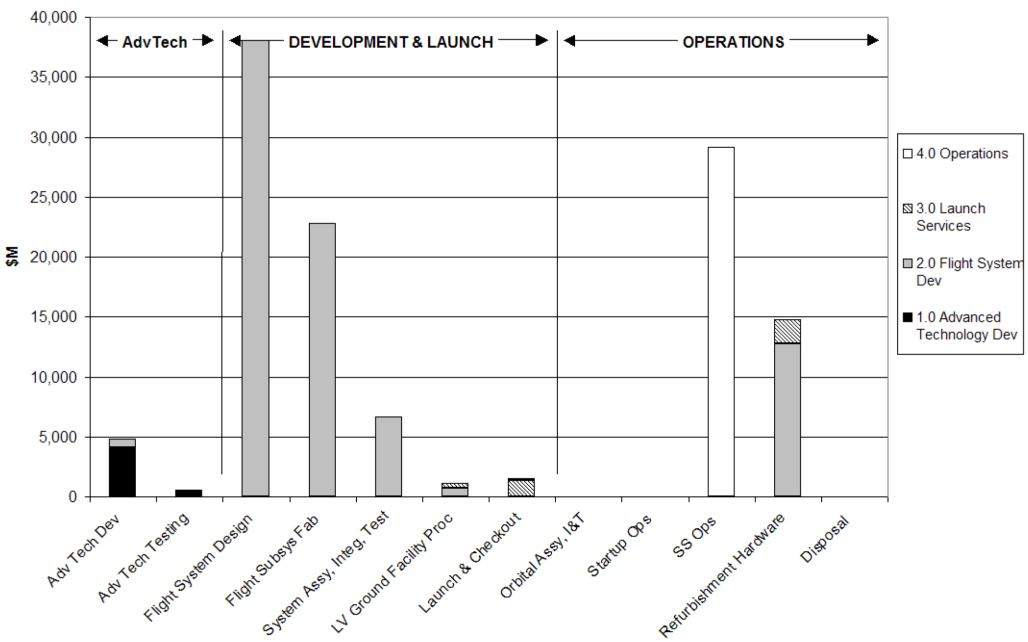

Figure 9-9 Case 1: Aldrin, ISRU, Solar, $2k/kg LV; Summary Cost Results

Table 9-16 Case 5: Aldrin, No ISRU, Solar, $2k/kg LV; Life Cycle Costs By WBS

Case5_Aldrin_2000_NoISRU_Solar_0326		15	= years of operation							Return			
	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	4,122	543	0	0	0	0	0	0	0	0	0	0	4,665
2.0 Flight System Development	687	0	38,086	22,787	6,633	714	0	0	0	0	12,776	0	81,683
3.0 Launch Services	0	0	0	0	0	421	1,338	0	0	0	1,994	0	3,753
4.0 Operations	0	0	0	0	0	0	162	0	0	29,126	0	0	29,288
LCC TOTAL	4,809	543	38,086	22,787	6,633	1,135	1,499	0	0	29,126	14,770	0	119,389
1.0 Advanced Technology Dev	4,122	543											4,665
1.1 General R&D	687	343											1,030
1.2 Facilities	3,435												3,435
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	687	0	38,086	22,787	6,633	714	0	0	0	0	12,776	0	81,683
2.1 Flight Elements	520	0	28,853	17,263	5,025	541	0	0	0	0	9,679	0	61,881
2.1.1 Astrotel	65	0	3,261	3,405	1,098	64	0	0	0	0	845	0	8,739
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	0	0	5,186
2.1.4 Mars Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	484	0	5,670
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	33	0	818	616	199	33	0	0	0	0	1,282	0	2,980
2.1.7 Astrotel Cargo Freighter	3	0	85	64	21	3	0	0	0	0	135	0	311
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	3	0	170	78	21	3	0	0	0	0	0	0	275
2.1.10 Mars Shuttle	12	0	600	602	160	22	0	0	0	0	535	0	1,932
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	283	0	14,135	8,750	2,319	316	0	0	0	0	4,755	0	30,559
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.15 L1 Water Electr & Cryo Storage	0	0	1	1	0	0	0	0	0	0	0	0	2
2.1.16 Phobos LOX Plant	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.17 Mars Surface Water Plant	0	0	1	1	0	0	0	0	0	0	0	0	2
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	26	26	8	1	0	0	0	0	17	0	78
2.2 Flight System Dev Support	52	0	2,885	1,726	503	54	0	0	0	0	968	0	6,188
2.3 Other Flight System Costs													0
2.4 Development Reserves	114	0	6,348	3,798	1,106	119	0	0	0	0	2,129	0	13,614
3.0 Launch Services						421	1,338				1,994		3,753
3.1 Launch Approval						381					128		509
3.2 Launch Processing						40					54		94
3.3 Launch Vehicle							1,338				1,812		3,150
3.4 Other Launch Services Costs													0
4.0 Operations							162			29,126			29,288
4.1 Operations Project Mgmt							16			2,913			2,929
4.2 Integrated Logistics							16			2,913			2,929
4.3 Flight Operations							16			2,913			2,929
4.4 Training Operations							16			2,913			2,929
4.5 Launch Operations							16			2,913			2,929
4.6 In-Space Crew Support							16			2,913			2,929
4.7 Comm/Data Handling Ops							16			2,913			2,929
4.8 Operations Proj Supp Costs							16			2,913			2,929
4.9 Other Operations Costs							16			2,913			2,929
4.10 Operations Reserves							16			2,913			2,929
LIFE CYCLE COST TOTAL	4,809	543	38,086	22,787	6,633	1,135	1,499	0	0	29,126		0	119,389

Table 9-17 Case 6: Aldrin, No ISRU, Solar, $10k/kg LV; Flight Element Masses

Case6_Aldrin_10000_NoISRU_Solar_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**668,800**	**906,060**
1	Astrotel	**2**	55,754	111,509	17,173
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*5,664*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	17,173
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	34,767	34,767	22,251
	Mars Cargo Freighter Xe Propellant				*583,497*
7	Astrotel Cargo Freighter	**1**	3,553	3,553	2,340
	Astrotel Cargo Freighter Xe Propellant				*56,688*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	5,970	5,970	0
10	Mars Shuttle	**1**	6,307	6,307	3,105
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	278,757	278,757	50,942
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	0	0	0
15	L1 Water Electrolysis & Cyro Storage	**1**	46	46	5
16	Phobos LOX Plant	**1**	0	0	0
17	Mars Surface Water Plant	**1**	46	46	9
18	Mars Spaceport LOX/LH2 Storage	**1**	1,430	1,430	300
19					
TOTALS	Flight Elements, dry			**668,800**	**159,888**
	Flight Elements, wet			**0**	**746,172**

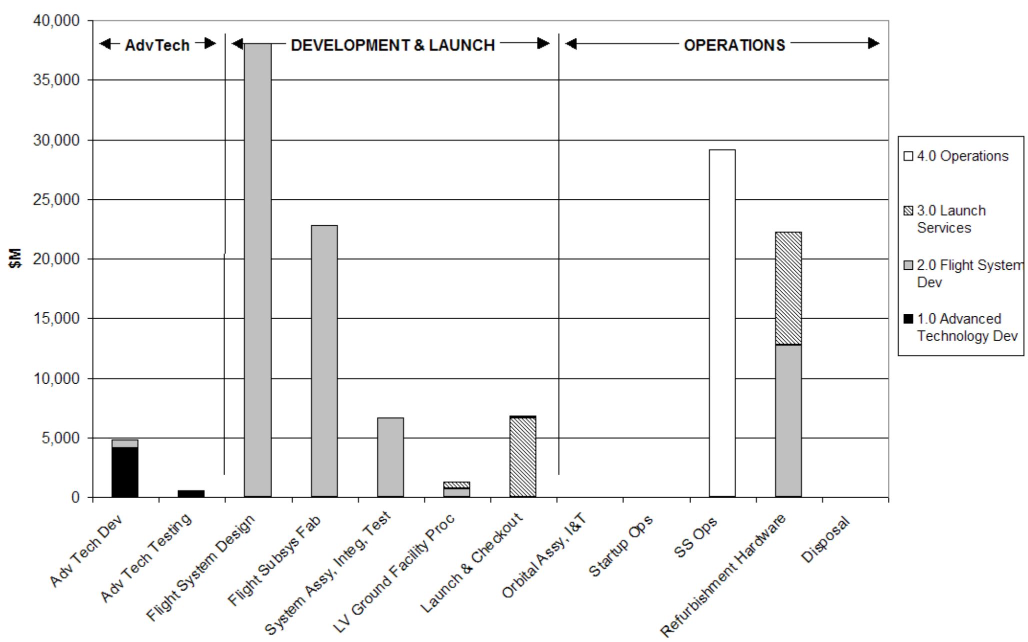

Figure 9-10 Case 6: Aldrin, No ISRU, Solar, $10k/kg LV; Summary Cost Results

Table 9-18 Case 6: Aldrin, No ISRU, Solar, $10k/kg LV; Life Cycle Costs By WBS

Case6_Aldrin_10000_NoISRU_Solar_0326

15 = years of operation

Return

Life Cycle Cost Phase, FY'00 $M **

Life Cycle Cost WBS Elements *	Adv Tech Dev		Development				Launch	Operations					
	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	4,122	543	0	0	0	0	0	0	0	0	0	0	4,665
2.0 Flight System Development	687	0	38,086	22,787	6,633	714	0	0	0	0	12,776	0	81,683
3.0 Launch Services	0	0	0	0	0	582	6,688	0	0	0	9,460	0	16,730
4.0 Operations	0	0	0	0	0	0	162	0	0	29,126	0	0	29,288
LCC TOTAL	4,809	543	38,086	22,787	6,633	1,295	6,850	0	0	29,126	22,236	0	132,366
1.0 Advanced Technology Dev	4,122	543											4,665
1.1 General R&D	687	343											1,030
1.2 Facilities	3,435												3,435
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	687	0	38,086	22,787	6,633	714	0	0	0	0	12,776	0	81,683
2.1 Flight Elements	520	0	28,853	17,263	5,025	541	0	0	0	0	9,679	0	61,881
2.1.1 Astrotel	65	0	3,261	3,405	1,098	64	0	0	0	0	845	0	8,739
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	0	0	5,186
2.1.4 Mars Spaceport	34	0	3,428	1,278	412	34	0	0	0	0	484	0	5,670
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	33	0	818	616	199	33	0	0	0	0	1,282	0	2,980
2.1.7 Astrotel Cargo Freighter	3	0	85	64	21	3	0	0	0	0	135	0	311
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	3	0	170	78	21	3	0	0	0	0	0	0	275
2.1.10 Mars Shuttle	12	0	600	602	160	22	0	0	0	0	535	0	1,932
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	283	0	14,135	8,750	2,319	316	0	0	0	0	4,755	0	30,559
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.15 L1 Water Electr & Cryo Storage	0	0	1	1	0	0	0	0	0	0	0	0	2
2.1.16 Phobos LOX Plant	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.17 Mars Surface Water Plant	0	0	1	1	0	0	0	0	0	0	0	0	2
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	26	26	8	1	0	0	0	0	17	0	78
2.2 Flight System Dev Support	52	0	2,885	1,726	503	54	0	0	0	0	968	0	6,188
2.3 Other Flight System Costs													0
2.4 Development Reserves	114	0	6,348	3,798	1,106	119	0	0	0	0	2,129	0	13,614
3.0 Launch Services						582	6,688				9,460		16,730
3.1 Launch Approval						381					128		509
3.2 Launch Processing						201					272		472
3.3 Launch Vehicle							6,688				9,061		15,749
3.4 Other Launch Services Costs													0
4.0 Operations							162			29,126			29,288
4.1 Operations Project Mgmt							16			2,913			2,929
4.2 Integrated Logistics							16			2,913			2,929
4.3 Flight Operations							16			2,913			2,929
4.4 Training Operations							16			2,913			2,929
4.5 Launch Operations							16			2,913			2,929
4.6 In-Space Crew Support							16			2,913			2,929
4.7 Comm/Data Handling Ops							16			2,913			2,929
4.8 Operations Proj Supp Costs							16			2,913			2,929
4.9 Other Operations Costs							16			2,913			2,929
4.10 Operations Reserves							16			2,913			2,929
LIFE CYCLE COST TOTAL	4,809	543	38,086	22,787	6,633	1,295	6,850	0	0	29,126		0	132,366

Table 9-19 Case 7: Aldrin, No ISRU, Nuclear, $2k/kg LV ; Flight Element Masses

Case7_Aldrin_2000_NoISRU_Nuclear_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**991,638**	**6,087,810**
1	Astrotel	**2**	60,547	121,094	26,758
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*6,063*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	26,758
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	329,716	329,716	211,018
	Mars Cargo Freighter Xe Propellant				*5,533,689*
7	Astrotel Cargo Freighter	**1**	3,841	3,841	2,529
	Astrotel Cargo Freighter Xe Propellant				*61,283*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	5,970	5,970	0
10	Mars Shuttle	**1**	6,307	6,307	3,105
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	283,191	283,191	55,796
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	6,815	6,815	6,815
15	L1 Water Electrolysis & Cyro Storage	**1**	46	46	5
16	Phobos LOX Plant	**1**	3,384	3,384	3,384
17	Mars Surface Water Plant	**1**	3,430	3,430	3,393
18	Mars Spaceport LOX/LH2 Storage	**1**	1,430	1,430	300
19					
TOTALS	Flight Elements, dry			**991,638**	**386,451**
	Flight Elements, wet			**0**	**5,701,358**

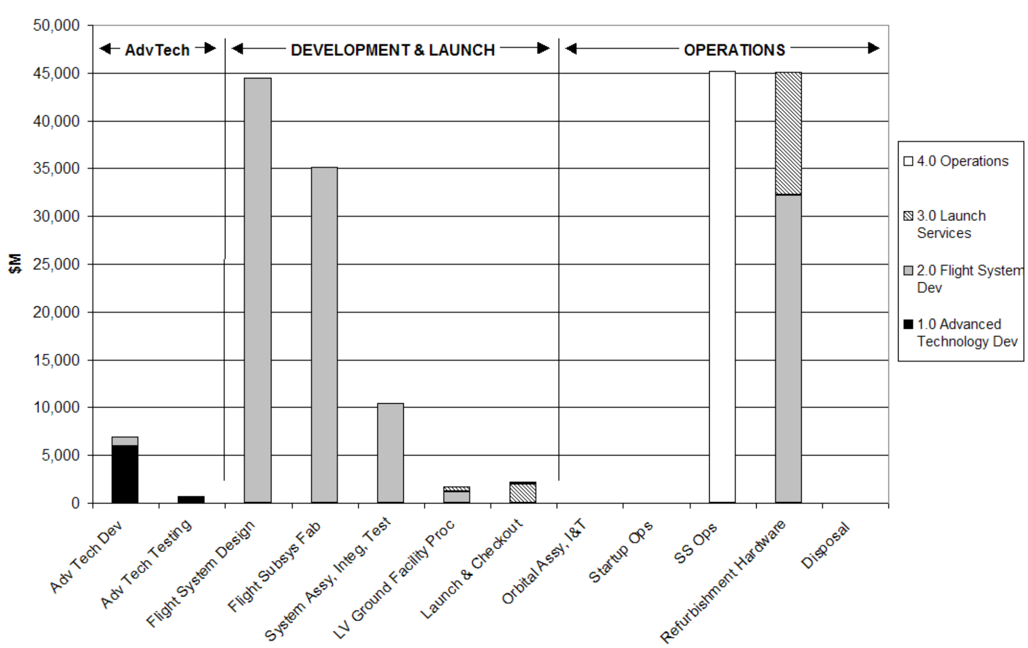

Figure 9-11 Case 7: Aldrin, No ISRU, Nuclear, $2k/kg LV; Summary Cost Results

Table 9-20 Case 7: Aldrin, No ISRU, Nuclear, $2k/kg LV; Life Cycle Costs By WBS

15 = years of operation

Return

Case7_Aldrin_2000_NoISRU_Nuclear_0326

	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	5,918	693	0	0	0	0	0	0	0	0	0	0	6,612
2.0 Flight System Development	986	0	44,513	35,159	10,478	1,201	0	0	0	0	32,230	0	124,567
3.0 Launch Services	0	0	0	0	0	505	1,983	0	0	0	12,863	0	15,351
4.0 Operations	0	0	0	0	0	0	251	0	0	45,181	0	0	45,432
LCC TOTAL	6,905	693	44,513	35,159	10,478	1,705	2,234	0	0	45,181	45,093	0	191,962
1.0 Advanced Technology Dev	5,918	693											6,612
1.1 General R&D	986	493											1,480
1.2 Facilities	4,932												4,932
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	986	0	44,513	35,159	10,478	1,201	0	0	0	0	32,230	0	124,567
2.1 Flight Elements	747	0	33,722	26,636	7,938	910	0	0	0	0	24,417	0	94,369
2.1.1 Astrotel	63	0	3,169	5,390	1,738	102	0	0	0	0	2,720	0	13,182
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	0	0	5,411
2.1.4 Mars Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	1,060	0	6,471
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	273	0	6,825	5,063	1,633	268	0	0	0	0	10,100	0	24,163
2.1.7 Astrotel Cargo Freighter	3	0	81	60	19	3	0	0	0	0	121	0	287
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	3	0	170	78	21	3	0	0	0	0	0	0	275
2.1.10 Mars Shuttle	12	0	600	602	160	22	0	0	0	0	535	0	1,932
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	273	0	13,635	8,894	2,357	321	0	0	0	0	5,052	0	30,532
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	3	0	133	588	156	21	0	0	0	0	1,062	0	1,962
2.1.15 L1 Water Electr & Cryo Storage	0	0	1	1	0	0	0	0	0	0	0	0	2
2.1.16 Phobos LOX Plant	3	0	133	588	156	21	0	0	0	0	1,062	0	1,962
2.1.17 Mars Surface Water Plant	3	0	134	588	156	21	0	0	0	0	1,062	0	1,964
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	26	26	8	1	0	0	0	0	17	0	78
2.2 Flight System Dev Support	75	0	3,372	2,664	794	91	0	0	0	0	2,442	0	9,437
2.3 Other Flight System Costs													0
2.4 Development Reserves	164	0	7,419	5,860	1,746	200	0	0	0	0	5,372	0	20,761
3.0 Launch Services						505	1,983				12,863		15,351
3.1 Launch Approval						445					322		767
3.2 Launch Processing						59					365		425
3.3 Launch Vehicle							1,983				12,176		14,159
3.4 Other Launch Services Costs													0
4.0 Operations							251			45,181			45,432
4.1 Operations Project Mgmt							25			4,518			4,543
4.2 Integrated Logistics							25			4,518			4,543
4.3 Flight Operations							25			4,518			4,543
4.4 Training Operations							25			4,518			4,543
4.5 Launch Operations							25			4,518			4,543
4.6 In-Space Crew Support							25			4,518			4,543
4.7 Comm/Data Handling Ops							25			4,518			4,543
4.8 Operations Proj Supp Costs							25			4,518			4,543
4.9 Other Operations Costs							25			4,518			4,543
4.10 Operations Reserves							25			4,518			4,543
LIFE CYCLE COST TOTAL	6,905	693	44,513	35,159	10,478	1,705	2,234	0	0	45,181		0	191,962

Table 9-21 Case 8: Aldrin, No ISRU, Nuclear, $10k/kg LV; Flight Element Masses

Case8_Aldrin_10000_NoISRU_Nuclear_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**991,638**	**6,087,810**
1	Astrotel	**2**	60,547	121,094	26,758
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*6,063*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	60,000	60,000	0
4	Mars Spaceport	**1**	60,000	60,000	26,758
5	Taxi	**2**	40,707	81,414	23,296
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	329,716	329,716	211,018
	Mars Cargo Freighter Xe Propellant				*5,533,689*
7	Astrotel Cargo Freighter	**1**	3,841	3,841	2,529
	Astrotel Cargo Freighter Xe Propellant				*61,283*
8	LEO Shuttle	**1**	15,000	15,000	23,296
9	Lunar Water Tanker	**1**	5,970	5,970	0
10	Mars Shuttle	**1**	6,307	6,307	3,105
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	283,191	283,191	55,796
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	6,815	6,815	6,815
15	L1 Water Electrolysis & Cyro Storage	**1**	46	46	5
16	Phobos LOX Plant	**1**	3,384	3,384	3,384
17	Mars Surface Water Plant	**1**	3,430	3,430	3,393
18	Mars Spaceport LOX/LH2 Storage	**1**	1,430	1,430	300
19					
TOTALS	Flight Elements, dry			**991,638**	**386,451**
	Flight Elements, wet			**0**	**5,701,358**

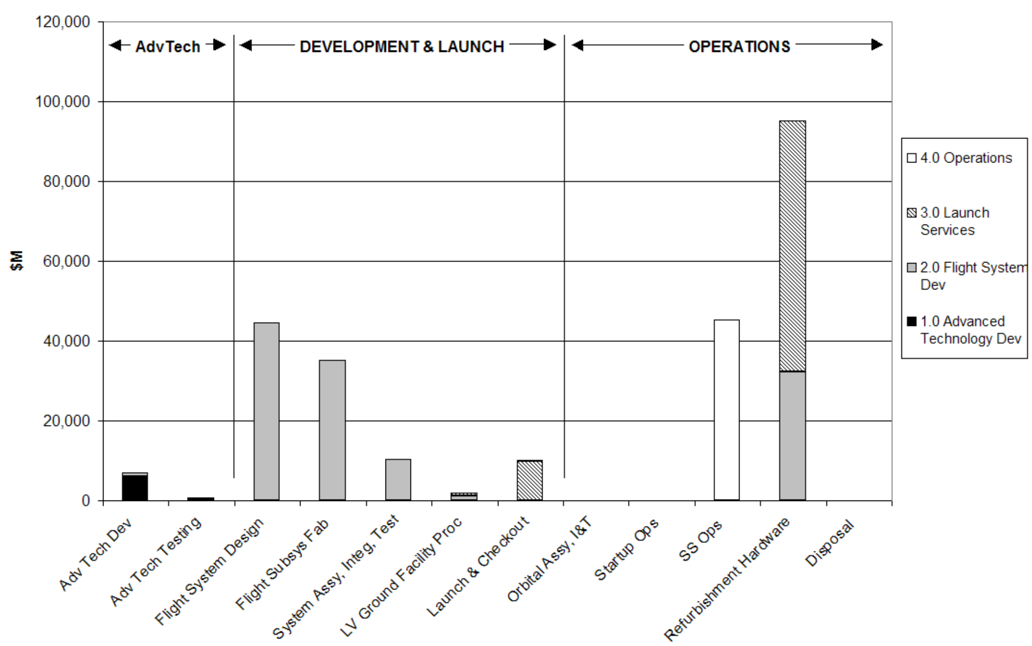

Figure 9-12 Case 8: Aldrin, No ISRU, Nuclear, $10k/kg LV; Summary Cost Results

Table 9-22 Case 8: Aldrin, No ISRU, Nuclear, $10k/kg LV; Life Cycle Costs By WBS

Case8_Aldrin_10000_NoISRU_Nuclear_0326		15	= years of operation							Return			
	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	5,918	693	0	0	0	0	0	0	0	0	0	0	6,612
2.0 Flight System Development	986	0	44,513	35,159	10,478	1,201	0	0	0	0	32,230	0	124,567
3.0 Launch Services	0	0	0	0	0	743	9,916	0	0	0	63,027	0	73,686
4.0 Operations	0	0	0	0	0	0	251	0	0	45,181	0	0	45,432
LCC TOTAL	6,905	693	44,513	35,159	10,478	1,943	10,167	0	0	45,181	95,257	0	250,297
1.0 Advanced Technology Dev	5,918	693											6,612
1.1 General R&D	986	493											1,480
1.2 Facilities	4,932												4,932
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	986	0	44,513	35,159	10,478	1,201	0	0	0	0	32,230	0	124,567
2.1 Flight Elements	747	0	33,722	26,636	7,938	910	0	0	0	0	24,417	0	94,369
2.1.1 Astrotel	63	0	3,169	5,390	1,738	102	0	0	0	0	2,720	0	13,182
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	0	0	5,411
2.1.4 Mars Spaceport	30	0	2,958	1,797	580	48	0	0	0	0	1,060	0	6,471
2.1.5 Taxi	13	0	670	615	198	12	0	0	0	0	410	0	1,919
2.1.6 Mars Cargo Freighter	273	0	6,825	5,063	1,633	268	0	0	0	0	10,100	0	24,163
2.1.7 Astrotel Cargo Freighter	3	0	81	60	19	3	0	0	0	0	121	0	287
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	1,215	0	3,427
2.1.9 Lunar Water Tanker	3	0	170	78	21	3	0	0	0	0	0	0	275
2.1.10 Mars Shuttle	12	0	600	602	160	22	0	0	0	0	535	0	1,932
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	273	0	13,635	8,894	2,357	321	0	0	0	0	5,052	0	30,532
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	3	0	133	588	156	21	0	0	0	0	1,062	0	1,962
2.1.15 L1 Water Electr & Cryo Storage	0	0	1	1	0	0	0	0	0	0	0	0	2
2.1.16 Phobos LOX Plant	3	0	133	588	156	21	0	0	0	0	1,062	0	1,962
2.1.17 Mars Surface Water Plant	3	0	134	588	156	21	0	0	0	0	1,062	0	1,964
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	26	26	8	1	0	0	0	0	17	0	78
2.2 Flight System Dev Support	75	0	3,372	2,664	794	91	0	0	0	0	2,442	0	9,437
2.3 Other Flight System Costs													0
2.4 Development Reserves	164	0	7,419	5,860	1,746	200	0	0	0	0	5,372	0	20,761
3.0 Launch Services						743	9,916				63,027		73,686
3.1 Launch Approval						445					322		767
3.2 Launch Processing						297					1,826		2,124
3.3 Launch Vehicle							9,916				60,878		70,794
3.4 Other Launch Services Costs													0
4.0 Operations							251			45,181			45,432
4.1 Operations Project Mgmt							25			4,518			4,543
4.2 Integrated Logistics							25			4,518			4,543
4.3 Flight Operations							25			4,518			4,543
4.4 Training Operations							25			4,518			4,543
4.5 Launch Operations							25			4,518			4,543
4.6 In-Space Crew Support							25			4,518			4,543
4.7 Comm/Data Handling Ops							25			4,518			4,543
4.8 Operations Proj Supp Costs							25			4,518			4,543
4.9 Other Operations Costs							25			4,518			4,543
4.10 Operations Reserves							25			4,518			4,543
LIFE CYCLE COST TOTAL	6,905	693	44,513	35,159	10,478	1,943	10,167	0	0	45,181		0	250,297

Table 9-23 Case 9: Stopover, ISRU, Solar, $2k/kg LV; Flight Element Masses

Case9_StopOver_2000_ISRU_Solar_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**564,258**	**1,203,725**
1	Astrotel	**2**	86,249	172,497	78,161
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*8,203*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	0	0	0
4	Mars Spaceport	**1**	0	0	78,161
5	Taxi	**2**	3,000	6,000	0
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	50,005	50,005	32,003
	Mars Cargo Freighter Xe Propellant				*839,240*
7	Astrotel Cargo Freighter	**1**	237	237	156
	Astrotel Cargo Freighter Xe Propellant				*3,775*
8	LEO Shuttle	**1**	15,000	15,000	0
9	Lunar Water Tanker	**1**	7,163	7,163	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	278,757	278,757	50,942
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	5,273	5,273	1,667
15	L1 Water Electrolysis & Cyro Storage	**1**	723	723	100
16	Phobos LOX Plant	**1**	7,817	7,817	5,980
17	Mars Surface Water Plant	**1**	5,261	5,261	3,712
18	Mars Spaceport LOX/LH2 Storage	**1**	2,384	2,384	522
19					
TOTALS	Flight Elements, dry			**564,258**	**252,183**
	Flight Elements, wet			**0**	**951,542**

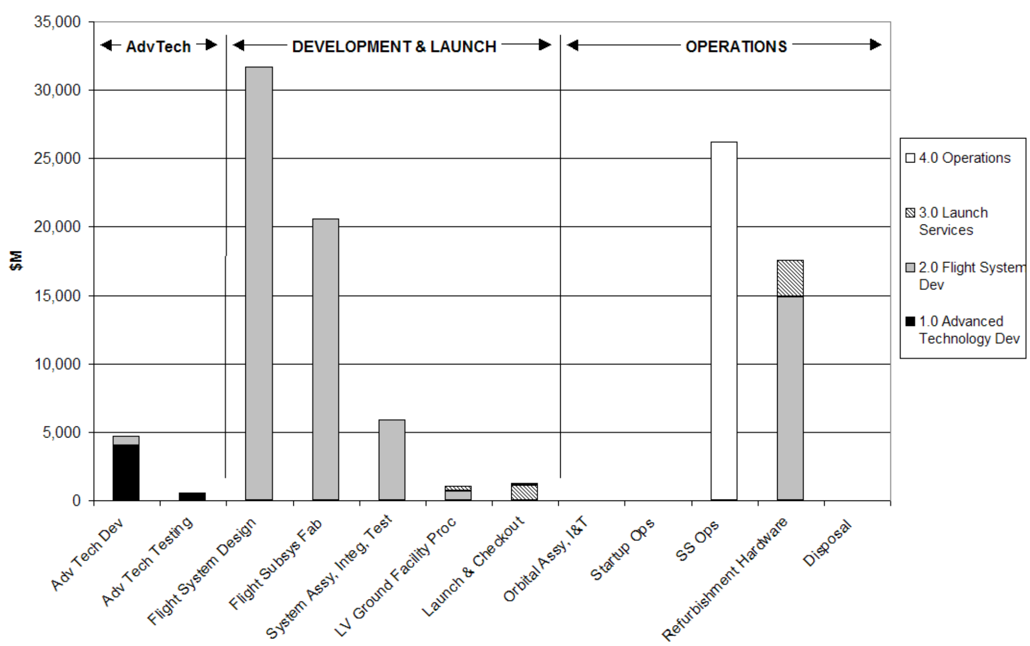

Figure 9-13 Case 9: Stopover, ISRU, Solar, $2k/kg LV; Summary Cost Results

Table 9-24 Case 9: Stopover, ISRU, Solar, $2k/kg LV; Life Cycle Costs by WBS

15 = years of operation

Return

Case9_StopOver_2000_ISRU_Solar_0326

Life Cycle Cost Phase, FY'00 $M **

	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	4,040	537	0	0	0	0	0	0	0	0	0	0	4,577
2.0 Flight System Development	673	0	31,663	20,585	5,874	695	0	0	0	0	14,908	0	74,399
3.0 Launch Services	0	0	0	0	0	350	1,129	0	0	0	2,629	0	4,108
4.0 Operations	0	0	0	0	0	0	146	0	0	26,195	0	0	26,340
LCC TOTAL	4,713	537	31,663	20,585	5,874	1,045	1,274	0	0	26,195	17,537	0	109,424
1.0 Advanced Technology Dev	4,040	537											4,577
1.1 General R&D	673	337											1,010
1.2 Facilities	3,367												3,367
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	673	0	31,663	20,585	5,874	695	0	0	0	0	14,908	0	74,399
2.1 Flight Elements	510	0	23,987	15,595	4,450	526	0	0	0	0	11,294	0	56,363
2.1.1 Astrotel	67	0	3,370	3,483	1,123	66	0	0	0	0	919	0	9,029
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.4 Mars Spaceport	0	0	0	0	0	0	0	0	0	0	1,251	0	1,251
2.1.5 Taxi	1	0	30	24	8	0	0	0	0	0	0	0	63
2.1.6 Mars Cargo Freighter	73	0	1,815	1,420	458	75	0	0	0	0	3,256	0	7,097
2.1.7 Astrotel Cargo Freighter	0	0	9	7	2	0	0	0	0	0	16	0	34
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	0	0	2,213
2.1.9 Lunar Water Tanker	4	0	191	95	25	3	0	0	0	0	0	0	319
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	283	0	14,135	8,750	2,319	316	0	0	0	0	4,755	0	30,559
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	11	0	562	218	58	8	0	0	0	0	128	0	985
2.1.15 L1 Water Electr & Cryo Storage	0	0	23	16	4	1	0	0	0	0	5	0	49
2.1.16 Phobos LOX Plant	15	0	769	349	93	13	0	0	0	0	467	0	1,706
2.1.17 Mars Surface Water Plant	9	0	427	215	57	8	0	0	0	0	278	0	994
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	42	44	14	2	0	0	0	0	30	0	132
2.2 Flight System Dev Support	51	0	2,399	1,559	445	53	0	0	0	0	1,129	0	5,636
2.3 Other Flight System Costs													0
2.4 Development Reserves	112	0	5,277	3,431	979	116	0	0	0	0	2,485	0	12,400
3.0 Launch Services						350	1,129				2,629		4,108
3.1 Launch Approval						317					149		466
3.2 Launch Processing						34					72		106
3.3 Launch Vehicle							1,129				2,407		3,536
3.4 Other Launch Services Costs													0
4.0 Operations							146			26,195			26,340
4.1 Operations Project Mgmt							15			2,619			2,634
4.2 Integrated Logistics							15			2,619			2,634
4.3 Flight Operations							15			2,619			2,634
4.4 Training Operations							15			2,619			2,634
4.5 Launch Operations							15			2,619			2,634
4.6 In-Space Crew Support							15			2,619			2,634
4.7 Comm/Data Handling Ops							15			2,619			2,634
4.8 Operations Proj Supp Costs							15			2,619			2,634
4.9 Other Operations Costs							15			2,619			2,634
4.10 Operations Reserves							15			2,619			2,634
LIFE CYCLE COST TOTAL	4,713	537	31,663	20,585	5,874	1,045	1,274	0	0	26,195		0	109,424

Table 9-25 Case 10: Stopover, ISRU, Solar, $10k/kg LV; Flight Element Masses

Case10_StopOver_10000_ISRU_Solar_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**564,258**	**1,203,725**
1	Astrotel	**2**	86,249	172,497	78,161
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*8,203*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	0	0	0
4	Mars Spaceport	**1**	0	0	78,161
5	Taxi	**2**	3,000	6,000	0
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	50,005	50,005	32,003
	Mars Cargo Freighter Xe Propellant				*839,240*
7	Astrotel Cargo Freighter	**1**	237	237	156
	Astrotel Cargo Freighter Xe Propellant				*3,775*
8	LEO Shuttle	**1**	15,000	15,000	0
9	Lunar Water Tanker	**1**	7,163	7,163	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	278,757	278,757	50,942
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	5,273	5,273	1,667
15	L1 Water Electrolysis & Cyro Storage	**1**	723	723	100
16	Phobos LOX Plant	**1**	7,817	7,817	5,980
17	Mars Surface Water Plant	**1**	5,261	5,261	3,712
18	Mars Spaceport LOX/LH2 Storage	**1**	2,384	2,384	522
19					
TOTALS	Flight Elements, dry			**564,258**	**252,183**
	Flight Elements, wet			**0**	**951,542**

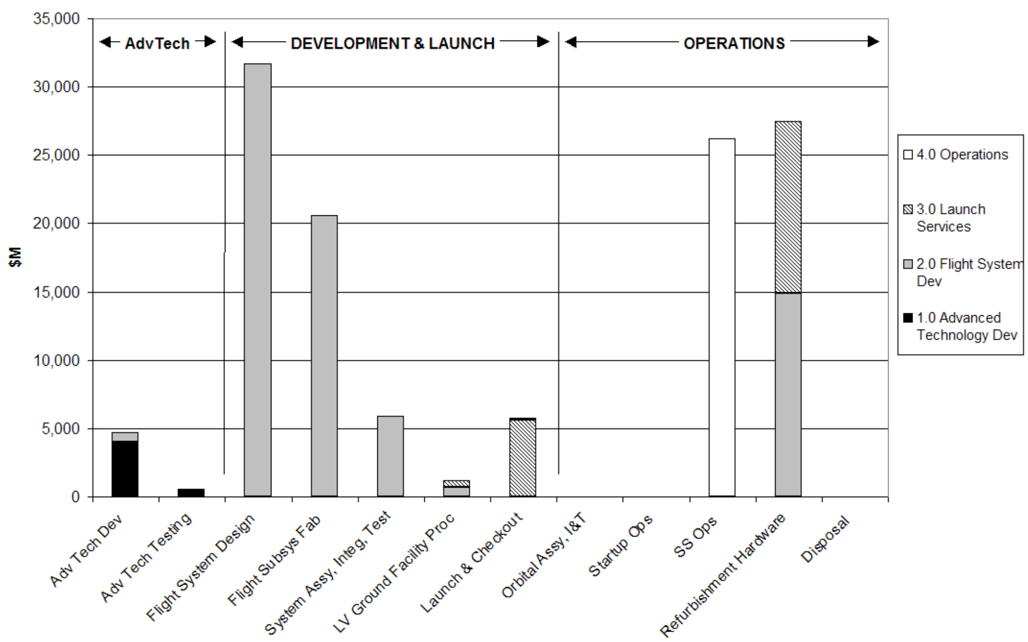

Figure 9-14 Case 10: Stopover, ISRU, Solar, $10k/kg LV; Summary Cost Results

Table 9-26 Case 10: Stopover, ISRU, Solar, $10k/kg LV; Life Cycle Costs by WBS

15 = years of operation

Return

Case10_StopOver_10000_ISRU_Solar_0326

Life Cycle Cost Phase, FY'00 $M **

	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	**LCC TOTAL**
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	4,040	537	0	0	0	0	0	0	0	0	0	0	4,577
2.0 Flight System Development	673	0	31,663	20,585	5,874	695	0	0	0	0	14,908	0	74,399
3.0 Launch Services	0	0	0	0	0	486	5,643	0	0	0	12,547	0	18,676
4.0 Operations	0	0	0	0	0	0	146	0	0	26,195	0	0	26,340
LCC TOTAL	4,713	537	31,663	20,585	5,874	1,181	5,788	0	0	26,195	27,456	0	123,992
1.0 Advanced Technology Dev	4,040	537											4,577
1.1 General R&D	673	337											1,010
1.2 Facilities	3,367												3,367
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	673	0	31,663	20,585	5,874	695	0	0	0	0	14,908	0	74,399
2.1 Flight Elements	510	0	23,987	15,595	4,450	526	0	0	0	0	11,294	0	56,363
2.1.1 Astrotel	67	0	3,370	3,483	1,123	66	0	0	0	0	919	0	9,029
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.4 Mars Spaceport	0	0	0	0	0	0	0	0	0	0	1,251	0	1,251
2.1.5 Taxi	1	0	30	24	8	0	0	0	0	0	0	0	63
2.1.6 Mars Cargo Freighter	73	0	1,815	1,420	458	75	0	0	0	0	3,256	0	7,097
2.1.7 Astrotel Cargo Freighter	0	0	9	7	2	0	0	0	0	0	16	0	34
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	0	0	2,213
2.1.9 Lunar Water Tanker	4	0	191	95	25	3	0	0	0	0	0	0	319
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	283	0	14,135	8,750	2,319	316	0	0	0	0	4,755	0	30,559
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	11	0	562	218	58	8	0	0	0	0	128	0	985
2.1.15 L1 Water Electr & Cryo Storage	0	0	23	16	4	1	0	0	0	0	5	0	49
2.1.16 Phobos LOX Plant	15	0	769	349	93	13	0	0	0	0	467	0	1,706
2.1.17 Mars Surface Water Plant	9	0	427	215	57	8	0	0	0	0	278	0	994
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	42	44	14	2	0	0	0	0	30	0	132
2.2 Flight System Dev Support	51	0	2,399	1,559	445	53	0	0	0	0	1,129	0	5,636
2.3 Other Flight System Costs													0
2.4 Development Reserves	112	0	5,277	3,431	979	116	0	0	0	0	2,485	0	12,400
3.0 Launch Services						486	5,643				12,547		18,676
3.1 Launch Approval						317					149		466
3.2 Launch Processing						169					361		530
3.3 Launch Vehicle							5,643				12,037		17,680
3.4 Other Launch Services Costs													0
4.0 Operations							146			26,195			26,340
4.1 Operations Project Mgmt							15			2,619			2,634
4.2 Integrated Logistics							15			2,619			2,634
4.3 Flight Operations							15			2,619			2,634
4.4 Training Operations							15			2,619			2,634
4.5 Launch Operations							15			2,619			2,634
4.6 In-Space Crew Support							15			2,619			2,634
4.7 Comm/Data Handling Ops							15			2,619			2,634
4.8 Operations Proj Supp Costs							15			2,619			2,634
4.9 Other Operations Costs							15			2,619			2,634
4.10 Operations Reserves							15			2,619			2,634
LIFE CYCLE COST TOTAL	4,713	537	31,663	20,585	5,874	1,181	5,788	0	0	26,195		0	123,992

Table 9-27 Case 11: Stopover, ISRU, Nuclear, $2k/kg LV; Flight Element Masses

Case11_StopOver_2000_ISRU_Nuclear_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**927,781**	**6,474,964**
1	Astrotel	**2**	91,037	182,075	87,739
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*8,601*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	0	0	0
4	Mars Spaceport	**1**	0	0	87,739
5	Taxi	**2**	3,000	6,000	0
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	348,210	348,210	222,854
	Mars Cargo Freighter Xe Propellant				*5,844,076*
7	Astrotel Cargo Freighter	**1**	248	248	163
	Astrotel Cargo Freighter Xe Propellant				*3,959*
8	LEO Shuttle	**1**	15,000	15,000	0
9	Lunar Water Tanker	**1**	7,220	7,220	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	283,191	283,191	55,796
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	8,571	8,571	7,524
15	L1 Water Electrolysis & Cyro Storage	**1**	756	756	105
16	Phobos LOX Plant	**1**	47,216	47,216	42,523
17	Mars Surface Water Plant	**1**	13,658	13,658	12,236
18	Mars Spaceport LOX/LH2 Storage	**1**	2,496	2,496	546
19					
TOTALS	Flight Elements, dry			**927,781**	**518,004**
	Flight Elements, wet			**0**	**5,956,960**

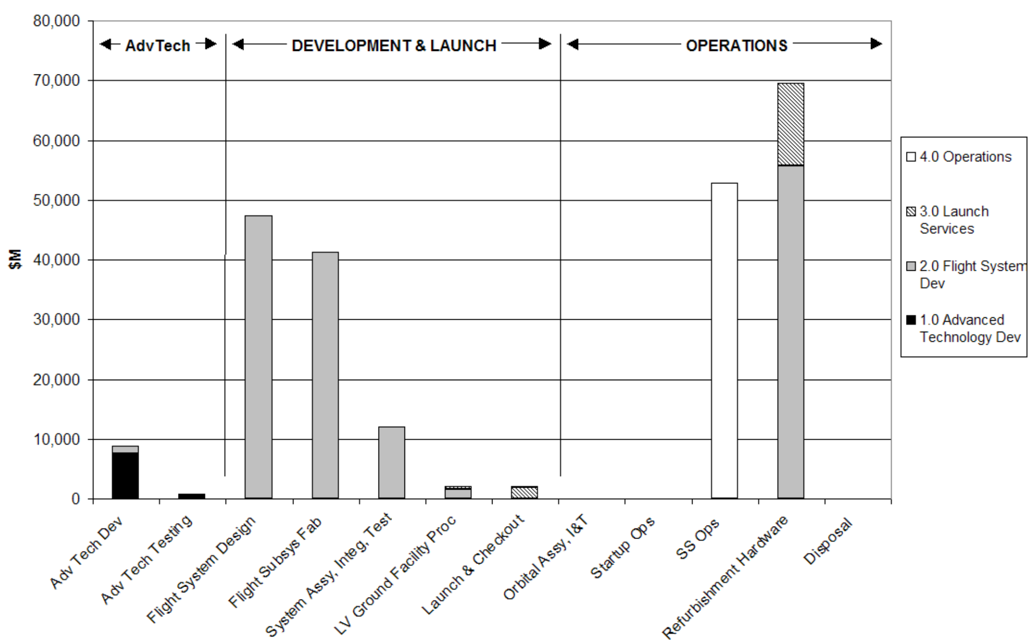

Figure 9-15 Case 11: Stopover, ISRU, Nuclear, $2k/kg LV; Summary Cost Results

Table 9-28 Case 11: Stopover, ISRU, Nuclear, $2k/kg LV; Life Cycle Costs by WBS

Case11_StopOver_2000_ISRU_Nuclear_0326		15	= years of operation								Return		
	Life Cycle Cost Phase, FY'00 $M **												
	Adv Tech Dev		Development				Launch	Operations					
Life Cycle Cost WBS Elements *	Adv Tech Dev	Adv Tech Testing	Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch & Checkout	Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	7,516	826	0	0	0	0	0	0	0	0	0	0	8,343
2.0 Flight System Development	1,253	0	47,340	41,236	12,092	1,569	0	0	0	0	55,731	0	159,222
3.0 Launch Services	0	0	0	0	0	529	1,856	0	0	0	13,896	0	16,280
4.0 Operations	0	0	0	0	0	0	293	0	0	52,795	0	0	53,088
LCC TOTAL	8,769	826	47,340	41,236	12,092	2,099	2,149	0	0	52,795	69,627	0	236,933
1.0 Advanced Technology Dev	7,516	826											8,343
1.1 General R&D	1,253	626											1,879
1.2 Facilities	6,264												6,264
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	1,253	0	47,340	41,236	12,092	1,569	0	0	0	0	55,731	0	159,222
2.1 Flight Elements	949	0	35,864	31,239	9,161	1,189	0	0	0	0	42,220	0	120,622
2.1.1 Astrotel	66	0	3,277	5,468	1,763	103	0	0	0	0	2,793	0	13,471
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.4 Mars Spaceport	0	0	0	0	0	0	0	0	0	0	2,060	0	2,060
2.1.5 Taxi	1	0	30	24	8	0	0	0	0	0	0	0	63
2.1.6 Mars Cargo Freighter	475	0	11,885	9,256	2,985	490	0	0	0	0	21,005	0	46,097
2.1.7 Astrotel Cargo Freighter	0	0	9	7	2	0	0	0	0	0	15	0	34
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	0	0	2,213
2.1.9 Lunar Water Tanker	4	0	192	96	26	3	0	0	0	0	0	0	321
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	273	0	13,635	8,894	2,357	321	0	0	0	0	5,052	0	30,532
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	11	0	562	1,320	350	48	0	0	0	0	2,316	0	4,607
2.1.15 L1 Water Electr & Cryo Storage	0	0	24	16	4	1	0	0	0	0	5	0	51
2.1.16 Phobos LOX Plant	59	0	2,932	3,855	1,022	139	0	0	0	0	6,532	0	14,540
2.1.17 Mars Surface Water Plant	13	0	660	1,284	340	46	0	0	0	0	2,221	0	4,564
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	44	46	15	2	0	0	0	0	31	0	138
2.2 Flight System Dev Support	95	0	3,586	3,124	916	119	0	0	0	0	4,222	0	12,062
2.3 Other Flight System Costs													0
2.4 Development Reserves	209	0	7,890	6,873	2,015	262	0	0	0	0	9,288	0	26,537
3.0 Launch Services						529	1,856				13,896		16,280
3.1 Launch Approval						473					557		1,031
3.2 Launch Processing						56					388		444
3.3 Launch Vehicle							1,856				12,950		14,805
3.4 Other Launch Services Costs													0
4.0 Operations							293			52,795			53,088
4.1 Operations Project Mgmt							29			5,279			5,309
4.2 Integrated Logistics							29			5,279			5,309
4.3 Flight Operations							29			5,279			5,309
4.4 Training Operations							29			5,279			5,309
4.5 Launch Operations							29			5,279			5,309
4.6 In-Space Crew Support							29			5,279			5,309
4.7 Comm/Data Handling Ops							29			5,279			5,309
4.8 Operations Proj Supp Costs							29			5,279			5,309
4.9 Other Operations Costs							29			5,279			5,309
4.10 Operations Reserves							29			5,279			5,309
LIFE CYCLE COST TOTAL	8,769	826	47,340	41,236	12,092	2,099	2,149	0	0	52,795		0	236,933

Table 9-29 Case 12: Stopover, ISRU, Nuclear, $10k/kg LV; Flight Element Masses

Case12_StopOver_10000_ISRU_Nuclear_0326

Component Number	System	Number of Units	Unit Mass, kg	Total Mass, kg	15yr Refurb Mass, kg
2.1 Flight Systems				**927,781**	**6,474,964**
1	Astrotel	**2**	91,037	182,075	87,739
	Astrotel Crew Consumables				*100,324*
	Astrotel Xe Propellant				*8,601*
2	Escape Pod	**1**	10,000	10,000	0
3	Earth Spaceport	**1**	0	0	0
4	Mars Spaceport	**1**	0	0	87,739
5	Taxi	**2**	3,000	6,000	0
	Taxi Augmentation Tanks	***1***			*0*
6	Mars Cargo Freighter	**1**	348,210	348,210	222,854
	Mars Cargo Freighter Xe Propellant				*5,844,076*
7	Astrotel Cargo Freighter	**1**	248	248	163
	Astrotel Cargo Freighter Xe Propellant				*3,959*
8	LEO Shuttle	**1**	15,000	15,000	0
9	Lunar Water Tanker	**1**	7,220	7,220	0
10	Mars Shuttle	**1**	3,141	3,141	779
11	Phobos LOX Tanker	**0**	0	0	0
12	Mars Base	**1**	283,191	283,191	55,796
13	Lunar Base	**0**	0	0	0
14	Lunar Water Mine	**1**	8,571	8,571	7,524
15	L1 Water Electrolysis & Cyro Storage	**1**	756	756	105
16	Phobos LOX Plant	**1**	47,216	47,216	42,523
17	Mars Surface Water Plant	**1**	13,658	13,658	12,236
18	Mars Spaceport LOX/LH2 Storage	**1**	2,496	2,496	546
19					
TOTALS	Flight Elements, dry			**927,781**	**518,004**
	Flight Elements, wet			**0**	**5,956,960**

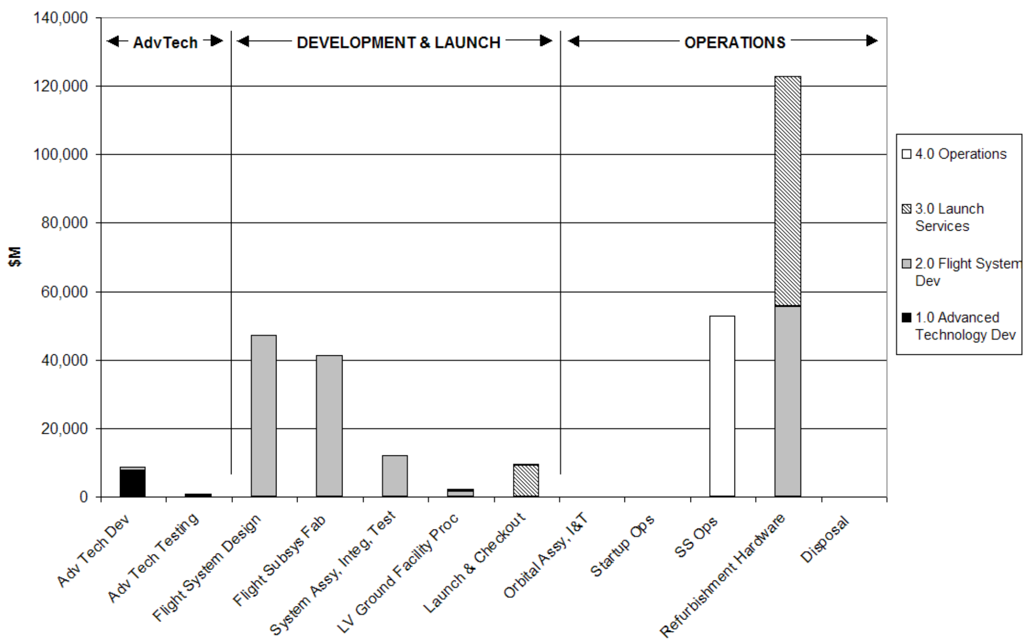

Figure 9-16 Case 12: Stopover, ISRU, Nuclear, $10k/kg LV; Summary Cost Results

Table 9-30 Case 12: Stopover, ISRU, Nuclear, $10k/kg LV; Life Cycle Costs by WBS

Case12_StopOver_10000_ISRU_Nuclear_0326 15 = years of operation Return

Life Cycle Cost WBS Elements *	Life Cycle Cost Phase, FY'00 $M ** Adv Tech Dev: Adv Tech Dev	Adv Tech Testing	Development: Flight System Design	Flight Subsys Fab	System Assy, Integ, Test	LV Ground Facility Proc	Launch: Launch & Checkout	Operations: Orbital Assy, I&T	Startup Ops	SS Ops	Refurbishment Hardware	Disposal	LCC TOTAL
LIFE CYCLE COST SUMMARY													
1.0 Advanced Technology Dev	7,516	826	0	0	0	0	0	0	0	0	0	0	8,343
2.0 Flight System Development	1,253	0	47,340	41,236	12,092	1,569	0	0	0	0	55,731	0	159,222
3.0 Launch Services	0	0	0	0	0	752	9,278	0	0	0	67,249	0	77,279
4.0 Operations	0	0	0	0	0	0	293	0	0	52,795	0	0	53,088
LCC TOTAL	8,769	826	47,340	41,236	12,092	2,321	9,571	0	0	52,795	122,980	0	297,932
1.0 Advanced Technology Dev	7,516	826											8,343
1.1 General R&D	1,253	626											1,879
1.2 Facilities	6,264												6,264
1.3 Flight Demos/Major Tests		100											100
1.4 Sys-Unique Test Facil/HW		100											100
1.5 Other ATD Costs													0
2.0 Flight System Development	1,253	0	47,340	41,236	12,092	1,569	0	0	0	0	55,731	0	159,222
2.1 Flight Elements	949	0	35,864	31,239	9,161	1,189	0	0	0	0	42,220	0	120,622
2.1.1 Astrotel	66	0	3,277	5,468	1,763	103	0	0	0	0	2,793	0	13,471
2.1.2 Escape Pod	6	0	613	135	43	4	0	0	0	0	0	0	802
2.1.3 Earth Spaceport	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.4 Mars Spaceport	0	0	0	0	0	0	0	0	0	0	2,060	0	2,060
2.1.5 Taxi	1	0	30	24	8	0	0	0	0	0	0	0	63
2.1.6 Mars Cargo Freighter	475	0	11,885	9,256	2,985	490	0	0	0	0	21,005	0	46,097
2.1.7 Astrotel Cargo Freighter	0	0	9	7	2	0	0	0	0	0	15	0	34
2.1.8 LEO Shuttle	32	0	1,617	414	134	16	0	0	0	0	0	0	2,213
2.1.9 Lunar Water Tanker	4	0	192	96	26	3	0	0	0	0	0	0	321
2.1.10 Mars Shuttle	8	0	382	424	112	15	0	0	0	0	189	0	1,131
2.1.11 Phobos LOX Tanker	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.12 Mars Base	273	0	13,635	8,894	2,357	321	0	0	0	0	5,052	0	30,532
2.1.13 Lunar Base	0	0	0	0	0	0	0	0	0	0	0	0	0
2.1.14 Lunar Water Mine	11	0	562	1,320	350	48	0	0	0	0	2,316	0	4,607
2.1.15 L1 Water Electr & Cryo Storage	0	0	24	16	4	1	0	0	0	0	5	0	51
2.1.16 Phobos LOX Plant	59	0	2,932	3,855	1,022	139	0	0	0	0	6,532	0	14,540
2.1.17 Mars Surface Water Plant	13	0	660	1,284	340	46	0	0	0	0	2,221	0	4,564
2.1.18 Mars Spaceport LOX/LH2 Storage	1	0	44	46	15	2	0	0	0	0	31	0	138
2.2 Flight System Dev Support	95	0	3,586	3,124	916	119	0	0	0	0	4,222	0	12,062
2.3 Other Flight System Costs													0
2.4 Development Reserves	209	0	7,890	6,873	2,015	262	0	0	0	0	9,288	0	26,537
3.0 Launch Services						752	9,278				67,249		77,279
3.1 Launch Approval						473					557		1,031
3.2 Launch Processing						278					1,942		2,221
3.3 Launch Vehicle							9,278				64,750		74,027
3.4 Other Launch Services Costs													0
4.0 Operations							293			52,795			53,088
4.1 Operations Project Mgmt							29			5,279			5,309
4.2 Integrated Logistics							29			5,279			5,309
4.3 Flight Operations							29			5,279			5,309
4.4 Training Operations							29			5,279			5,309
4.5 Launch Operations							29			5,279			5,309
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4.7 Comm/Data Handling Ops							29			5,279			5,309
4.8 Operations Proj Supp Costs							29			5,279			5,309
4.9 Other Operations Costs							29			5,279			5,309
4.10 Operations Reserves							29			5,279			5,309
LIFE CYCLE COST TOTAL	8,769	826	47,340	41,236	12,092	2,321	9,571	0	0	52,795		0	297,932

9.3 Summary of MAMA Trade Study Results

This section summarizes the results of the various trade studies that were performed. Summary results are shown in Table 9-31 and Figure 9-17 and Figure 9-18. ***Note***, the detail numbers in Figure 9-17 are different than in the "Life Cycle Costs by WBS" charts. In the WBS charts the refurbishment hardware in accounted for in Development costs and their launch costs are in Launch costs. Because we want to correctly differentiate between non-recurring and sustaining, recurring costs we included the refurbishment hardware costs and their launch in Operations costs. The launch costs in Figure 9-17 are only "Initial Launch" costs required to put the initial flight systems in space.

Table 9-31 MAMA Trade Studies and Summary Results

Case	Architecture	Launch Costs	ISRU	Power		RESULTS LCC	Total Mass, kg	15yr Refurb Mass, kg
1	Aldrin	2,000	yes	Solar		117,549	653,037	451,220
2	Aldrin	10,000	yes	Solar		126,648		
3	Aldrin	2,000	yes	Nuclear		204,394	998,339	5,667,161
4	Aldrin	10,000	yes	Nuclear		259,318		
5	Aldrin	2,000	no	Solar		119,389	668,800	906,060
6	Aldrin	10,000	no	Solar		132,366		
7	Aldrin	2,000	no	Nuclear		191,962	991,638	6,087,810
8	Aldrin	10,000	no	Nuclear		250,297		
9	Stopover	2,000	yes	Solar		109,424	564,258	1,203,725
10	Stopover	10,000	yes	Solar		123,992		
11	Stopover	2,000	yes	Nuclear		236,933	927,781	6,474,964
12	Stopover	10,000	yes	Nuclear		297,932		

Several findings can be made from the trade study results:

1) Launch Costs are only a minor portion of the life-cycle costs (LCC) as long as costs are \$2k/kg. The largest LCC element is Development followed by costs for 15 years of Operations;

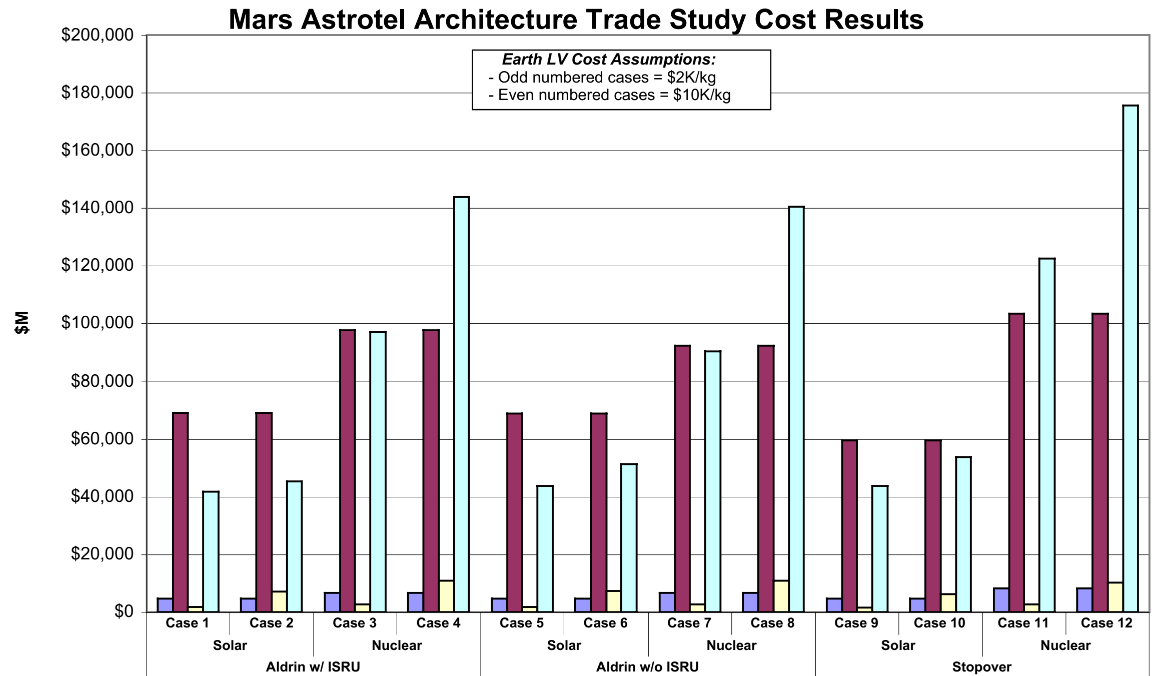
2) Operations costs range from \$41.7-43.9B for Solar plus \$2k/kg options and \$45.5-53.7B for Solar and \$10k/kg options. Operations costs for Nuclear options equal (at \$2k/kg) or exceed (at \$10k/kg) Development costs (however, nuclear systems were replaced at the same rate as solar systems which may skew the nuclear recurring costs too high);

3) ISRU benefits are tightly tied to launch costs. Operations costs for the Aldrin cycler option with ISRU about \$2B less than the no ISRU option when launch costs are \$2,000/kg, however this difference become \$6B assuming \$10,000/kg. The Aldrin cycler option without ISRU but with Solar would need launch costs to be less than \$865/kg to be competitive with ISRU. This is a much lower launch cost than the \$2,000/kg low-end of the range used for these trade studies. This launch cost analysis did not incorporate cost impacts for developing any new launch vehicles and treated launch costs as a service entirely captured by the \$/kg value. Including these costs will increase \$/kg values;

4) LCC for the Solar options is significantly lower than for the Nuclear options. This conclusion is strongly driven by the power levels used for a given architecture design; Since the Astrotels operated at only 160 kW_e and most of the surface systems were on the order of a MW_e or less, the nuclear reactor design was penalized with a large minimum mass; The power level at which the nuclear option would be less massive than the solar option was not explored in this study;

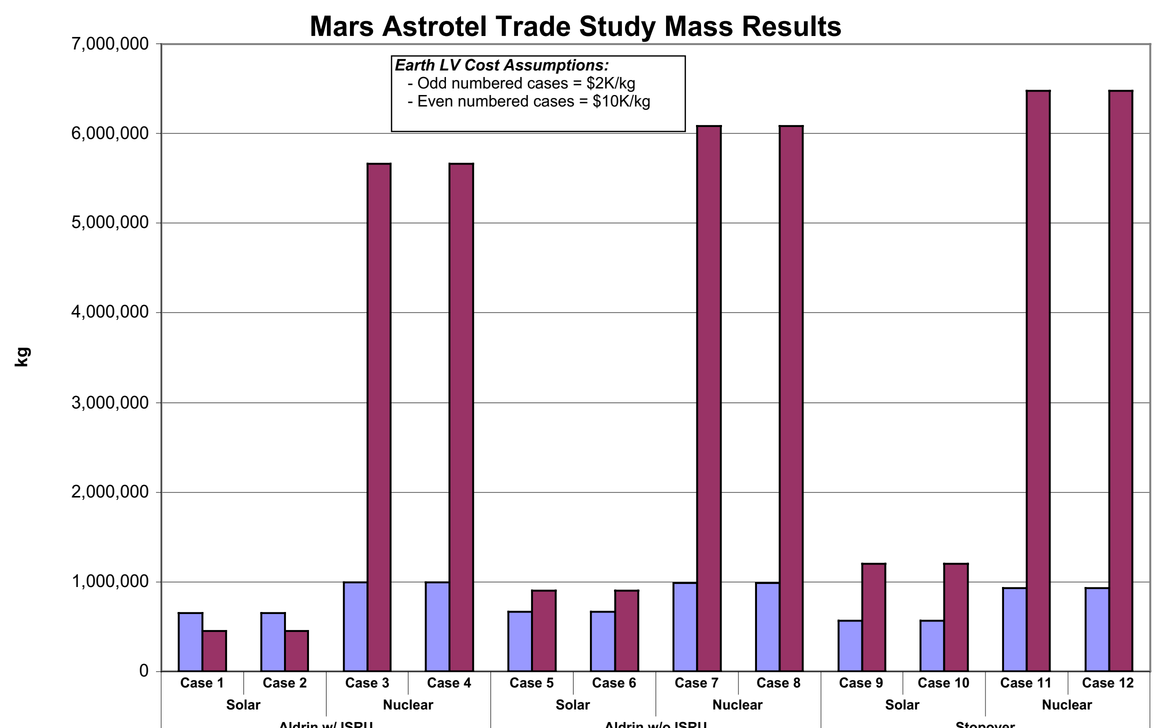
5) The Stopover architecture LCC (Case 1) is \$8B less than the Aldrin Low-thrust architecture LCC (Case 9), however the Operations costs are about \$2B more. When launch costs are \$10,000/kg, the Stopover Operations costs are about \$12B more expensive than the Aldrin Low-thrust architecture.

6) Launch cost assumptions play a major role in determining which subsystem and architecture options are better.



	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6	Case 7	Case 8	Case 9	Case 10	Case 11	Case 12
Adv Tech	$4,626	$4,626	$6,765	$6,765	$4,665	$4,665	$6,612	$6,612	$4,577	$4,577	$8,343	$8,343
Dev	$69,208	$69,208	$97,746	$97,746	$68,907	$68,907	$92,337	$92,337	$59,491	$59,491	$103,491	$103,491
Initial Launch	$1,892	$7,273	$2,790	$11,017	$1,920	$7,431	$2,739	$10,910	$1,625	$6,274	$2,678	$10,323
Ops	$41,662	$45,541	$97,093	$143,790	$43,896	$51,362	$90,274	$140,438	$43,732	$53,651	$122,422	$175,775

Figure 9-17 Trade Study Cost Results



	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6	Case 7	Case 8	Case 9	Case 10	Case 11	Case 12
Initial Flight HW	653,037	653,037	998,339	998,339	668,800	668,800	991,638	991,638	564,258	564,258	927,781	927,781
Refurb HW	451,220	451,220	5,667,161	5,667,161	906,060	906,060	6,087,810	6,087,810	1,203,725	1,203,725	6,474,964	6,474,964

Figure 9-18 Trade Study Mass Results

9.4 Potential Applications/Extensions of the Modeling Concept

The MAMA modeling approach could be applied to support systems analyses of complex concepts for a variety of different applications. The key benefits are improved understanding of life cycle impacts from alternative system design approaches and different technologies. Sensitivity studies exploring benefits from technology improvements can be used to help direct advanced technology development efforts towards the highest leverage investments.

10 Astrotel Phase II Publications, Presentations, and Press Releases

10.1 Technical Papers or Articles

- "Frequent, Fast Trips to and from Mars via Astrotels," Paper and Presentation to the Space Studies Institute/Princeton Conference on Space Manufacturing: The High Frontier Conference XV, May 8, 2001
- CSM prepared and presented a paper titled "Conceptual Design of ISRU Propellant Storage Depots" at the 3rd Annual Space Resources Roundtable at CSM on October 25, 2001
- "Hyperbolic Rendezvous for Earth-Mars Cycler Missions," Paper and Presentation at AAS/AIAA Space Flight Mechanics Meeting, San Antonio, Texas, 27-30 January 2002.
- "The Astrotel Scenario for Mars Exploration," in Space Resources Roundtable Newsletter, Vol. 1, No. 1, April 6, 2002.
- "Earth-Mars Transportation Using Stop-over Cyclers," paper and presentation to AIAA Astrodynamics Conference, Monterrey, CA, August 2002.
- "A Mars Cycler Architecture Utilizing Low-Thrust Propulsion," paper and presentation to AIAA Astrodynamics Conference, Monterrey, CA, August 2002.
- "An Interplanetary Rapid Transit System Between Earth and Mars," paper and presentation to the Space Technology & Applications International Forum, Albuquerque, NM, February 2-5, 2003.
- "Interplanetary Rapid Transit to Mars," paper and presentation to International Conference on Environmental Systems, Vancouver, Canada, July 7-10, 2003 (planned).

10.2 Media Interviews and Articles

- Space.com Interview, May 8, 2001 and Article in October 2001
- Interview for Swedish Educational TV, "University TV," August 16, 2001
- "Pendelsystem zwischen Erde und Mars," in http://www.scienceticker.de/news/EpkFlpZlAl.shtml.
- "Main Belt transportation system?" in June 2002 Asteroid/Comet News in http://www.hohmanntransfer.com/news/0206.htm.
- "Earth-Mars Interplanetary Rapid Transit System," in Cosmiverse, June 12, 2002 in http://www3.cosmiverse.com/news/space/space06120205.html.
- "2035-ben u ˝rhotel,u ˝rtaxi és u ˝rkiköto ˝a Marson," in Tudomány, June 20, 2002, http://www.origo.hu/tudomany/vilagur/200206202035ben.html.
- "Astrotels might aid explorers' Mars trips," Floridatoday.com, July 2, 2002 in http://www.floridatoday.com/news/space/stories/2002a/070202astrotels.htm.
- "Cyclical visits to Mars via astronaut hotels," in Scientecmatrix.com, http://www.scientecmatrix.com/seghers/tecma/scientecmatrix.nsf/_/E70654D46DCC705DC1256BD500372664.

10.3 Briefings and Presentations

- Briefing to the NASA Institute for Advanced Concepts (NIAC) 3rd Annual Meeting: Visions of the Future in Aeronautics and Space, June 5, 2001
- Astrotel Team Meeting and Presentations at the Colorado School of Mines, July 13, 2001
- Presentation prepared for the NASA Associate Administrator for Human Exploration and the Development of Space, September 11, 2001 (Meeting Cancelled)
- Briefing to the Exploration Office NASA Johnson Space Center, November 27, 2001
- Briefing to the NASA Institute for Advanced Concepts (NIAC) 4th Annual Meeting, Houston, TX, June 12, 2002
- Briefing to the Exploration Office NASA Johnson Space Center, June 10, 2002
- "Cyclic Visits To Mars Via Astronaut Hotels or The Interplanetary Rapid Transit (IRT) System," presentation to Transformational Space Concepts and Technologies (TSCT) for Space Missions Human Exploration and Development of Space Enterprise (HEDS) Technology Interchange Meeting, Pasadena, CA, January 14-16, 2003.

11 Summary

This report describes the work accomplished and results obtained during the Phase II of the development of an innovative and new concept for a Cyclical Visits to Mars via Astronaut Hotels in support of the NASA Institute for Advanced Concepts. During the Phase II, we have made significant progress as evidenced by the following summary of key accomplishments:

- Written and presented 8 technical papers at a variety of conferences,
- interviewed for Swedish TV and Space.com plus several media contacts resulting in web or print media coverage,
- selected a lunar orbit radius (LOR) location for the Earth Spaceport,
- completed a comprehensive analysis, modeling, and design of ISRU systems,
- developed concepts for providing reasonable launch periods for Taxi vehicles undergoing hyperbolic rendezvous with Astrotels,
- developed, analyzed and optimized the Mars Shuttle entry and ascent trajectories,
- completed detailed design studies of Mars Shuttle vehicle,
- completed detailed design studies of three-stage Taxi vehicle,
- briefed the NASA JSC Exploration Office on the concept and the progress of the study,
- developed several architecture and subsystem options and compared their masses and costs with MAMA,
- completed this Final Report.

During Phase II, we estimate that we have met nearly 100% of our objectives though there is still much to be done.

Appendix: 33rd International Conference on Environmental Systems Paper

Paper Number: 03 ICES-295

Interplanetary Rapid Transit to Mars

Kerry Nock, Angus McRonald, Paul Penzo, and Chris Wyszkowski
Global Aerospace Corporation

Michael Duke, Robert King, Lee Johnson
Colorado School of Mines

Mark Jacobs, Jerry Rauwolf
Science Applications International Corporation

ABSTRACT

A revolutionary interplanetary rapid transit concept for transporting scientists and explorers between Earth and Mars is presented by Global Aerospace Corporation under funding from the NASA Institute for Advanced Concepts (NIAC) with support from the Colorado School of Mines, and Science Applications International Corporation. We describe an architecture that uses highly autonomous spaceships, dubbed Astrotels; small Taxis for trips between Astrotels and planetary Spaceports; Shuttles that transport crews to and from orbital space stations and planetary surfaces; and low-thrust cargo freighters. In addition we discuss the production of rocket fuels using extraterrestrial materials; aerocapture to slow Taxis at the planets; and finally describe a number of trade studies and their life-cycle cost results.

INTRODUCTION

Someday scientists and explorers will regularly travel to Mars for research and exploration as they now travel to Antarctica. Knowing this eventuality enables us to plan for the future. As with the South Pole Base at Antarctica, an efficient transportation system will be needed to rotate crews back and forth between Earth and Mars and to resupply equipment and fuels.

Global Aerospace Corporation and its partners have developed an innovative architecture that uses highly autonomous, solar-powered, xenon ion-propelled spaceships, dubbed Astrotels; small Taxis spaceships for trips between Astrotels and planetary Spaceports; Shuttles that transport crews to and from orbital space stations and planetary surfaces; and low-thrust cargo freighters that deliver hardware, fuels and consumables to Astrotels and Spaceports. Astrotels can orbit the Sun in cyclic orbits between Earth and Mars and Taxis fly hyperbolic planetary trajectories between Astrotel and Spaceport rendezvous. Together these vehicles transport replacement crews of 10 people on frequent, short trips between Earth and Mars. Two crews work on Mars with alternating periods of duty, each spending about 4 years there with crew transfers occurring about every two years. The production of rocket fuels has been studied using materials mined from the surfaces of the Moon, Mars and the Martian satellites. The use of the atmospheres of the planets themselves to slow Taxis, called aerocapture, has been developed and analyzed. A tool has been constructed that can estimate the life-cycle cost of a transportation architecture and its various options.

This concept provides a framework and context for future technology advance and robotic mission exploration. The human exploration program could benefit from a focus on permanent Mars habitation, instead of brief and expensive expeditions; lunar and Phobos exploration as steps to Mars; and evolutionary vehicle and system development toward a Mars transportation infrastructure. The inevitability of human Mars exploration will be much closer once we begin taking these steps.

MARS BASE TRANSIT STOP

The level of capability envisioned at the Mars Base supports significant surface activities in the areas of science exploration, resource surveys, life-cycle maintenance, propellant production, and materials processing and fabrication. These activities will take place at one or two fixed-site facilities on Mars and on distant traverses from the base. Such operations will require a high degree of mobility, appropriate levels of automation with efficient man-machine interfaces, and they require crews that combine the need for individual specialization with job sharing abilities. A crew complement of 20 on the Martian surface will carry out these activities; the resident population at any time could fluctuate substantially from the average depending on the phase of the crew rotation cycle dictated by the interplanetary transportation orbit options. The Mars surface is assumed to be continually inhabited thereby requiring staggered crew rotations and, thus, overlap between "experienced" and "fresh" personnel. Figure 1 illustrates one base concept and illustrates an example equipment list (where mt is metric tonne or 1000 kg).

The Mars Base is nearly self-sufficient and it maximizes its use of in situ resources with minimal replenishment from Earth. Robotics and automation activities are focused on in situ resource, refurbishment, repair and upgrade (RRU), power generation, and life support monitoring functions. The environmental control and life support systems are regenerative to a large degree but not entirely closed. Life support gases and water will be extracted from the soil and atmosphere as needed. Agriculture, in greenhouses, and aquaculture will supply plants and perhaps animals for food. Propellants for mobility systems on the surface and in the atmosphere and for rocket transportation between the Mars Base and the Mars Spaceport are created in situ. The entire Mars Base requires delivery of about 280 metric tonnes (mt) of hardware to the surface in the build up phase and about 50 mt of RRU hardware every 15 years. Table 1 summarizes the mass of the various base elements. To support the Mars Base a means of transporting crews and RRU equipment between the planets is needed. It is the crew and logistical support to this base that is the driver for this Mars transportation system architecture.

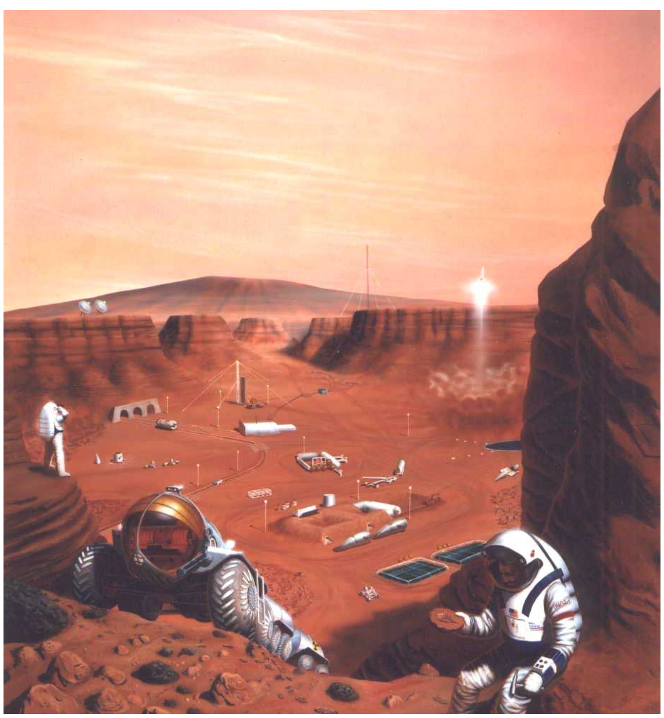


Figure 1 Artist Concept of a Mars Base

Table 1 Mars Base Systems Mass Summary

Mars Base Systems	# of Units	Unit Mass (mt)	Total Mass (mt)
Life Critical Systems			
Habitat	4	38.5	154.0
Washdown facility	2	0.9	1.8
Subtotal			**155.8**
Mission Support Systems			
120 kW Solar Array - @100W/kg	2	1.2	2.4
Power Management, & Distribution	2	0.3	0.6
Energy Storage (NRFC packages)	2	1.0	2.1
Suitup/Maintenance Facility	2	1.8	3.6
Pressurized Transporter	3	9.1	27.3
Open Rovers	3	1.0	3.0
Inflatable Shelter w/Airlock	10	0.5	5.0
Communication Satellites	3	0.8	2.4
Crane	2	5.0	10.0
Trailer	2	2.0	4.0
Subtotal			**60.4**
Science and Exploration Systems			
Base Laboratory	2	13.6	27.2
Mobile Laboratory	3	9.1	27.3
200 m Drill	1	2.3	2.3
10 m Drill	3	0.1	0.3
UAV	3	0.3	0.9
Robotic Rovers	10	0.2	2.0
Weather Station	5	0.2	1.0
Survey Orbiters	2	0.8	1.6
Subtotal			**62.6**
Total			**278.8**

SPACE TRACKS

Cycling orbits can be designed to enable sustained human interplanetary transportation through regular encounters with Earth and the target planet or between Earth and the Moon. Several interplanetary cycler orbit concepts have been developed over the last two decades to support studies of sustained Mars operations. Cyclers (Aldrin, 1985; Hoffman, 1986; and Nock, 1987) and the classic Stopover trajectories (Penzo, 2002b) are two types of orbits that have received high interest for use in Mars transportation scenarios. The Aldrin cycler orbit type can be seen as viewed from the North Ecliptic Pole in Figure 2.

The Aldrin Cycler orbits have a period that is approximately equal to the Earth-Mars synodic period (26 months) and, when the line of apsides is rotated by gravity assist methods (average of about 51.4° each orbit), will enable Earth-to-Mars and Mars-to-Earth transfers every 26 months. Aldrin Cycler orbits come in two types, an Up Cycler and a Down Cycler orbit. The Up Cycler has the fast transfer occurring on the Earth to Mars leg while the Down Cycler is just the reverse. Fig.2 illustrates both orbit transfer geometries. When two Astrotels are used, an Aldrin Cycler provides relatively short transit times (~5 months) and regular transit opportunities. However, the planetary encounters occur at high relative velocities and typically, impose harsher requirements on the Taxi craft than other cyclers. Also, the Aldrin Cycler requires a modest mid-course correction on 3 out of 7 orbits to maintain the proper orbit orientation. These delta-Vs will be carried out using low-thrust, solar powered ion propulsion systems (IPS).

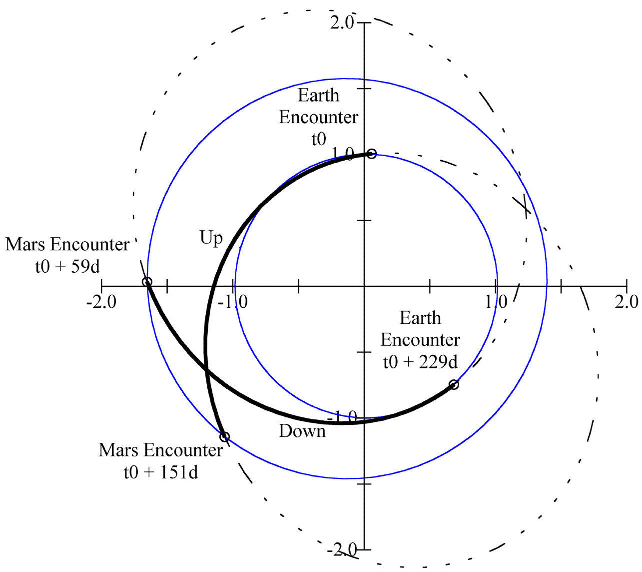


Figure 2 Aldrin Cycler Orbits

Stopover Cyclers are direct transfers from Earth to Mars with high-thrust propulsive maneuvers at both ends of the trajectory and a stop at each planet. Stopover Cyclers require two Astrotels operating. Flight time varies between 4-7 months depending on opportunity and propellant loading. Stay time at Mars is identical to the Semi-cyclers or about 1.5 years. Advantages of Stopover Cyclers are low departure and arrival velocities for a given flight time, flexible launch and arrival dates, elimination of the hyperbolic rendezvous, close vicinity of the station to the planet for replenishment and refurbishment, and alternate mission uses for the stations while in orbit about each planet, waiting for the next opportunity to return.

At this time, the Aldrin Cycler orbits have been selected as the reference because of several key advantages. One advantage is that Astrotels do not require high-thrust chemical propulsion systems, whereas, in the Stopover Cycler concept, high-thrust propulsion must be used to keep the flight time short. Another important advantage of Aldrin Cyclers is that the Astrotels never stop. The implication of this advantage, combined with the use of low-thrust systems, is that one can incrementally increase the Astrotel capability over time with very little propulsion cost. Example increased capabilities include more radiation shielding, incorporation of artificial gravity if desired, redundancy in the form of additional Taxi and/or escape vehicles, and a growing cache of repair hardware, propellants and consumables at the Astrotel. Finally, by using the low-thrust Aldrin Cyclers, only two Astrotel vehicles need to be constructed and maintained.

GETTING ON AND OFF THE TRAIN

Cycler orbits with Earth and Mars hyperbolic flybys necessitate transfers between a planetary Spaceport and an Astrotel via a Taxi vehicle. The Astrotel flyby is completely constrained in periapsis date, distance, and inclination, since it must continue to travel on its desired cycler path between the planets. One of the major concerns in the use of cyclers for human transportation has been the hyperbolic rendezvous where the Taxi departs the Earth with a near instantaneous launch period without any margin for error or hardware delay. The primary restriction here is that the rendezvous must take place within about 7 days from the time of departure from the Spaceport because Taxi vehicles have limited consumables and life support and are lightly shielded against radiation. Figure 3 (adapted from Penzo and Nock 2002a) shows the Spaceport orbit, the Astrotel flyby and three Taxi hyperbolic rendezvous options. We have selected the 3-burn option for the reference because it requires a low total delta-V and a short flight time. The 4-burn option has a lower delta-V however a prohibitively long transit time for a Taxi. The 3-burn option consists of a Taxi departure maneuver, ΔV_1, to lower periapsis altitude for the injection delta-V, ΔV_2, followed several days later by the rendezvous maneuver, ΔV_3.

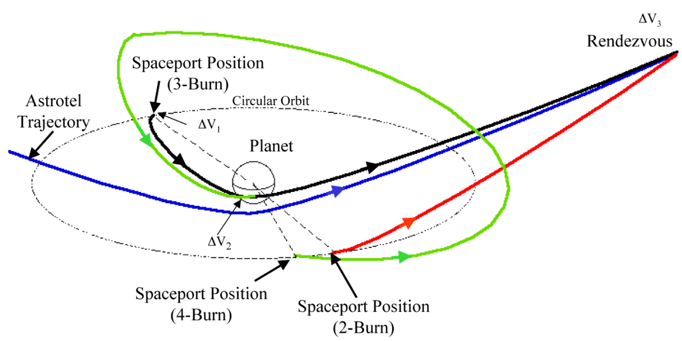


Figure 3 Hyperbolic Rendezvous Options

Having to solve the hyperbolic rendezvous problem provides insight into the desired location of the Earth Spaceport. Plane changes are best made far from the planet where the orbital velocities are low. Velocity changes are best made close to the planet, i.e. within the gravity well. In LEO, the orbital velocity is almost 8 km/s, whereas at lunar distance, the velocity is about 1 km/s, which is a better place to make a required plane change.

TRANSIT STATION LOCATIONS

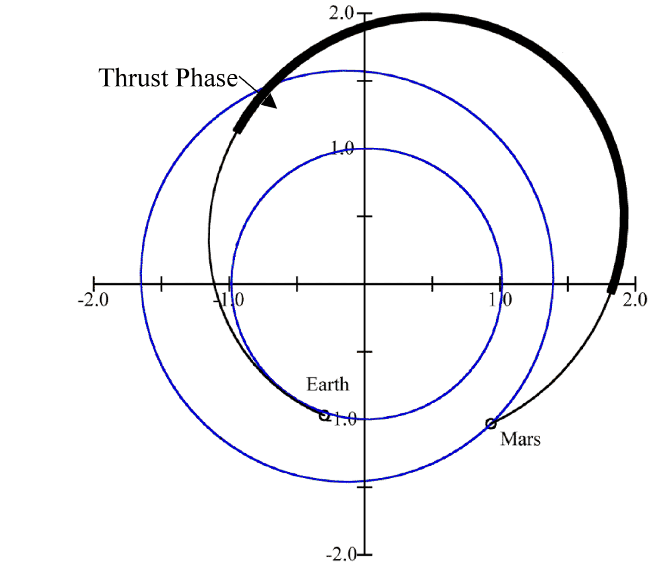
The Earth-Moon L–1 libration point, which has the Moon's period, but is closer to the Earth, has been considered in the past for the Earth Spaceport location. The advantage of L-1 is a lower orbital velocity than the Moon, and therefore a lower required plane change delta-V. The problem with L-1 for Earth-to-Mars transportation, however, is that it is tied to the Moon's geometry; having the same period of about 28 days. For this reason, it is almost always in the wrong position in its orbit for the first maneuver for a hyperbolic rendezvous sequence. For the L-1 location, the Earth Spaceport could be almost a month off from its required position. This position mismatch can be mitigated by either high delta-V, which negatively impacts mission performance, or very long phasing orbits, which require excessive crew time in the Taxi.

An ***active*** Earth Spaceport is needed in a relatively high orbit, which can move itself into the optimum hyperbolic departure position at the correct departure time. This positioning is accomplished by changing the period of the Earth Spaceport to cause it to drift to the required longitude over a period of months, and then reverting back to its original period. This phasing velocity is proportional to drift time, about 1 m/s for 1 deg over a period of a month, and can easily be carried out with low-thrust IPS. We have chosen Lunar orbit radius (LOR) as the Earth Spaceport location for the current studies since we are using lunar resources, though other high Earth orbits are also candidate Spaceport locations.

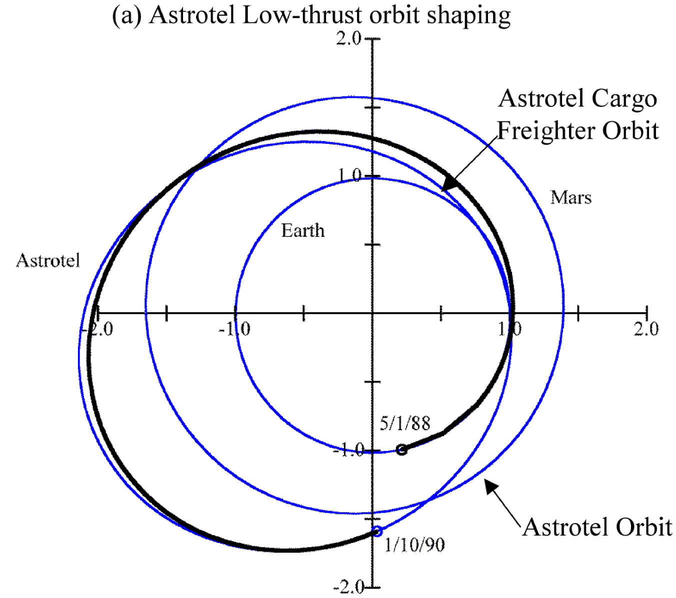
The Mars Spaceport is located near Phobos (to be near its resources). Because the Phobos orbit period is only about 7.5 hr, it's period is short enough to accommodate significant launch time variations for Taxis departing Mars. In other words, launch periods of hours to days are feasible leaving Mars.

ION DRIVE ENGINES

Low-thrust, ion drive propulsion, utilizing mass-efficient solar powered ion engines, is applied to the Astrotel architecture in four areas: (1) midcourse shaping maneuvers of the Astrotel orbits; (2) spaceport orbit phasing maneuvers, (3) round-trip cargo freighters to resupply the Astrotel vehicles in transit; and (4) round-trip cargo freighters to resupply the infrastructure at Mars. Figure 4 illustrates the use of low-thrust propulsion on such two orbits (with representative dates). Figure 4a shows the region of the Astrotel orbit where low-thrust is applied in order to carry out the periodic shaping maneuvers. Figure 4b shows the trajectory of the Astrotel Cargo Freighter from its Earth spiral departure until it rendezvous with the Astrotel. The use of low-thrust delta-V is ideal for these orbits where high-thrust is not needed, flight time is not critical, and considerable savings in propellant mass can be achieved.



(a) Astrotel Low-thrust orbit shaping



(b) Astrotel Cargo Freighter low-thrust maneuvering and rendezvous

Figure 4 Low-Thrust Maneuvering on Orbits

HOTEL, TRANSIT STATION, TAXI, SHUTTLE AND FREIGHTER DESIGNS

This section describes various elements of the interplanetary transit system beginning with Astrotels.

ASTRONAUT HO**TEL** OR ASTROTEL

Astrotels are highly autonomous and transport only human and other high value cargo, use highly efficient solar electric propulsion for periodic orbit shaping maneuvers, and do not require artificial gravity. These features keep the size of these vehicles down to about 70 mt including IPS, radiation shielding, habitation, storage, power, and emergency escape pod. Reducing its mass significantly reduces the total propulsive energy budget required for course corrections to the 2767-kg propellant required for all major corrections over 15 years. The 70-mt mass includes a habitability module for a crew of ten. The size and volume of this system would provide a crew volume of about 6-times that available to today's Space Shuttle crew. Figure 5 is a schematic of one concept for an Astrotel that is approaching Mars. The two smaller modules between the TransHab and the solar array are cargo bays. The Astrotel Cargo Freighter autonomously delivers all cargo to the Astrotel contained within a standard cargo bay. These are pressurized modules to facilitate crew unloading of consumables and RRU hardware. Once emptied the cargo bay could be discarded or used to provide added crew volume. Table 3 summarizes the Astrotel components and their masses.

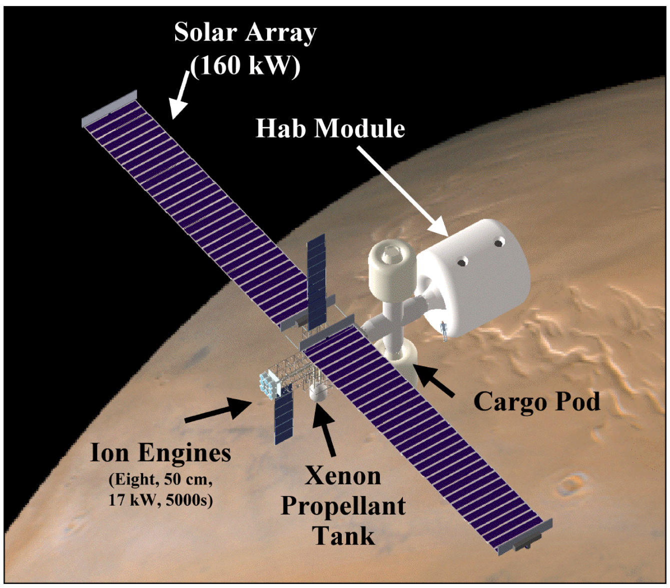


Figure 5 Astrotel Concept

TRANSIT STATIONS

Spaceports are collection points for the arrival and distribution of humans, cargo and propellants destined for transport to planet or natural satellite surfaces or to cycling Astrotels. In past architectures such Spaceports were large, rotating, permanently crewed platforms. In this new concept, a Spaceport is based on the Astrotel design philosophy. Crew stay times are limited in order to minimize effects of zero-g. Crew maintenance is minimized by maximum application of autonomy in order to shorten stay times. Station-keeping, orbit corrections, orbit-phasing delta-Vs could easily be performed by the same or even smaller IPS system envisioned for the Astrotels.

Table 2 Astrotel Equipment Mass Summary

Subsystem or Item	Dry Mass	Consumables	Subtotal Mass
Physical/Chemical Life Support	2,778	3,840	6,618
Crew Accomodation	5,000	4,224	9,224
Structure	5,500		5,500
EVA Equipment and Consumables	1,183	446	1,629
Communications and Information	320		320
Thermal Control	550		550
Power	785		785
Propulsion	644		644
Attitude Control	500		500
Radiation Shielding	9,254		9,254
Escape Pod and Reserve	22,000		22,000
Crew	1,200		1,200
Utility Module Base	5,000		
Permanent Cargo Bay	3,000		
Spares	2,100		2,100
Total Mass	**59,814**	**8,510**	**68,324**

TAXIS AND USING THE ATMOSPHERE TO PUT THE BRAKES ON

Taxis provide transportation between Spaceports and Astrotels. In order to minimize propulsive energy use, Taxis use advanced aeroassist technologies for planetary orbit capture. Aerocapture takes maximum advantage of planetary atmospheric drag to slow the vehicle on its approach from planetary space. The key sizing assumptions are: a.) Minimal radiation protection for the crew is provided since transfer times to/from the Astrotels is less than 7 days, b.) No cargo is transported to the Astrotel by the Taxi, c.) 15% of the entry mass is aeroshell, d.) LOX/LH propulsion system at I_{sp} of 460 s and thrust of 60,000 lbs./engine, e.) Fuel cell energy storage, no solar array power source, and f.) Propellant tank augmentation (expendable drop tanks and in some cases additional engines) is required at Mars. Taxis escape planets and are placed onto hyperbolic rendezvous trajectories with Astrotels. Rendezvous time to Astrotels is measured in days in order to reduce the duration of crew time in the expected cramped quarters, since crew volume is comparable to Apollo. Figure 6 illustrates the Taxi departing Earth. Figure 7 illustrates the common crew module.

Table 3 summarizes the system mass of the common crew module. This crew module is used in both the Taxi and the Mars Shuttle, to be discussed later. Table 4 summarizes the overall mass breakdown of the Taxi system.

The Taxi vehicle uses aerodynamic orbit capture (aerocapture) at both Earth and Mars. The entry speed at Earth is modest and the velocity to be lost is consistent with a relatively short-duration aerocapture flight. At Mars, the entry speed is much larger than the exit speed desired, so that the aerocapture vehicle has to cruise around the planet at nearly constant altitude for a relatively long period. A vehicle with relatively high lift-to-drag ratio is required at the start of the cruise in order to supply the required centripetal acceleration and to stay under a total g-load of about 5. The current baseline Taxi vehicle is known as an elliptical raked cone (Scott, 1985) which has a maximum lift-to-drag ratio of 0.63. The crew is provided g-seats that rotate in order to accommodate the varying g-load direction and the quite different thrust direction during propulsive maneuvers than for aerocapture maneuvers. The base vehicle is about 20 mt, dry. Fully loaded Taxis vary in mass from the single stage low delta-V Mars and Earth configurations of ~40 mt to the three stage high delta-V Mars configuration of ~300 mt most of which is propellant.

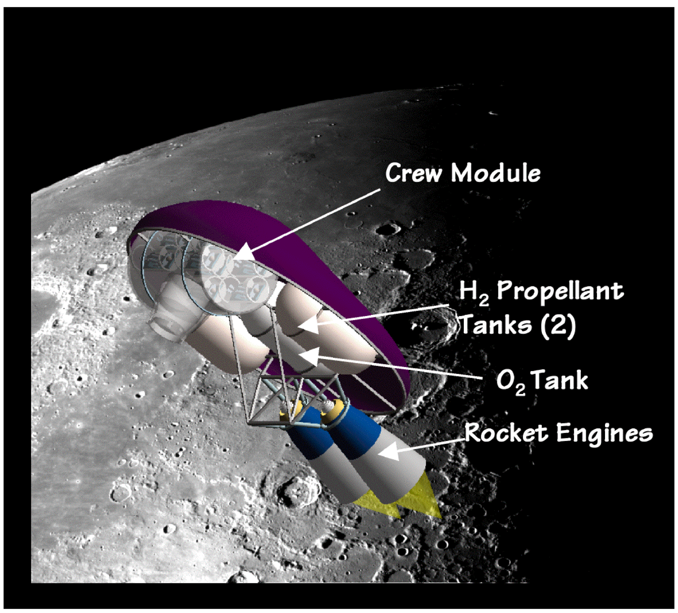


Figure 6 Taxi Leaving Earth

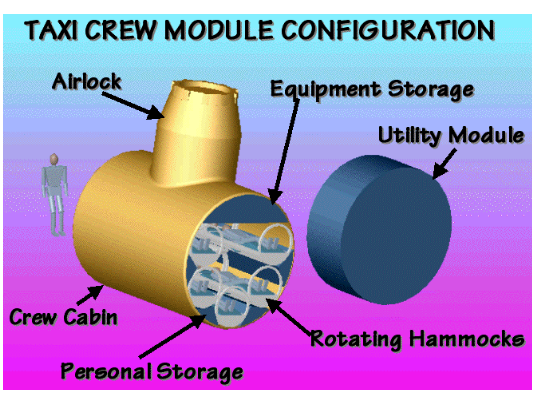


Figure 7 Common Crew Module Cut-away

Table 3 Crew Module Mass Summary

Crew Module System Element	Mass, kg
Crew Cabin	
Structure	1,431
Airlock plus Tunnel	810
Insulation, 30mm	188
Nav	100
Telem	100
Elect	100
Comm	50
Crew Accom	694
Crew Mass	818
Misc	200
	4,490
Utility Module	
ECLSS	121
Electrical Power	171
Subtotal	292
Total Mass	**4,783**

Table 4 Taxi System Mass Summary

Taxi System Element	Dry Mass, kg
Crew Module	7,207
Primary Structure	1,000
Propulsion	4,407
Subtotal	12,614
Aeroshell	2,967
Grand Total Dry Mass	**15,581**

The Taxi on departure from Mars is either a two stage vehicle (3 of 7 opportunities) or three stage vehicle (4 of 7 opportunities) of which the last stage is the basic vehicle similar to that shown in Figure 6. Figure 8 shows the 1st and 2nd stages of the three stage Mars Taxi with their augmentation tankage.

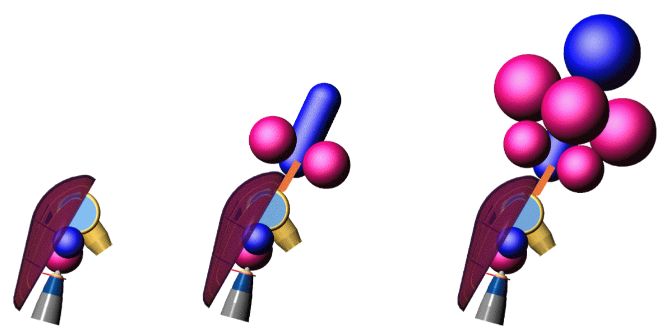


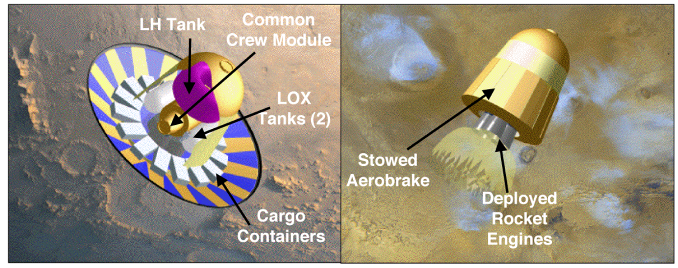
Figure 8 1st, 2nd, and 3rd Stage of Three Stage Taxi Departing Mars

The propellant tank configuration (blue = LOX and red = LH) was selected after evaluation of several solutions with the lowest mass of tankage and structure, compactness and simplicity of design in mind. The thrust vector of the engines passes through the vehicle's center of gravity requiring minimum engine gimbaling (within ±5 deg) during operation of all three stages, including 3rd stage with empty tanks. All propellant tanks except the 2nd stage LOX tank are spherical. The 2nd stage LOX tank is cylindrical with hemispherical heads and performs dual functions: besides storing LOX it is used as the main structural element supporting all 1st stage tanks and 2nd stage LH tanks and transferring resulting forces to the 3rd stage. Aluminum tank jackets includes multi-layer insulation (MLI). The 3rd stage tanks (two LOX and two LH tanks) are supported by Aeroshell structure (reinforcing ribs). The shells are reinforced to properly distribute dynamic pressure and concentrated support loads. The 2nd stage LOX tank is connected to two central Aeroshell reinforcing ribs similar to a boat keel. The attachments, besides small bending moment, are loaded with dynamic force resulting from the acceleration of the 1st and 2nd stages, transferred through 2nd stage LOX tank shell and reinforcing rings. Four 2nd stage LH tanks are attached to front reinforcing ring of the 2nd stage LOX tank by tension members and supported on its rear reinforcing ring. Tension members are tangential to the respective tank shells. The 1st stage LH tanks are similarly attached to the 1st stage LOX tank and supported on the 2nd stage LOX tank front reinforcing ring. The 1st stage LOX tank, by far the heaviest component of the system, is supported from the 2nd stage LOX tank front reinforcing ring. All the tanks are internally and/or externally reinforced so the concentrated as well as dynamic pressure loads are properly distributed to the shell.

The Taxi is propelled with three Pratt & Whitney RL 60 engines, rated at 60,000 lbs. each. The engines may be gimbaled ±5°, both vertically and horizontally. All three engines are shown installed in a common frame but in the future individual gimbals and actuators for each engine will be designed (an earlier two engine version of the Taxi is shown in Figure 6).

MARS SHUTTLE

The Mars Shuttle transports a crew of 10 to and from the Mars Base and the Mars Spaceport near Phobos. The Mars Shuttle supports crew needs during the very short transit (<1 days) between the Mars Base and the Mars Spaceport. In addition, the Mars Shuttle carries out delta-V maneuvers, performs aero-entry and landing maneuvers within the Martian atmosphere, navigates autonomously during all maneuvers, provides electrical power to its subsystems and carries RRU cargo from the Mars Spaceport to the Mars Base. The Mars Shuttle is designed to travel only between the Mars surface and the Mars Spaceport at Phobos. The basic vehicle is a low lift/drag ratio design with a deployable 20-m diameter aerobrake used during entry and landing. At take-off, the aerobrake is stowed to reduce atmospheric cross-section and minimize drag. The low lift/drag ratio design offers reduced mass, ease of fabrication, reduced cost and growth accommodation over higher lift/drag designs. The Mars Shuttle mass is 67 mt fully loaded, 42 mt at entry, 32 mt landed and 22 mt dry. Figure 9 illustrates computer-generated designs of the Mars Shuttle in its entry and launch configurations.



(a) Mars shuttle at entry (b) Mars shuttle after launch

Figure 9 Mars Shuttle Configuration

TURNING PLANET DIRT INTO ROCKET FUEL

The use of planetary resources significantly reduces the material that needs to be brought up through the gravity well of the Earth and delivered to a planetary transportation node. The energy required for transportation of propellant is proportional to the square of the velocity change that it must undergo. For example, the energy required conveying propellant from the Moon to L-1 is approximately 1/30th of that required from the Earth's surface to L-1 and requires a much simpler spacecraft. The transportation architecture includes the use of *in situ* resources at the Moon (lunar polar water ice), at Phobos (O_2 production from carbon reduction of the regolith), and on the surface of Mars (heat extraction of water from regolith, electrolysis, O_2/H_2 liquefaction and storage). In addition, there are off-world processing and storage facilities including a water electrolysis, O_2/H_2 liquefaction and storage at the Earth Spaceport and O_2/H_2 storage at the Mars Spaceport. The baseline architecture requires resource production rates of 15.4 kg/hr of Phobos LOX, 6.6 kg/hr of Martian water, 10.2 kg/hr of lunar water, and 1.6 kg/hr of LOX/LH from lunar water at the Earth Spaceport. Production rates account for the lower duty cycle of planetary solar power (Phobos, Moon, and Mars).

Excavation

It is a challenge to design excavation and extraction systems for Phobos, Mars and the Moon since they lack significant, or any, atmospheres, gravity is low to extremely low, and temperature variations are extremely high. In addition to operating under these extreme environments, a Phobos excavation system requires obstacle avoidance, rock sorting, continuous excavation duty cycle, excavator flexibility, and a low mass. A bucket-wheel excavator system (BWE) specifically designed for extraterrestrial environments, as shown in the left of Figure 10, was found to meet all these requirements (Johnson, 2002). The BWE excavates continuously and simultaneously transports the materials to storage. Excavation forces are primarily horizontal and provided by the mass of the entire excavator instead of only the bucket mass allowing the BWE to work in extremely low gravity without exterior anchoring, provided its mass provides ample traction for excavation and forward movement.

Extractors and Reactors

On the Moon and Mars relatively low temperatures (100°C to 500°C) are required to boil off the water from the soil (assumed in a 1% abundance by weight) after which is collected, liquefied and stored in preparation either for transport to the Earth Spaceport or for further processing into rocket fuel at Mars. At Phobos the soil is placed in a Carbothermal reactor where combined with high temperatures (1700 °C) and hydrogen and carbon, oxygen is produced. After producing the oxygen it is liquefied for eventual transport to the Mars Spaceport for long-term storage.

Processing

Except for rocket fuels needed to launch from the surface of the Moon, producing LOX/LH from lunar water is done at the Earth Spaceport (by means of electrolysis and liquefaction) where an abundant supply of solar energy is available. Processing of Martian water occurs near the Mars Base after which LOX/LH are stored for use by the Mars Shuttle.

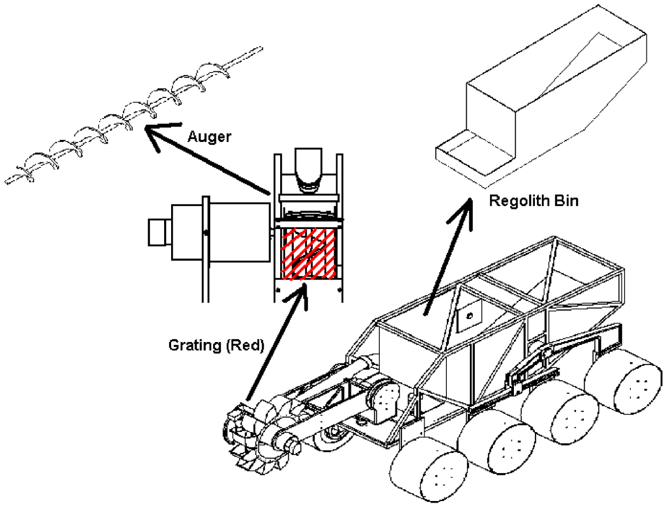


Figure 10 Bucket Wheel Excavator and Transporter

WHAT'S THE BEST TRANSIT SYSTEM AND HOW MUCH WILL IT COST TO OPERATE?

A computerized model has been developed that describes the baseline architecture and a number of options and generates life-cycle cost (LCC) estimates. These life-cycle estimates are best used to compare competing options rather than establishing credible estimates of the real total cost of such an architecture. The model is highly integrated and interrelated including transportation vehicles; ground systems; subsystem technology assumptions; *in situ* resource assumptions and systems; and celestial mechanics analysis. The model was developed to facilitate integration of various system elements, to facilitate overall architecture trade studies and to support life cycle cost analysis. This approach allowed independent development of individual elements and supporting analyses by focusing on the relationships among the system elements and establishing element-to-element links for selected inputs/outputs. A detailed work breakdown structure was developed including Advanced Technology Development, Flight System Development, Launch and Operations. Costs were tracked at one and two levels below the main categories. Cost references were a mix of actual data from past missions and component-level performance parametrics developed by technology specialists in NASA, industry and academia. The 15-year operations costs for the transportation architecture options are displayed. This model can perform trade studies so one can vary system capabilities or architecture assumptions and hence compare cargo mass and *in situ* resource requirements and eventually life-cycle costs. In the current version of the model, there are over 100 individual sub elements

This model is best at comparing life-cycle costs of different architecture options. Several cost estimates were generated varying a number of assumptions including (a) basic trajectory type (Aldrin Low-thrust Cycler vs. Stopover Cycler), (b) launch costs ($2k or $10k), (c) with and without ISRU, and (d) use of solar or nuclear for surface and space power sources. These cost estimates are compared in Figure 11 below. It is interesting to note that the LCC for the solar option are substantially lower than for the nuclear option. The reasons for this are that the solar array power requirements, masses and costs are low and the nuclear systems are very expensive to develop even if small. If the power requirements had been substantially higher the nuclear option would look better. Another interesting thing to note is that for launch costs of $2k/kg there is not a significant benefit of ISRU in lowering LCC (one still needs to look at the operations costs to see if sustained operations costs would be significantly lower). However, when launch costs are $10k/kg there difference is much greater between the ISRU and non-ISRU cases. Figure 12 compares cost elements between the Aldrin and Stopover architectures. Of interest is the fact that the development costs are about $10 B less for the Stopover architecture, however the operations costs are higher.

The baseline Mars Astrotel scenario (Aldrin, $2k, solar, plus ISRU) life-cycle cost of $117 B is split between $5 B for advanced technical development, $69 B for flight system development, $1 B for launch services and $42 B for operations over 15 years. The total sustained operations costs are estimated at $2.8 B per year, or about 20% of the current NASA budget.

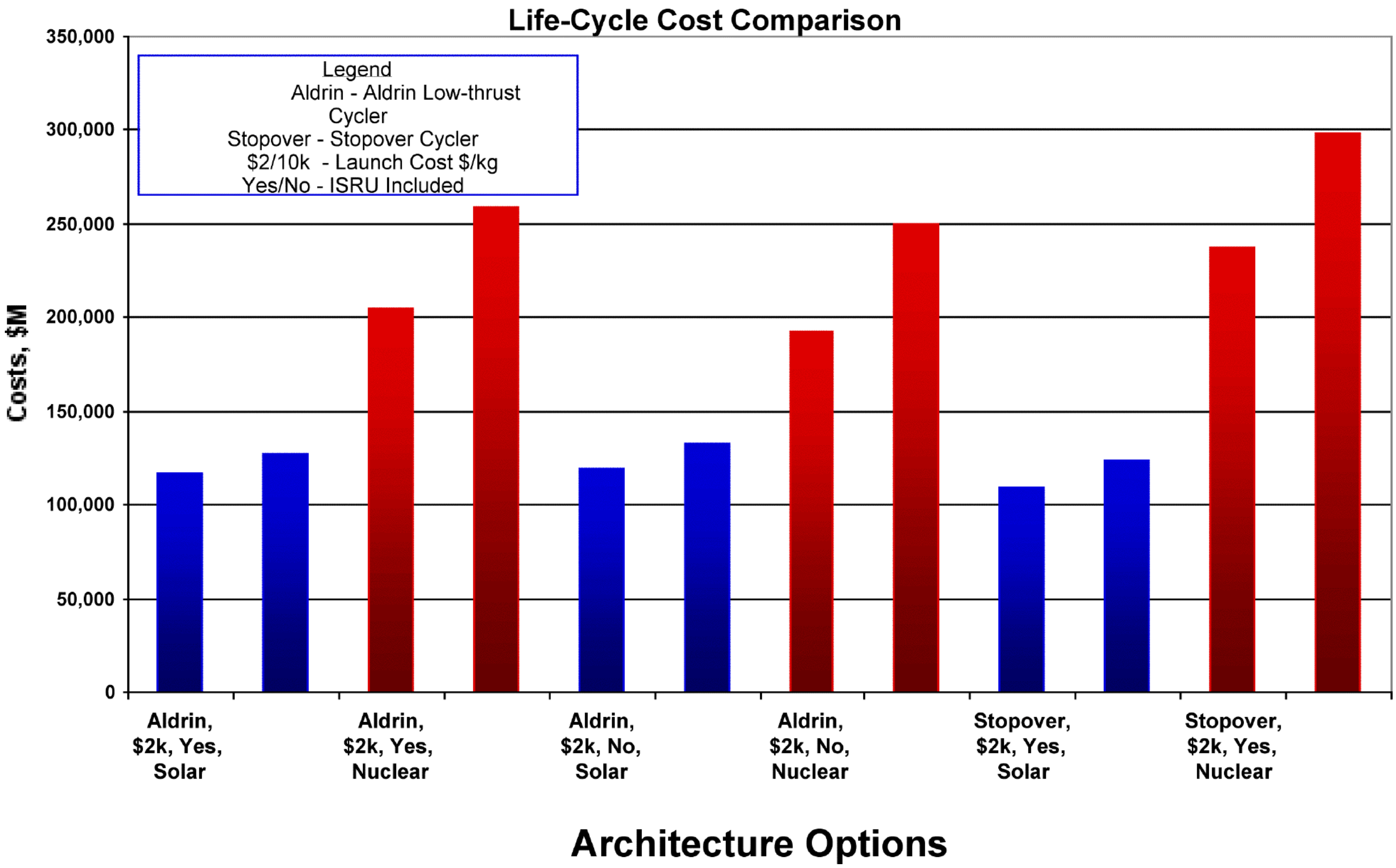


Figure 11 Life-Cycle Cost Comparison with Architecture Options

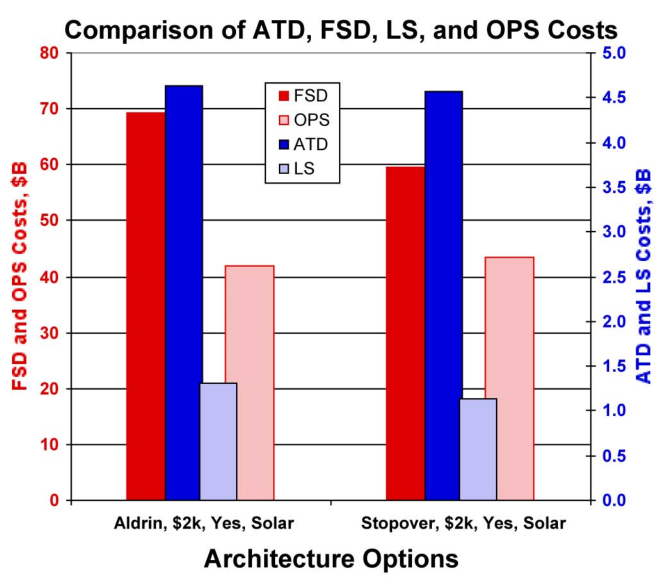


Figure 12 Comparison of Cost Elements between the Aldrin and Stopover Architectures

SUMMARY

There are a number of choices in future Mars exploration planning. One can follow the Apollo expedition approach, characterized as "flags and footprints", that has resulted in an absence of human exploration of the Moon for over a quarter century. Or one can develop an evolutionary approach to Mars exploration by establishing and using elements of the infrastructure needed to eventually maintain a human presence on Mars. The first choice will get humans to Mars but in the process we may damage political and public support to the extent that, like Apollo, Mars exploration is eventually abandoned. The latter choice lays the foundation for low life-cycle cost transportation that has a much higher probability of sustained political and public support.

If we know what systems and vehicles will be required in the future to support a Mars transportation architecture, the steps to the development of this infrastructure can be constructed including the long-range planning and costing, advanced technology development, advanced system development and flight-testing. Intermediate vehicle or surface systems can begin to be used, perhaps for lunar or Phobos exploration, as they are developed with the knowledge that their efficiency is driven not necessarily by their immediate use but by their eventual application in the overall infrastructure.

A key example of providing a context for future technology development is the assessment of the need for and development cost of nuclear power generation systems in a future architecture. If such systems are not absolutely required, or result in higher life-cycle costs than solar systems, as discussed above, then we can save considerable time and precious resources by not pursing the nuclear option.

A key question is the need for ISRU systems and whether they are cost effective. The answer to the question will require a better understanding of the future of launch costs. If launch costs can really be reduced to the order of $2k/kg the argument for ISRU may be less strong than if these costs are really $10k/kg. In addition, how ISRU fits into future near-Earth operations needs to be evaluated in order to determine if the development, and a portion of the operations costs, can be amortized beyond the Earth to Mars transportation needs.

The concepts envisioned by this systems architecture have a potential role to play in the expedition phase of Mars exploration. The application of these orbit and systems concepts in the expedition phase of Mars exploration may serve to reduce overall mission development costs and improve overall mission reliability and safety. Once launched into cycling orbits, Astrotels can orbit indefinitely as long as they are periodically maintained, improved and supplied with orbit correction propellants. In addition, the result of embracing such a mission concept early in an expedition phase means that a permanent inhabitation phase of Mars is closer. An implication of pursuing this path toward a Mars transportation architecture is near-term development of intermediate systems of immediate benefit to human space exploration, which have a role to play in the expedition phase of human Mars exploration.

By establishing the goal of developing a Mars transportation architecture, planning and mission context is provided for robotic and human space exploration. Without this framework, we may expend valuable resources on extraneous technologies and dead-end system developments.

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