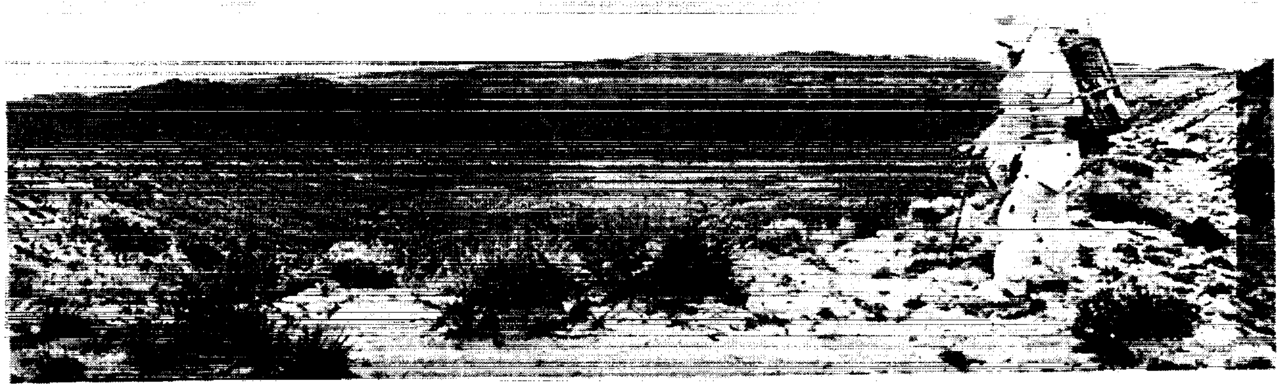
SECOND ANNUAL HEDS-UP FORUM



May 6–7, 1999
Lunar and Planetary Institute, Houston, Texas

LPI Contribution No. 979

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Edited by
Michael B. Duke

Sponsored by
Lunar and Planetary Institute
National Aeronautics and Space Administration

Lunar and Planetary Institute 3600 Bay Area Boulevard Houston TX 77058-1113

LPI Contribution No. 979

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Cover: Mars space suit tests, conducted at Meteor Crater, Arizona, are part of the preparation for the human exploration of Mars.

PREFACE

HEDS-UP (Human Exploration and Development of Space–University Partners) conducted its second annual forum on May 6–7, 1999, at the Lunar and Planetary Institute in Houston. This year, the topics focused on human exploration of Mars, including considerations ranging from systems analysis of the transportation and surface architecture to very detailed considerations of surface elements such as greenhouses, rovers, and EVA suits. Ten undergraduate projects and four graduate level projects were presented with a total of 13 universities from around the country. Over 200 students participated on the study teams and nearly 100 students attended the forum meeting. The overall quality of reports and presentations was extremely high, with most projects requiring that the students dig into space systems concepts, designs, and technologies in detail. University team outreach projects also reached approximately 1500 people through articles and Web sites developed by the students. Several of the teams had NASA or industry mentors and included visits to NASA centers as part of their class activities.

Awards were made to the three top undergraduate teams and the top team of graduate students. The first-place award went to a team from Wichita State University, Wichita, Kansas. Their faculty advisor was Dr. Gawad Nagati of the Department of Aerospace Engineering. Second place went to a team from the California Institute of Technology, Pasadena, California, with Dr. James Burke of the Jet Propulsion Laboratory as advisor. Third place was awarded to the University of Houston in Houston, Texas, where Dr. David Zimmerman was the faculty sponsor. The graduate award was made to a team from the University of Maryland, College Park, Maryland, under the sponsorship of Dr. David Akin.

Besides presenting their study results at the forum, the students were updated on exploration themes by NASA or NASA-related personnel who discussed current projects or views of human exploration. John Young gave a keynote address, recounting his lunar missions and encouraging the students to focus on the exploration of space as one of the key steps to preserving the future of humans on Earth. Steven Hoffman (JSC/SAIC) discussed current NASA concepts for the "surface mission," the set of activities that astronauts on Mars will undertake on early missions. George Parma (JSC) described the Transhab project, an inflatable habitat for the International Space Station and the human exploration of Mars. Dean Eppler (JSC/SAIC) discussed the recent field projects to define requirements for Mars EVA suits. Ron White (National Space Biomedical Research Institute) concluded the presentations by describing efforts underway to understand the adaptation of people to space in the context of missions to Mars. Steve Squyres (Cornell University) contributed an invited talk on recent findings from robotic missions and plans for subsequent robotic exploration of Mars.

The papers resulting from the investigations are collected in this report, along with selected contributions from invited speakers. These reports also are available on the HEDS-UP Web site (http://cass.jsc.nasa.gov/lpi/HEDS-UP/).

Many good comments about the program have been received, which will be used to improve and strengthen the program. Participants asked for mechanisms for greater interaction between the participating universities and between the universities and NASA. The Web site will be used next year to provide these linkages. An expanded program will be conducted next year, with up to about 20 university teams.

Two aspects of the university program bear special mention. Many of the concepts developed in the student design studies should be of considerable interest to NASA engineers and managers as missions to Mars are contemplated. HEDS-UP will work to make sure that the students' work is considered by NASA. The other aspect of interest is the outreach programs conducted by the universities, which included public Web sites, presentations to university and public audiences, and visits to elementary-school classrooms. Conservatively, 1500 additional people were directly reached by the outreach activities of the university teams.

HEDS-UP is off to a good start in building communities of interest in universities dedicated to advancing the human exploration of space. We are thankful to the sponsors of the project, particularly Lewis Peach of NASA's Office of Space Flight, for the opportunity to conduct the program. We thank all of the universities who participated so enthusiastically in the forum and for their work. And we thank the efforts of the LPI staff, who made the forum run smoothly and effectively.

Michael B. Duke
Houston, Texas
August 1999

Nancy Ann Budden (Lunar and Planetary Institute) displays the first-place plaque awarded to the team from Wichita State University, Wichita, Kansas.

Nancy Ann Budden and Mike Duke (Lunar and Planetary Institute), Joyce Carpenter (NASA Johnson Space Center), and Lewis Peach (NASA Headquarters) meet with instructor and students from the University of Southern California.

CONTENTS

AGENDA

May 6, 1999

8:00 a.m.	Registration, Coffee and Donuts
8:30	Welcome (M. Duke, LPI; J. Young, JSC)
9:00	The Mars Surface Mission (S. Hoffman, SAIC)
9:30	Surface Infrastructure Systems (Cornell University)
10:15	Search for Life (Metropolitan State College, Denver)
11:00	Habitats, Space Suits and Power Supplies (University of California, Berkeley)
11:45	Lunch Break
1:00 p.m.	Recent Results from Robotic Exploration Missions (Steve Squyres, Cornell University)
1:30	Surface Systems (University of Southern California)
2:15	Mars Greenhouse (University of Texas, San Antonio)
3:00	Break
3:15	Pressurized Rover (Wichita State University)
4:00	Transhab Design for the Space Station (George Parma, JSC)
4:45	Reception and Posters

May 7, 1999

8:00 a.m.	Coffee and Donuts
8:30	Astronaut Space Suit Field Demonstrations (D. Eppler, SAIC)
9:00	Mars Ballistic Exploration Vehicle (University of Maryland, undergraduate)
9:45	Mars Direct Revisited (California Institute of Technology)
10:30	Mars Transportation System (University of Texas, Austin)
11:15	Earth-Mars Transportation System (Georgia Institute of Technology)
12:00	Lunch Break
1:15 p.m.	Human Adaptation for Mars Missions (R. White, Space Biomedical Research Institute)
2:00	Wind Power on Mars (University of Houston)
2:45	Mars Ballistic Exploration Vehicle (University of Maryland, graduate)
3:30	Mars Field Geology Simulations (Arizona State University)
4:15	Mars Lander (Texas A&M University)
5:00	Wrap Up and Suggestions for the Future
5:15	Adjourn

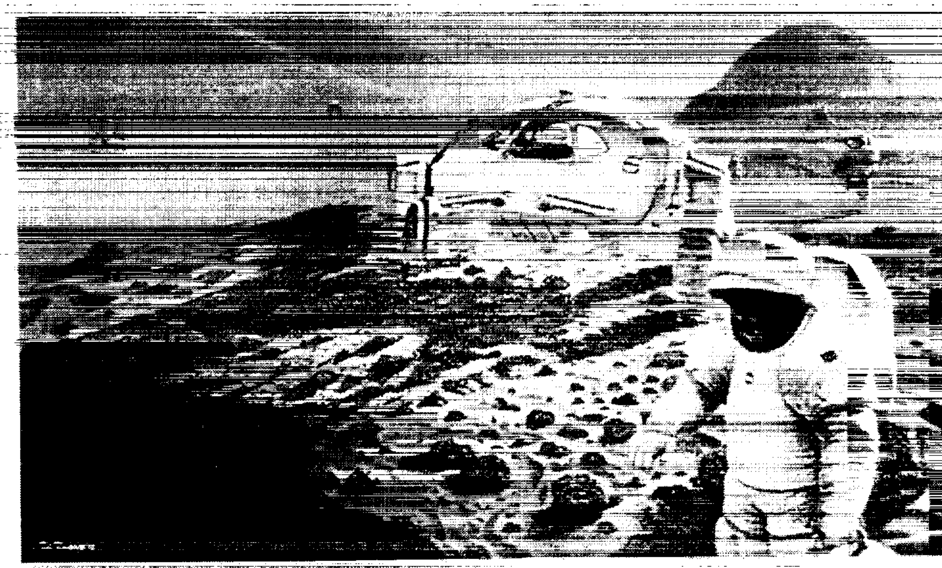
INVITED TECHNICAL PRESENTATIONS

The Mars Surface Mission

Steven J. Hoffman
Science Applications International Corporation

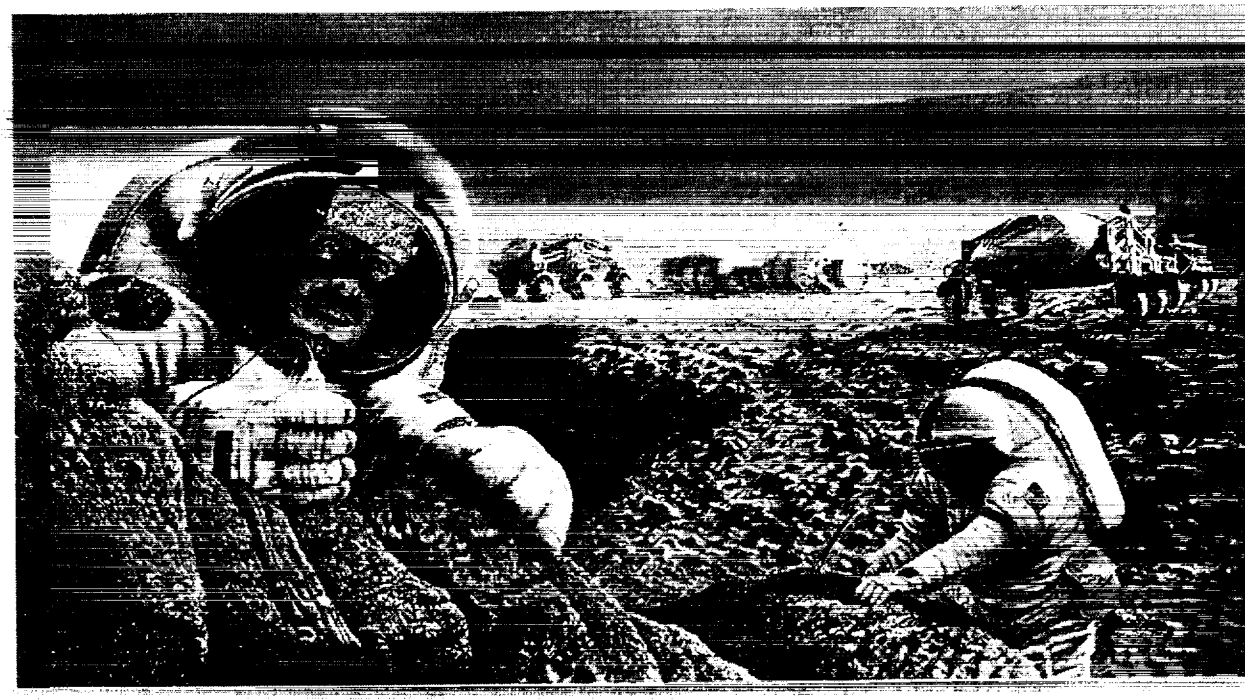
An overview is provided for the Mars Surface Mission, which designates that part of the human exploration of Mars conducted by astronauts on the surface. The surface missions will involve six crew members for about 500 days and will be focused on scientific exploration as well as learning to "live off the land." Some key points are illustrated here.

The Explorers



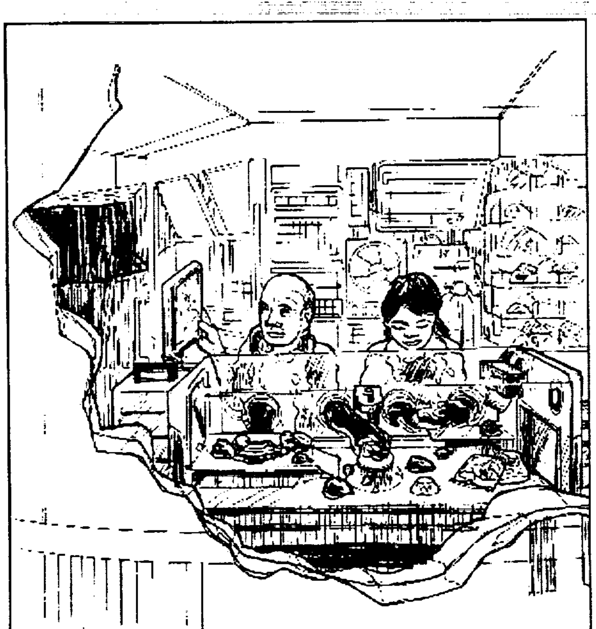
Exploring the region surrounding the landing site will be a key objective of the human crew sent to Mars. This image illustrates the EVA crew members beginning to explore the region in the immediate vicinity of the landing site. Pressurized rovers, such as the one illustrated here, will be used for a variety of tasks both close to and distant from the pressurized habitat. These rovers will have the capability to allow the crew to conduct EVAs, as required, in the vicinity of the rover. These pressurized rovers and teleoperated robots will allow the crew to explore regions well beyond "walk back" distances from their landing site.

A Martian Field Camp



Providing the systems necessary to set up and operate a remote field camp makes possible another means for extending the exploration range of the Mars crew. Crews operating from a field camp will allow interesting sites to be explored in more detail than would be possible if the EVA were staged from the landing site.

A Laboratory on Mars

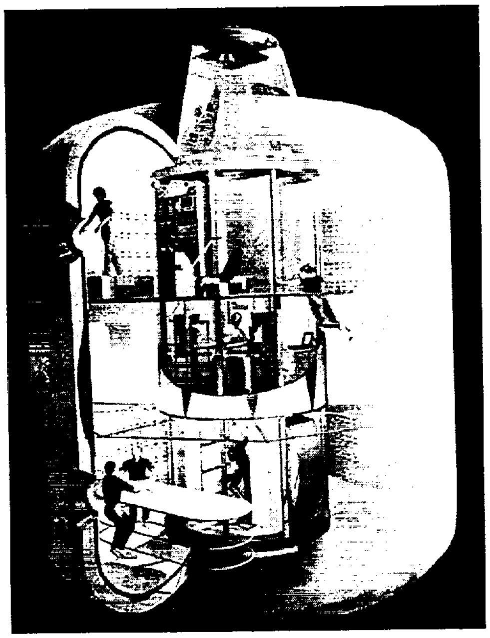


In addition to exploring the surface in the vicinity of their landing vehicles, Mars crews will require the capability to examine samples they have gathered in the field. This image illustrates crew members examining a number of collected surface samples inside a glovebox facility. This facility will not only protect the crew from potential hazards associated with the sample, but will also protect the sample from contamination by the crew. This preliminary examination of samples will be important for two reasons. First, a significant scientific objective will be to search for signs of indigenous life. Such an identification will be important before bringing these samples back to Earth. The second is to sort through the gathered samples and select only the most important for return to Earth. There will be mass restrictions on what can be returned with the crew and the samples will be no exception.

TransHAB: An Inflatable Habitation Module for ISS and Other Space Applications

George Parma
NASA Johnson Space Center

ISS TransHAB



Level 4: Pressurized Tunnel

Level 3: Crew Health Care

Level 2: Crew Quarters and Mechanical Room

Level 1: Galley and Wardroom

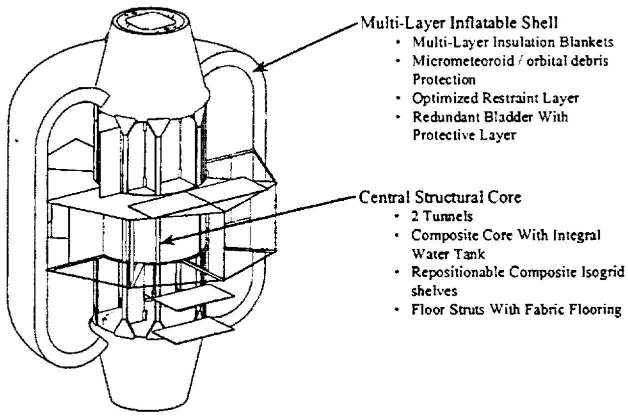
What is TransHAB?

- TransHab is a light weight inflatable habitation module for space applications
- Original concept for light weight module for transit to Mars
- Proposed to the International Space Station (ISS) Program as a replacement for the current Hab Module

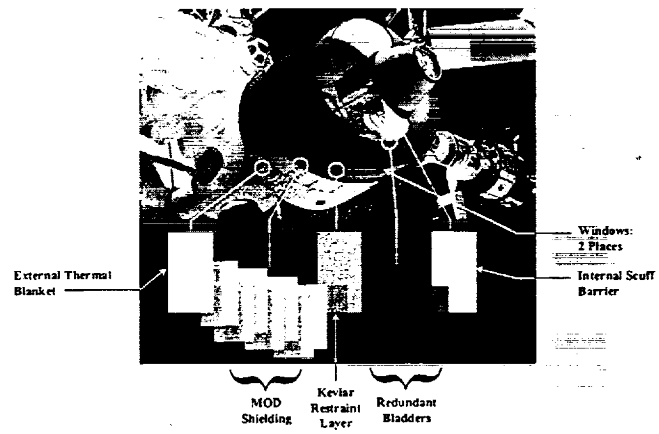
ISS TransHAB Functions

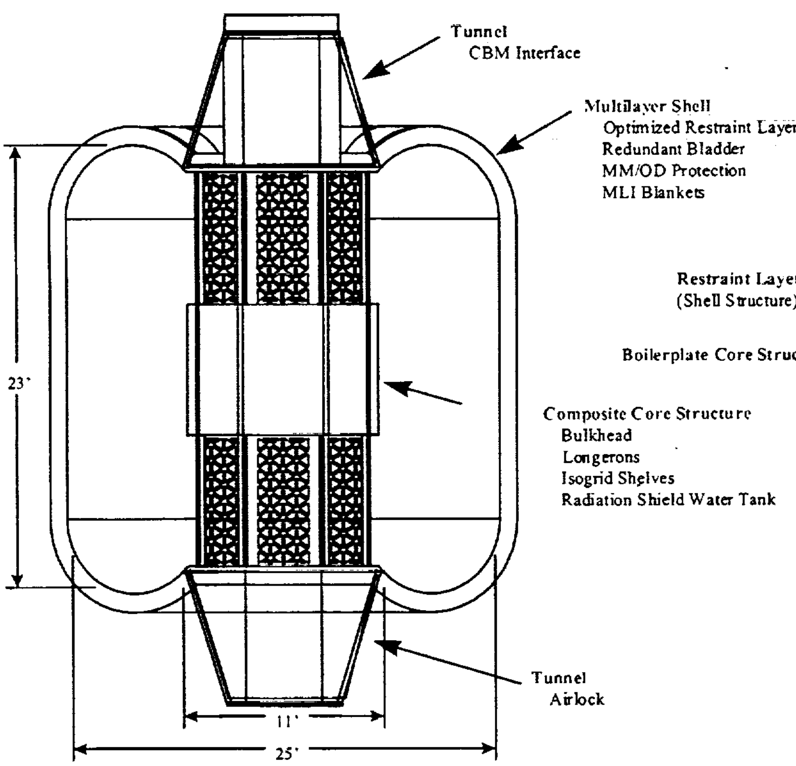
- Private Crew Quarters
- Galley & Dining
- Meeting area for entire ISS crew
- Health Care & Exercise
- Hygiene
- Stowage
- Crew Accommodations
- Environmental Control & Life Support System (ECLSS)
- Communications
- Command, Control & Data Handling
- Protection during Solar Particle Events

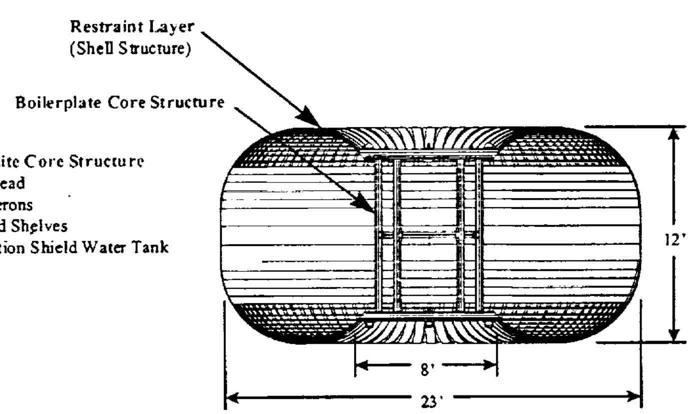
General Structural Configuration



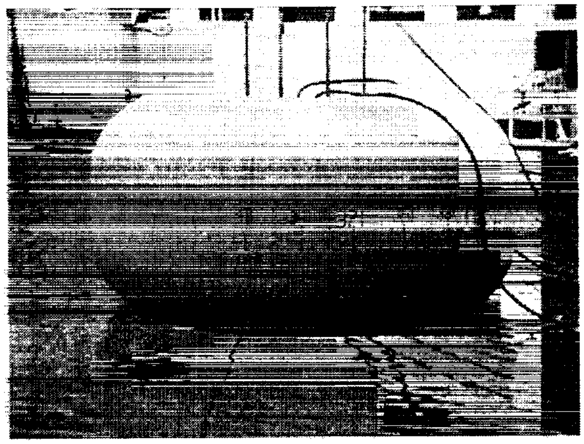
Multilayer Inflatable Shell Overview





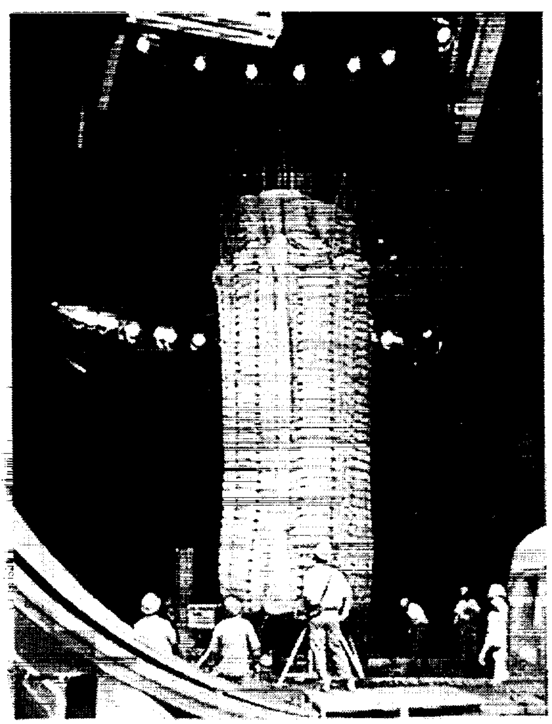


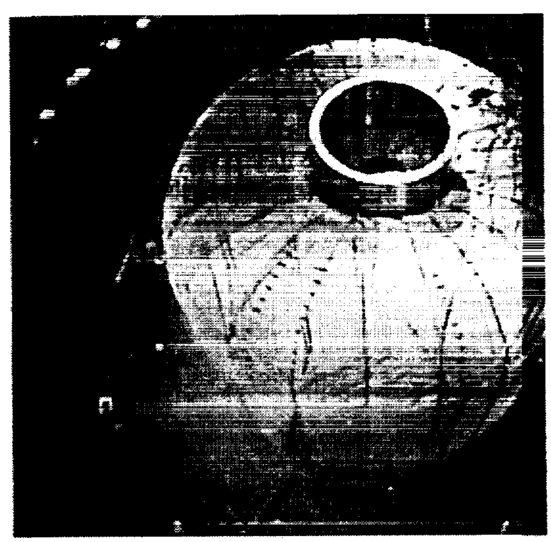
TransHAB Shell Development Unit 2



September, 12, 1998
Structural Integrity Verified to a Factor of Safety of 4.0

TransHAB Full Scale Shell Development Unit (SDU-3)



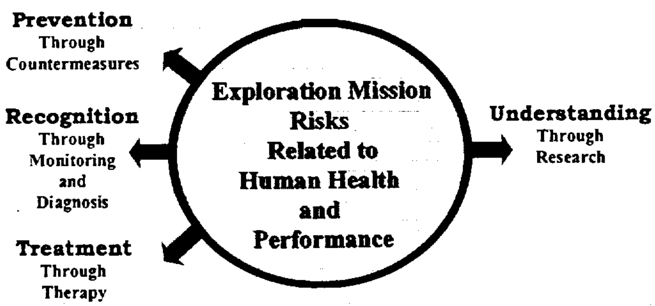


December 21, 1998
Vacuum Deployment Test

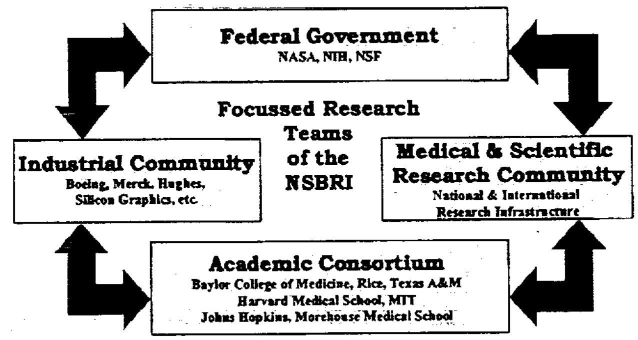
Human Adaptation: Mars Missions

R. J. White
National Space Biomedical Research Institute

NSBRI Leadership Role in Space Biomedical Research



NSBRI Parterners in Research



Current NSBRI Research Teams

Bone Demineralization
J. R. Shapiro, M.D.
USUHS

Cardiovascular Alterations
R. J. Cohen, M.D., Ph.D.
MIT

Human Performance
C. A. Czeisler, M. D., Ph.D.
Harvard

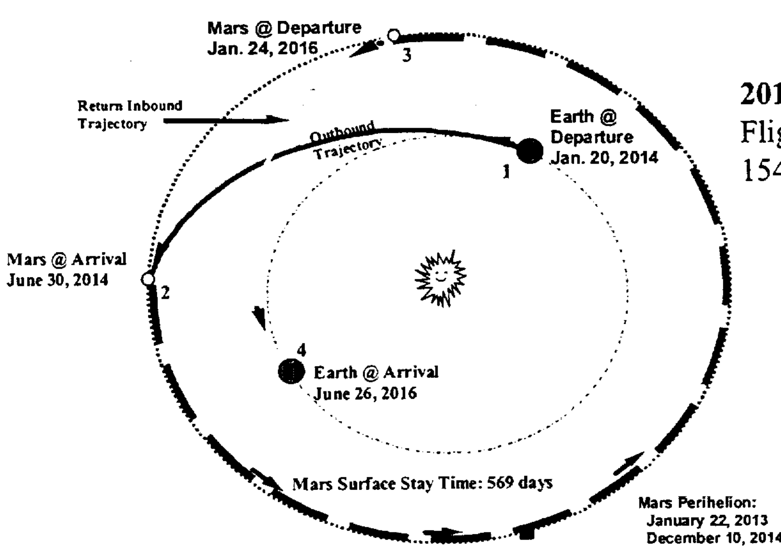
Immunology, Infection & Hematology
W. T. Shearer, M. D., Ph.D.
Baylor

Muscle Atrophy
R. J. Schwartz, Ph.D.
Baylor

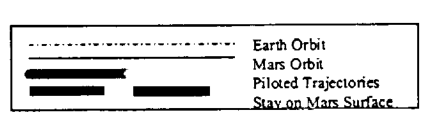
Neurovestibular Adaptation
C. M. Oman, Ph.D.
MIT

Radiation Effects
J. F. Dicello, Ph.D.
Johns Hopkins

Technology Development
V. L. Pisacane, Ph.D.
Johns Hopkins Applied Physics Laboratory



2014 Human Mars Mission Trajectory
Flight Profile: 161 day Transit Out,
154 day Return



In the Human Life Sciences, in order to prepare and conduct human exploration missions risks, we must: (1) assess, (2) understand, (3) mitigate, and (4) manage.

Tier I = CRITICAL RISKS: Severe risks characterized by
- potentially the most significant impacts to mission objectives and/or crew health, **AND**
- the existence of scientific evidence verifying these risks, **AND**
- the absence of effective countermeasures reducing these risks to an acceptable level

Tier II = VERY SERIOUS RISKS: Risks characterized by
- potentially significant impacts to mission objectives and/or crew health, **AND/OR**
- the necessity of gathering further scientific evidence verifying these risks, **AND/OR**
- the absence of a plan to validate the effectiveness of countermeasures reducing these risks to an acceptable level

Tier III = SERIOUS RISKS: Risks characterized by
- potentially serious impacts to mission objectives and/or crew health, **AND/OR**
- the absence of scientific evidence allowing these risks to be properly evaluated, **AND/OR**
- the existence of countermeasures to reduce these risks to an acceptable level

Risks may be moved from tier to tier as a result of implementing the plans:
- from critical risks to very serious risks through the implementation of a feasible plan to develop countermeasures
- from very serious risks to serious risks by the testing and validation of appropriate countermeasures
- from serious risks to manageable risks through the development of end items

BONE LOSS RISKS

Tier I - Critical Risks
- Acceleration of Age-Related Osteoporosis (1)

Tier II - Very Serious Risks
- Fractures (Traumatic, Stress, Avulsion) & Impaired Fracture Healing (2)

Tier III - Serious Risks
- Injury to Soft Connective Tissue or Joint Cartilage, and/or Intervertebral Disc Rupture With or Without Neurological Complications (3)
- Renal Stone Formation (4)

CARDIOVASCULAR ALTERATIONS RISKS

Tier I - Critical Risks
- None

Tier II - Very Serious Risks
- Occurrence of Serious Cardiac Dysrhythmias (1)
- Impaired Cardiovascular Response to Orthostatic Stress (1)

Tier III - Serious Risks
- Diminished Cardiac Function (2)
- Manifestation of Previously Asymptomatic Cardiovascular Disease (3)
- Impaired Cardiovascular Response to Exercise Stress (4)

HUMAN BEHAVIOR AND PERFOMANCE RISKS

Tier I - Critical Risks
- Human Performance Failure Because of Poor Psychosocial Adaptation (1)

Tier II - Very Serious Risks
- Human Performance Failure Because of Sleep and Circadian Rhythm Problems (2)

Tier III - Serious Risks
- Human Performance Failure Because of Human System Interface Problems and Ineffective Habitat & Equipment Design, etc. (3)
- Human Performance Failure Because of Behavioral Illness (e.g., Depression, Anxiety, Trauma, Psychiatric Dysfunction) (4)

IMMUNOLOGY, INFECTION AND HEMATOLOGY RISKS

Tier I - Critical Risks
- None

Tier II - Very Serious Risks
- None

Tier III - Serious Risks
- Infections (1)
- Carcinogenesis Caused by Immune System Changes (1)
- Altered Hemodynamics Caused by Changes in Blood Components (1)
- Altered Wound Healing (2)
- Altered Host-Microbial Interactions (3)
- Allergies and Hypersensitivity Reactions (3)

MUSCLE ALTERATIONS AND ATROPHY RISKS

Tier I - Critical Risks
- None

Tier II - Very Serious Risks
- Loss of Skeletal Muscle Mass, Strength, and/or Endurance (1)
- Inability to Adequately Perform Tasks Due to Motor Performance, Muscle Endurance, and Disruptions in Structural and Functional Properties of Soft and Hard Connective Tissues of the Axial Skeleton (1)
- Inability to Sustain Muscle Performance Levels to Meet Demands of Performing Activities of Varying Intensities (2)

Tier III - Serious Risks
- Propensity to Develop Muscle Injury, Connective Tissue Dysfunction, and Bone Fractures Due to Deficiencies in Motor Skill, Muscle Strength and Muscular Fatigue (3)
- Impact of Deficits in Skeletal Muscle Structure and Function on Other Systems (NR)

NEUROVESTIBULAR ADAPTATION RISKS

Tier I - Critical Risks
- None

Tier II - Very Serious Risks
- Disorientation and Inability to Perform Landing, Egress or Other Physical Tasks, Especially During/After g-level Changes (1)
- Impaired Neuromuscular Coordination and/or Strength (2)

Tier III - Serious Risks
- Impaired Cognitive and/or Physical Performance Due to Motion Sickness Symptoms or Treatments, Especially During/After g-level Changes (3)
- Vestibular Contribution to Cardioregulatory Dysfunction (4)
- Possible Chronic Impairment of Orientation or Balance Function Due to Microgravity or Radiation (5)

RADIATION EFFECTS RISKS

Tier I - Critical Risks
- Carcinogenesis Caused by Radiation (1)

Tier II - Very Serious Risks
- Damage to Central Nervous System from Radiation Exposure (2)
- Synergistic Effects from Exposure to Radiation, Microgravity and Other Spacecraft Environmental Factors (3)
- Early or Acute Effects from Radiation Exposure (4)

Tier III - Serious Risks
- Radiation Effects on Fertility, Sterility and Heredity (5)

CLINICAL CAPABILITY RISKS

Tier I - Critical Risks
- Trauma & Acute Surgical Problems (1)

Tier II - Very Serious Risks
- Toxic Exposure (2)
- Altered Pharmacodynamics and Adverse Drug Reactions (3)

Tier III - Serious Risks
- Illness and Ambulatory Health Problems (4)
- Development and Treatment of Decompression Illness Complicated by Microgravity-Induced Deconditioning (5)
- Difficulty of Rehabilitation Following Landing (6)

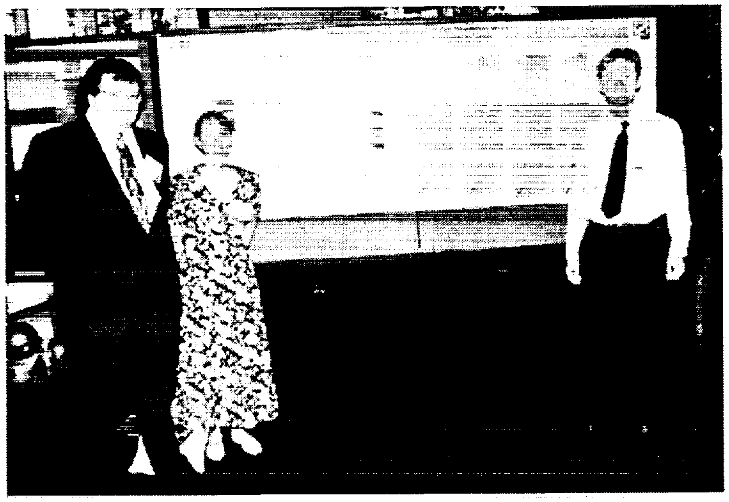
OTHER RISKS

Tier I - Critical Risks
- None

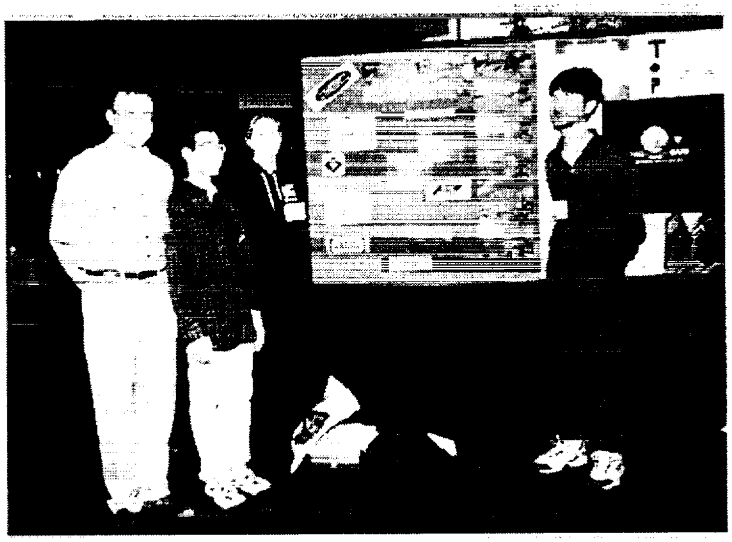
Tier II - Very Serious Risks
- Inadequate Nutrition (Malnutrition?)
- Post-Landing Alterations in Various Systems Resulting in Severe Performance Decrements and Injuries

Tier III - Serious Risks
- None

UNIVERSITY DESIGN STUDIES



Arizona State University



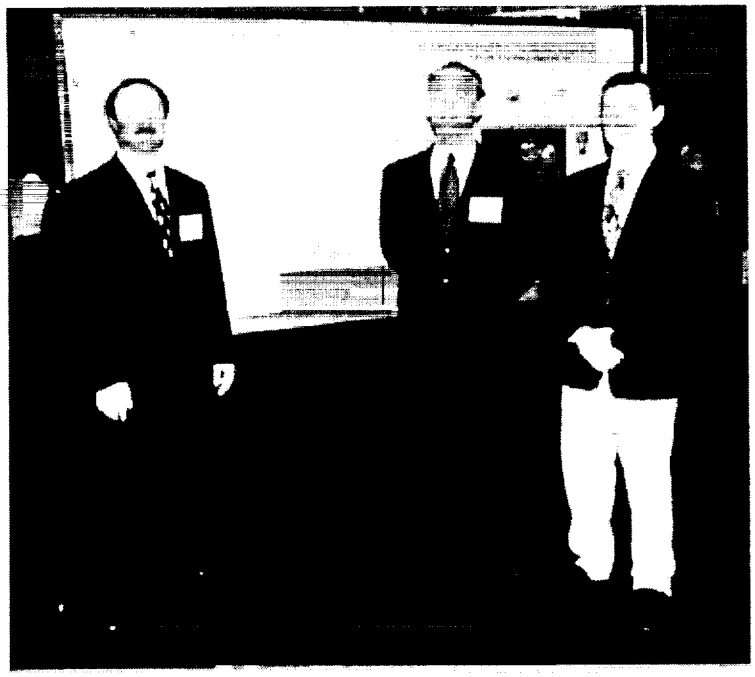
California Institute of Technology

Cornell University

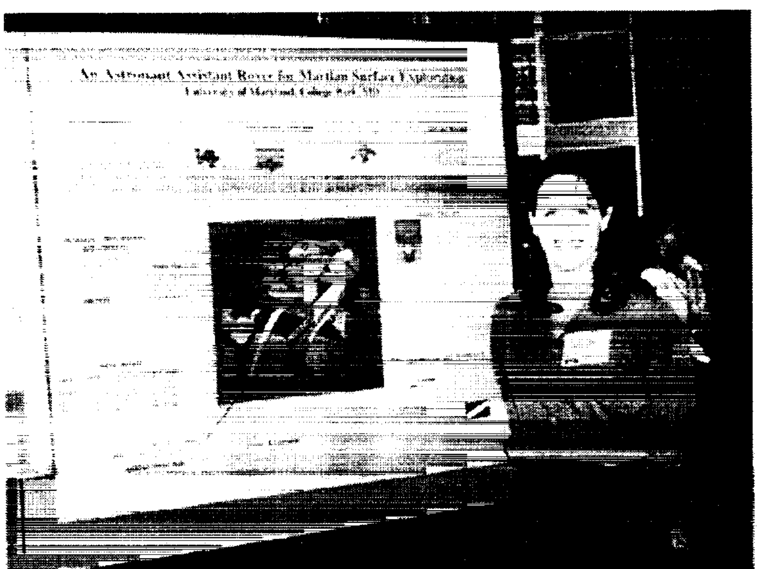
Georgia Institute of Technology

University of California, Berkeley

University of Houston

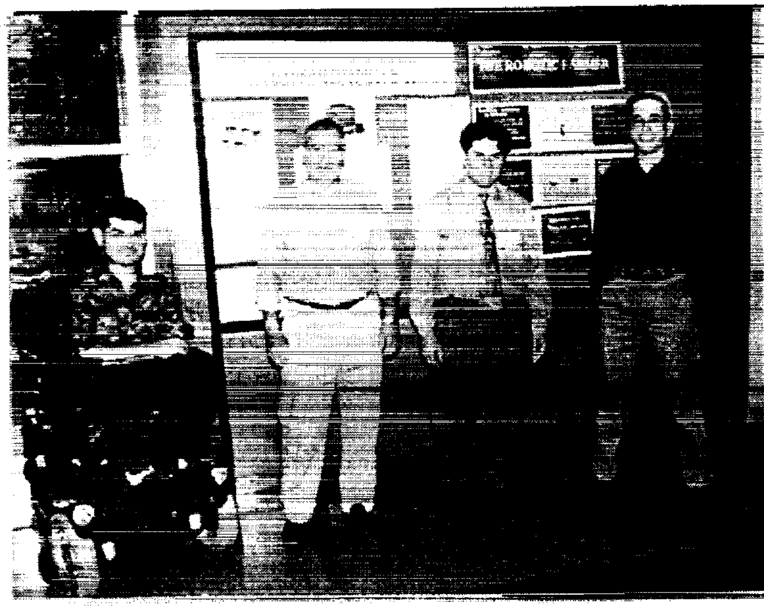


University of Maryland (Undergraduate)



University of Maryland (Graduate)

University of Texas at Austin



University of Texas at San Antonio

Wichita State University

Geologic Studies in Support of Manned Martian Exploration

Perry Frix, Katherine McCloskey, and Lynn D. V. Neakrase
Department of Geology, Arizona State University, Tempe

Ronald Greeley
Faculty Advisor, Department of Geology, Arizona State University, Tempe

Abstract

With the advent of the space exploration of the middle part of this century, Mars has become a tangible target for manned space flight missions in the upcoming decades. The goals of Mars exploration focus mainly on the presence of water and the geologic features associated with it. To explore the feasibility of a manned mission, a field analog project was conducted. The project began by examining a series of aerial photographs representing "descent" space craft images. From the photographs, local and regional geology of the two "landing" sites was determined and several "targets of interest" were chosen. The targets were prioritized based on relevance to achieving the goals of the project and Mars exploration. Traverses to each target, as well as measurements and sample collections were planned, and a timeline for the exercise was created. From this it was found that for any mission to be successful, a balance must be discovered between keeping to the planned timeline schedule, and impromptu revision of the mission to allow for conflicts, problems and other adjustments necessary due to greater information gathered upon arrival at the landing site. At the conclusion of the field exercise, it was determined that a valuable resource for mission planning is high resolution remote sensing of the landing area. This led us to conduct a study to determine what ranges of resolution are necessary to observe geology features important to achieving the goals of Mars exploration. The procedure used involved degrading a set of images to differing resolutions, which were then examined to determine what features could be seen and interpreted. The features were rated for recognizability, the results were tabulated, and a minimum necessary resolution was determined. Our study found that for the streams, boulders, bedrock, and volcanic features that we observed, a resolution of at least 1 meter/pixel is necessary. We note though that this resolution depends on the size of the feature being observed, and thus for Mars the resolution may be lower due to the larger size of some features. With this new information, we then examined the highest resolution images taken to date by the Mars Orbital Camera on board the Mars Global Surveyor, and planned a manned mission. We chose our site keeping in mind the goals for Mars exploration, then determined the local and regional geology of the "landing" area. Prioritization was then done on the geologic features seen and traverses were planned to various "targets of interest". A schedule for each traverse stop, including what measurements and samples were to be taken, and a timeline for the mission was then created with ample time allowed for revisions of plans, new discoveries, and possible complications.

1. Introduction

The intensive study of Mars through exploration by humans is a long-term goal of the United States space program, although a definite schedule has not been established for this to take place. The cost of a human mission to Mars will be very high – certainly billions of dollars. The mission must therefore be very efficient in achieving the goals for Martian exploration. The primary goals, as prescribed by NASA (McCleese 1998), encompass many issues from scientific to human exploration, and are summarized as follows; the search for water (when, where, form, and amount), the search for life (extinct or extant), the examination of climate (weather processes and history), and the exploration of resources (environment and its possible usefulness to future missions). The question of the availability of water on the surface of Mars is relevant to all the objectives. The possibility of the development of life, whether past or present, and how water has shaped the geologic and climatologic histories of the planet are both dependent on the presence of liquid water. The history of Mars and the corresponding implications of the potential for life could be useful in the study of how life evolved on Earth and interacted with its primeval surroundings. The latter goal is concerned with the availability and feasibility of exploiting in situ resources for future missions. In situ resources would greatly diminish the cost of exploration by providing the raw materials for fuel synthesis, construction of settlements, and the ability to initiate and maintain closed, self-sustaining life support systems.

In order to ensure the success and efficiency of a human mission to Mars, it is essential that the mission be planned and practiced in detail with simulations on Earth. This approach will provide astronauts rigorous training that will make routine activities time effective. At the same time, the simulation should help identify potential unexpected conditions and hazards, as well as assist astronauts in responding effectively to these conditions. Simulations on Earth also permit testing of equipment and procedures, and enable subsequent modifications at a small relative cost. Earth can provide an excellent analog environment, because its geology approximates some of that expected to be found on Mars.

In this report, we discuss our involvement in a Mars Field Analog Project conducted in October and November of 1998. We briefly explain our methods used in this exercise, as well as our results and conclusions derived from the information gathered in the field. We also discuss the procedure and results collected from a resolution study, which stemmed from our findings in the Field Analog Project. The third section of this report summarizes our planning of a manned mission to Mars using the information gained from the field exercise and the resolution study. We explain the procedures used as well as our reasoning behind our methods.

2. Field Analog Project

2.1 *Project Objectives*

The Field Analog Project was created to allow students to experience all aspects involved in sending humans to Mars, including planning, executing and examining the data gathered from a manned mission. The project not only explores what goes into planning and executing a mission, but also what problems can occur and what steps are necessary to plan, correct and/or adjust for those problems. The objectives of this project were to become more familiarized with Mars and its geology, to be able to use and interpret remote sensing data, to plan and run a field exercise, and to analyze the results from that exercise.

2.2 *Approach*

By taking into consideration the goals stated for Mars exploration, a simulated manned mission was conducted using two teams of geologists at two locations in northern Arizona, to study the feasibility of such a mission to Mars. One stipulation of the exercise was to have no prior knowledge of the site other than aerial photographs, taken at different scales, which simulated a sequence of space craft descent pictures. These pictures were used to plan and execute a manned mission as if on Mars. The local and regional geology of the two areas were determined and several "targets of interest" were chosen and prioritized based on their relevance to achieving the goals of the mission. Traverses were planned to each site and a timeline for the intire mission was created. The benefits of a manned mission over robotic missions were addressed including the problem-solving abilities of humans versus the programmability, expendability, and environmental tolerances of robots. Robotic missions, although expendable, are inherently less efficient due to the communications lag times between Earth and Mars that are involved. The use of experienced humans allows split-second, on site decision-making. The ability of communicating rapidly is useful in coping with potential uncertainties or problems encountered by a robotic mission.

2.3 *Procedure*

The first step in planning for the field exercise was the analysis of four or five photos of each respective landing area. These photographs simulated the increase in image resolutions that would be collected during the descent of a vehicle landing on Mars, in addition to the orbital images that are typically utilized in actual mission planning. Geologic features were identified according to the Mars exploration goals (McCleese 1998). Topography was estimated from the shadow lengths of features and the time of day of the photography.

After key geologic targets were identified, preliminary geologic maps and histories of each site were developed. Given the perspective that precise landing sites are rarely known until it occurs, prioritization of features was based on perceived accessibility and scientific value. Possible traverse routes and schedules for the six-hour exercise were devised. Prioritization was based on relevance to the Mars exploration goals, as stated above, accessibility to the features based on calculated topography and traverse length, and mission length or how much total time was available. Therefore, features that encompassed more than one objective were given a higher priority over other features. Alternate and contingency traverses for time or environmentally restricted mission profiles were also planned according to the prioritization used above. Time was also added into the schedule for the possible "collection" of rock and soil samples, as well as atmospheric content samples and temperature readings.

Upon arrival at the simulated landing site, one member of each team was chosen as the "geologist", another as the "backup geologist", and the third as CAPCOM. The geologist was required to wear a heavy frame backpack, large heavy-duty outdoor gloves, and a plexiglass visor to simulate the life support system (space suit) that the astronauts would be required to wear during their traverses. Once in the field, the first problem addressed was orientation to the surroundings. Although the true landing points were only off by tens of meters, no one 'landed' precisely where their traverses were planned to begin, and revisions were hastily devised according to the alternate prioritizations. Additionally, actually surveying the terrain predicated some traverse alterations solely for safety concerns. Initial observations in the field also prompted radical reworking of target prioritization in most cases. Over the course of the exercise innumerable 'targets of opportunity' that had not been foreseen in the preliminary remote sensing arose as well as emergency situations, which placed time constraints on the project. Final analysis invariably resulted in revised site geologic histories and maps, and the formulation of laboratory analyses of proposed returned field samples. The actual collecting of samples was precluded by the property owners, the Navajo Nation.

2.4 *Geology of the Landing Sites*

2.4.1 Site A Sedimentology and Structures

The geology of Site A based on the remote sensing showed some specific dominant features. The lowest resolution image showed that the general geologic context of the area as predominantly flat-lying terrain with little or no change in relief. Wind streaks, soil variations, and some larger channels were the dominant features at this scale, but many of the more detailed features were not identifiable. Channel structures could be traced through the field area on the higher resolution photos, and the shading and texture of the surface suggested relatively recent debris from running water. Most of the area between the channels seemed to be flat but strewn with cobbles of various sizes. A few of these cobbles seemed to have a wood-like texture. Near the western edge of the area, a large hill or possibly a butte dominated the terrain. Its summit showed signs of weathering in the broken nature of the boulders, which were interpreted to be volcanic. The areas between the hill and the channels seemed have a substantial, fine soil cover indicated by half-buried clasts and mud cracks. The shading of the main surface material changed over the entire area from the channels to the hill, which could indicate differences in composition or differences in weathering and/or soil formation.

4.2.1 Site A Igneous Petrology

If indeed the hill was volcanic in nature, then most of the rocks in the area should have compositions that fit with this hypothesis. The soils were thus assumed to be clays and muds directly associated with the weathering of volcanics. The specific types of weathering products from different types of extrusive igneous rocks are well defined based on starting composition of the source rock (Blatt & Tracy, 1996). By determining the types of volcanics that the hill is comprised of, the nature and composition of the soils, clays and muds could then be inferred and tested. The change in color of the soil material may reflect the compositional differences from the weather of the hill material as uposed to other sources. The fractures observed in the photos should have also been consistent with known jointing patterns of the appropriate volcanic rocks, such as columnar jointing in basalts. Together with these attributes, the albedo or coloring of the rocks at the site should have also been appropriate for the type of rocks.

2.4.3 Planning the Site A Traverses

Prioritization of features to examine in the area was completed after the preliminary photogeologic analyses. The main goal for Site A was to visit the channels that were probable locales for the preservation of water related structures. Adhering once again to the main mission goals for Martian exploration, the presence of water-sculpted landforms was the first step to detecting the possible existence of liquid water at the surface. This in turn could indicate that paleoclimatic conditions were once warm enough to support life. The traverse from the landing site to the main channel would be the most important for these reasons. The second most important goal involved determining the composition of the hill. Was it indeed volcanic or was it something else? The final area of interest was the soil and its formation. Could the weathering be traced through the differences in the soil types? Many locales across the field area were chosen as alternates for the various priorities since accessibility to these sites would be limited by time and landing location.

2.4.4 Actual Site A Geology in the Field

The geology of the site provided some unexpected surprises when compared to the photogeologic study. The first observations were of the actual landing site, where the clasts that were observed on the photos were identified. The soil had a hard, reddish covering and was strewn by pieces of petrified wood, sandstone, and jasper. There was no real distinction between particles of different sizes, indicating the sorting was poor. Mud cracks were nearly ubiquitous. The channels provided a look at the sediment being carried downstream. The middle of the channels was coated in a fine-grained gray sediment that seemed to match the layers seen in the distance near the hill (Figures 2.3, 2.4). The size of these sediments was a fine grained to silty sand. There were areas in or near the channels that seemed to be where water had pooled and then evaporated leaving the behind evaporate residue. In the largest channel there was a cut bank where the stratigraphy could easily be seen (Figure 2.5). At this locale the red surface soil was measured to be 13.5 cm thick. It appeared to be an oxidized zone with a slight fining upward sequence in the distribution of the coarser grained particles, although the sorting was still poor. Below this layer was a predominately greenish-gray layer approximately 30-35 cm thick. The sorting in this layer was better developed and also contained possible root casts that were filled with red sediment from the above layer (Figure 2.3). The hill in the field area that appeared to be volcanic rock was actually a sandstone layer lying on top of softer shale or mudstone layers. The less resistant layers were eroding causing the upper, more resistant sandstone layers to fracture. The result was the creation of fairly large sandstone boulders, some of which could be found on the plain where the channels were located. In general, the topography and geologic features found at the site were visible and identifiable on the "descent" images. The one main difficulty discovered was determining the compositional nature of the hill and surrounding areas. Other discrepancies mainly occurred with the size of the features seen. Smaller features, such as the stratigraphy within the large channel were not as visible in the images due to the resolution used. Another discrepancy we found dealt with estimating the topography of the area from the "descent" sequences. In many cases, the steepness of the landscape was greater than originally perceived from the images.

Table 2.1 Site A geologic predictions and observations.

Predictions from photogeology	Observations from field work	Rating of Accuracy
Boulders and hill were volcanic	Boulders and hill were sedimentary	1
Small clasts in photos looked like wood	Petrified wood, jasper, sandstone	7
Channels - braided	Braided streams	10
Fine grained sediment at bottom of channel	Fining upward sequence and evaporites	8
Channel walls have some stratigraphy	Two main layers, red and gray with root cas	5
Soil differences with changes in shading	Two main soil types, red and gray	8
Mottled surface of soil - mudcracks	Mudcracks and uneven soil surface	9
Some vegitation and animal tracks	More extensive vegitation and animal tracks	7

Rating of 1 is lowest accuracy and 10 is the highest.

2.4.5 Site B Sedimentology and Structures

The reconnaissance photographs of the landing site showed that the general geologic context of the area as predominantly flat-lying terrain with little or no change in relief. Wind streaks, soil variations, and some larger channels were the dominant features at this scale, but many of the more detailed features were not identifiable. The high resolution photographs showed that the region consisted mostly of thin bedded, light colored sedimentary rocks with several dry streambeds that cut into the sedimentary layers at the southern end of the study area, possibly as much as one to two meters, creating ledges and small overhangs. It seemed possible that the dry beds were once braided streams due to their winding channels and uneven floors. The region appeared to dip SSW to SW at a low angle, and displayed evidence of aeolian erosion from a SE direction based on wind shadows present behind outcrops and other obstructions. One region to the north of the proposed landing site had several large boulders of possible granitic composition on a small plateau. It was hoped that these were transported features that would offer insight into regions well beyond the travel constraints of the mission. Numerous small, shallow tributaries crossed the relatively flat surface to "feed" into the larger streambeds. These tributary paths were faint, but branched several times, possibly indicating that whatever amount of water present at the site was drained out of the region. The large number of tributaries seemed to indicate that the transport volume of each stream was very low. These features led to the hypothesis that the region was the modern remnant of an ancient floodplain.

2.4.6 Igneous Petrology

If the boulders located to the north edge of the site location were granitic in composition, it would explain their greater resistance to weathering than the surrounding sedimentary units. The boulders also have a very rounded appearance, consistent with the spheroidal exfoliation weathering pattern of granite. Quartz and feldspars would then be assumed to be present, due to the extent of weathering of the boulders, and soil samples were planned to be collected at and near that location (Blatt & Tracy, 1996). Also, the sandstones seemed to be darker in color (possibly red) which may indicate a more arkosic composition. This would agree with the granitic composition of the boulders since they could then become the source of the feldspars necessary for the arkose sandstones.

2.4.7 Planning the Site B Traverses

Prioritization of features to examine in the field was completed after the preliminary photogeologic analyses. It was decided that the main points of interest were the large channel to the south of the photos as well as the boulder region to the north. The large channel, being the most likely source of liquid water, was chosen as a high priority to look for evidence of life, as well as give clues to the climatological history of the area. The boulder region was chosen as a priority in order to determine if they were the same composition as the flat sediments, if they were an intrusive igneous body, thus creating the possibility of hydrothermal environments for life, or if they were transported from another location. With these areas in mind, several traverses were planned according to possible landing locations and differing time constraints.

2.4.8 Actual Site B Geology in the Field

Upon the preliminary observations from the landing site the original traverses were altered. It was determined that the boulders to the north were not transported from another location, and were of sedimentary origin. The large streambed to the south would thus be the most likely source of finding a diversity of samples. Roughly clockwise traverse plans starting in the eastern region of the study area were adopted, and the researchers extensively studied the large southeastern outflow channel, the upper end of the flow system in the south west and offered cursory observations of the boulder-filled northern region.

Most of the rocks at Site B were red to tan, fine-grained sandstones. The sandstones contained cross beds of various sizes indicating more than one possible depositional environment. Larger cross beds may have represented ancient sand dunes and would indicate a desert environment, while more recent fine mica-bearing cross beds seemed to be deltaic deposits indicating an aqueous environment. The intermixing of the two cross bed features meant that the amount of water entering the area fluctuated several times during the emplacement of the sediments. Some of the sandstones contained carbonate material weathered from the rocks in the form of weathered out holes (Figure 2.7), and solution pits on the exposed surfaces. The general reddish color of all of the rocks of the region indicated an oxidizing environment typical of a deltaic system. Samples documented at the site included extensive transported chert from the southeastern outflow channel, and small pieces of coal and petrified wood from the southwestern channel system. Evidence of recent fluvial activity in the region included ripple marks (Figure 2.6), puddles of water in the bases of some streambeds (Figure 2.8), and mud cracks in other streambeds where water has seeped into the ground or evaporated. Vegetation and signs of animal life were observed at the site, and vegetation growth patterns were found to coincide with a regional vertical joint set oriented towards the NW. Again, the topography and geologic features found at the site were visible and identifiable on the "descent" images. The main difficulty discovered was determining the compositional nature of the boulders and surrounding areas. Other discrepancies mainly occurred with the sizes of features that were detectable. Smaller features, such as the stratigraphy within the large channels were not as visible in the images due to the resolution used. Another discrepancy we found dealt with estimating the topography of the area from the "descent" sequences. In many cases, the steepness of the landscape was greater than originally perceived from the images.

Table 2.2 Site B geologic predictions and observations.

Predictions from photogeology	Observations from field work	Rating of Accuracy
Thinly bedded sedimentary rocks	Thinly bedded sandstones with cross-beddir	10
Streams braided	Not very braided, only in places	6
Large boulders igneous or sedimentary	Sedimentary boulders (sandstones)	5
White spots on photos: weathering rinds	Caliche, evaporites	8
Oxidation not really apparent	Red, oxidized coating widespread	4
Small clasts, volcanic or sedimentary	Chert, coal, petrified wood	3
Sedimentary structures not identifiable	Ripple marks, puddles, mudcracks	2
Some vegetation and animal tracks	More extensive vegetation and animal track:	7
Vegetation lineations mirror joints	Vegetation followed vertical jointing of rock	6

Rating of 1 is lowest accuracy and 10 is the highest.

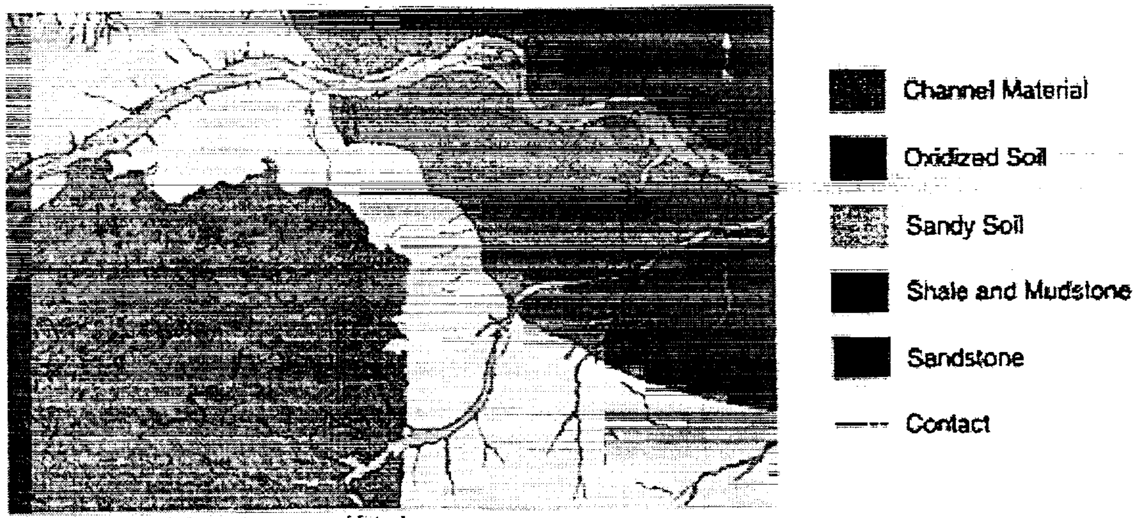


Figure 2.1 *Post-field geologic map of Site A*

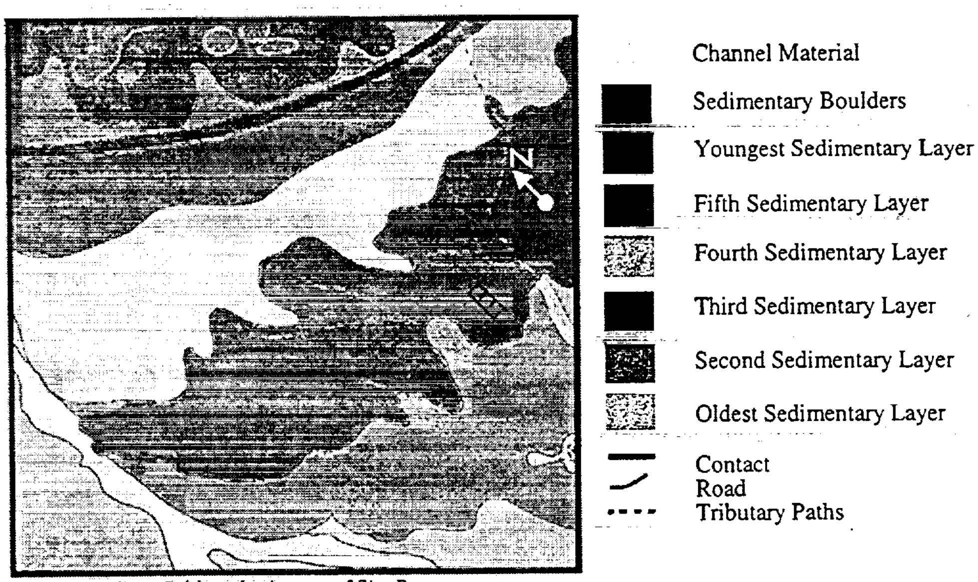


Figure 2.2 *Post-field geologic map of Site B*

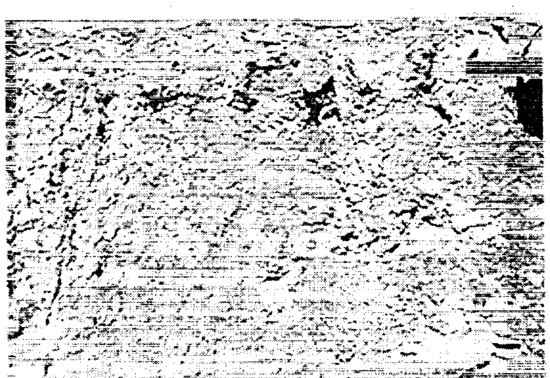


Figure 2.3 *Soil stratigraphy from Site A channel showing root casts.*

Figure 2.4 *Channel from Site A showing extent of soil horizons.*

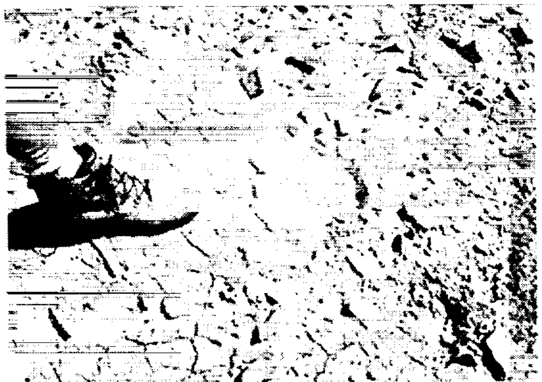


Figure 2.5 Evaporite deposite from Site A channel.

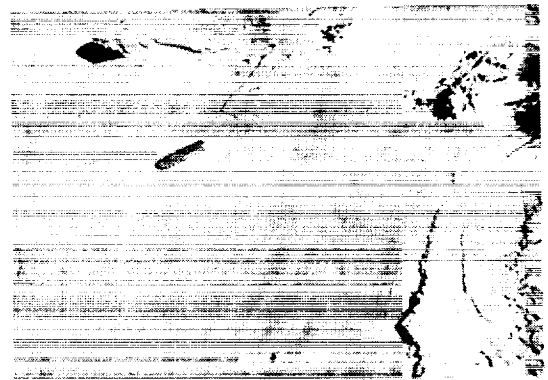


Figure 2.6 Ripple marks in sediment at Site B.

Figure 2.7 Weathered sandstone with carbonate at Site B.

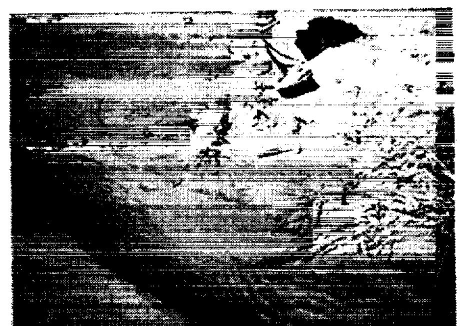


Figure 2.8 Standing liquid water and ripple marks at Site B.

2.5 *Discussion and Implications for Manned Missions*

2.5.1 Results of the Mars Analog Field Exercise

Given the resolution of the remote sensing and the cursory field survey parameters of the exercise, it was shown that a degree of successful 'mission' planning can be accomplished using the techniques mentioned above. A major success was the ability to identify large scale geologic structures such as fluvial systems or sedimentary strata in their study areas with only minor difficulties. Also, relatively precise planning of preliminary transit routes and traverse timing resulted from accurately mapped horizontal distances in each of the study areas. One discovery made during the exercise was the necessary balance between keeping to the planned timeline schedule, and impromptu revision of the planned mission to allow for conflicts and problems. A good deal of planning needed to be done before arrival in the field, not only of the traverses, but also of what experiments we wanted to conduct, and samples we wanted to take. Once in the field however, many of these plans needed to be corrected, due to the greater amount of information gathered about the area from field observations. Many areas of interest that were discovered upon landing at the site were not visible in the 'descent' photos due to the resolution of the images, and thus could not be planned for, causing minor alterations to the planned traverses. (It should be noted that the photographic resolution is only limited by grain size or the emoltion of the film used.) Also, some areas turned out to be of more scientific value than originally thought and thus, due to time constraints, other areas had to be cut from the schedule. These decisions could only be made upon arrival, and thus any planned mission must allow for these types of changes in its schedule. The people involved should be qualified and able to make the decisions necessary to compensate for these unforeseen discoveries.

2.5.2 Complications Experienced During Field Exercise

Numerous problems occurred during the exercise, many of which remained completely unforeseen during planning. Most notable amongst the failures was the unreliable estimation of topography based on shadows in the reconnaissance photos. In most cases distinction of shadow from equally dark features was very difficult, making accurate measurement of shadow lengths very nearly impossible. In some instances, solar geometry in the photos completely concealed topography that at other times would have been clearly visible. These two factors combined to produce gross underestimates of vertical transit possibilities at both study areas.

Field documentation proved cumbersome throughout the exercise. Maintaining the analog environment of the project, the actual field geologist on Mars would be in an environmental suit with bulky gloves and a visor; precise, delicate movements would be severely limited. Two different methods of taking field notes were utilized; the use of strictly audio-recorded notes compared to that of traditional written field notes. The use of written field documentation, such as those methods acceptable in traditional field geology prove to be far too time consuming in the restricted EVA environment. One way around this shortcoming was to include photo documented samples that would have been collected and returned to a laboratory environment. By voice annotating the samples and photos, a very clear picture of the collection environment could be maintained without the use of notes. Rock and soil samples were not allowed to be taken from the study areas as required by the landowners.

Proper spatial orientation at any given time also proved to be an onerous issue. Orientation for site documentation purposes was done using the descent photographs, and was an experience that consumed the attention of all present. The final accuracy of these impromptu map reading sessions was usually highly questionable; at a minimum, teams became slightly disoriented, but perhaps most problematic was the inference of local geology to regional geology. By not knowing exact locations, the field observations allowed only cautious conclusions concerning the macro-geology on a regional scale of tens to hundreds of kilometers.

2.5.3 Proposed Field Mission Revisions

Mission planners should make extensive accommodations for time management in field operations. From a technological standpoint this should include mission specific implements such as specialized geologic tools. During the field exercise, the geologists' simulated life support system rendered most standard geologic tools worthless. Simple items such as hammers, sample containers, hand lenses, and cameras must be carefully designed in conjunction with EVA equipment. Inclusion of some sort of absolute location system analogous to terrestrial GPS would be beneficial to efficient field operations. The researchers in the field exercise also discovered that at least minimal geochemical analysis, such as detecting carbonates with hydrochloric acid, would have been highly beneficial in sample selection.

Secondly, simulated astronaut injuries indicated that several of the team members should be trained in multiple disciplines in order to compensate for unforeseen accidents or problems experienced during the course of the mission. During the field exercise, the "astronaut geologist" was injured, thus making field operations difficult. The other team participants had to then try and assume a major portion of the role of "geologist" that could not be performed by the injured person. Having the astronauts trained in a variety of disciplines insures that under such conditions, mission activities would continue with little to no loss of time or resources.

Currently in manned space exploration, the only information available to the astronauts for mission planning is the images taken by remote spacecraft. Thus, the remote sensing should be of the highest resolution possible in order to obtain the greatest amount of information about the landing site. Due to weight, size and cost limitations in spacecraft designs, the equipment used to produce the highest resolution imagery is not always chosen. With this in mind, it is necessary to determine the lowest resolution needed to view the regional and local geologic features that pertain to the objectives of Mars exploration with enough accuracy to adequately plan a manned mission. Therefore an in-depth study was done to quantify the resolution limitations of a variety of specific geologic features important to this type of mission.

3. Image Resolution Study

3.1 *Objective*

From our experiences during the Field Analog Project, we discovered that the one piece of information most useful to us were the 'descent images' of the proposed landing sites. However, the resolution of these images (3mm/pixel), though much higher than those used in current space exploration (not including the most recent Mars Orbital Camera (MOC) images), were not as adequate as we would have liked for determining the local geology of the site. The main reason for this section of research was to determine what resolution was necessary to view geologic features of differing size scales, which are important to achieving the main goals for Mars exploration.

3.2 *Approach*

For this study, we took a set of images and degraded each of them to differing resolutions, creating a resolution sequence. The sequences were then examined to try and determine what geologic features could be detected or interpreted at each resolution within the sequence. Each geologic feature was rated, from 1 to 10, as to its visibility or recognizibility at each resolution stage. The results were then tabulated with the hopes that a minimum necessary resolution could be determined.

3.3 *Procedure*
The method implemented in this research involved taking the highest resolution images of each "decent" set from the Field Analog Study and images from other analog sites elsewhere. These images were digitized on a HP DeskJet III flatbed scanner with HP's DeskScan II software and were stored as TIFFs for processing with Adobe Photoshop 5.0.1. Due to the size of the original photographs, they were scanned in sections and then hand mosaicked together using Photoshop, and some corrections were made to even out the contrast and brightness variation that occurred during the scanning process. The composite images were cropped to remove interfering overlays or objects along the edges of the images. Raw image resolutions were computed using scale references (the 3 meter cloth and 10.1meter width of road in a few of the images), and appropriate scale bars were produced. The images were then saved as Photoshop PSD files. Image dimensions were then calculated for new image sizes that would represent scaled degradations of the raw image.

$$\frac{\text{Raw Resolution (m/pixel)}}{\text{New Resolution (m/pixel)}} * \text{Original width (pixels)} = \text{New width (pixels)}$$

(note: the width was auto-scaled to height in Photoshop so the other dimension was accounted for).

This was done to produce sets of degraded images with resolutions ranging from 3 centimeters/pixel to 25 meters/pixel, depending on the original starting resolution of the image used. A resizing command was executed, scaling the images to greatly reduced pixel scales using Photoshop's Bicubic interpolation algorithm, which is "for the slowest, but most precise, method resulting in the smoothest tonal gradations" (Adobe Tech Document #315461, 1999). By this operation, pixel information was averaged, removing pixels from the overall image to simulate a lower resolution. The degraded images were next rescaled, again with the Bicubic algorithm, to the exact printable dimensions of their original images, which removed most of the pixellation and ensured that all the images were at uniform scales for analysis. This operation used the reduced image to interpolate the now missing pixels back into the image. Appropriate scale bars were added to the new images and then saved as Photoshop documents. Each degradation stage image was then viewed to try and determine what geologic features could be seen and interpreted, and the features were rated on a scale of 1 to 10 for visibility/recognizibility. The results were then tabled and a necessary resolution threshold was determined. This was done using a general gradient from the lowest resolution to the highest, which was based on an increasing assurance of feature recognition.

3.6 *Results*
From examining the degraded images and tabulating our results, it was determined that a fundamental resolution of approximately 1 meter/pixel was necessary to observe the general topography. By grouping the geologic features seen into fluvial, bedrock, and volcanic categories, the necessary minimum resolution can be narrowed down for each group. It is important to note, however, that this minimum resolution depends strongly on the size of the feature observed. For fluvial features, large streams and tributaries could be identified with a resolution of 3 meters/pixel while smaller features such as small streams, braided floors and smaller tributaries were identifiable at higher resolutions of approximately 70 to 50 centimeters/pixel. The volcanic features included fissures and fractures, which could be seen at 3 to 1 meter/pixel, flows and polygonal cracks, which could be seen at 70 to 50 centimeters/pixel, and small folds and bombs, which could be seen at 30 to 10 centimeters/pixel resolution. Boulders generally ranged in resolution from 3 meters/pixel to 50 centimeters/pixel, again depending on the size of the boulders (see Figure 3.1).

3.7 *Discussion & Implications*
Water is the key to finding life on Mars, and water related landforms can be divided into three main categories that are specific to different aspects of the search for life. The first of these categories involves looking for areas that are known to have had liquid water at some point in time. Locations of this nature would be useful in determining the time frame on the development of life and possibly some of the most recent occurrences. Sites of liquid water would include valleys with incised channels, where some stratigraphy could be observed in the walls, lacustrine or lake settings, with fine-grained sediment, and flood plains were intermittent finer- and coarser-grained materials could be good for preserving fossil evidence of early life.

The second category of locales involves a type of environment that is just recently becoming better understood on Earth. Hydrothermal settings would be good places to look for life on Mars because they are local areas of specialized temperature, pressure, and chemical composition. Locations on Earth that fit this description include places like the hot springs in Yellowstone National Park or black smokers at vents at the bottom of the ocean. These locales have the unique property that their extreme compositions and temperature and pressure regimes actually serve to protect organisms from surrounding hostile conditions. In the case of the black smokers, the heat and gases provide a region around the vent that is habitable for certain types of organisms that normally could not survive at the extreme high pressures and low temperatures of the sea floor. Regions such as the hot springs in Yellowstone could easily exist on Mars around thermal vents or volcanic settings. Geothermal heat and sulfides would be perfect for anaerobic microorganisms to thrive, much like they do on Earth (Farmer 1996). The mineral concretions formed in these settings have the ability to entomb any organisms providing a potential good fossil record (Cady and Farmer 1996).

The third potential area of interest would be anywhere where there is substantial amounts of ice. Places that would include this are presently limited to the polar ice caps. However, there are a couple of other areas that are possible locations of ice. Some of the water that once was on the surface has been hypothesized to have migrated to subterranean reservoirs. Here under the surface, reaping the benefits of some lithostatic pressure from the rocks above, ice could theoretically exist. Ice of this nature could potentially provide protection for the remains of organisms that were carried there by liquid water originally and then frozen in the subsurface. Another possibility for finding evidence of life is that of glacial landforms. Places where ice moving in the form of glaciers could have carried life with it and then deposited it in the fine-grained till material left after the glacier had vanished (Kargel and Strom 1992).

Geologic features that are found in these settings would most likely be important indicators of the presence of water and/or other potentially important resources. Some places that are indicative of water can then be possible locales for preservation of extinct life. By using the results of the Image Resolution Study, limitations can be placed on the types of imaging systems needed for the detection of these geologic features. Minimal resolutions for the positive identification of most moderately sized features have been determined to be around 1 meter/pixel. This estimate is dependent on the magnitude of the features being examined and is therefore biased against moderate and smaller features.

It is likely that MOC or MOC variants will be the standard of manned-mission planning data, because the highest resolution of the instrument corresponds well to the experimentally determined resolution limitations of the important geologic features, although, better resolution is still needed to adequately view features on the surface. Transporting significantly more robust imaging systems to Mars that are larger and thus capable of higher, multispectral resolutions, however, is not a realistic option given the cost and payload constraints of Discovery-class missions and the marginal benefits such an instrument would provide. Constraints on landing site selection can be better accomplished because the mapping resolution of MOC is higher than that necessary to identify geologic features. This increased resolution makes observation of higher order features possible, which allows for better prioritization of possible landing sites and traverses.

4. Planning a Mars Mission

4.1 *Introduction*

The goals for Mars exploration, as stated earlier, include the search for water, the search for extant and extinct life, and the exploration of resources for use in future missions. In achieving these goals, it is believed that manned missions would be more useful and more successful than robotic missions for several reasons. One benefit of sending humans over robots is the problem solving ability and reasoning skills versus the programmability and autonomy. Robotic missions, although more expendable, are less efficient when dealing with time constraints due to communication lag times between Earth and Mars. The ability of rapid communication is extremely useful in coping with potential uncertainties or problems and split-second and on site decision making could reduce damages to both the people and equipment involved.

4.2 *Approach*

This section of research was conducted to simulate the planning and initial exploration phase of a Mars Manned Mission using actual Mars data and images taken by the Mars Global Surveyor (MGS) and the Mars Orbital Camera (MOC). In this exercise, we attempted to perform all the steps involved in planning, preparing for and executing a human mission, including examining images for possible landing locations, determining local geology from MOC images of landing area, prioritizing possible sample collecting sites and geologic features, planning traverses, and creating plausible timelines for planned activities.

4.3 *Procedure*

The first step of this exercise was to look through the images of the surface of Mars taken by the Mars Orbital Camera (MOC) which were located on the JPL Photojournal website. These images were taken at 1.5 meters/pixel. We chose our site, a small crater within the larger Alexey Tolstoy crater, keeping in mind the goals for Mars exploration. We wanted an area that a spacecraft would be capable of landing at, that showed some evidence of liquid water in the area, and that had local geology that could potentially harbor life. Once the site was chosen, MOC photos of the site, as well as regional photos taken by *Viking 1* were examined for local and regional geology of the area, and preliminary climatological and geological histories were constructed. For Alexey Tolstoy crater, there are breaks in the crater wall which resemble places where channels might have eroded through the wall, allowing water to flow into the crater itself. This main crater is large, about 94 kilometers, which implies that it should be very deep. The observed floor of the crater is not much deeper than the rim, suggesting significant amounts of fill material. With evidence of possible channels and significant deposition of material in the crater, Alexey Tolstoy could be a good place for the preservation of fossil life forms. Looking at the smaller crater contained with in the walls of Alexey Tolstoy, this gives a natural window into the stratigraphy of the crater fill material.

Once the local geology was determined, the next step was to prioritize the geologic features seen in terms of meeting the goals and objectives of the mission. Features that would possibly give information about the sedimentary and aeolian materials within the crater floor were chosen for sample collection and other measurements to be conducted. Traverses were then planned to the various features discovered and locations for measurement and sample collections were determined. A schedule for each traverse stop and a timeline for the mission were constructed, with ample time allowed for revisions of plans, new discoveries and possible problems or complications.

4.4 *Results and Discussion*

4.4.1 Regional Geology

Alexey Tolstoy crater is located at approximately 47°S latitude, 235°W longitude, is 94 kilometers in diameter, and is situated in Promethei Terra on the boundary between the southern highlands and the northern lowlands. Looking at the crater area in the *Viking 1* orbiter image, the region consists mostly of cratered terrains believed to be volcanic in origin. There is a gradation from young to old craters, but the classification is based more on morphology than superposition/chronology. The oldest craters are very irregular in shape, subdued, have very eroded and degraded rims, and lack visible ejecta blankets, while the younger craters are more circular in shape, have good rims and small ejecta blankets. Many of the larger craters have slump material from the crater walls into the interior floor, possibly covering up any evidence of a central peak structure. In several places along the rim of Alexey Tolstoy, there appears to be areas where the rim has been down cut by some process, possibly by water. These small channels could have supplied water and sediment into the crater, filling it in and evening out the topography on the crater floor. A majority of these craters have accumulated aeolian material on their floors, creating a flatter appearance. There are several large scarps and wrinkle ridges trending northeast and a few edges of large ejecta blankets can be traced out along the landscape. In the west and southwest there are numerous knobs that rise approximately 5-10 kilometers above the local topography, and possess debris aprons around their bases. These knobs may be either remnants of basement material, emplaced by fracturing or faulting, which is currently being eroded and covered by aeolian material.

4.4.2 Local Geology

The image taken by the MOC, is a close up of the interior of Alexey Tolstoy crater. The area seems to have two material units: a darker, rougher, more knobby material located in the upper region of the image, and a lighter, smoother material located in the lower region. It appears that the lighter material has been eroded back, exposing both the darker underlying material and a small crater, approximately 850 meters in diameter, within the floor of Alexey Tolstoy. A majority of the ejecta blanket from this small, unnamed crater is visible, except the lower right portion, and the center of the crater is completely filled in with the lighter material. The rim is partially exposed, and it seems to be slightly eroded, having some breaks across it, possibly indicating cut channels into the floor of the small crater. There are a few smaller, possibly secondary impact craters in the area, but they are not as visible due to the smooth nature of the lighter material. There is a smaller section in the lower left corner of the image that is light in color but slightly rougher in texture than the rest of the light colored material.

4.4.3 Mapping Units

In the *Viking 1* Orbiter image, the slightly rough material from which the majority of the knobs penetrate was mapped as Old Knobby Terrain, whereas areas that were smoother and were devoid of knobs were mapped as Smooth Old Terrain. These two areas contain many scarps and extremely eroded craters. The younger craters were mapped according to relative age based on superposition of units. Crater floor material, slump material and ejecta blankets were assigned to each crater when possible (see Figure 4.1). For the MOC image, the three main materials were mapped into separate units based on surface roughness and albedo (dark rough unit, light rough unit, and light smooth unit). The main crater materials were also mapped into rim material, floor material and ejecta material, which were separated based on observed topography and surface roughness. (see Figure 4.2)

4.4.4 Feature prioritization, traverse locations and timeline

When examining the small crater of the MOC image, we determined that two traverses were necessary to cover the areas of interest found. First, we want to take samples of the material filling in the small crater as well as the excavated material from below. This material is thought to be comprised of sedimentary and aeolian deposits emplaced by the small channels that cut the rim of Alexey Tolstoy and flood the floor of the large crater. To sample some of the excavated material, we chose a sample collection site along the ejecta blanket for the crater (see figure 4.3 for traverse map). A stop along the rim of the crater was also planned to sample and to examine any possible stratigraphy of the underlying layers. We also want to take samples of the dark, rougher material exposed to the north, the light, rougher material in the lower left corner of the image, and the light, smooth material burying the crater. In examining these samples we will try to determine their compositions and see if the albedo differences seen are due to compositional or textural differences, or some combination of both. We also hope to find some evidence of extant or extinct life preserved within the sedimentary deposits. By estimating the amount of time necessary to travel to each site, to collect each sample and take other measurements, a rough timeline was devised for the two traverses and this portion of the mission. From these samples, we hope to be able to construct a more detailed geologic and climatic history of Alexey Tolstoy Crater and the surrounding area.

Timeline for Small Crater Traverse

Day 1 Central Traverse to Crater

00:00	Egress
00:00-00:30	**Landing Site**: Photography Contingency *Collections*
00:30-01:30	Assembly of Automated Meteorological Systems Assembly of Geophysical Systems
01:30-02:00	Transit to Site 4
02:00-02:45	**Site 4**: Photography Sample Collection of Ejecta Material
02:45-03:15	Transit to Site 5
03:15-04:00	**Site 5**: Photography Sample Collection of Rim Material Floor Material
04:00-04:45	Transit to Base Camp
04:45	Ingress

5. Conclusions

Currently, Mars exploration has four main goals that focus primarily on the presence of water, the geologic features associated with it, and their relationships to the possibilities of life. It is believed that manned missions would be more useful and more successful than robotic missions for several reasons that were discussed in this paper. To explore the feasibility of a manned mission, a Field Analog Project was conducted. The project began by examining a series of aerial photographs representing "descent" space craft images from which the local and regional geology of the two "landing" sites was determined and several "targets of interest" were chosen. The targets were prioritized based on their relevance to achieving the goals of the project and Mars exploration in general. Traverses to each target, as well as measurements and sample collections were planned, and a timeline for the exercise was created. From this it was found that for any mission to be successful, a balance must be discovered between adherence to the planned timeline schedule, and impromptu revision of the mission to allow for conflicts, problems and other adjustments necessary due to greater information gathered upon arrival at the landing site. At the conclusion of the field exercise, it was determined that a valuable resource for mission planning is high resolution remote sensing of the landing area.

This led us to conduct a study to determine what ranges of resolution are necessary to observe geology features important to achieving the goals of Mars exploration. The procedure implemented involved degrading a set of images to differing resolutions, which were then examined to determine what features could be seen and interpreted at each degradation stage. The features were rated for recognizability, the results were tabulated, and a minimum necessary resolution was determined. Our study found that for the streams, boulders, bedrock, and volcanic features that we observed, a resolution of at least 1 meter/pixel is necessary. We note however that this resolution depends on the size of the feature being observed, and thus for Mars the necessary resolution may be lower due to the larger size of some features.

With this new information, we then examined the highest resolution images taken to date by the Mars Orbital Camera (at 1.5 meters/pixel) on board the Mars Global Surveyor, and planned a manned mission. We chose our site, keeping in mind the goals for Mars exploration, then determined the local and regional geology of the "landing" area. Prioritization was then done on the geologic features seen and traverses were planned to various "targets of interest". A schedule for each traverse stop, including what measurements and samples were to be taken, and a timeline for the mission was then created with ample time allowed for revisions of plans, new discoveries, and possible complications. One point to note with this exercise is that with the increased resolution given by the MOC, you are seeing the features in greater detail, however, the size of the area you are viewing is much smaller than with lower resolution images, such as the *Viking 1* orbiter.

This exercise has given us an opportunity to experience what is necessary to plan and execute a manned mission. It also exposed us to the problems and complications that are associated with these types of operations, as well as the mechanical, technological, and design constraints for accomplishing the scientific goals of a mission of this type. Improvements in these areas, however, are allowing more scientific work to be done in greater quantities and at finer detail than previously accomplished. Perhaps these advancements will give a better understanding of the Martian geologic history that might, in turn, further the understanding of the beginnings of life on Earth.

6.0 *Acknowledgements*

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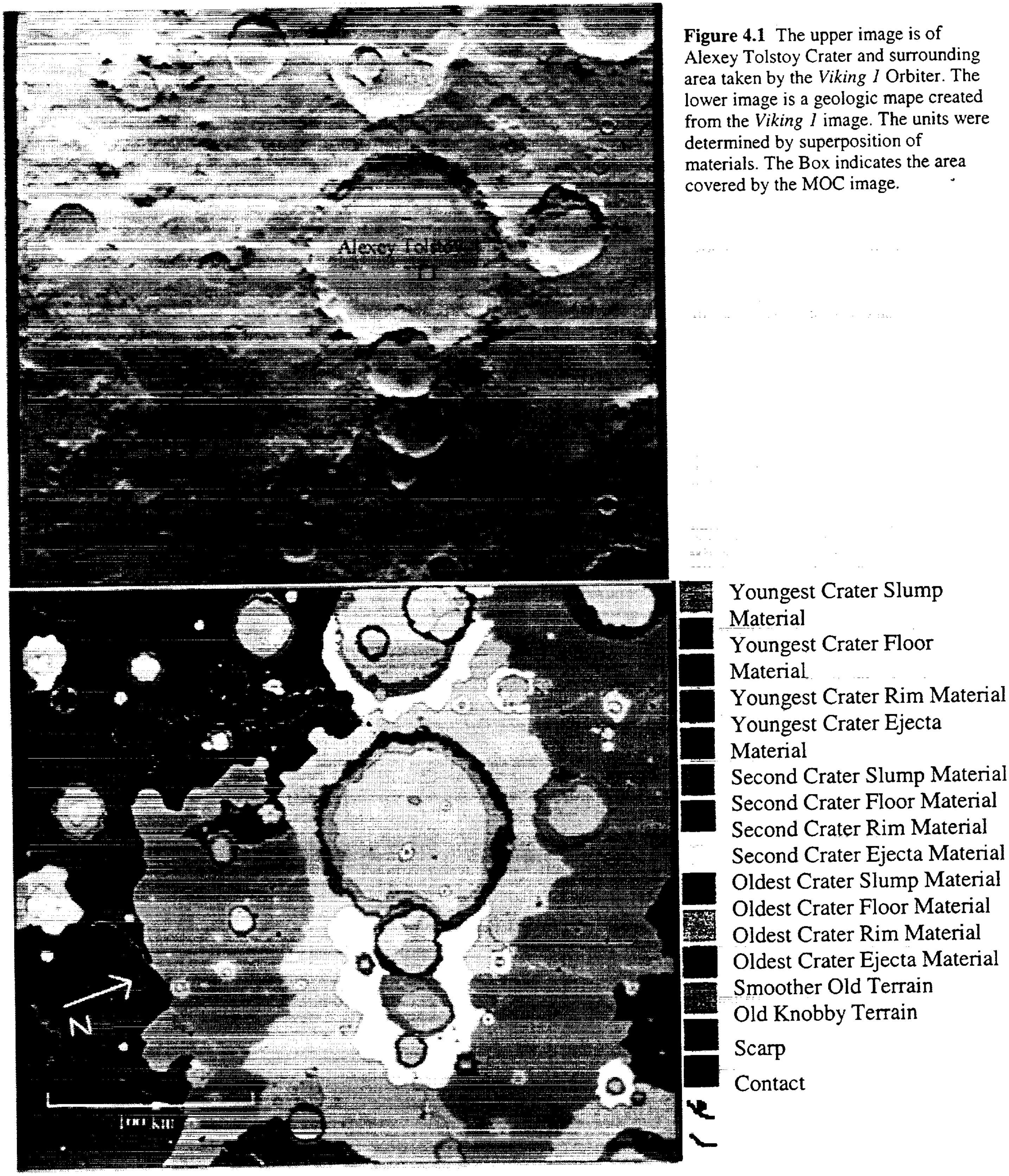

Figure 4.1 The upper image is of Alexey Tolstoy Crater and surrounding area taken by the *Viking 1* Orbiter. The lower image is a geologic mape created from the *Viking 1* image. The units were determined by superposition of materials. The Box indicates the area covered by the MOC image.

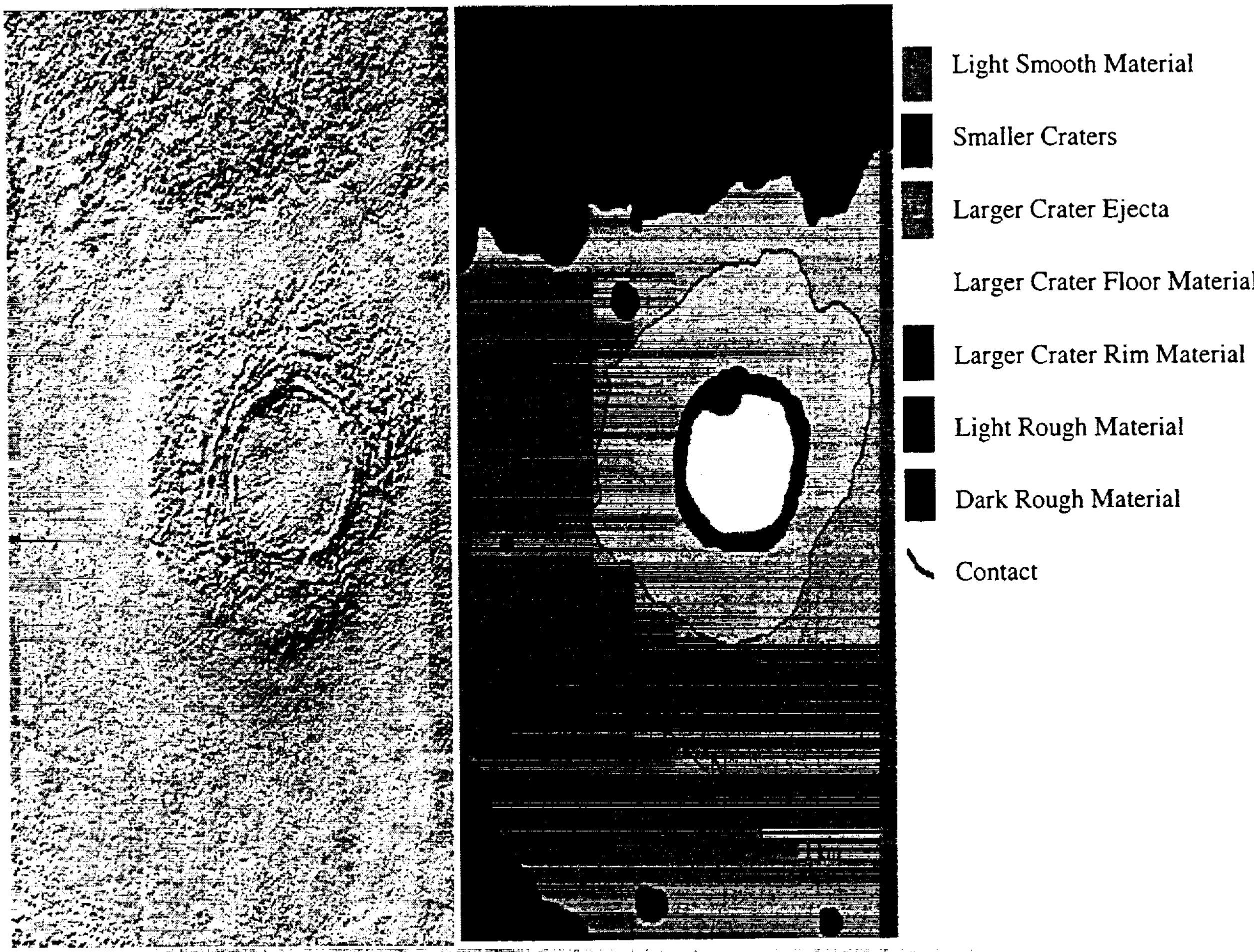

Figure 4.2 The image on the left is of the proposed landing area within Alexey Tolstoy crater taken by MOC. The image on the right is a geologic map created from the MOC image. The units were determined by superposition of materials as well as textural and albedo differences.

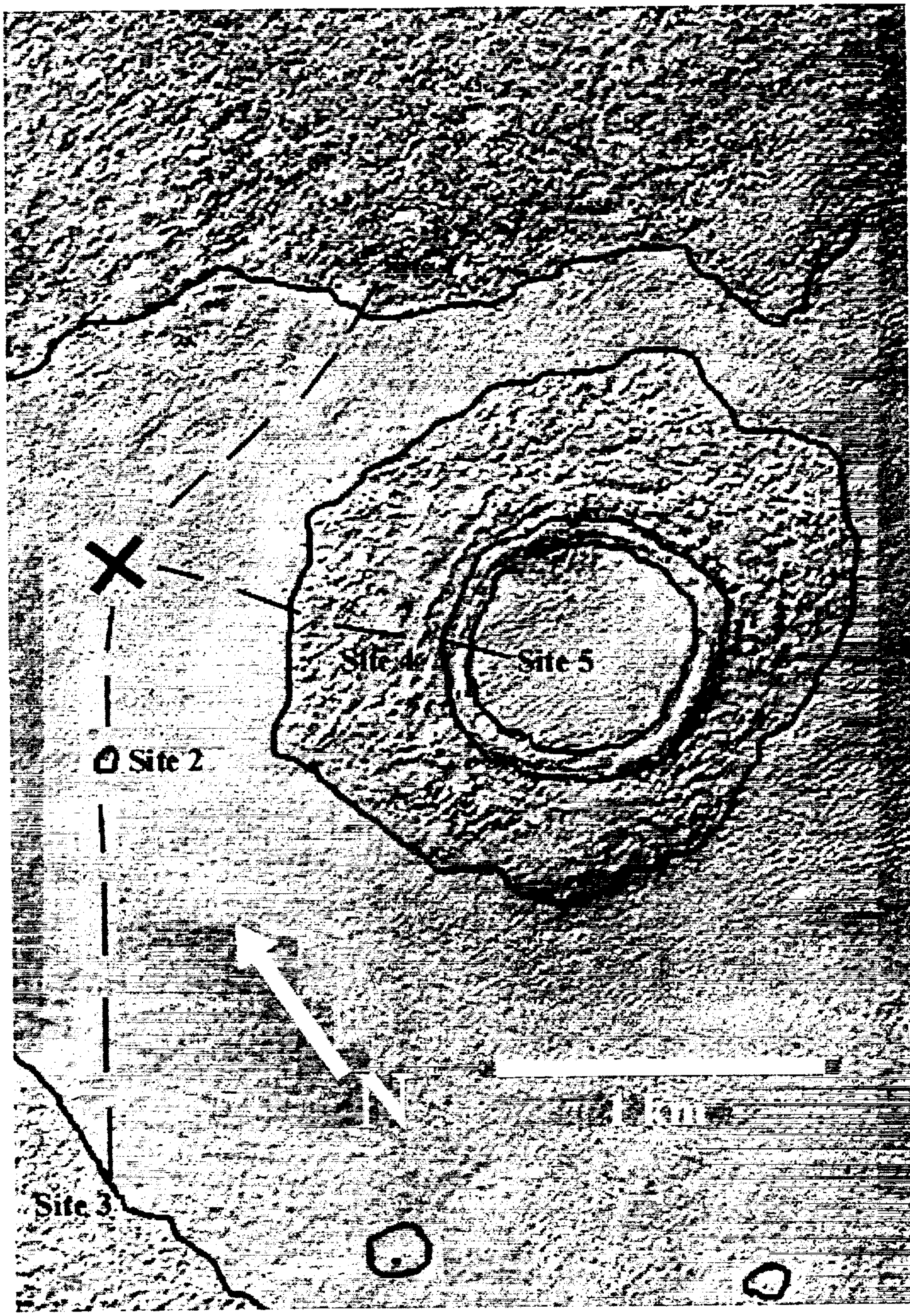

Figure 4.3 Planned traverses of the proposed landing site within Alexey Tolstoy Crater. The sites labeled are locations where samples of the materials and measurements will be taken to examine the local sedimentary and aeolian units of the area.

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A New Plan for Sending Humans to Mars: The Mars Society Mission

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Abstract

Optimal cost and safety will be instrumental not only in sending humans to Mars, but also in achieving the political support and scientific consensus that will allow such an endeavor to begin. The Mars Society Mission (MSM) was created to improve upon the safety, cost, and political viability of previous plans, with emphasis on the NASA Reference Mission 3.0 (RM 3.0). The Mars Society Mission is a complete description of a possible 5-human expedition to the Red Planet targeted at the 2011 (cargo) and 2014 (crew) launch opportunities. All components are capable of performing in any succeeding launch windows. The Mars Society Mission features:

1. **Increased redundancy of design for reduced development**. For instance, the Mars Society Mission's Mars Ascent Vehicle (MAV) and Earth Return Vehicle (ERV) derive from a common Crew Return Vehicle (CRV, distinct from that planned for the International Space Station).
2. **Increased redundancy for maximum safety**. A CRV will accompany the outbound habitat module, and in the event of habitat failure would be able to support the crew until arrival on Mars or Earth. After a 612-day surface stay, both the MAV and ERV will accompany the crew during the return to Earth. If either ERV or MAV fails, or Mars orbital rendezvous does not take place, either component could return the crew.
3. **The Qahira Interplanetary Transportation System** (QITS, pronounced "Keats"). QITS is based on the Qahira launch vehicle, a Delta-IV inspired heavy lifter with only two new components, the Qahira Booster Core (QBC) with 4 RS-68 engines, and the Qahira Upper Stage (QUS), with 1 RD0120 engine. The maximum configuration, the Q3041, is capable of sending 55 MT trans-Mars.
4. **Detailed and improved trajectories**, including a 3/2 Hohmann transfer orbit for the ERV that minimizes propellant boil-off and reduces launch facility strain, as well as optimal trajectories for cargo, free-return, and return from Mars surface.
5. **"Piggyback" payload capabilities** to reduce launch costs and encourage additional planetary science missions.
6. **Minimal assembly in Earth orbit**, specifically no more than one Earth orbit rendezvous.
7. **No nuclear thermal rocketry**, and no activation of nuclear power sources until Mars surface.
8. **A large science payload**, with 13.7 MT available for the 2014 mission.

The 2014 Mars Society Mission will consist of five launches, four of which will use the Q3041 configuration: Payload (A), ERV in June 2011; (B) Cargo including power, hydrogen and science in October 2011; (C) MAV and In-Situ Resource Utilization in November, 2011; and (D) Habitat module in January 2014. A Q1310 configuration will launch Payload (E), Crew in CRV, also in January, 2014. Extensive computer programming and simulation were used to design launch vehicles and trajectories. Comparative risk analysis indicates that the Mars Society Mission has significantly less risk of failure than the Reference Mission 3.0 or Mars Direct.

1. Motivation For Design

Among the requirements listed by Goldin for the commencement of a human Mars mission is "an affordable mission scenario that can be accomplished in about one decade."[1] Recent adoption of in-situ resource utilization (ISRU) and other cost-lowering technologies have come close to achieving this, but a politically and scientifically viable mission with unified support has yet to be realized. The Mars Society Mission resolves this problem by addressing the safety and scientific shortcomings of the Reference Mission 3.0 and Mars Direct.

2. Approach to Mission Design

The Mars Society Mission was designed with the final goal of a complete and workable human Mars mission, with the constraints of: (1) technological simplicity, with few new technologies required; maximization of (2) crew survival options and (3) science return; and minimization of (4) politically sensitive technologies without significant increase in cost. The team achieved this goal through computer simulation, spreadsheet, and in-depth final system selection.

2.1. Computer simulation

Three computer simulations were designed using the C programming language to address trajectory, launch capability, and aerocapture.

2.1.1. Trajectory Program

The trajectory program analyzed mean Keplerian orbital elements of Earth and Mars and assumed a heliocentric conic section transfer orbit. Within these approximations, trajectories were calculated exactly. The program was validated by comparison to previous interplanetary probes.[2,3] Table 2.1.1. displays this validation, using C_3 as the benchmark trajectory feature. After establishing this small absolute error in C_3 for recent Mars trajectories, the program was deemed valid for calculations used in designing the Mars Society Mission.

Table 2.1.1. Validation of Trajectory Program

Probe	Predicted C_3 (km^2/s^2)	Actual C_3 (km^2/s^2)	Absolute Error
Mars Global Surveyor	9.9846	10.0194	0.0348
Mars Climate Orbiter	10.93	11.19	0.26

2.1.2. Launch Vehicle Program

The launch vehicle program assumed a gravity turn trajectory, thrust, and a simple model for air drag. Within these approximations, the payload capacity to low-Earth orbit (LEO) was calculated exactly. The Space Shuttle was used as a test case for the launch vehicle program, which predicted a payload capacity of 28.442 MT to LEO, as opposed to an actual 29.5 MT,[4] an error of 3.59%. Given that the error is expected to be greatest for vehicles where the payload is a small fraction of the mass at burnout (such as the Shuttle, and not the Qahira launch vehicles proposed in Section 3.2.), this program was considered valid for use in designing the mission architecture.

2.1.3. Aerocapture Program

The Mars aerocapture phase of the mission was investigated using a computer model of the habitat module with aeroshell entering the Mars atmosphere at speeds of 8 to 13 km/s. Although it could not produce data on the thermal and aerodynamic issues, as it simply assumed lift and drag coefficients, it was useful in determining the deceleration loads on the crew and lift-to-drag ratio required for a successful aerocapture.

2.2. Spreadsheets

Microsoft Excel was used to create spreadsheets to calculate mass and power budgets, ISRU requirements, health effects, risk, and trans-Earth injection. The trans-Earth injection (TEI) spreadsheet was the most complicated; it accepted data from the trajectory program and returned the parameters of TEI such as the required change in velocity (ΔV) and direction of burns.

3. How the Mars Society Mission Will Send the First Humans to Mars

A step-by-step description of the MSM is provided below. Further details on each system and mission phase are given in the appropriate sub-section.

2011: On July 1, 2011 a Q3041 Launch Vehicle, part of the Qahira Interplanetary Transportation System, will send an Earth Return Vehicle on a 3/2-orbit, low energy trans-Mars trajectory. The Earth Return Vehicle consists of a Crew Return Vehicle and a methane-oxygen rocket stage. A second Q3041 launch on October 27, 2011 sends to Mars a cargo vehicle with surface mobility containing liquid hydrogen feedstock, a 160-kWe nuclear reactor, and science equipment. A third Q3041 launches the Mars Ascent Vehicle (MAV) on November 11, 2011. The MAV consists of a CRV and methane-oxygen rocket stage, identical to that on the ERV, with attached first stage and ISRU unit.

2012: On August 24, 2012, the cargo payload aerobrakes into Mars orbit, then aerobrakes again to reach the Martian surface once satisfactory conditions have been ascertained. After a quick preliminary scout, it will deploy the nuclear reactor, and place a radio beacon at a site appropriate for the landing of the MAV, which will join the landed cargo on September 7, 2012. The ISRU unit aboard the MAV will activate and be connected to the nuclear reactor and hydrogen feedstock, and begin making necessary methane, oxygen, and life support reserves.

2013: The Earth Return Vehicle aerobrakes into a Mars orbit of slightly less than $C_3=0$ and of period 3 sols on July 15, 2013. On November 20, 2013, a second MAV intended for the 2016 mission and including a large science payload departs Earth on a Q3041, followed by a Q3041 cargo launch containing liquid hydrogen for the 2016 mission and the pressurized rover to be used by both the 2014 and 2016 missions.

2014: By early January, the ISRU unit's propellant manufacture is complete and it detaches to be used for other MAV's. Back on Earth, verification that the ISRU has succeeded is followed by the launch to low-Earth orbit of an unmanned habitat module aboard a Q3041. After a final check of the MAV's fitness for their return, the crew launches in a Crew Return Vehicle aboard a Q1310 equipped with a Launch Escape System (LES) capable of accelerating crew and CRV to safety in the event of launch failure. After rendezvous in low-Earth orbit, on January 11 the upper stages of first the Q1310 and then the Q3041 push the crew, CRV, and hab module into a 134 day trajectory trans-Mars. After trans-Mars injection, the CRV separates and from a slight distance accompanies the hab, which deploys a 125-meter truss for artificial gravity at the other end of which lies the burnt out QUS, and at the center of which lie the solar panels that will provide 30 kWe power for the crew until they reach Mars. On May 25, the hab and CRV reach Mars. The CRV will be reused and returns to Earth on a free return trajectory, while the habitat with the crew lands on the Martian surface after aerobraking, several orbits of Mars, and finally descent to the surface. On July 4, the 2016 Mission cargo payload with pressurized rover and hydrogen arrives. The 2016 mission MAV lands on September 15.

2016: In January 2016, the 2014 Mission crew board the MAV, which launches them to rendezvous with the ERV parked in Mars orbit. Then, on January 27, firing first the ERV's and then the MAV's engines, both ERV and MAV accompany the crew back to Earth. On December 25, 2015, the CRV that escorted the 2014 Mission returns on its free return trajectory from Mars and aerobrakes into a C_3=0 parking orbit, where it awaits the launch of the 2016 Mission crew in a stripped down ("ghost") CRV aboard a Q1010. The 2016 Mission crew and hab launch aboard a Q1010 and Q3041, respectively. The crew transfers to the hab after docking, after which the ghost CRV's QUS propels hab and ghost CRV partway to C_3=0 for the rendezvous with the free-returned, fully functional 2014 CRV. After the ghost CRV's QUS burnout, the ghost CRV separates and later falls back to Earth, while the hab's QUS completes the rendezvous with the fully functional 2014 CRV. After docking with the 2014 CRV, the hab's QUS sends the crew, 2014 CRV, and hab module on their way to Mars. On June 4, the first crew returns to Earth, descending to the surface in either their ERV or MAV, while the other component aerobrakes into a C_3=0 orbit, there to await its next task, accompanying the crew of the 2018 mission.

The mass scrub of each of the payloads used for the 2014 mission is shown below.

Table 3.A. Habitat Mass Budget

	Mars Direct	Reference Mission	MSM	Explanation for MSM Figures
Habitat Module structure	5.000	5.500	4.751	Reference Mission figure linearly scaled by surface area ratio (.7511) + margin of 15%.
Life-support system	3.000	4.661	3.796	Figure of current NASA model for crew of six.
Consumables	7.000		3.236	Using 98% closed H_2O/O_2 + Food = .000630 MT per person per day. 900 days total.
Descent Power (fuel cell + radiator)	1.000	2.974	1.292	Adapted from Reference Mission 1.0, p. 3-96.
Reaction control system	0.500		0.500	From Mars Direct—not included in RM.
Comm/info	0.200	0.320	0.320	Exact figure from RM.
Science	1.000			On cargo landers – see Section 3.8.2.
Crew	0.400	0.500	0.417	Crew of 184 lbs each.
EVA suits (4 in Mars Direct, 6 in RM and Mars Society Mission)	0.400	0.969	0.969	Reference Mission figure for six suits, thus including a spare.
Furniture and interior	1.000		1.500	Arbitrary.
Open rovers (2 in MD, 1 in MSM)	0.800	0.500		Mass budgeted with surface power.
Pressurized Rover	1.400			Not included in hab payload.
Hydrogen & LSS ISRU			0.406	(.211 MT H_2 Requirement) x (1.468 Tank) x (1.15 for Boiloff) +.05 MT for micro ISRU unit. To provide additional LSS on Mars surface.
Spares and margin (16%)	3.500	0.000		Included in individual listings.
Health care			1.250	Arbitrary.
Thermal		0.550	0.475	Reference Mission figure linearly scaled by surface area ratio (.7511) + margin of 15%.
Crew Accommodations		11.504		Included elsewhere in habitat mass budget.
Surface Power (RM uses RFCs + "keep alive" solar)		1.700	5.000	At least 25 kWe needed. Reference Mission specifies 5.7 MT for 50 kWe nuclear reactor.
EVA Consumables		2.300		Produced by ISRU on MAV and Hab.
Power Distribution		0.275	0.316	Reference Mission figure scaled up by 15%.
Total Landed	25.200	31.753	24.228	Total of above.
Terminal Propulsion + Propellant			5.330	See section 3.7.4.
Parachutes		.7	0.525	¾ RM's 4 parachutes needed.
Orbital Power (solar)			1.682	Assumes rigid panels from DRMv1.
Aeroshell Structure & TPS			9.530	30% of the above mass
Artificial Gravity Truss (125 m)			1.381	See Section 3.7.2.
Transit Power (solar)			1.682	Assumes rigid panels from DRMv1.
Reaction control propellant	Above		1.666	Calculated in section 3.7.2.2.1.
Total Injected			**46.024**	

Table 3.B. 2014 Cargo Lander Mass Budget

Component	Mass (MT)	Explanation
Nuclear reactor	9.300	Reference Mission 3.0. Uses lander mobility for deployment.
Hydrogen	11.760	Stoichiometry.
Tank	4.704	40% of liquid hydrogen mass.
Power Line from Reactor	0.837	Reference Mission 3.0 Mass Scrub
Science & Exploration	4.692	Based on remaining launch to Mars surface capability.
Fuel cell	0.347	5 kWe power (Reference Mission 1.0.)
Cargo lander mobility	5.544	Assumed 15% of total landed mass.
Descent Propulsion	0.612	Four RL-10M engines.
Descent Propellant	7.170	For 632 m/s Delta V.
Propellant Tanks	0.645	9% of Propellant.
Parachutes	0.700	Reference Mission 3.0
Aeroshell	8.185	18% of Payload.
Transit Power 5 kWe solar	0.480	Reference Mission 1.0
Interplanetary RCS	0.800	Provides 45 m/s Delta V.
Total	54.940	Sum of above.

Table 3.C. Total MAV Payload Mass Budget

Component	Mass/MT	Explanation
MAV	15.042	Table 3.5.1. figure minus mass of crew and Mars rocks
ISRU	9.010	See Table 3.5.2.2.B.
1st Stage	12.380	9% of propellant mass + 14 RL-10M engines.
2nd Stage	2.377	9% of propellant mass + 2 RL-10M engines.
Fuel Cell	0.347	Reference Mission v. 1.0 figure[5]
Landing Propellant	3.670	Enough to provide ΔV 324 m/s (see Section 3.4).
Parachutes	0.700	Reference Mission v. 3.0 figure
Aeroshell	8.372	18% of payload
Interplanetary RCS	0.800	Provides 45 m/s ΔV
Transit Solar Power	0.480	Reference Mission v. 1.0 figure
Total	53.178	Sum of above figures

Table 3.D. Overall ERV Payload Mass Budget

Component	Mass/MT	Explanation
CRV	15.459	See Table 3.5.1.
TEI stage structure	2.377	9% of propellant mass + 2 RL-10M engines
TEI stage propellant	23.025	Propellant needed to return crew to Earth
Power supply	0.827	NASA Reference Mission 1.0
Aerobrake	7.504	18% of Payload mass
Interplanetary RCS	0.800	Provides 45 m/s ΔV.
Total	49.992	Sum of above figures

3.1. Crew Size

A crew of five was determined for the Mars Society Mission based on the minimum of four for adequate science return and system maintenance advocated by Mars Direct,[6] with the addition of a crew member for medical duties as advocated by the Reference Mission.[7] Instead of sending two medical crew members as in the Reference Mission, however, at least one of the science crew will be able to supply medical treatment in the event of the medical officer's illness or injury. Five crew members were thus determined to be sufficient for science return, maintenance of systems, and medical upkeep of the crew, a view supported as plausible by Connolly.[8] The rapid accumulation of habitats and other infrastructure at a single point on the Martian surface and the availability of additional CRV's for later missions (see Section 3.10.3.3.) could allow for greater crew size on succeeding missions.

3.2. Qahira Interplanetary Transportation System

The Qahira Interplanetary Transportation System (QITS) will send all crew and cargo to Mars, using two liquid O_2/H_2 components in two main configurations. The smaller configuration, used for the 2014 Crew/CRV launch, will feature three Castor-120 solid rocket boosters attached to the first stage.

3.2.1. Components

The three components of QITS are the Qahira Booster Core (QBC), the Qahira Upper Stage (QUS), and the currently available Castor-120 solid rocket.

Table 3.2.1. Performance Description of QITS Components

	Wet Mass (MT)	Dry Mass (MT)	Thrust, vac. (Mlbs)	Thrust, Sea level (Mlbs)	I_{sp}, vac. (s)	I_{sp}, Sea level (s)	Burn time (s)
RS-68[9]		6.618	.745	.650	410	365	
QBC	560	65.9	2.98	2.60	410	365	151
RD0120[10]		3.44	.44		455		
QUS	205	18.0	.44		455		425
Castor 120[11]	54.1	5.39	.363	.323*	277.9	247*	82.5

* Estimated from vacuum values

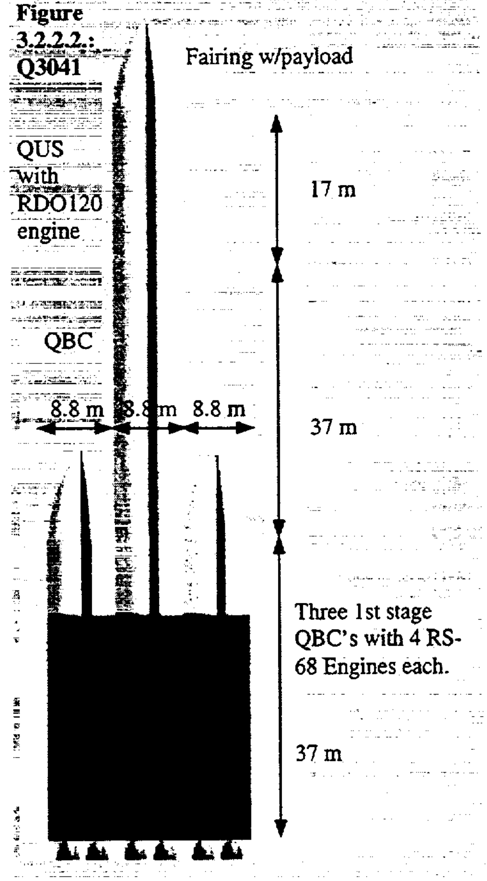
3.2.1.1. Qahira Booster Core

The Qahira Booster Core is powered by four LOX/Hydrogen RS-68 engines. The RS-68 (mixture ratio 6.0) engines were chosen because (1) they are simple; (2) they will have been used and presumably validated many times prior to 2011; and (3) their specifications match those needed for the launch vehicle (moderate to high I_{sp} and high thrust). The QBC has a diameter of 8.8 m and a length of 37 m to provide enough volume for propellant.

3.2.1.2. Qahira Upper Stage

The Qahira Upper Stage serves as a second or third stage for QITS payloads, depending on the configuration. It uses a single RD0120 (mixture ratio 6.0) engine fueled by LOX/Hydrogen. The RDO120 was selected for its high specific impulse and moderately high thrust. The QUS has a diameter of 8.8 m (to match the QBC) and a length of 17 m.

Figure 3.2.2.2: Q3041



3.2.1.3. Castor-120

The 2014 Crew/CRV launch will use three Castor-120 solid rockets. These rockets were selected because (1) they are inexpensive, (2) simple, and (3) they provide the necessary extra acceleration to allow the QUS of the 2014 Crew/CRV launch to arrive in LEO with sufficient propellant to push the hab and CRV through trans-Mars injection.

3.2.2. Configurations

An advantage of QITS is the ability to mix and match its components to best accommodate a given payload. The two configurations used for the first Mars Society Mission are the Q1310 and Q3041, with the Q1010 and Q3041 used on succeeding missions.

3.2.2.1. Terminology

The terminology for the various QITS configurations is as follows: in the four-digit code, the 1st digit represents the number of QBCs on the 1st stage; the 2nd digit represents the number of solid boosters; the 3rd digit represents the number of engines on the 2nd stage; and the 4th digit represents the number of engines on the 3rd stage. For example, the Q3041 is composed of a 3-QBC 1st stage, 0 solid boosters, a 4-engine 2nd stage, and a 1-engine 3rd stage.

3.2.2.2. Q3041 Configuration

This configuration consists of a three QBC 1st stage, a single QBC 2nd stage, and a single QUS as the 3rd stage. This configuration will be used for all launches except the CRV/Crew payload, because it provides maximum lift capability to Mars, 55 metric tons on a cargo trajectory. The total mass at take-off (including payload) is 2500 MT. Figure 3.2.2.2 depicts the Q3041.

3.2.2.3. Q1310 Configuration

This configuration consists of a single QBC 1st stage and a QUS 2nd stage, with three already-existing Castor-120 solid rocket boosters attached to the 1st stage. Its purpose is to launch payloads which are heavier than what current launch vehicles such as the Proton can handle, but are so light that to place them on a Q3041 would be wasteful. In the MSM, the only use of the Q1310 is to lift the 2014 CRV/Crew payload. A solid-propelled Launch Escape System (LES) similar to that used for the Apollo and Soyuz programs is provided through first stage burnout so that the crew can be recovered in the event of a launch failure. The total mass at take-off is 946 MT.

3.2.2.4. Q1010 Configuration

The Q1010 configuration is simply the Q1310 without the three solid-fueled booster rockets. It is used for the 2016 and succeeding missions.

3.2.3. "Piggyback" payloads

The ERV launch leaves 5 MT of payload capacity unused. Rather than wasting this capacity, the Mars Society Mission uses this space for scientific "piggyback" payloads. These secondary payloads would be released after the upper stage of the Q3041 had burned out. These could include: unmanned spacecraft destined for Mars or its moons (possibly communications or navigation satellites); other planetary science spacecraft, which would be propelled to their destinations by solar-electric propulsion or Mars gravity assist; astronomy satellites that must operate away from Earth, a major infrared source; or space physics satellites to investigate the interplanetary medium. Additional piggyback payloads could be budgeted from within the large science and exploration allotments of the MSM's cargo launches.

3.2.4. Payload packaging

Payload packaging of the MSM differs from the Reference Mission in the orientation of the habitat module. While for the ERV, cargo, and MAV/ISRU launches it is best to have the 9.2-m diameter aeroshell double as a launch shroud as in the NASA Reference Mission, for the human mission this packaging configuration presented difficulties. It prevented the docking port on top of the hab module from easily connecting to the CRV in low-Earth orbit: the hab's aeroshell would be in the way. Therefore, the habitat will launch with the forward part of the aeroshell pointed down into the QUS, with the hab module right-side up and its docking port on top, protected by a light payload fairing. This packaging of the hab module also avoids a sudden reversal of heavy deceleration during Mars entry, as occurs when the Reference Mission hab rotates 180° to fire its retro rockets. Using the MSM launch configuration, the hab module's retro rockets are already properly oriented at this stage of Mars entry. The CRV that will accompany the 2014 Mission outbound crew will launch Apollo-style on the Q1310 with the Launch Escape System on top.

3.2.5. Comparisons with Magnum, NTR, and other Launch Options

The MSM uses the Q3041 heavy lift vehicle, and its smaller siblings, the Q1310 and Q1010, with cryogenic propellants for Trans Mars Injection (TMI), but the decision for this type of launch system came only after consideration of several other launch options. These alternatives for Earth-to-Orbit transportation included a small launch vehicle such as the Proton (20 MT to LEO);[12] a medium launch vehicle in the 40 metric ton range; a Magnum-sized vehicle capable of lifting 80 MT to LEO;[13] and the Q3041 or Saturn-class rocket (151 MT to LEO). The cryogenic propellants for TMI were compared with storable (hydrazine/N_2O_4), nuclear thermal, nuclear electric, and solar electric alternatives.

The use of cryogenic (LOX/hydrogen) propulsion for TMI was decided early in the MSM design process. Nuclear-thermal rocketry (NTR) was rejected for three reasons: (1) NTR presents severe political difficulties; (2) A massive tank is needed to hold the all-hydrogen propellant; and (3) The necessity of reaching a nuclear-safe orbit (at least 407[14]-800 km[15]) before using any NTR stage nearly offset the advantage of its higher specific impulse. Combined with the development costs of the new technology, this argues against nuclear propulsion as an option for human Mars missions possible within a decade.

Ion propulsion was investigated as well, but it was dropped for a multitude of reasons. First, while the exhaust velocities given for ion engines are 5-10 times better than the best cryogenically fueled rockets,[16] ion propulsion requires that the spacecraft slowly spiral away from Earth, then drop back for a chemical rocket stage to provide TMI. The alternative—hundreds or thousands of perigee burns with the ion rocket—would take far too long, and the ion engine would have to provide about 8 km/s of ΔV compared to 3.2 km/s for a LOX/Hydrogen rocket. For solar electric propulsion, the necessary arrays of solar cells would make the reduction in the total mass sent to LEO modest, at best. Finally, for both nuclear and solar electric propulsion, the ion propellant xenon is rare in Earth's atmosphere and therefore costs as much to produce as it does to launch into orbit. The cost savings from ion propulsion may therefore be outweighed by the manufacturing cost of the noble gas.

Finally, storable propellants were found to have too low a specific impulse. Table 3.2.5 summarizes these conclusions; the numbers in each cell of the table indicate the reasons why that possibility was rejected.

Table 3.2.5. Launch Vehicle Rationale

		Launch Vehicle with payload to LEO			
		20 MT Proton	41 MT Q1010	80 MT Magnum	151 MT Q3041
Propulsion Method	Storable (N_2O_4/hydrazine)	7	2	2, 4	2
	Cryogenic (LOX/Hydrogen)	5, 7	6	4	√
	Nuclear Thermal	1, 7	1	1, 4	1
	Ion	3, 7	3	3, 4	3
1. Political difficulties, large tank requirement, reduction in usefulness due to nuclear-safe orbit requirement.					
2. Storable propellants too low in I_{sp} to use unless necessary to prevent boiloff during LEO assembly.					
3. Cost of xenon, offset of I_{sp} by high-ΔV trajectory, requirement of large solar cells for solar electric propulsion.					
4. Liquid Fly-Back Boosters have possible cost, technical issues, and still require QBC and QUS type components.					
5. Boil off of cryogens due to long duration of LEO assembly.					
6. Same development costs for QUS/QBC components as 151 MT vehicle; inefficient to use only smaller configuration.					
7. Long term costs, efficiency, and launch facility strain.					

The launch system that avoids these arguments is the Qahira Interplanetary Transportation System based on the Q3041 with the Q1310 and Q1010 as byproducts, and therefore this system was selected for the Mars Society Mission.

3.2.6. Launch Facilities

The size and placement of QITS' engine and stage components will require either modification of an existing launch pad or construction of a new launch pad. A 90-m high tower was designed for QITS. This tower is high enough to refuel the Q3041's QUS; it is not necessary to have a walkway for crew to Q3041's payload, as only the Q1310 and Q1010 will carry crew, for which a walkway is provided.

3.2.7 Timeline of Earth Ascent

The description of launch events from T+0:00.0 to TMI for the MSM launches is provided in the following sections.

3.2.7.1 Q3041 Earth Ascent Timeline for ERV, MAV/ISRU, and Cargo Launches

Table 3.2.7.1 describes the path of the ERV, MAV/ISRU, and Cargo launches into LEO.

Table 3.2.7.1. Cargo Trajectory-Type Ascent of Q3041

Time	X/km	Y/km	Velocity (km/s)	∠-Vert. (°)	Event
T+0:00.0	0	0	0	0	Liftoff @ 1.4 g
T+0:06.5	0	.09	.028	0	Clear tower
T+0:52.0	1.8	7.1	.331	25	Speed of Sound/Mach 1
T+2:31.0	94	67	2.33	68	Stage 1 separation @ 3.9 g
T+5:02.0	646	163	5.74	86	Stage 2 separation @ 4.1 g
T+8:09.0	1860	199	7.39	90	Stage 3 shut down @ 1.1 g/ enter LEO

The TMI burn is calculated to C_3=15 km^2/s^2, to reach up to 1.77 AU aphelion in the plane of Earth's orbit and sufficient to reach Mars with a wide launch window in any opportunity. TMI for the 2014 Mission launches of 2011 lasts 3 min 57 s for a change in velocity ΔV = 3.91 km/s. Total mass injected is 55 MT.

3.2.7.2. Q3041 Earth Ascent Timeline for Habitat Launch

The habitat launch to LEO differs from the other Q3041 launches because it carries 47 MT of payload for TMI (which will be supplemented by 16 MT carried by the sister Q1310 launch). A dual launch strategy using first the Q3041 to boost the habitat into LEO and then the Q1310 to boost its CRV with the crew was chosen because the Q3041 cannot send the habitat on a fast free-return trajectory to Mars. Upon the Stage 3 QUS shutdown, the Hab gets ready to dock with the CRV containing the crew. Orbit circularization takes place at 360 km for a duration of 6.8 seconds, for a change in velocity ΔV = 0.08 km/s. The 170 MT vehicle is now ready to rendezvous with the crewed CRV. Its mass breaks down into the 47 MT habitat payload and the 18 MT dry QUS with 105 MT propellant. Propellant boiloff is assumed to be kept to 2%, leaving 103 MT.

3.2.7.3. Q1310 Earth Ascent Timeline for 2014 CRV with Crew and LES Launch

Table 3.2.7.3. describes the path of the CRV/crew launch aboard a Q1310 to LEO, for rendezvous with the hab module at 360 km.

Table 3.2.7.3. Ascent of Q1310 with CRV and Crew

Time	X/km	Y/km	Velocity (km/s)	∠-Vert. (°)	Event
T+0:00	0	0	0	0	Liftoff
T+0:05	0	.09	.036	0	Clear tower
T+0:36	1.5	5.3	.331	24	Speed of Sound/Mach 1
T+1:22	21	29	1.12	48	Jettison 3 Castor-120's
T+2:31	133	98	3.00	74	Stage 1 separation @ 4.6 g, jettison LES
T+8:05	1650	296	7.34	89.5	2nd stage shut down @ 2.7 g/ enter transfer

After the initial QUS shutdown, orbit circularization occurs at 360 km and lasts 3.0 seconds, for a change in velocity ΔV = 0.08 km/s. The 73 MT vehicle is now ready to rendezvous with the habitat. Its mass breaks down into the 16 MT CRV and the 18 MT dry QUS with 39 MT propellant.

3.2.7.4. Trans-Mars Injection of 2014 Habitat with Crew and CRV

The smaller vehicle—that is, that launched by the Q1310 consisting of CRV, crew, and QUS—carries out up to 130 m/s of rendezvous maneuvers using 2 MT propellant and leaving 37 MT. The docking occurs in an orbit of 360 km at velocity 7.69 km/s with period 1 hr 32 min. The fully assembled TMI vehicle then has a mass 239 MT, of which 140 MT is propellant.

Of the two QUS stages at each end of the vehicle, with the hab and CRV sandwiched between, the Q1310-launched QUS fires first. It fires for 1 min 24 s, providing 0.75 km/s ΔV and raising the orbit to 360x3830 km with a period of 2 hr 10 min. This QUS is then released. The Q3041-launched QUS fires second, for 3 min 54 s, providing 3.66 km/s Delta V and raising the orbit to C_3=28.1 km^2/s^2.

Rationale for dual launch strategy can be found in Section 3.7.1.

3.3. Trajectories

Exact calculation of trajectories to and from Mars was necessary to determine exact payload capabilities and the time the crew will spend in interplanetary space. Results for outbound flights for all vehicles arriving before the end of 2014 Mission are shown in Table 3.3 in order of arrival.

Table 3.3. Trajectories trans-Mars

		Launch Date	Arrival Date	Transit time/days	C_3 (km^2/s^2)	Perihelion (AU)	Aphelion (AU)	Inclination (degrees)
2014 Mission	Cargo	10/27/2011	8/24/2012	302	10.2	0.98	1.52	0.9
	MAV	11/11/2011	9/7/2012	301	9.0	0.99	1.52	1.7
	ERV	7/1/2011	7/15/2013	731	10.6	1.01	1.52	1.3
	Crew	1/11/2014	5/25/2014	134	26.0	0.98	2.23	0.3
2016 Mission	Cargo	12/8/2013	7/4/2014	208	13.1	0.98	1.60	1.4
	MAV	11/20/2013	9/15/2014	307	13.7	0.96	1.45	1.9

3.3.1. ERV Trajectory

The ERV trajectory from Earth to Mars after launch aboard a Q3041 was calculated as a 3/2-orbit minimum energy trajectory. This decision was made because (1) the same maximum payload capacity is available in both 3/2 and standard Type I & II transfers; (2) the 3/2 trajectory results in a later arrival date around Mars, which means less exposure to the infrared radiation from the Martian surface that is the leading cause of propellant boiloff[17]; and (3) the earlier ERV launch places fewer constraints on launches scheduled for the Type I & II launch windows, and reduces the chance that additional launch facilities would need to be constructed.

3.3.2. MAV/ISRU and Cargo trajectories

The MAV/ISRU cargo payload payloads arrive at Mars using standard Hohmann transfer orbits to maximize their payload capacities, as detailed in Table 3.3.

3.3.3. Free Return Trajectories for Crewed Flights

Evaluation of the Reference Mission and Mars Direct found little detail concerning free return trajectories planned for crewed flights. Mars Direct specifies the same two-year free return trajectory used by the MSM habitat, but gives a transit time to Mars of 180 days,[18] whereas the actual two-year free return trajectory takes a maximum of 154 days. The importance of free return trajectory data for appropriate determination of payload capacity and crew safety called for extensive scrutiny of free return options for the Mars Society Mission. Of the different free return trajectories available, the fastest—lasting two years after Mars flyby—was selected for its hastening of crew return in the event of an emergency. (Actually, faster returns to Earth are possible on trajectories using a Venus flyby on the outbound transit; however, such trajectories are not available in all launch opportunities.) The free return trajectory chosen for the crew launches on January 11, 2014; encounters Mars on May 25, 2014; and returns to Earth on December 25, 2015. Free return trajectories were also important to the MSM for returning to Earth the CRV that escorts the 2014 habitat on the way to Mars. Unless an emergency occurs requiring the use of the CRV beforehand, the CRV will escort the 2016 outbound habitat and crew, as well.

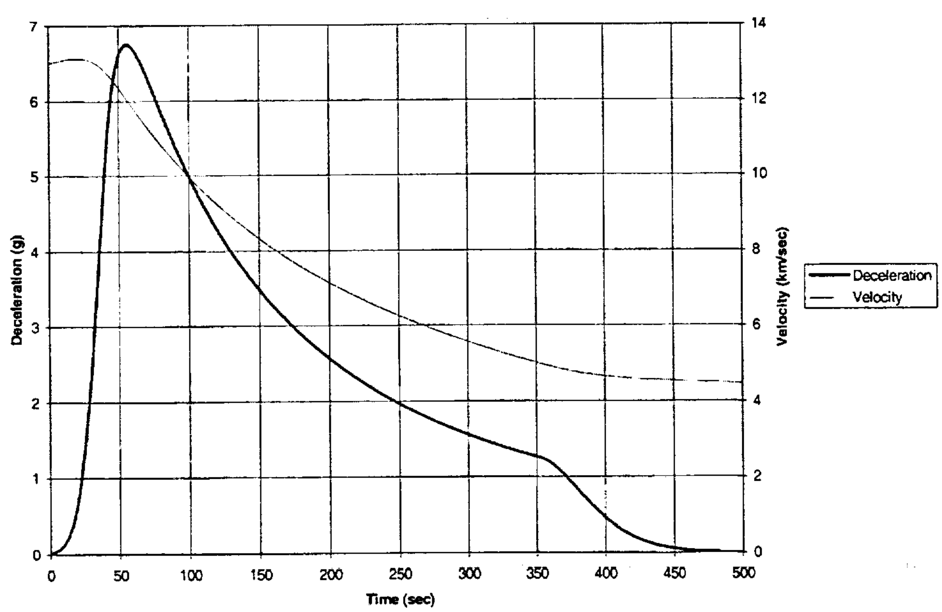
3.3.4. Return trajectories from Mars

The Mars Society Mission's 2014 crew will depart Mars on January 27, 2016. First the ERV's rocket engines fire, putting the crew on a 146-day return trajectory toward Earth with departure C_3=15.7 km^2/s^2. The MAV rocket engine can then fire and provide 0.83 km/sec of ΔV to reduce the transit time to 129 days, returning the crew to Earth on June 4, 2016.

3.4. Aerocapture and Descent

To reduce total mass, aerocapture will be used to insert all vehicles into Mars orbit. The ERV, MAV, and Cargo vehicles use a common biconic aeroshell design that is 13 meters long and 9 meters in diameter at its widest point to capture into Mars orbit. The habitat, being sent on a much faster trajectory and entering the Mars atmosphere at 12.2 km/s, must have a different aeroshell optimized for these higher velocities. Computer simulations of the aerocapture suggested that lift-to-drag ratios of 0.6-0.7 are sufficient for the habitat entry, producing decelerations that peak around 6-7 g. (The deceleration rises quickly at first, going from 1 g to its peak value in ~ 30 seconds. After an additional two minutes, the deceleration has been reduced to 3 g. It is only above 5 g for one minute.)

Figure 3.4. Mars Habitat Aerocapture @ 13 km/sec

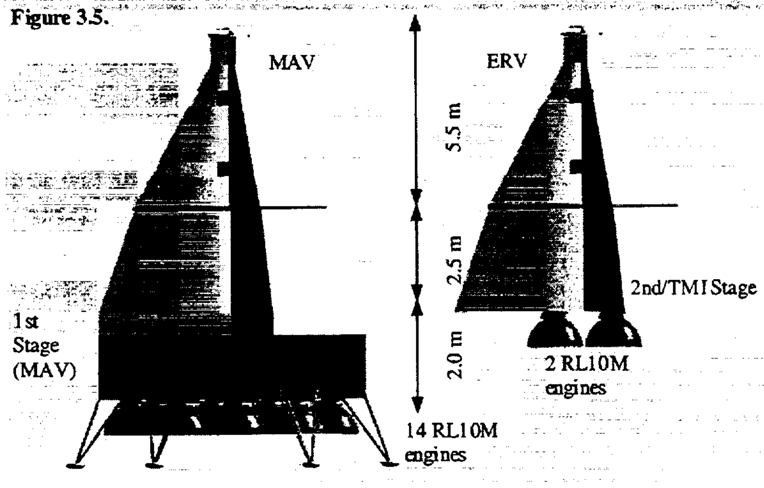


After jettison of the aeroshell, landing is performed using three 50-m diameter parachutes for the hab, four for the MAV and cargo. (The Reference Mission uses four parachutes for every landed payload.[19]) The habitat and Cargo landers burn methane/oxygen bipropellant in four RL-10M engines to provide a final 632 m/s of ΔV and achieve a soft Mars landing,[20] while the MAV lander burns methane and oxygen in its 14 RL-10M engines (see Section 3.5.2.1.). Because of the very high thrust of the MAV lander and its correspondingly greater deceleration, less ΔV is needed on this landing. The 60,000-pound thrust and 40-MT mass figures in Reference Mission 3.0 were used to calculate a terminal descent velocity of 280 m/s with parachutes of the NASA cargo vehicles; for the 240,000-pound thrust used by our MAV lander, we determined that the ΔV needed to land would be only 324 m/s.

3.5. Crew Return Vehicle

The Crew Return Vehicle is the basis for both the Mars Ascent Vehicle and Earth Return Vehicle. Either MAV or ERV can return the crew, the difference being from where: Mars surface for the MAV, low-Mars orbit for the ERV. The generic term "Crew Return Vehicle" is used when the specific designation "ERV" or "MAV" does not matter, such as when discussing the return vehicle's shape, or as during descent to Earth surface, when either MAV or ERV might do the job. Figure 3.5. is a side by side depiction of the CRV's ERV and MAV configurations.

Figure 3.5.



3.5.1. ERV & MAV Mass Budgets

The similarities between the Mars Society Mission ERV and MAV are best illustrated by their mass budgets. Table 3.5.1. itemizes the ERV mass budget of the Reference Mission[21] and the ERV and MAV mass budgets of the Mars Society Mission. The MSM CRV is conical with a base diameter of 6.5 m and a height of 5.5 m. The ratio of the surface area of the Reference Mission ERV to that of the MSM CRV is 0.4295, and this figure was used to linearly scale several mass budget items from the RM to the MSM. Several devices, such as the Reference Mission life-support system, were not scaled down from a RM crew of six to an MSM crew of five, which can be approximated as a 20% margin for the MSM. The "CRV Earth Entry" column describes the mass budget of the CRV immediately prior to re-entering the Earth atmosphere with the crew, the importance of which lay in calculating the aeroshell necessary for returning to Earth. The "CRV Earth Capture" column describes the mass budget of the CRV that will enter a C_3=0 parking orbit immediately after its aerobraking for use in later missions.

Table 3.5.1. ERV/MAV Mass Budgets

	RM ERV	MSM MAV	MSM ERV	CRV Earth Entry	CRV Earth Capture	Comments
Structure	5.500	2.574	2.574	2.574	2.574	Reference Mission 3.0 figure linearly scaled by surface area ratio (.4295) +15%.
Thermal	0.550	0.257	0.257	0.257	0.257	Scaled from RM 3.0 by S.A., +15%.
Power + dist.	3.249	3.249	3.249	1.292	3.249	Directly from RM 3.0; solar arrays are retractable.
Comm/info	0.320	0.320	0.320	0.320	0.320	Directly from RM 3.0.
Spacesuits	0.243	0.300	0.000	0.300	0.000	MSM figure is for 5 pressure suits.
Crew		0.409	0.000	0.409	0.000	.184 MT each for 5 crew members.
Life support system	3.796	3.796	3.796	2.094	3.796	Exact figure of current NASA model for crew of six.
Food	12.058	0.567	0.000	0.000	0.000	.000630 MT per day per person. RM figure has crew accommodations.
Water & oxygen	0.000	0.095	0.378	0.000	0.095	Assuming 2% open loop for ERV, .5% for MAV (150 day return).
RCS	0.000	0.600	0.600	0.600	0.600	Not listed in Reference Mission.
Mars samples	0.000	0.500	0.000	0.500	0.000	Arbitrary.
Health care	0.000	0.000	1.000	0.000	1.000	Arbitrary.
Science Equipment	0.600	0.000	0.000	0.000	0.000	No need for returning science equipment to Earth.
Spares	1.924	0.000	0.000	0.000	0.000	Accounted for in each item.
Furniture & interior	0.000	0.800	0.800	0.800	0.800	Arbitrary
Subtotal	29.105	13.467	12.975	9.146	12.691	Total excluding Earth landing needs.
Earth Landing Parachute	0.000	0.200	0.200	0.200	0.200	
Aeroshell	0.000	2.284	2.284	2.284	2.284	18% of CRV Earth capture mass
Descent capsule	4.829	0.000	0.000	0.000	0.000	In RM, same unit used for Mars Ascent
Total	33.934	15.951	15.459			

3.5.2. MAV

The Mars Ascent Vehicle is designed to either dock with the ERV in Mars orbit and jointly return the crew to Earth, or to independently return the crew. Its primary difference from the ERV is its second stage.

3.5.2.1. Use of 1st Ascent Stage as Descent Retro-Rocket

The RL-10M engines of the MAV 1st stage will be used to slow the MAV to a gentle touchdown on the Martian surface. This was decided on because (1) the mass of the required propellant (3.67 MT) is far less than a distinct retro-rocket system with or without parachutes and (2) this use allows verification of the MAV 1st stage's functionality all the way to the Mars surface.

3.5.2.2. In-Situ Resource Utilization

The in-situ resource utilization chemical plant is carried aboard the MAV payload attached to the underside of the first stage. Data from TEI calculations were used to find the exact amount of propellant needed. Mass and power needs of the ISRU device for creating life support surpluses was carried over exactly from the Reference Mission, producing an additional 20% margin for the MSM due to the MSM's five member crew size. Since life support figures are unchanged, the mass and power needs of these components was not scaled. Total ISRU output is itemized in Table 3.5.2.2.A.

Table 3.5.2.2.A. Total ISRU Quantities

DRM 3.0	Mass (MT)	MSM	Mass (MT)
O_2	30.33	O_2	106.38
CH_4	8.67	CH_4	30.40
Consumables	23.00	Consumables	23.00
Total	62.00	Total	159.78

The ratio of propellant mass required in the MSM to that required in the RM (3.5385) was used to linearly scale mass and power requirements of propellant-related components for ISRU. The results of this scaling (with comparison to the Reference Mission) are itemized in Table 3.5.2.2.B.

Table 3.5.2.2.B.

	Reference Mission v3.0[22]				**Mars Society Mission**			
	Subsystem Mass (MT)		*Subsystem Power (kWe)*		*Subsystem Mass (MT)*		*Subsystem Power (kWe)*	
Mass Elements	Propellants	Life Support	Propellants	Life Support	Propellants	Life Support	Propellants	Life Support
Compressor	0.496	0.193	5.645	2.893	1.755	0.193	19.975	2.893
Sabatier Reactor	0.060	0.050	0	0	0.212	0.050	0	0
Hydrogen Membrane Separator	0.029	0.023	0.288	0.225	0.103	0.023	1.019	0.225
Methane Water Separator	0.394	0.315		1.69	1.394	0.315	0	1.69
Pyrolysis Unit	0.711	1.172	3.397	3.911	2.516	1.172	12.020	3.911
Electrolysis Unit	0.277		18.734		0.980	0	66.290	0
Oxygen Liquefier	0.043		2.215		0.152	0	7.838	0
Methane Liquefier	0.041		2.093		0.145	0	7.406	0
Subtotal	2.051	1.753	32.372	8.719	7.257	1.753	114.547	8.719
Total	3.804		41.091		9.010		123.266	

After completion of the 2014 mission's MAV fuel production and arrival of the 2014 crew, the ISRU detaches and is moved by the 2014 crew to the 2016 MAV, where propellant production begins again.

3.5.2.3. Two-Stage Mars Ascent System

The MAV uses two stages to either reach the ERV or inject itself trans-Earth. The first stage, which is unique to the MAV, is a cylinder 9 m in diameter and 3 m in height. The MAV 1st stage contains fourteen RL-10M engines arranged in a hexagonal configuration. The MAV 2nd stage is a frustum, going from a 9 m diameter base (to match the 1st stage) to a 6.5 m diameter top (to match the MAV) over a height of 2.5 m. It contains two RL-10M engines.

3.5.2.4. Description of Mars Ascent

The MAV reaches the ERV as described in Table 3.5.2.4.

Table 3.5.2.4. Ascent of MAV

Time	X/km	Y/km	Velocity (km/s)	∠-Vert. (°)	Event
T+0:00	0	0	0	0	Liftoff
T+0:03	0	.013	.009	0	Clear 13 m
T+1:14	2.9	7.6	.238	32	Speed of Sound/Mach 1
T+6:05	370	100	2.99	83	Stage 1 shutdown @ 1.7 g
T+13:01	1580	194	2.88	88.3	Stage 1 restart with 1 engine @ .15g
T+17:20	2340	201	3.24	90	Stage 1 separation @ .16 g/ enter LEO

After a coast period of up to two hours to attain proper orientation relative to the ERV, one RL-10M on 2nd stage of the MAV will ignite for 9 min 33 s and reach the ERV.

3.5.2.5. Back-up Options

The two-stage MAV and adaptability of the ERV provide a number of backup options.

3.5.2.5.1. Failure of First Stage

Immediately prior to T+0:00, the fourteen RL-10M engines can be tested at 30% capacity. If twelve or more are fully functional, the crew can increase the engine performance to 100% and launch; they then have enough thrust to reach Mars orbit, rendezvous with the orbiting ERV, and return home in the event of a docking failure. The crew can still reach the ERV if they have ten of the fourteen engines working by jettisoning the MAV first stage sub-orbital and firing their second stage to reach the ERV orbit, at which point the Mars orbit rendezvous would have to work. If even more engines failed, the crew could still reach low Mars orbit and the ERV could descend to meet them; however, such an event is extremely unlikely – if the probability of an RL-10M engine failing is 2%, then losing five or more engines will occur on 1 in 150,000 missions.

3.5.2.5.2. Failure of Second Stage

If the MAV second stage fails completely, then the crew remains in low Mars orbit while the ERV aerobrakes down to their altitude. Then the crew transfers to the ERV, separates from the MAV, and fires the ERV rocket stage to return to Earth in 148 days.

3.5.2.5.3. Failure of Orbital Rendezvous

In the event of failure of the Mars orbital rendezvous, the MAV has enough propellant to return to Earth by itself on a 146-day return trajectory to Earth.

3.5.2.6. Overall mass budget for MAV/ISRU payload

The mass budget for the MAV and ISRU payload to be launched from Earth on a Qahira 3041 is given in Table 3.C. (Section 3 introduction.)

3.5.3. ERV

The ERV is designed to return the crew from anywhere in Mars orbit. It consists of a CRV-based ERV, and a trans-Earth injection stage that is identical MAV's second stage, the only difference being that the methane/oxygen bipropellant of the ERV will come from Earth.

3.5.3.1. Description of ERV and MAV Combined Trans-Earth Injection

The combined MAV/ERV will inject itself trans-Earth as described in Section 3.3.4.

3.5.3.2. Back-up Options

The presence of both an ERV and MAV capable of returning the crew provides for a number of contingencies.

3.5.3.2.1. Failure of ERV Trans-Earth Injection Stage

If the ERV's trans-Earth injection stage fails, the crew abandon the ERV and continue on to Earth in the MAV alone, using a firing sequence similar to that in Section 3.5.2.5.3.

3.5.3.2.2. Failure of ERV Critical Systems

If the ERV's life support, communications system, or other critical system is disabled and the ERV is rendered unable to support the crew before or after TEI, the ERV will still accompany the crew. This is because (1) a faster trajectory is possible using both ERV and MAV stages, regardless of life support capabilities and (2) the ERV could still provide spare parts to the MAV or be repaired after aerobraking into Earth orbit.

3.5.3.3. Rationale for Freefall During Transit to Earth

Freefall during return from Mars was deemed acceptable because (1) the deceleration upon entering Earth's atmosphere would not be as great as that experienced during Mars entry; (2) full medical support would be available to the crew upon arrival on Earth, with no physical activity immediately required, and (3) the 129 day return trajectory from Mars is comparable to time in freefall experienced by previous astronauts with no long-term ill effects.

3.5.3.4. Overall mass budget for ERV payload

The mass budget for the ERV payload to be launched from Earth on a Qahira 3041 is given in Table 3.D. (Section 3 introduction.)

3.6. Cargo Lander

The Mars Society Mission will launch hydrogen feedstock, a nuclear reactor, and additional science and exploration equipment aboard a Q3041 on October 27, 2011. This cargo lander is mobile to allow easy deployment of the nuclear reactor and transportation of the liquid hydrogen to the MAV.

3.6.1. Power

Power needs were calculated based on the requirements for ISRU, MAV, and habitat module. The power requirement of 123 kWe for ISRU are detailed in Section 3.5.2.2. The hab power requirement of under 25 kWe is justified in section 3.7.2. The MAV will be primarily in utility mode during its stay on the Mars surface, requiring around 5 kWe of power.[23] A SP-100 type nuclear reactor capable of 160 kWe and massing 9.3 MT, is more than able to meet these power needs.[24] With the addition of the hab 30 kWe nuclear reactor, a total of 190 kWe is available, providing a surplus of around 37 kWe to recharge the rover (see Section 3.8) and power science equipment. The nuclear reactor aboard the cargo flight will be deployed using the cargo lander mobility.

3.6.2. Liquid Hydrogen

For an ISRU unit to create 30.67 MT of methane and 23 MT of water, 10.22 MT of feedstock hydrogen are needed. To provide for losses of 15% in the ISRU plant and through boiloff, the MSM sends 11.76 MT of liquid hydrogen in the cargo lander. The hydrogen tank (with mass 40% that of the hydrogen) will be moved to the ISRU unit attached to the MAV using unpressurized rovers.

3.6.3. Cargo Lander Mass Budget

The mass budget for the cargo payload to be launched from Earth on a Qahira 3041 is given in Table 3.B. (Section 3.0).

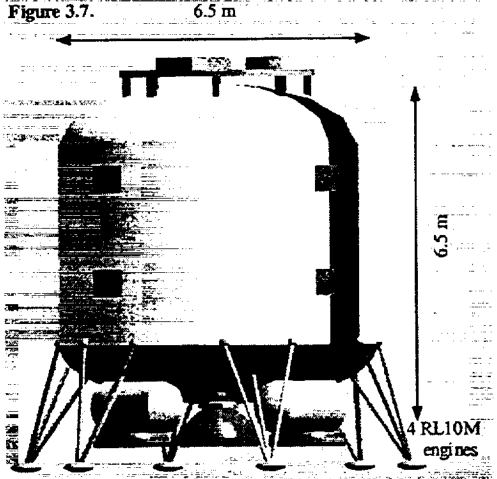
3.6.4. Science and Exploration Equipment

The Mars Society Mission will use 4.692 MT available on the cargo launch for science and exploration equipment, which can also include science support and mobility components such as unpressurized rovers. This space was initially considered for the pressurized rover, but because many rover designs exceed this mass allotment and it is desirable to send the rover to Mars as a single piece, the rover was moved to the cargo launch for the 2016 mission. The mass thus made available to science equipment is 2.6 times the amount available (1.77 MT) in the Reference Mission 3.0 for Mars surface science and exploration.[25] This large amount of science equipment provides a clear indication that science is the mission focus, and will be instrumental in establishing the mission's scientific credibility. Additional science equipment is available to the 2014 Mission crew from the 2016 Mission launches. See Section 3.8.2 for a discussion of exactly which scientific equipment can be sent and what it can do.

3.7. Habitat

The habitat module used in the Mars Society Mission is similar in concept to that of Mars Direct and the Reference Mission. Figure 3.7. depicts the hab module as it would appear landed on Mars.

Figure 3.7.



3.7.1. Dual Launch Strategy

A dual launch strategy was initially decided upon because the desired mass of the hab module to be injected (47 MT) exceeded the capabilities of what Q3041 could send on the 134 day free return trajectory. This was deemed preferable to a larger launch vehicle or additional new components because (1) a smaller Q1310 was a simpler solution and required no development of new components and (2) the secondary Q1310 launch allowed enough additional payload capacity for the inclusion of a 100% redundant backup in the form of a CRV.

3.7.2. Artificial Gravity System

An artificial gravity system was deemed necessary for the MSM's outbound hab flight to (1) minimize bone loss and other effects of freefall; (2) reduce the shock of deceleration during Mars aerobraking; and (3) have optimal crew capabilities immediately upon Mars landing. Experience with astronauts and cosmonauts who spent many months on Mir suggests that if the crew is not provided with artificial gravity on the way to Mars, they will arrive on another planet physically weak. This is obviously not desirable. Countermeasures to freefall are suggested as a means of solving this problem, but are not very effective at present. Unless a set of countermeasures that can reduce physiological degradation in microgravity to acceptable levels is developed, the only real alternatives to a vehicle that spins for artificial gravity are futuristic spacecraft that can accelerate (and then decelerate) fast enough to reach Mars in weeks, not months. To save on mass, the MSM uses an artificial gravity system with the habitat counterbalanced by a burned-out QUS, as in Mars Direct.

3.7.2.1. Rationale For Truss System

If 3 rpm is taken as the maximum rotational rate that we may subject humans to for long-duration missions, and Mars gravity (which is easier to provide than Earth gravity, but will of course condition the crew for the gravitational environment of their destination) is desired, the distance between the spacecraft and its burned-out upper stage is calculated to be 125 meters. The mass of an aluminum truss was calculated according to the equation $M = (6gml\rho)/s_t$, where the number 6 is for margin, deployment mechanism, and cross-struts, g is the desired acceleration, m is the mass at one end of the truss which experiences this acceleration, l is the length of the truss, ρ is the density of the truss material (in this case aluminum, 2,700 kg/m^3), and s_t is the tensile strength of the truss material (for aluminum 220,000 N/m^2).[26] Oscillation was not deemed problematic because the oscillation frequencies of the truss in all cases were much higher than the frequency of the system's rotation.

A truss connection between the hab and burnt-out QUS was chosen over a tether because the truss had (1) a much lower risk of failure when impacted by a micrometeorite; (2) no risk of snag; (3) less energy stored in the tension of the connecting structure which could be potentially damaging if released. An artificial acceleration due to gravity of 3.7 m/s^2 was chosen as a compromise between desired fitness of the crew and mass and mass budget concerns stemming from a larger truss. The final mass budgeted to the artificial gravity system is enough for a truss system capable of bearing 6 times the expected load.

3.7.2.2. Artificial Gravity, CRV Escort, and Reaction Control

After trans-Mars injection and transfer of the crew from the CRV escort in which they launched to the hab, the hab will separate from the CRV, which will use its reaction control system to move a short distance away from the hab/QUS and out of the system's plane of rotation, to guarantee the CRV's safety in the unlikely event of truss failure. In case of emergency requiring relocation to the CRV, the hab will have to spin down and dock. With modern technology, this process could be automated.

3.7.2.2.1. Reaction Control System (RCS) Propellant Needs

In addition to hydrazine reaction control propellant already needed for guidance and maneuvering, the artificial gravity system introduces the requirement of RCS propellant for spin up, spin down, and turning of the entire system to allow the solar panels to face the sun continuously. The figure for necessary RCS propellant produced by this reasoning was increased by 20% so that the mission could still be carried out if one of the six RCS tanks was destroyed. The total propellant need for a normal mission was 1282 kg (646 kg for the artificial gravity system), so 1666 kg was budgeted for reaction control. In the case of the free return trajectory, the habitat's RCS propellant budget is pushed closer to the total amount available. However, in this case the CRV used to escort the hab to Mars could provide some of the reaction control needs.

3.7.2.2.2. CRV Escort

The CRV is tagging along with the habitat on the trip to Mars to provide the crew with a backup spacecraft which can keep them alive in the event a critical system on the habitat fails. One instance in which the CRV would be used is failure of the habitat life support. The hab life support system was designed with surpluses for 98% closed loop operation. This means that for up to 2% of the 900-day potential operational lifetime—18 days total—the hab life support system can malfunction without endangering the crew. If the malfunction cannot be repaired, the 18 day allowance grants ample time to spin down the hab, dock with the CRV, and transfer the crew.

In the unlikely event of unrepairable habitat breach, the critical factor is the length of time it takes the crew to don their spacesuits. Even if no artificial gravity were present and the CRV and hab had remained docked, the port between the two would be closed to prevent a single breach from robbing air from both components. Continuing the explanation of why the freefall situation is not necessarily safer than the artificial gravity situation, the crew in both cases would perform a brief EVA traveling between the airlocks of each vehicle in order to minimize further air loss (the docking port is not an airlock; it is meant to contain the air pressure of its respective vehicle while closed, and allow travel between two pressurized vehicles when open). True, the crew of the hab with artificial gravity would have the added complication of de-spinning and docking with the CRV, but once in spacesuits the operational lifetimes of the suits would provide more than ample time for the automated despinning and docking procedure. Finally, the likeliest form of hab breach is one that is small and repairable, and if the breach is not repairable, it is likely to be at least slow enough to allow adequate time for spacesuits, despin, and docking.

3.7.3. Hab power systems

The habitat module will use solar panels during its transit to Mars, regenerative fuel cells during power disruptions due to events such as aerobraking , and nuclear power during the Mars surface mission. The NASA Reference Mission version 1.0 specified a 29.4 kWe power need for the habitat[27]; however, the life support power requirements have been reduced from 12 kW to 5831 watts.[28] This reduces the total power requirement to below the 25 kWe level assumed for normal habitat operation.

3.7.3.1. En route power

The Habitat module has two identical sets of solar panels, which are identical to those of RM 3.0 including their power output of 30kWe. The first is deployed at as an attachment to the center of the artificial gravity truss between the hab and QUS. It was decided to abandon this power along with the truss prior to Mars aerobraking because (1) the mass of an additional set of solar panels should the hab need to remain in orbit was considered negligible, (2) a system to retrieve the first solar array would be complex, and (3) the mass of the retrieval system would cancel any mass benefit from not needing a second solar array. After detachment of the truss and its solar array, the hab will most likely aerocapture directly to Mars surface. However, if a dust storm or other complication precludes direct aerocapture, the hab will deploy its second set of solar panels upon leaving the atmosphere and attaining parking orbit, subsisting on its regenerative fuel cells during aerobraking and passage over the nighttime hemisphere of Mars.

3.7.3.2. Surface power and deployment

Upon reaching the Mars surface, the habitat will deploy a 30 kWe nuclear reactor which will be deposited by a robotic rover in a crater or similar shielded area. A nuclear power source was chosen as superior to solar panels or RFC's on the Martian surface. Solar arrays are not efficient on the Mars surface where full sunlight is 485 W/m^2 at aphelion, and low angles of the sun, night, and atmospheric dust reduce the amount of light reaching solar arrays even further, causing the mass of the surface solar power system to become prohibitive.[29] RFC's were rejected for hab contingency power because the lifetime of the mass of RFC's budgeted for the Reference Mission is less than 24 hours, which requires that the crew tap into the existing Martian power grid in an unreasonably short amount of time. This would be problematic in the event of hab landing farther than 1 km from the target. Furthermore, longer lasting RFC's would have too large a mass penalty. Use of a small nuclear reactor as in the Mars Society Mission provides virtually unlimited time to complete surface rendezvous. If future development of solar arrays allows the substitution of solar for nuclear power on the surface, then such a substitution should be made.

3.7.4. Habitat descent system

The habitat will decelerate to a soft landing on the Martian surface by (1) aerobraking with its aeroshell, (2) deploying three 50-m diameter parachutes, and (3) firing its four RL-10M engines to produce 80,000 pounds of thrust during the landing. The propellant masses were calculated to produce 632 meters per second of ΔV required during the landing.[30] The habitat is designed with six landing legs (as are all landed components) rather than three or four so that it can still land if one of the legs fails to deploy. This would prevent it from suffering the fate of the DC-XA experimental single stage rocket, which tipped over in 1996 and was destroyed during a vertical-landing attempt when one of its four legs did not deploy properly.[31]

3.7.5. Habitat in-situ resource utilization

The habitat carries 211 kg of liquid hydrogen to the Mars surface for use in life support in-situ resource utilization. Although it is not essential for the mission, inclusion of this hydrogen and a small ISRU plant for generating oxygen and water is an important safety feature, as it allows the crew to produce 1.9 MT of water and 0.1 MT of oxygen, sufficient for the crew to survive for 19 days on the surface of Mars running on open-loop life support, even if they do not land next to their MAV and cargo lander. The water and oxygen produced could also be used to support the crew on the Mars surface for 630 days in the event that the life support system loses efficiency and can only achieve 97% closure for water and oxygen loops.

3.7.6. Habitat mass budget

The MSM habitat module launched in 2014 and its successors consist of a 4.1 m tall cylinder 6.5 m in diameter capped on top and bottom by hemi-ellipsoids that increase the total height to 6.5 m. The ratio of the surface area of the MSM hab to that of the Reference Mission is .7511, and this figure was used to linearly scale several mass budget items from RM to MSM. Several devices, such as the Reference Mission life-support system, were not scaled down from a RM crew of six to an MSM crew of five, which can be approximated as a 20% margin for the MSM. Table 3.A (Section 3.0) itemizes the habitat mass budgets of Mars Direct[32], the Reference Mission[33], and the Mars Society Mission.

3.8. Resources Available from the 2016 Mission Launches

The 2016 Mission launches to the Mars surface include MAV and Cargo payloads, both of which are launched in December 2013, and arrive on Mars in July 2014, within 40 days of the 2014 Mission's May 25, 2014 crew arrival date. As both vehicles land at the 2014 Mission landing site, additional equipment is available to the crew.

3.8.1. CRV-Derived Pressurized Rover

The Mars Society Mission lands a CRV-derived pressurized rover in July 2014, as part of the cargo payload for the 2016 mission. Table 3.8.1. illustrates mass available on this cargo payload.

Table 3.8.1. 2016 Cargo Payload Mass Budget

Component	Mass (MT)	Explanation
Hydrogen	11.760	Stoichiometry
Tank	4.704	40% of liquid hydrogen mass
Descent power (fuel cell)	0.000	Replaced by Rover RFC's
Rover	**19.536**	**Based on remaining launch to Mars surface capability**
Descent Propulsion	0.612	4x RL-10M
Descent Propellant	7.103	For 632 m/s Delta V
Propellant Tanks	0.639	9% of Propellant
Parachutes	0.700	RM 3.0
Aeroshell	8.110	18% of Payload
Transit Power 5 kWe solar	0.480	RM 1.0
Interplanetary RCS	0.800	Provides 45 m/s Delta V
Total	**54.444**	

The essential component of the 2016 Mission cargo flight, the 17 MT of feedstock hydrogen with tank for the 2016 MAV, leaves 19 MT for a rover's structure and power. The Mars Society Mission's considerations for the rover were (1) minimizing radiation on the launchpad on Earth and (2) minimizing rover mass while (3) maximizing range and carrying capacity. Minimizing radiation was decided upon as the most important factor to avoid the political issues that might prevent a human Mars mission from happening at all.

3.8.1.1. Rover Power

The Mars Society Mission recommends a rover powered by regenerative fuel cells (RFC's). The decision for the RFC system over a Dynamic Isotope Power System (DIPS) was made for political expediency, as the low mass and theoretically infinite range of a DIPS system make it the undisputed choice from a scientific standpoint. A 10 kWe DIPS system converting heat to electricity at 25% efficiency has an activity of 870 kCi; this amount of radiation on the launch pad would reduce the political viability of initiating a human Mars mission. The DIPS radiation level compares poorly to the combined launch pad radiation levels of the 30 and 160 kWe nuclear reactors; even though the SP-100 type reactors generate substantial penetrating β- and γ-particle radiation upon activation on the Mars surface, their launch pad levels of < 1 Ci make them less threatening to the public. The α-particles emitted by the DIPS ^{238}Pu are easy to shield, but given the recent experience of Cassini, the 71 kg of ^{238}Pu required for a DIPS rover should not be made a prerequisite to human exploration of Mars.

Unfortunately, the RFC-powered rover has a limited range because it must return to the nuclear reactor for recharging. The mass available for RFCs on our rover is 7.387 MT, greater than the 6.5 MT[34] estimated to be necessary for the rover to travel 500 km from its base and return. Therefore, the MSM rover is suitable for use in regional exploration.

3.8.1.2. Rover Mass Budget

The Reference Mission 1.0 assigns 16.5 MT rover to the pressurized rover, with 1.1 MT for DIPS.[35] The Mars Society Mission's CRV-derived rover improves upon the structural mass of the rover by using a modified CRV, but the use of a heavier power system, RFC's, raises the total rover mass to 19.536 MT. The CRV's mass budget is available in Section 3.5. The rover's mass budget is itemized in Table 3.8.1.2.

Table 3.8.1.2. CRV-Derived Rover Mass Budget

Component	CRV-Based Rover	Explanation
Structure	2.574	CRV figure
Thermal	0.257	CRV figure
Power + dist.	7.387	Remaining available for RFC's
Comm/info	0.320	CRV figure
Life support system	3.796	CRV figure
Food	0.095	.000630 MT for 5 people for 30 days
Water & oxygen	1.051	Enough for 10 days of open loop operation
Furniture & interior	0.800	CRV figure
Subtotal	16.280	Sum of the above
Wheels & Mobility	3.256	20% of subtotal
Total	19.536	Total rover mass

The MSM rover lands with water and oxygen for 10 days, in case it has to be used as an emergency vehicle immediately. This water and oxygen allowance, supplemented from ISRU-created stores after losses from EVA and other leakage, means that all five crew members can be 10 days away from base camp when life support fails and successfully return with a 20% margin. The mass of the wheels, carriage, and other mobility requirements was taken as 20% of the landed mass.

3.8.2. Science and Exploration Equipment

The crew of the first human Mars mission will be on Mars for 612 days. During this time, they will conduct scientific investigations of Mars and perform experiments that pave the way for the construction of a Mars base. To do so, they will make use of the 13.7 MT of science equipment sent to Mars on the 2014 cargo and 2016 MAV flights. The 13.7 MT figure is dictated by the amount of space that arises naturally since the MSM vehicles are not all the same size and the launch system is designed to deliver the largest one. It was not chosen due to a desire to include that much payload.

Nevertheless, this payload space is there and it would be foolish to waste it. Therefore, it was devoted to science equipment. The NASA Reference Mission version 3.0 budgets 1.77 MT for scientific equipment including a field geology package, geoscience laboratory, exobiology laboratory, traverse geophysical instruments, geophysical/meteorology instruments, a 10-meter drill, meterology balloons, and a biomedical/biosciences lab. In addition it allots 600 kg of instrumentation for cruise science (space physics, solar studies, and astronomy). Such instrumentation could be included on the ERV, which weighs 5 MT less than the Q3041 payload capacity; however, it should be noted that sending such instruments into Mars orbit and then bringing them back onto a trans-Earth trajectory is wasteful of propellant. It might be better to piggyback a robotic vehicle with this equipment on the ERV flight. This is especially true since, unlike geological investigations on Mars, cruise science can be automated.

For the MSM, the 13.7 MT of surface science equipment can include a 9 MT drill (capable of reaching hundreds of meters depth), and 4.7 MT of other equipment. This can include the 1114 kg of exobiological, geological, and meteorological equipment specified by the MERLIN study[36]; the 1770 kg science package in the NASA Reference Mission (which provides some overlap with the MERLIN equipment)[37]; and the 1000 kg advanced meteorology laboratory planned for the Reference Mission's third crew.[38] After this, there are still 800 kg left over for discretionary science.

3.9. Risk analysis

When sending humans to Mars, it is desirable to know the level of risk to human life in each plan. In this case, risk estimation is an inexact science that is made even more rough by the fact that many of the relevant systems do not yet exist. What is needed is a means of comparing mission architectures. The best means of doing this are different for the outbound, surface, and return phases.

Since the outbound and surface phases of the mission are similar among the Mars Direct, Reference Mission, and MSM proposals, we did not study them extensively. However, there is good reason to believe that the MSM is the safest during these phases. The CRV escort provides backup to the habitat in many critical functions during the outbound transit, and can completely replace the hab in the event of hab failure; also, the launch vehicle is equipped with an Apollo-style Launch Escape System, which is not the case in either Mars Direct or the Reference Mission. Finally, a serious question about the Reference Mission's safety is surface power, which the habitat cannot provide. If the Mars surface rendezvous fails, even by a few kilometers, the habitat would be stuck on another planet without power. The crew, incapacitated by months of microgravity, would likely perish.

The main safety drawbacks of the MSM are that the crew is placed on a new launch vehicle, and that the Mars aerocapture is performed at high speed. Unfortunately, at the present time, we can say very little about the reliability of planetary aerocapture fifteen years in the future; any numerical estimates would be very speculative. Partial answers to these issues are that the presence of an escape tower and the use of all-proven engines on the Q1310 might outweigh the "new launch vehicle" issue, and that a free return trajectory and fast transit time are worth the slightly riskier aerocapture.

It should be emphasized that the risk analysis shown below was comparative. An absolute estimation would require knowledge of the exact systems to be used; since a mission architecture does not include such specific components, a bottoms-up risk analysis of this kind was impossible. Absolute risk estimation would also require analysis of factors such as radiation, which do not vary appreciably between the three architectures evaluated here, although it should be noted, the MSM CRV provides approximately 13 g/cm^2 of shielding, the hab, 17 g/cm^2. To compare mission architectures, it is appropriate to model the risk instead based on failures at the level of spacecraft, rocket engines, and the tasks which these are expected to perform.

It was decided to estimate the probability of losing the crew during Earth return (that is, the inbound phase) as a function of six parameters: the probability per engine of a rocket engine having to be shut down, r; the probability per engine of a rocket stage exploding when fired, R; the probability of a Mars orbit rendezvous failing, f; the probability of a CRV or CRV-type capsule failing, p; the probability of a habitat failing, q; and the probability of losing the crew during Earth aeroentry, K. To lowest order, the resulting equations were:

Table 3.9.A. Methods for Calculating Risk

Plan	Probability of Not Returning the Crew from Mars Surface
Mars Direct*	$7R+r+15r^2+p+K$
Reference Mission	$4R+2r+r^2+f+q+K$
Mars Society Mission	$18R+r^2f+2r^2p+p^2-fp^2+fp+K$
MSM minus ERV	$16R+r^2+364r^3+p+K$
*Requiring 5 of 6 engines on first stage, 1 of 1 on second.	

The starting assumption was for r=0.002, R=0.0005, f=0.01, p=0.02, q=0.01, and K=0.005, yielding risks of 3.1% each for Mars Direct and the Reference Mission and 1.5% for the MSM. However, a variety of other possibilities were examined, as seen in Table 3.9.B.

Table 3.9.B. Effect of Different Risk Assumptions on Total Mission Risk

Possibility	Assumption(s)	Risk of Mission:			
		Mars Direct	NASA DRM	MSM minus ERV	**Mars Society**
Starting assumptions		3.1%	3.1%	3.3%	**1.5%**
Mass budget forces less redundant CRV*	p=0.08	9.1%	3.1%	9.3%	**2.1%**
Mars orbit rendezvous considered risky	f=0.05	3.1%	7.1%	3.3%	**1.5%**
Mars orbit rendezvous nearly guaranteed	f=0.0005	3.1%	2.2%	3.3%	**1.4%**
Unreliable engines	r=0.005, R=0.001	3.7%	3.9%	4.1%	**2.4%**
CRV almost as good as habitat	p=0.0133	2.4%	3.1%	2.6%	**1.4%**
*A concern that was raised about Mars Direct					

As can be seen, the MSM is the safest mission in all of these scenarios - as long as the ERV is included. It is because of this risk reduction and the comparatively low development costs that the ERV was included in the Mars Society Mission in the first place. Several times, our team considered removing it from the plan. However, since it cuts the return risk by approximately half, and also provides a CRV to accompany the next outbound crew, the ERV was left in the mission plan.

3.10. Beyond the First Mission

The flexibility of many Mars Society Mission aspects allows for a number of options for the 2016 and later missions.

3.10.1. Incorporation of Future Technologies into QITS

While the LOX/Hydrogen based QITS is all that is needed for the Mars Society Mission plan, and is the MSM's recommended launch system, it would be fully compatible with upgrades such as Magnum-style liquid flyback boosters (LFBB's), which in any case require a core stage such as the QBC and QUS. One such configuration, playfully dubbed the FatCat, could use two pairs of LFBB's arranged in two catamarans on both sides of a QBC as its 1st stage, a second QBC as its 2nd stage, and a single QUS as its 3rd stage to send 73 MT trans-Mars on a single launch. However, it should be emphasized that upgrades such as the FatCat's LFBB's are neither necessary nor directly planned for by MSM, and thus should not be included in calculating the MSM's development cost. A nuclear thermal rocket (NTR) as a fourth stage could increase the Q3041 lift capability to Mars to approximately 70 MT.

3.10.2. Crewed Launches of 2016 and Later

This section is designed to explore additional possibilities for later crewed launches using QITS.

While it is possible to launch the crews of succeeding missions in the same manner as the 2014 Mission, a more economical option is to launch crews aboard a lightweight, "ghost" CRV stripped of life support and food, all of which will not be necessary during the brief transit from Earth to the orbiting hab. Instead, the same reserves that allow 3.8 days of open loop operation during a CRV's return from Mars provide ample life support ascent to the hab. This lighter vehicle allows for the use of a Q1010 and elimination of the Castor-120 solid rocket boosters.

3.10.2.1. Ghost CRV
The ghost CRV is identical to the fully functional CRV's used elsewhere in the Mars Society Mission (see Section 3.5.) but without food or a recycling life support system, giving it a mass of 11.088 MT as opposed to 15.951 MT for a fully functional CRV.

3.10.2.2. Q1010 Earth Ascent Timeline for 2016 Ghost CRV with Crew and LES Launch
The 2016 crew launch involves first a Q3041 launch with a habitat, as in the 2014 crew launch. However, the crew launches in an 11.088 MT ghost CRV that is light enough to be launched by a Q1010 - that is, a Q1310 without the solid boosters. After the initial QUS shutdown, orbit circularization occurs at 360 km and lasts 1.3 seconds, for a change in velocity ΔV = 0.04 km/s. The 63.6 MT vehicle is now ready to rendezvous with the habitat. Its mass breaks down into the 11.1 MT CRV and the 18.0 MT dry QUS with 34.5 MT propellant.

The smaller vehicle—that is, that launched by the Q1010 consisting of ghost CRV, crew, and QUS—carries out up to 130 m/s of rendezvous maneuvers using 2 MT propellant and leaving 32.5 MT. The docking occurs in an orbit of 360 km at velocity 7.69 km/s with period 1 hr 32 min. The fully assembled TMI vehicle then has mass 229.6 MT, of which 135.5 MT is propellant.

3.10.2.3. Trans-Mars Injection of 2016 Habitat with Crew and 2014 CRV
Of the two QUS stages at each end of the vehicle, with the hab and ghost CRV sandwiched between, the Q1010-launched QUS fires first. It fires for 1 min 14 s, providing 0.68 km/s ΔV and raising the orbit to 360x3410 km with a period of 2 hr 5 min. The crew transfers to the hab and the QUS and its ghost CRV are then released. The Q3041-launched payload (QUS and hab) is then accelerated by the QUS's single RD0120 engine. This fires for 2 min 44 s, providing 2.51 km/s Delta V and raising the orbit to C_3=0. Here the crew can dock with the CRV left behind at that energy by a previous mission, using up to 130 m/s of ΔV for rendezvous maneuvers. After docking, the vehicle consists of a QUS (18 MT dry with 28 MT propellant), 47 MT hab, and 16 MT full CRV. The QUS then fires its engine again for 1 min 4 sec, burning all its remaining propellant to provide ΔV=1.32 km/sec. As the ghost CRV is designed to carry the crew back to the Earth's surface intact, there is no reason not to land the ghost CRV intact even without crew. When the ghost CRV's QUS burns out and the CRV jettisoned, the perigee of its orbit is only 360 km. If the ghost CRV's orbit is allowed to decay in a controlled manner, it can be recovered, refurbished (necessary only for the aeroshell), and reused to launch the 2018 and succeeding crews.

3.10.3. Additional Applications of the CRV
The redundancy of the CRV used by the Mars Society Mission is the basis for additional applications unrelated or merely incidental to sending humans to Mars.

3.10.3.1. Lifetime of a Typical CRV
The availability of CRV's for additional applications is exemplified by the CRV that accompanies the crew on the 1st mission in 2014. On December 25, 2015, the CRV returns on a free return trajectory from its mission as an escort to the outbound 2014 crew. It is then available to act as escort again for the 2016 crew. After free-returning to Earth and aerobraking a second time, its aeroshell is likely to have ablated to where it cannot aerobrake again. Two such CRV's will be produced by the 1st mission: the aforementioned outbound escort to be used by the 2014 and 2016 outbound crews, and the inbound escort from the 2014 mission (of the ERV and MAV, whichever is not used to land the crew on Earth), which will be used as an outbound escort by the 2018 mission. Starting with the 2016 mission, one re-usable CRV will be produced per mission, each to be used by the mission four years later. After the second aerobraking at Earth, the CRV can be used for other purposes.

3.10.3.2. Space Station
CRV's retired from interplanetary space could augment the International Space Station. For every CRV added, the ISS benefits from increased (1) crew capacity by five, (2) power by 81 kWe (the same array that produces 30 kWe at Mars aphelion produced 81 kWe at Earth aphelion), and (3) volume by 55 m^3 minus space already taken up by equipment. Compared to the planned capabilities of the ISS, a single CRV would increase the crew capacity, power, and volume of the ISS by 71%, 88%, and 5% respectively.[39]

3.10.3.3. Increased crew capacity
The aeroshells of twice-aerobraked CRV's could be refurbished for additional Earth aerobraking. The CRV's could then be used to expand future Mars missions by sending additional crew, possibly by launching crew in fully functional (instead of ghost) CRV's aboard Q1310's. As only the hab is equipped to land on Mars with people, the crew would spend the time in transit in the CRV, then transfer to the hab briefly to aerobrake to Mars surface while the CRV free-returns to Earth. The hab would carry all 10 astronauts to the Martian surface, where the extra five could occupy habs from previous missions, inflatable structures, or structures created on the Martian surface as part of an expanding Mars base.

3.10.3.4. Lunar missions
CRV's could also be sent to Luna. They are potentially useful as return vehicles from the lunar surface, as they could keep their crew alive for months in lunar orbit (i.e., until a rescue mission could arrive) in the event that Trans Earth Injection from lunar orbit were a failure. A Q3041 equipped with a LOX/hydrogen upper stage (for lunar landing) could deliver a CRV with a storable propellant (N_2O_4/hydrazine) trans-Earth/ascent stage to the lunar surface. A ghost CRV also has sufficient open-loop life support (3.8 days) to be used as a lunar return module.

4. Conclusions

The Mars Society Mission has the potential to send humans to Mars with reduced risk using fewer components and requiring less development than the NASA Reference Mission 3.0. NASA should evaluate the Mars Society Mission and consider it for adoption as the basis for the Design Reference Mission 4.0.

5. Future Studies and Lessons Learned

In relation to the Mars Society Mission, a number of paths are still to be explored. As our goal was not to create a project for a class, but to develop a comprehensive infrastructure and strategy for human interplanetary exploration, we are striving to improve the existing MSM. As with any humans-to-Mars mission design, the specifics of all components are being continuously improved upon, for instance, we are designing the exact fuel tank configuration of the QBC, QUS, and MAV stages, and considering new launch vehicle and Mars ascent ideas. We are also determining the applicability of MSM components to human and robotic missions to Luna, Saturn, and the asteroids, and will plan new components.

6. Outreach Efforts

As members of an organization dedicated to furthering the robotic and human exploration of Mars, our team has done considerable public outreach, and will continue these in the future. Efforts specific to the Mars Society Mission have included:

- Presentations of the Mars Society Mission to the public at Caltech on May 2nd and May 4th, with a total of about 150 attendees.
- A comprehensive web site explaining the details of both the Mars Society Mission and the general aspects of Mars exploration, including computer-generated movies and a comprehensive resource area with fliers, banners, and other downloads that everyone can use to rally support for Mars exploration. The site has been uniquely accessed over 5500 times since it was created on September 1, 1998. See http://www.cco.caltech.edu/~mars.

Our four person team also has several additional activities scheduled for the coming year, with several events which we will be personally organizing:

- June 11-13, 1999: **Booth and Presentation at AgamemCon Sci-Fi convention** in Anaheim, CA, featuring a large amount of literature distribution, Mars Society membership drive, and a talk on behalf of future Mars exploration.
- July 20th, 1999: **30th Anniversary of the Apollo 11 Lunar Landing**. We are planning an event to attract media attention to the fact that, 30 years after Apollo, we have yet to reach Mars. We will be collaborating with the Fifth International Conference on Mars, which will be held at Caltech during the weekend of the Anniversary.
- August 12th - 15th, 1999. **Presentation of the MSM at the National Convention of the Mars Society.**
- December 3rd, 1999: **Mars Polar Lander in the Southern Layered Terrain**. This event will be a celebration of the (hopefully!) successful arrival on Mars of the Mars Polar Lander. The Mars Society will use the interest generated by this 1999 landing to increase membership and awareness.
- April, 2000: **Membership Drives and Public Awareness Campaign** in conjunction with the release of the major motion picture *Mars*.
- May 25, 2000: **The Next First Step T-14 Years**. According to the Mars Society Mission trajectories, the first human landing on Mars will take place on May of the year 2014. This event will call attention to the effort to make sure the scheduled date is achieved.

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Extravehicular Activity Suit Systems Design: How to Walk, Talk, and Breathe on Mars

Cornell University
1999

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Abstract

Design parameters for a Mars Extravehicular Mobility Unit (EMU) are different from current space shuttle and past Apollo EMU designs. This report derives functional requirements for the life support, communication, and power subsystems of a Mars EMU from the HEDS reference mission and Mars surface conditions and proposes a design that satisfies all of the currently understood functional requirements for each subsystem. Design for the life support system incorporates O_2 storage, possible O_2 production, CO_2 absorption, humidity control, thermal regulation, and radiation protection. The communication system design centers on a reconfigurable wireless network, virtual retinal display, and emergency locator beacons. Portable power options are analyzed, and Direct Methanol Liquid Feed Fuel cells are selected for use in a design that satisfies the power requirements. Mass, cost, and technological readiness are considered for each system. This paper concludes with a recommended combination of subsystem designs that combine to form the primary subsystems of a Mars EMU.

1.0 Introduction

Mankind has the ability to safely send humans into Earth orbit and to the Moon. We have sent telemetry-controlled robots to the far reaches of our solar system as our hands, eyes and ears. We will gain further knowledge about our past and the nature of the universe by sending a human mission to Mars. To accomplish the scientific objectives that help achieve this goal, a human must interact with the Mars surface in real time. This requires an Extravehicular Mobility Unit (EMU) that will ensure the safe and comfortable survival of the human during Extravehicular Activities (EVAs).

Key parameters in designing systems for use on a Mars EMU include the planned length of the mission, the number of EVAs per EMU, and the indigenous resources and physical limitations of Mars. The required EVA duration from the HEDS Reference Mission is 4 hrs, with a goal of 8 hrs. Assuming a 6 person crew, a 500 day max surface stay [1], and that each astronaut performs two EVAs every three days on average (0.66 EVAs per day per person), this leads to 2000 total individual EVAs. If each person has one suit, it will have to withstand use on 333 EVAs. Less exhausting EVA scenarios are outlined below (Table 1).

Table 1: EMU use with respect to # of EVAs

Days surface stay	*Average EVAs per day per person*	*EMUs per person*	*EVAs an EMU withstands*	*Total Hours for 4hr EVA or 8hr EVA*
500	0.66	1	333	1330 or 2660 hrs
500	0.66	2	167	670 or 1330 hrs
500	0.5 (includes a day off)	1	250	1000 or 2000 hrs
500	0.5 (includes a day off)	2	125	500 or 1000 hrs

Further design constraints are introduced by environmental parameters (see Table 2) that are significantly different on Mars compared to on Earth or in Earth orbit.

Table 2: Comparison of parameters: Mars, Micro-gravity, Earth [2]

Parameter	*Mars*	*Micro-gravity*	*Earth Standard*
Temperature	130K to 300K	Insulated	288K mean
Pressure	.01 atm (1% Earth pressure)	--	1 atm
Gravity	$3.73ms^{-2}$ (39% Earth)	--	$9.80ms^{-2}$
Magnetic field	No current field	Missions within Earth field	Magnetic field
Radiation	About 5-15 rems/yr	Same as Earth	About 0.4 rems/yr
Atm. Composition	CO_2, N_2, Ar, O_2	Not applicable	N_2, O_2, H_20, Ar
Solar constant	590 W/m^2 mean	Same as Earth	1371 W/m^2 mean

The mass of the current space shuttle EMU is 113 kg [23], which would translate into a weight of about 44 kg for a Mars EMU. This is an unacceptable amount for a person to carry. An acceptable weight to carry would be about half that, or 22 kg. A martian weight of 22 kg is equivalent to a system mass of about 58 kg, which is the mass limit that we adopt for this design. From the 58kg, 20kg is allocated for upper body EMU structure, 10kg is allocated for the legs/boots and 28kg is allocated for the life support, power, and communications subsystems.

The goal of the design is to meet the following functional requirements while staying within the 28kg mass allocation (Table 3).

Table 3: System Functional Requirements

Life Support System	*Communication System*	*Power System*
Suit pressure 8.3 psi (including N_2 buffer gas)	30km radius range	Provide 150W
O_2 partial pressure 4.1psi O_2 flow rate 0.074kg/hr	Provide biomedical and diagnostic information	Potential 18V
CO_2 flow rate 0.2035kg/hr	Audio and 1-way video	30 minute min. backup power
Temp. 283-317K (9.85-43.85°C)	Scientific information	
Total dose radiation < 10rems	Independent backup 3km radius range	

Most EMU subsystems require modification from those used on current/past EMUs in order to satisfy both the functional requirements and operate under martian conditions (Table 4).

Table 4: Subsystems Affected by Mars Conditions

Subsystem	*Gravity*	*Temp.*	*Press.*	*Atm.*	*B-field*	*Solar*	*Dust*	*Duration*	*# EVAs*
Gas Exchange	✓	✓	✓	✓			✓		
Thermal Regulation	✓	✓	✓	✓			✓		
Radiation Protection	✓			✓	✓	✓		✓	✓
Int. Communication	✓								
Ext. Communication	✓				c		✓		
Backup Comm.	✓							✓	
Primary Power	✓	✓				✓	✓	✓	✓
Backup Power	✓							✓	✓

2.0 Design Approach

2.1 Life Support System

The components that require modification from the existing micro-gravity EMU life support system for use on Mars include oxygen storage and production, carbon dioxide removal, humidity and temperature control, and radiation level monitoring.

The primary design constraint for a life support system is EVA duration. The oxygen required for 4 hr to 8 hr EVAs varies from .092 kg to 1.816 kg depending on EVA length and activity level. Here 0.595kg O_2 is used to supply an 8 hr EVA with average exertion. [3] High-pressure oxygen storage parameters are compared with cryogenic oxygen storage.

High-pressure oxygen gas storage on the current space shuttle EMU can be modified for the Martian environment. The current system contains high-pressure canisters for the storage of oxygen in the Primary Life Support System (PLSS). Two rechargeable primary tanks contain all oxygen needed for the astronaut to breathe during an 8 hr EVA at 6.2Mpa (900psia). In the event of primary system failure, two smaller tanks charged to 41.4 MPa (6000 psia) are backup. These provide oxygen to the astronaut at a much higher rate in purge mode for up to 30 minutes. These are not rechargeable, the equipment to re-pressurize them with the necessary amount of oxygen is prohibitively heavy. (For 60-minute backup, the mass would be at 2.38kg and volume at around 5.10L at 6000psia.)

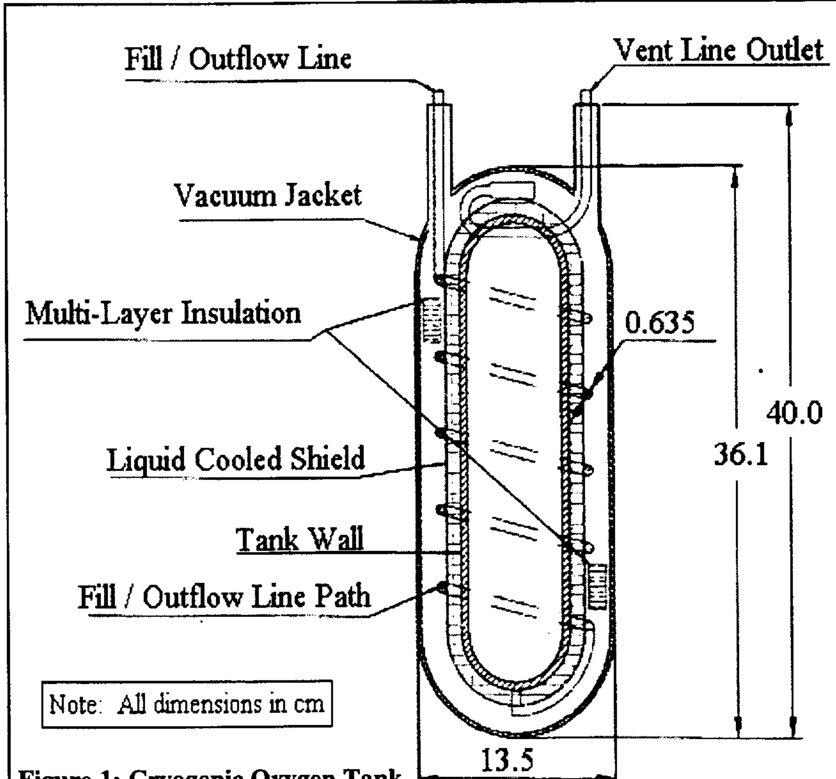


Figure 1: Cryogenic Oxygen Tank

This storage system has limitations. The thick walls necessary to contain the high pressures, while not a concern in micro-gravity, are too massive to use on Mars. The life support system must be physically small, requiring a more volume-efficient method of storing oxygen. Finally, the high pressure for oxygen storage on the current EMU are hazardous if a malfunction occurs. Leakage and bursting are dangers due to the explosiveness of the pressure as well as the flame-enhancing characteristics of oxygen.

A second proposed method of oxygen storage uses cryogenic tanks to store liquid oxygen. Such a system could use a single oxygen storage device for an 8 hr EVA as well as the 30 minutes of backup. Because it requires too much energy to warm a large quantity of supercritical oxygen passing through the suit during a purge situation, secondary oxygen tanks like those used in current EMUs should provide backup supply. For rechargeability, the secondary canisters may be modified to be filled with liquid oxygen and then warmed to ambient temperature, at which the oxygen will boil, pressurizing the tank.

The proposed tank design is adapted from one suggested by Lockheed Martin for the space shuttle EMU. [3] This consists of an inner tank containing cryogenic liquid surrounded by a liquid-cooled shield (LCS) (Figure 1). This is in turn surrounded by multi-layer insulation (MLI) and a vacuum jacket. This system is designed to minimize heat flow into the liquid so that little vapor venting is required to relieve boiloff pressure. The LCS is key. Outflow liquid oxygen from the bottom of the tank is routed around the LCS to cool it to subcritical temperatures and absorb any heat transferred into the system before it can warm the fluid within. This design reduces heat input to almost zero, and relies on a liquid positioning device (LPD) to keep the cryogenic oxygen over the outlet at the bottom, however, Mars gravity makes this precaution unnecessary. Upon exiting the LCS, the oxygen is warmed to breathing temperature through heat exchange with the power source and liquid cooling ventilation garment (LCVG) explained below.

This cryogenic system addresses the concerns of a portable oxygen supply system in a Martian EVA suit. Using a cryogenic storage method reduces both tank mass and volume over traditional high-pressure systems. This allows for the expansion of the system to carry more oxygen if a longer EVA is desired. Both the cryogenic and high pressure systems can satisfy the life support oxygen flow rate, pressure, and partial pressure requirements, but the cryogenic system can do so with less mass and more oxygen.

Table 5: Comparison Data for two different oxygen storage systems.

	Primary System Mass (empty)	Primary System Volume	Total System Volume	Primary Tank O_2 Mass/Mass	Primary Tank Pressure
High Pressure O_2 System 4 Tanks total	4.4 kg	16.88 L	19.83 L	0.125	6.2 MPa 900 psia
Cryogenic Oxygen System 3 Tanks total	4.0 kg	7.87 L	10.82 L	0.149	< 930 kPa < 135 psia

For subsequent missions, a self-contained oxygen production system that uses the abundant CO_2 in the Mars atmosphere is desirable to produce breathable oxygen dynamically on the EMU. One technology that can achieve this is solid oxide electrolysis. A prototype solid oxide electrolysis unit was demonstrated by University of Arizona Space Technologies Laboratory [4]. Their system heats CO_2 to an operating core temperature of 1023K (750°C), dissociating two molecules of CO_2 into two carbon monoxide molecules and one molecule of oxygen (Figure 2). An electric potential dissociates molecular oxygen into two oxygen ions, which diffuse through an oxygen-permeable yttrium-stabilized zirconia membrane. The ions recombine on the other side of the membrane into molecular oxygen. The prototype mass and volume is 1kg and 3.9L with a steady state power requirement of 9.5W and 15W as the start up power requirement. Reflective ceramic insulation keeps the external surface temperature below 313K (40°C), and oxygen output was 0.5 cm^3/min. [4] The oxygen production is less than the minimum required for human consumption, 0.53 cm^3/min, but of the same order of magnitude. In the future, the production level is expected to rise to provide enough oxygen for dynamic consumption. As a result, suit power will become the only limiting factor for the length of an EVA. Inefficiency from heat exchange with the Mars atmosphere can be decreased through the application of new insulation technologies. For example, silica aerogels

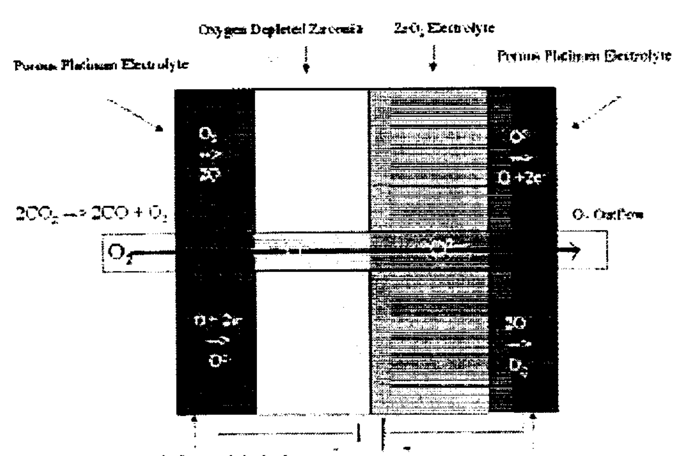


Figure 2: Oxygen Production, Solid Oxide Electrolysis

are extremely lightweight and can be made to be quite strong while possessing an average thermal conductivity of 0.017 W/mK to better insulate the oxygen production cell. [5]

The prototype will be flight tested as part of the MIP (Mars In-Situ Propellant Production Precursor) on the Mars Surveyor Program Lander in 2001[5]. Eventually, further miniaturization and insulation advances may allow the unit to become a standard component in the EMU life support system.

Glow discharge and permeation is another way to produce oxygen from the Mars atmosphere. A reaction chamber heats gas from the Martian atmosphere to 450°C. A glow discharge is generated from a silver electrode that disassociates carbon dioxide into carbon monoxide and atomic oxygen. Oxygen is separated using a silver membrane. A silver lattice structure is selectively permeable to the atomic oxygen, allowing it to pass across the membrane and recombine to molecular oxygen on the other side, where it can be accumulated and used for the astronaut's needs. Carbon monoxide is vented to the atmosphere [6]. The system is at a low level of technological readiness and currently requires 2 kW to produce 1kg of usable oxygen in one day. However, the system does not bring with it any complications involving dust collection and CO_2 pumping, and operates at significantly lower temperatures than solid oxide electrolysis. The system's current status implies that this technology will someday be competitive with other oxygen production techniques for Martian exploration [7].

Several methods exist to remove carbon dioxide. Lithium hydroxide scrubbers have been used extensively on nuclear submarines and past space missions including the Apollo program. Lithium hydroxide (LiOH) spontaneously and exothermically reacts with carbon dioxide (CO_2) to produce solid lithium carbonate (Li_2CO_3).

$$2LiOH_{(s)} + 2H_2O_{(g)} \quad 2\,LiOH \bullet H_2O_{(s)}$$
$$2LiOH \bullet H_2O_{(s)} + CO_{2(g)} \quad Li_2CO_{3(s)} + 3H_2O_{(g)}$$
$$\Delta H^{o} = -25.2\text{kcal/mol}$$

This system has been successfully used to capture exhaled CO_2 and convert it into solid Li_2CO_3 on past space flights. However, the reaction chemistry causes regenerating LiOH to be difficult, making LiOH scrubbers non-reusable. The technique is less than ideal for a prolonged mission where scrubbers for hundreds of EVAs would have to be brought from Earth.

Metal oxides have also been used in past missions. This system relies on the reaction chemistry of metals to take carbon dioxide out of the system. While metal oxide canisters are reusable, the heavy metal substrates cause significant increases in system mass and volume. This problem can be ignored when the system is deployed in micro-gravity, but it makes the system impractical in environments where mass is a limiting factor. [12]

The DARA system, a carbon dioxide removal technique that utilizes solid amines, is the better option. This system, co-developed by the European Space Agency and the German National Space Agency, uses a porous resin as a carrier for series of weak basic amine groups. The mechanism of reaction is:

$$(3)(R\text{-}NH_2) + CO_2 + H_2O \qquad (R\text{-}NH_3^{+} + HCO_3^{-})$$

The solid amine matrix (type DOR-SA-028), produced by Bayer A.G. is composed of a extremely porous polystyrene. Particle size ranges from 0.5 to 1.2 mm, and it is regenerative. When CO_2 load capacity has been reached, 38.1 kg of CO_2 per kilogram of amine at 4kPa partial pressure, the carbon dioxide bonded to the polystyrene matrix can be released by altering the equilibrium of the reaction through a change in pressure or the addition of heat. The high loading capacity allows the total mass of the system to be low enough to make the system practical for EMU CO_2 removal. It is also stable; solid amine active groups and material properties remain intact after 15,000 hours of operation. Even after two years in storage, there is no evidence of material degradation. The reproducibility of the solid amine product is at a replicable quality level. [13] The result of test trials with the system can be used to predict a weight of 2.6 kg for two solid amine canisters. However, power requirements for management and maintenance to the system must be reduced to make the system practical for use on an EMU. [90]

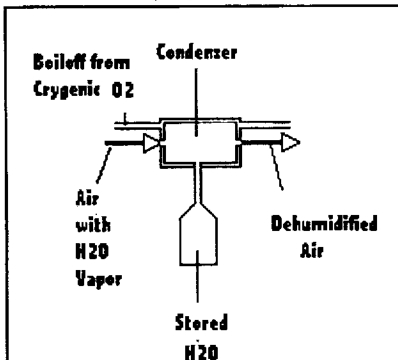


Figure 3: Sample Humidity Removal System

Exhaled water vapor must be removed and recycled. An adult male exhales between 0.15 and 1.5 grams of water per minute [12], so the EMU must remove water vapor at this. NASA currently uses sublimators and lithium hydroxide (LiOH) scrubbers. It may also be possible to use on Mars the same basic methods used in dehumidifiers on Earth: both desiccant- and cooling-based dehumidifier systems might be possibilities.

Lithium hydroxide scrubbers are efficient, but not recyclable. The mission would need to bring enough scrubbers for over two years worth of EVAs. Finally, desiccant dehumidifier materials have a high affinity for water vapor such as lithium chloride that combines with water to form a liquid solution and continues to absorb water after solution has formed [15]. After use, the solution can be heated to regenerate the LiCl and water vapor. However, this process requires a fan and volume to hold the solution.

The cooling-based dehumidifier is the most practical because it requires no regeneration of expendables and no extra volumes, and can be used to supplement other parts of the EMU: liquid oxygen tubes may be used as coolant, and oxygen could be heated also in this process, for breathing. This would satisfy the life support requirement to filter out humidity.

A critical life support consideration is maintaining a thermal balance within the Mars EMU. Current space suits are designed to function in a vacuum. However, for the Mars EMU, convective heat loss through the atmosphere must be considered. The range of temperatures comfortably tolerated by humans is about 18°C to 27°C [18].

Current EMU's consist of a Liquid Cooling and Ventilation Garment (LCVG) and insulating materials [19]. Cold water, fed through the LCVG tubes, picks up heat as it circulates throughout the body [87]. The LCVG then separates into two streams, one directed to a sublimator and the other directed to a contaminant control system [24]. The sublimator convects heat and water vapor to the atmosphere [87]. The insulating materials consist of aluminized Mylar plastic, unwoven Dacron, and Orthofabric; these synthetics can protect from a temperature range of -129°C to 148°C, which is sufficient for the Mars temperatures.

Challenges in developing a Mars EMU include heat convection to the Mars atmosphere and EMU thermal accumulation [18]. The EMU heat sources and heat sinks are listed as follows.

Table 6: EMU Heat Sources and Sinks

Heat Sources		Heat Losses	
Body Heat	0 – 560 W	Wind	300 –730 W
Fuel Cell	0 – 150 W	Cryogenics	7-15 W
Solar Heat	0 – 120 Wm^{-2} ± 20%	Boots	Minimized
Total Range	**0 – 850 W**	**Total Range**	**300 – 750 W**

Solar heat on Mars is nominally 590 Wm^{-2} with ±20% variation due to the perihelion-aphelion positions. The white exterior of the suit will absorb only an estimated 25% of the heat, reducing the solar effect. The EMU power supply (here a fuel cell) generates excess heat that must be relieved for efficient operation. Convection heat losses due to atmosphere through the EMU surface can range from 300 W to 725 W [23]. Oxygen from cryogenic storage must be heated to breathe. In response to the cold martian temperatures, the solution proposed by Hamilton Standard is the creation of an external thermal garment. However, for thermal insulation to be effective on Mars, layering up to four inches thick will be required [23].

According to Hamilton Standard, the solution to thermal regulation in the suit is passive heat rejection. EMU insulation is minimized and allows heat to escape to the Mars atmosphere. If heat loss is too great, a thermal overgarment, stored on the EMU support cart, can be donned. Hamilton Standard conducted tests demonstrating the ease of donning and doffing the external thermal garments [23] but thick garments hamper mobility. If the LCVG unit is to be used as needed on the Mars EMU, a sublimator cannot be used as there is a problem with its heat exchange mechanics on Mars. [87] Wind speed may not dissipate sufficient heat, requiring an auxiliary cooling device. The sublimator successfully dissipated heat for current EMUs, but it is impractical for use on Mars. The porous plate on the sublimator would get clogged by dust. It is designed for a vacuum and the atmospheric pressure on Mars, 1% of Earth pressure, will inhibit sublimation [87]. A sublimator also vents valuable water, preventing the Mars EMU from remaining a closed system. A convection radiator is another alternative, warm water from the LCVG circulates through a finned radiator on the EMU backpack. The radiator convects heat to the Mars atmosphere but requires a large finned radiator array.

Table 7: Sample Metal Hydride Heat Pump System (MHHP)

	HCI Tests	Mars Mission Requirements
Dimensions	0.305 m × 0.610 m × 0.914 m	0.305 m × 0.457 m × 0.080 m
Radiative Surface Area	1.49 m^2	0.261 m^2
Mass	112.2 kg	7.35 kg
Radiator Temperature	56°C	80°C
Heat Radiation	440 W	125 W
Duration	4hrs	4hrs, replacement in cart
Power required (approx.)	20 W with ~10% efficiency	20 W with ~10% efficiency

A Regenerable Nonventing Thermal Sink (RNTS) as proposed by Hydrogen Consultants, Inc (HCI) is a practical option. The system uses Metal Hydride Heat Pumps (MHHP) and a blackbody-type radiator [89]. The low temperatures on Mars would facilitate heat radiation to the ambient atmosphere.

Mars heat radiation requirements are lower because the bulk of EMU cooling is from atmospheric convection. The amount of heat that the MHHP can dissipate varies with the ambient temperature on Mars and the temperature of the radiator surface. The following figure illustrates different radiator temperatures with corresponding heat dissipation. The MHHP can replace sublimator as the cooling mechanism for the LCVG. The MHHP consists of an aluminum radiator lined with tubes of hydride A ($La_{1.1}Ni_{4.6}Sn_{0.4}$) [89]. Tubes of hyrdride B (MM $Ni_{4.5}Al_{0.5}$) are placed in the radiator cavity. Warm water from the LCVG runs over the hydride B tubes and heats the metal hydrides, causing the release of hydrogen. This hydrogen is fed into the hydride A tubes and is deposited onto hydride A, increasing radiator surface temperature which dissipates heat to the Mars atmosphere [89]. Hydride B is the cooling source and hence its temperature be kept just above 273 K to prevent the cooling water from freezing. Using two containers with hydrides A and B eliminates venting; the containers can also be recharged at the base [89]. Compared to the current EMU, the LCVG configuration will remain unchanged. The only major change is using the MHHP in place of the water-fed sublimator.

Thus a solution to thermal regulation to stay well within the required temperature range of 9.85°C to 43.85°C involves utilizing the environment as well as implementing an active auxiliary thermal control system. Areas of continued research include other sources of heat loss, MHHP power requirements, and effectiveness of multiple insulative layers.

The radiation environment on the surface of Mars is more difficult to deal with than for previous manned missions for two reasons. First, the radiation that astronauts will be exposed to will be of a different variety than was previously encountered. Also, the energies and fluxes of the radiation will be much higher than designers have had to previously consider. Complicating this fact is the extended duration of the mission. Missions to Low Earth Orbit (LEO) or to the surface of the moon were of short enough duration that weight savings on radiation insulation could be justified by the brevity of the mission. [2] There are three kinds of radiation on Mars:

- Ultraviolet radiation consists of high frequency electromagnetic waves traveling through space at the speed of light. The fact that this type of radiation is composed entirely of energy (and therefore has no mass) makes it relatively easy to counteract. This type of radiation is a familiar concern on Earth, and significant research has been done into inexpensive and effective methods of blocking it.

- Another type of radiation is solar particles, mostly protons, and due to their particulate nature these particles will be inherently more difficult to block. On Earth, much of this radiation is deflected by the magnetic field – protons have a charge, and are deflected by the large field produced by the Earth. Mars does not have any appreciable magnetic field. Solar Particle Events (SPEs), when the sun periodically releases high concentrations of high-energy particles in the form of solar flares and solar storms. are the real danger.

- Very high-energy heavy particles coming from neighboring galaxies are commonly referred to as Galactic Cosmic Radiation (GCR). Although this radiation spreads throughout the universe at a constant rate, surface doses fluctuate in response to solar activity, solar minimums corresponding to the highest levels of GCR and vice versa. These particles will not be detected in large quantities when compared to the normal flux of solar radiation; however, their extremely high velocity and larger mass make them a serious consideration. Again, these particles are not a concern on Earth, as the magnetic field and thick atmosphere deflect most dangerous levels.

Table 8: NASA radiation exposure limits for LEO missions [5]

Exposure Interval	Blood Forming Organs	Ocular Lens	Skin
30 Days	25	100	150
1 Year	50	200	300
Career	100-400	400	600

The normal background radiation exposure on Earth is about 0.4 rem/yr. The occupational limit for high risk jobs is 5 rem/yr. A once in a lifetime emergency exposure of 25 rem is not fatal, but 500 rem over the course of a human lifetime will be lethal (this figure is dependent on many physical characteristics and could vary by as much as a factor of two). [3] NASA has also set limits for radiation exposure for missions into low earth orbit (LEO).

Although no limits have been set for a Mars mission, reasonable estimates can be derived, assuming an overall dosage maximum of 100 rem for the entire mission, of which, 5-10 rem will come from exposure during EVAs. Technology for blocking ultraviolet radiation already has been developed to a relatively high degree. Quality plastics are good enough to stop even the high levels of UV radiation on Mars, and protective coatings can be applied to almost any surface.

Background solar radiation, although dangerous if unshielded, is well blocked by relatively thin layers of shield material. Taking SPEs into account complicates the situation. It would not be practical to provide the astronauts with enough shielding to withstand SPEs at all times. Fortunately, SPEs can currently be predicted up to a day in advance. After one is detected from Earth, an alert to Mars will give the astronauts about 15 minutes to retire to a designated "storm shelter" set up to shield them from radiation storms.

GCR requires the most innovative thought. Unlike SPE radiation that causes damage simply by colliding with molecules in its way, GCR arrives with such momentum that it breaks apart atoms of the shield materials producing secondary radiation particles. In this scenario, small quantities of shielding are worse than no shielding at all. The GCR component of the background radiation on Mars is too energetic to be shielded against without unacceptable quantities of material [7]. Moreover, these fluxes are low enough to justify the omission of this extra mass. Solar particle events can be protected against through the use of a storm shelter. By shielding against UV and background solar radiation, predicting SPEs, and calculating that the GCR radiation is not enough to be harmful, the requirement to protect the astronauts from radiation is satisfied.

2.2 Communications

In the design of this external-to-EMU communications system, the assumption was made that there would be no existing infrastructure for communications (such as a satellite network or local area network) for the first manned mission. This "starting from scratch" approach led to the evaluation of the following system possibilities in selecting a suitable communications network for an EMU and its data interface to interact with.

Table 9: Communications System Comparison

Communications Network	Infrared	Fiber Optic	Satellite	Reconfigurable Wireless Network
Supports navigation	✓	--	✓	✓
Mobile/Flexible	✓	--	✓	✓
Robust	✓	--	✓	✓
Allows easy repair	✓	--	--	✓
Practical setup	--	--	--	✓
Upgrade/Extendable	--	--	✓	✓
Flight Tested	--	Not large scale	✓	--
Max. Range	--	4km*	3900km	30km
Network Mass	--	175kg/km	About 8000 kg	About 180kg
Power Requirement	--	--	Solar/battery	10W (EMU)
Mars Dust Factor	Not good	Not good	--	Good

*100 fiber single mode loose tube cable without splicing

Assuming that the first manned Mars mission will have EVA range limited by either the distance the astronaut can travel on foot or by a small rover during a 4 hr (or a goal of 8 hr) EVA, a range of 30km for a surface communication system is adequate. If the astronaut is traveling at a quick clip of 5mph for 4 hrs (this is, for the goal 8hr EVA = 4 hr out and 4hr back), that is only a distance of 20 miles or about 32km. It is only necessary then, to have a ground-based communication system with a range of 30km for the first mission. This saves cost and mass on satellites.

A wireless radio system was selected from the options (Table 9) as being the most mass and cost efficient for a primary mission. In this RF system, the loss experienced by the carrier signal and the range to which it can be detected is dependent on the surrounding terrain. [37] The frequency at which the system operates is in the VHF band (100MHz to 450MHz). A non-mountainous terrain strewn with boulders is assumed. [38] The wavelengths corresponding to the 100 MHz to 450 MHz range are 3 m to 0.66 m. The signal power must never fall to less than 3 dB within the mobile receiver area regardless of the terrain. [34, 39]

Table 10: Revised Communication Design

Current EMU Design		Revised Design for Mars EMU	
External System	*Internal System*	*External System*	*Internal System*
No ground network	"Snoopy Cap"	Local Wireless Network	Virtual Retinal Display
Backup tethering system	Extravehicular Comm.	Mobile base/repeaters	Component relocation
	System control box with data interface display	Remote emergency locators	

The Reconfigurable Wireless Network (RWN) developed by Cornell Professor Zygmunt Haas and collaborator Siamak Tabrizi satisfies the requirements for a ground network. [36] This network is expandable for the increased demands that future missions may have. In addition, it minimizes the power required for transmissions, allowing for hand-held systems (or EMU systems) of a practical size and weight. There is no single point of failure since the RWN is organized in a flat configuration, all users with the same equipment rather than certain transmitters acting as centralized relay points.

Cellular phones on Earth rely on being within range of a base station at all times. Because Mars is not yet populated with base stations at regular intervals, cellular networks cannot be used. RWN can adapt to a changing network topology. This involves adapting to roaming base stations as well as compensating if one or more of them should fail. Haas and Tabrizi [2] propose having each mobile unit function also as a base station, negating the need for mobiles to remain within a certain radius of a fixed base station. [35]

The use of mobile base stations presents another challenge. On Mars, this allows the astronauts to communicate around obstacles and even out of line-of-sight of the lander base. Because of this mobility, a more sophisticated routing protocol that accounts for a changing network topology is required.

There are two general alternatives for routing a protocol. Proactive protocols continuously evaluate the network topology and update this information so that whenever a call needs to be made, the correct route can be immediately determined. Reactive protocols do a global search for the correct route at the time the call is requested. Clearly proactive protocols have a faster response time for calls that are made, yet require a constant flurry of information being sent to update the routes even when no calls are being made. Reactive protocols, because of their idleness, do not tie up the transmission medium until a call is required, yet may cause a significant delay in the establishment of a call.

For RWN purposes, Haas and Tabrizi propose a hybrid of these two extremes - Zone Routing Protocol (ZRP). This protocol performs proactive routing in the local neighborhood of a transmitter, but uses reactive methods for any long-distance communications. This limits the high traffic demands because the continual updates only occur in a limited area (relative to each transmitter), and also avoids the big delays associated with a purely reactive protocol.

Haas [38] has tested this protocol in simulation by using a 10 by 10 mile grid. With randomly distributed 'dark territories' that block communication, he simulated 51 mobile units. When each mobile unit was allowed to move at up to 50 mph, and given a communication radius of 5 miles, the percentage of calls blocked was nearly 0%. For Mars EMU communication, it is unlikely that more than 5 mobile units will be active at a time. Top speed will also be far under 50 mph. Using 10 Watts of power for communication purposes on the EMU is enough to provide a range of at least 10 km at a transmission frequency of 100 MHz.

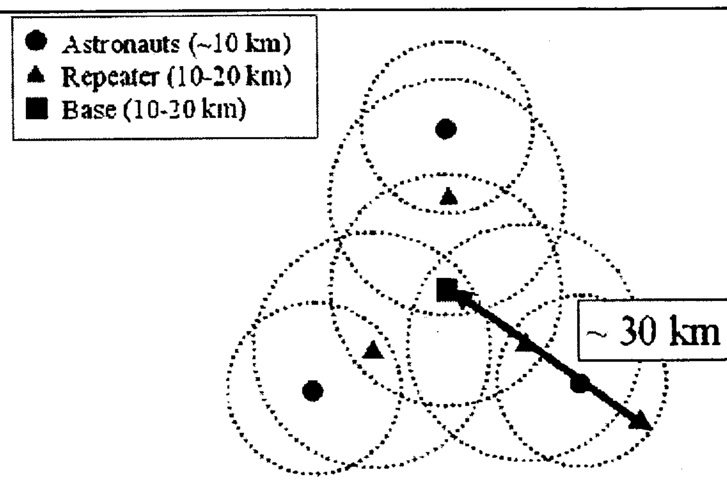


Figure 5: Triangular Ranging using repeater stations

While accommodation for mobility is a very attractive feature of ZRP, it naturally is not restricted to mobile units and can also take advantage of fixed stations. To this end, ZRP can also make use of deployable repeater stations that could be included to lengthen transmission range. A RWN is designed to handle more than our current mission needs and can easily be extended to handle the greater demands of a larger crew.

For decades, the U.S. military, federal agencies and scientific research groups have utilized repeater stations for their receiver-transmitter communication needs. Their scientific uses have proven especially viable in harsh conditions such as those found in Antarctica, where the ruggedness and isolation of the region make a robust communication network necessary. [40] There are two general varieties of repeater stations: active and passive. For use on Mars, an active station is best since passive stations tend to have high attenuation because it only reflects the signal received instead of amplifying it before transmitting, as an active repeater does. Also, a repeater station with duplexing capabilities will be useful, as it will allow the station to receive and transmit signals at the same time. [41] The ideal repeater station will be lightweight and capable of being used for navigation. Examples of commerical and military navigation systems include VHF Omni Range (VOR), Distance Measuring Equipment, (DME), or the common global positioning system (GPS). With the range restriction of 10km, three repeater stations can be deployed to create a triangular ranging area. During EVAs, astronauts will be able to relay signals between the repeater station via their transceivers. Figure 5 shows a sample network scenario with triangular ranging. The astronauts are out of range of the base but within range of the repeater stations, allowing relay to the base

Commercial repeater models like the Motorola GR900 typically have a range of 3-4km [42]. However, at free space attenuation, the reliability of such models is decreased because of their small design. [43] On Mars, a more robust and reliable system is needed such as the MastrIII repeater station designed by General Electric.[44] Such models are frequently used by military agencies because they guarantee not only a range of at least 10 km but are also able to withstand harsh and unusual conditions. [43] Most commercial repeater stations offer a variety of frequency ranges, in both UHF and VHF range. For the purposes of mobile communication on Mars, a range of 150 Mhz in VHF will suffice, as this is the range standard repeater stations operate on. [44]

Table 12: Repeater Stations

Commercial Models	Frequency Range (VHF)	Mass	Duplexer	Transmission Range
GE MastrIII	150 – 174 Mhz	50 - 60 kg	Yes	> 10 km
Motorola GR 300/900	136 – 174 Mhz	10 - 20 kg	Yes	< 5 km

The RF power output of average repeater stations is about 100 watts. [44] This can vary however, depending on the size of the station; the smaller, desktop models output 10 - 25 watts of power. [44] Through the use of a simple high-mounted antenna, this power output can be almost doubled. [43]

The drawback in using repeater stations is simply a matter of mass. Reliable models used in scientific fieldwork and government operations are about 60 kg each. [43] Setting up repeater stations can be the first step to creating an entire relay network on Mars. This approach should be considered first-generation and only necessary because of the reliable backup it offers. The primary alternative, satellites, are much more massive and not easily repaired. They cover a lot more area, but that may not be necessary on a first manned mission.

While the ground-based system we have proposed is ideal for initia l mission constraints, a satellite infrastructure would extend the range of communications and navigation functions to the level required for the significant scientific exploration proposed for future missions. Assuming the habitat delivery vehicle [1] provides a satellite in geostationary Mars orbit (GMO) for a continuous link to Earth, a single GMO satellite could provide reliable communications for a range of approximately one third the total surface area of Mars[70].

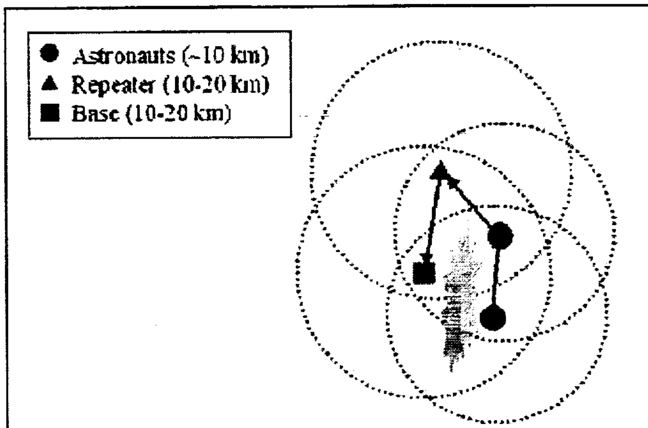


Figure 6: Sample Network Scenario with large obstacle

Extending the range of expeditions allowed by a ground-based infrastructure requires the addition of many perimeter stations as well as many active intermediate repeater stations to amplify communications and navigation signals. In addition, the presence of surface obstacles requires the redirection of ground-based communication and navigation signals with even more repeater stations. In the long run, repeater stations add considerably to the mission payload, the necessary power support (since each station must be individually powered), and the groundwork required to establish the network.

The current terrestrial GPS satellites weigh 1667 kg each and employ Delta II rockets for launch support into geosynchronous Earth orbit [68]. Adding three LPS technology satellites (the orbiter can act as the fourth) would increase the total mission payload by nearly 5000 kg. For the initial mission, with "roaming" expeditions probably limited to within 10 km, this payload mass greatly exceeds the mass associated with the repeater stations and navigational beacons of the ground-based architecture proposed.

One solution is a radical downsizing of the satellite components in order to provide navigation and satellite communications from GMO orbital altitudes without exceeding the practical payload mass limits. This is the most technologically demanding system proposed for Mars communications and navigation, requiring the most research and development for realization. A mass-based classification scheme has been established for small satellites:

Large Satellite: >1000kg
Small Satellite: 500-1000kg
Mini-satellite: 100-500kg
Micro-satellite: 10-100kg
Nano-satellite: <10kg

In order to match and compete with the payload of the initial ground-based communications network, the small satellites combined with their receiving/transmitting ground support equipment should be limited to a total on the order of 400-500kg.

Another design criterion for communication on Mars is a viable EMU data interface system for use during EVAs. The current space shuttle EVA suits used by NASA implement a communications system which consists of five parts: a headpiece, or "Snoopy Cap," a helmet-mounted video camera, a biomedical monitoring system, a control pad, and an extravehicular communicator that sits on top of the primary life-support system (PLSS). [45]

The Snoopy Cap is a fabric hood that can fit over the head of the astronaut during an EVA. The hood contains an earpiece and microphone as well as a link to an external video camera mounted on the helmet for one-way video transmission from astronaut to base. The earpiece, microphone, and video camera are connected through the suit's hard upper torso (HUT) to the extravehicular communicator via a pass-through. [45] The biomedical monitoring system functions so that both the astronaut and the base may monitor the astronaut's physical status. Electrocardiographic (EKG) information is transmitted in the same manner as audio and visual information: through the extravehicular communicator mounted atop the suit's PLSS. [45] Current designs for the extravehicular communicator used on space shuttle EVA suits are 30.4 cm long, 10.9 cm high, and 8.8 cm wide, with a mass of 3.9 kg. [45] The communicator utilizes two single-channel UHF transmitters and three single-channel UHF receivers for radio communications. In addition, the controls for the communications system are located on the front of the HUT in the suit's display and control module.

Hamilton Standard Space Systems International, Inc., the company responsible for the designs of the current space shuttle EVA suits, recommends base-lining a communications system that is similar to the current space shuttle EVA suit communications system, but with the radio communications components integrated into the HUT of the suit. [23]

The current communication system consisting of an audio transmitter/receiver, a video transmitter, and an EKG monitor operating under the current specifications would probably be sufficient for short-range use on the surface of Mars. But to do better long-range exploring, a more current communications support system, including networking, long-range capabilities, and navigation is needed. A visual "heads-up" display, much like the helmet-mounted displays (HMDs) used by military fighter pilots, would also be useful and keep the astronaut's hands free.[47]

Figure 7: Sample VRD View of Mars

Microvision Inc. has developed a specific application heads-up display for military aviators and ground troop commanders which uses a laser, monocle-size optical and tiny scanners to "paint" an image on the eye by moving the laser beam across and down the retina. Their screenless device, called a Virtual Retinal Display (VRD), allows the pilot or commander to see the surrounding environment while also accessing digital navigation cues and images that appear to float several feet away, even in bright sunlight. A single electronically encoded, low-power laser beam projects rows of pixels directly on the user's eye, creating a high-resolution, full motion image directly on the retina. [46]

VRD components are tiny and lightweight, allowing the device to integrate into small, highly portable packaging configurations, such as a helmet or hard upper torso of the suit. The light sources and scanners use very little power to project images on the retina. VRD is able to achieve a wider range of the color palette than any other display technology, modulating light sources to vary the intensity of red, green and blue light. It is capable of interfacing with head tracking systems, video sensor, and display controls which would enhance interaction in the Martian environment. Figure 7 shows a sample VRD display as seen by the astronaut. This includes a basic time and sol number count in the upper right-hand corner. In the lower right-hand corner is a biomedical monitoring table with heart rate and blood pressure data, as well as a distance marking from the nearest repeater or base station. The map allows for navigational tracking, with features such as the base, repeater stations and other astronauts clearly displayed. Navigation parameters are also marked. The left side of the screen includes gas level, temperature and pressure readings. At pr sent, new innovations in miniaturization are shrinking the hardware needed to generate the VRD. Tiny laser diodes will replace larger conventional lasers and handheld displays are being produced in microscopic size.

Table 13: Comparing Virtual Retinal Display to other visual display components

Display Source	Resolution (Pixel Size)	Luminance	Color	Weight	Power Consumption
VRD	.5 Micron	Unlimited brightness	Full color with no loss in resolution	Low	Low
CRT (Cathode Ray Tube)	25 Micron	Up to 1,000 fL	Only with sequential	High (with cabling)	High
AMLCD	12 Micron	Poor – backlight dependent	Yes in 6 VGA resolution	Low	High with backlight
Ferro-Electric LCD	13 Micron	Poor - 20fL	Yes with field sequential LEDs	Low	Low
Thin Film Electro-Luminescent	24 Micron	Poor - 60fL	Yes with field sequential shutters, small color depth	Low	High
Field Emission Display (FED)	16 Micron	300 fL	Yes with low resolution	Low	High
AMLCD on CMOS	12 Micron	Poor - 30fL	Yes	Low	Low

During planetary exploration, there are considerable risks because of the unfamiliarity of the terrain. Simple navigation and emergency-alert systems can be deployed for backup. There are many available methods on Earth, from the Cospas-Sarsat Personal Locator Beacon (PLB) [48,49] system to the avalanche beacons [50] which have been recommended by the International Commission for Alpine Rescue (ICAR). [51] Because it is impractical to assume immediate satellite coverage on Mars due to mass and cost restrictions, a light, simple, low-power homing beacon would be ideal. To achieve the best blend of beacon characteristics, a combination of the Cospas-Sarsat and avalanche beacons should be used.

PLBs have a 406MHz digital or 121.5MHz analog satellite signal as well as a homing beacon. Although their efficiency would increase through satellite use, 121.5MHz homing beacons are viable alone. They have a range of 3-5km, and, if necessary, can be sent in Morse code to include more information. [52] Avalanche beacons are light (230 grams), small (130 x 80 x 25 mm), and have a working life of about 250 hours on 3V batteries. They have high-impact strength and shock resistant casing, can operate between -30°C and 50°C, and can be connected to an earphone, allowing for audio transmission. However, avalanche beacons only have a range of 80m. [50]

An ideal beacon would combine the PLB homing beacon with an avalanche beacon. With this combination, a 3-5 km radius could be covered to locate an astronaut. In an emergency, a rescue team would need to come within 45m of the astronaut and then the avalanche beacon could pinpoint his location with an accuracy of 70cm. [53] This simple beacon could be triggered either manually or automatically (by shock), and would send a signal out which could be received by both other astronauts and the base station. All EVA suites should have both a transmitter and a receiver to allow the fastest possible astronaut rescue. An accompanying rover/cart should also have a receiver like the Cospas-Sarsat Repeater Unit to relay the message back to the base. [54]

2.3 Power

The power section compares the practicality of current portable power options and focuses on a direct methanol liquid feed fuel cell (DMLFFC) as the main power source for a Mars EMU with a small battery as backup. Reactant production, reactant storage, fuel cell materials and thermal distribution are analyzed.

The power system for the Mars EMU will be required to satisfy the maximum possible power demand over the duration of a 4 hour EVA (goal of 8 hours). While the Ag-Zn battery currently used to power the shuttle EMU is capable of supplying approximately 70 watts at 17 volts [1], the design proposed here requires ~150 Watts. A new portable, reusable power source is needed to satisfy functional requirements for a Mars EMU:

- 150 W (at 18V)
- Low mass
- Sealed from μm dust
- 4 hr supply (8 hr goal)
- Maximum 333 EVAs use for a 500 day surface stay

The values below break down the total power draw of a Mars EMU as estimated by engineers at Hamilton Standard [2]. Note that these values are only rough estimates that use as a baseline the current shuttle EMU subsystem power requirements.

Table 14: Mars EMU Subsystem Power Breakdown (estimates by Hamilton Standard)

Subsystem	Power required (estimated)
Communications	10 W
Cooling fluid (LCVG) circulation	30 W
Life support	5 W
Lighting / Ventilation	30 W
Instrumentation	5 W
Active Control Valves	5 W
Control / Monitoring System	5 W
Dynamic H_2O separation	10 W
Heaters	35 W
Information Display	10 W
Total Power	145 W

Portable power technology candidates for use on a Mars EVA suit are numerous. However, many are not desirable because they have low power/mass ratios, cannot be reused over many cycles, or are potentially toxic to the astronaut. Nuclear power is quickly ruled out as a portable power candidate, as is solar power because to meet the power requirement, more surface area of solar array would be needed than there is surface area of an EMU.

Table 15: Portable primary power options

Type	Power Profile	Advantages	Disadvantages
Solar	~ 50 W/m^2 (Mars)	Power density Unlimited power	~ 3 m^2 for 150 W Fragile Dust accumulation
Battery			
Nickel Metal Hydride	~ 55 Wh/kg	~ 3000 recharges	~ 10.9 kg
Lithium-Ion	~ 250 Wh/kg	~ 2.4 kg for primary	Not flight qual. for EMU
Silver-Zinc	~ 90 Wh/kg	~ Flight qualified	~ 6.7 kg, ~100 recharges
Nuclear	High	Power density	Not flight qual. for EMU
Fuel Cell			
H_2-O_2	~ 300-600 Wh/kg (achieved)	Power density Low mass	H_2 storage
CH_3OH	~ 500-1000 Wh/kg (expected)	Power density ~ 1kg mass Mars resources	Still being developed

The most reasonable portable power options are of the battery or fuel cell type. With fuel cells, the logistics that are involved with using a hydrogen-oxygen fuel cell, namely the difficult provision and storage of the hydrogen, prohibit its use as an on-back portable power supply for this application. Batteries seem like a viable solution, with lithium-polymer or lithium-ion batteries as practical choices. These would require electrical recharging, and if they are used as primary on-back power supplies, it would be difficult to get NASA approval for them because lithium is volatile should it come in contact with water, and the cells would be in close contact with a water-based human. It would be easier to get approval for a smaller lithium ion battery to be used as backup. The current silver-zinc (Ag-Zn) battery is practical for use in a micro-gravity EMU, but to provide the additional primary power that would be needed on a Mars EMU would significantly increase its mass (at 283 A h/kg).

The HEDS reference mission mentions batteries as a possible power supply for the EMU. Batteries can be recharged a limited number of times while a fuel cell may produce electricity as long as fuel is supplied. Extra batteries would need to be brought to meet mission duration and backup requirements. A silver-zinc battery powers shuttle EMUs and must be stored dry, filled, sealed and charged prior to flight. Ag-Zn batteries are dense and impractical due to mass constraints. Fuel cells are lightweight compared to the required number of Ag-Zn batteries.

It is desirable to use indigenous resources. Mars has an atmospheric pressure that is 1% of Earth's and consists of 95% carbon dioxide. [60] The HEDS reference mission outlines the use of this carbon dioxide to produce methane for the Earth return vehicle (ERV) propellant using the Sabatier reaction. [1] Robert Zubrin has developed a working model in Mars-like conditions. [61] In order for methane production to be practical, hydrogen must be sent to Mars since no significant source of hydrogen is known to exist on Mars. Conceptual missions have a methane production plant and an earth return vehicle sent to Mars a year before any crew is sent. The plant produces methane from the transported hydrogen and carbon dioxide in the Mars' atmosphere. Once enough fuel is confirmed to be available for a crew to return to Earth from Mars, a crew may be sent. Once methane is available, methanol may be produced by modified standard-industry processes [61] or new processes developed for the automobile industry. This small amount of methanol can then be used as fuel for DMLFFCs. It is possible to synthesize methanol directly from CO_2 and water as shown in Figure 8. [1]

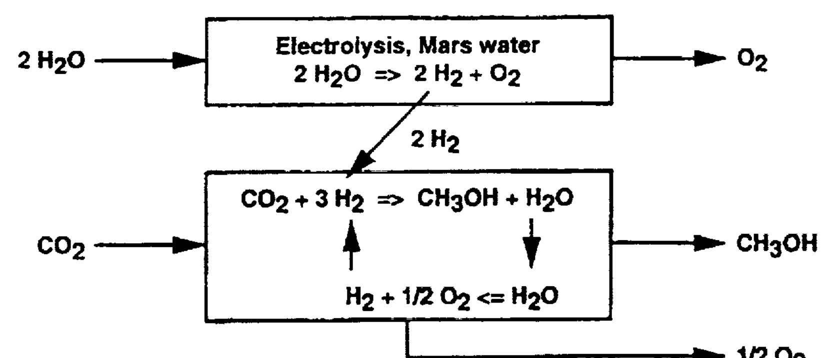


Figure 8: Methanol synthesis from CO_2

A fuel cell stack is built from a number of cells arranged in series. Each cell works as follows: oxygen is pumped into the cathode side, and a methanol/water solution is fed into the anode side, where the anode catalyst strips hydrogen from the methanol. The catalyst atomizes and then dissociates the hydrogen into protons and electrons. The electrons become the generated electricity. This anode reaction produces carbon dioxide, which can be collected or vented out of the system. The protons are then conducted through the membrane-electrode assembly (MEA), made of the anode, Proton Exchange Membrane, (PEM) and cathode, to the cathode side, where they react with the atomized oxygen and incoming electrons to form water. The water can then be recycled or stored for later use. Each cell is separated by a bipolar plate that acts as both the anode for one cell and as the cathode for the neighboring cell. These plates have channels in their surface which distribute the reactants across the membrane assembly. [62]

The anode partial reaction is: CH3OH + H2O à CO2 + 6H+ + 6e-.

The cathode partial reaction is: (3/2)O2 + 6H+ + 6e- à 3H2O.

The overall cell reaction is: CH3OH + (3/2)O2 à CO2 + 2H2O. [63]

The PEM is a polymer film that blocks the passage of gases and electrons but allows hydrogen ions (protons) to pass. [64] Current DMLFFC technology uses DuPont's commercial Nafion, a perfluorinated ionomer with thechemical composition below:

$$-O-CF_2-CF(CF_3)-O-CF_2-CF_2-SO_3H. \quad [65]$$

Nafion exhibits relatively good proton conductivity but also allows some methanol to cross over from the anode to the cathode side. Researchers are learning about PEMs. Sen, et al. compared Nafion-115 to Dow's membrane material, PFSA-800, to learn in what way water content affects membrane conductivity. They found that resistivity of the membranes decreases sharply with temperature up to 60°C, reaches a minimum near 80°C and then increases up to 100°C. The Dow membrane has a lower resistivity than Nafion-115 over the entire range. They also found that water content has a significant impact on membrane resistance. The resistivity decreases by approximately two orders of magnitude between 0 and 100% [relative humidity] at room temperature. [65]

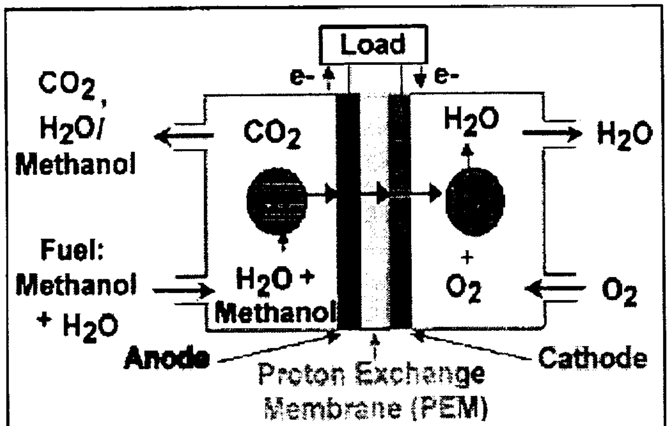


Figure 9. DMLFFC Schematic

The anode catalyst is responsible for stripping hydrogen from methanol, and the cathode catalyst reduces oxygen. The highest performing anode material is the one that demonstrates the highest activity for methanol electro-oxidation. Anode and cathode catalysts may be supported or unsupported. A support is a structural backing such as porous carbon which is transparent to the conduction of protons. Chu et al. tested unsupported alloys of platinum (Pt) and ruthenium (Ru) of different compositions and at different temperatures for use as an anode catalyst. They found that Ru was inactive below 25°C but became active from 40 to 80°C, and that for a voltage of 0.3V, a 50:50 composition provides the best results on an electrode [geometric] area basis. [65] Subsequent research by other groups using various methanol concentrations has confirmed a 1:1 ratio for Pt-Ru as optimal.

Bipolar plates separate individual fuel cells and distribute fuel or oxygen to their active surface areas. Individual fuel cells must be arranged in stacks to achieve a usable voltage and current. The plates function as anode to one cell and the cathode to the neighboring cell, allowing efficient packing of cells. Borup and Vanderborgy outline design criteria for plate materials. The design constraints they consider include electronic conductivity, gas diffusivity, chemical compatibility, cost, weight, volume, strength, and thermal control. [66]

Table 16. System mass design

Stack Size (# cells)	30	Mass of PEM (g)	12.75
Cell Length (cm)	9	Mass of Anode (g)	9.22
Cell Width (cm)	5.25	Mass of Cathode (g)	9.22
Stack Pwr (W)	153	Mass of Plates (g)	790.97
Stack Vltg (V)	18	Sys. Mass (g)	822.16
Pwr Density (mW/cm²)	150		
Voltage per cell	0.6	Sys. thickness (cm)	6.60
Pwr rqmnts (W)	*150*	Sys. volume (cm³)	311.85
Voltage rqmnts (V)	*18*		

Fuel cells in a series stack add their voltages and power outputs. Stack output power is a function of stack voltage and current density, but stack output voltage depends on the sum of individual cell voltages. Hence, minimizing the number of fuel cells in the stack can be accomplished only by maximizing the voltage output of each cell. There are three main ways to accomplish this: (1) increasing cell operating temperature, (2) using pure oxygen at the cell cathode, and (3) careful construction of the cell's membrane and electrode assemblies.

Table 17: Fuel Cell Specifications vs. EMU Needs

Fuel Cell Specs:*		**EMU Needs:**	
Output Voltage:	0.6 V	Power Req'd:	150 W
Current Density:	150 mA/cm²	Voltage Req'd:	18 V
Power Density:	90 mW/cm²	Required Duration:	4 hours
Overall Efficiency:	35 %	Goal Duration:	8 hours

* At 60 °C, operating on Earth air at 20 psig and a flow rate 3 to 5 times stoichiometric. [80

]Increasing the cell operating temperature from 60°C to 90°C can increase cell output voltage by almost 50% over the data given above. However, there are three problems with this approach. First, a higher operating temperature increases the rate of reactant crossover in the fuel cell, resulting in a loss of output current and a drop in fuel cell efficiency. [80] The cold environment on Mars will make it difficult to insulate a cell and guarantee an operating temperature of 90°C. The most significant concern is that a high operating temperature increases the thermal stress on fuel cell systems. This leads to early dehydration of each cell's PEM, rendering the cells inoperative and useless. [79] Such failure compromises crew safety and also requires a large reserve stock of replacement fuel cells.

Using pure oxygen at the cell cathode to react with methanol at the anode can increase cell voltage by 15% to 20%. [84] On an EMU, pure oxygen would be used anyway. This method of increasing cell voltage is probably the easiest and presents no major problems. The DMLFFC was originally designed to react liquid methanol and pure gaseous oxygen directly, but scientists at JPL used a high-flow air supply to provide oxygen for fuel cell testing since oxygen is present in Earth's atmosphere in significant quantities. Careful preparation of the membrane-electrode assemblies with catalyst material can also contribute another 15% to 20% increase in fuel cell performance. [84] This is a fairly time-consuming process and increases fuel cell cost, but it need only be done once, before the cell is brought into operation for the first time.

A combination of pure oxygen usage and catalyst preparation can thus provide an overall increase in cell output voltage of 30% to 40% to about 0.63 V, which leads to a proportional decrease in the number of cells required for the EMU's power stack. In addition, increasing cell output voltage also leads to an increase in cell output power. To satisfy the EMU power requirements, fuel cells are rated at 0.6V each at a power density of 150mW/cm

In addition to providing on-back power for the EMU, a fuel cell stack can generate a significant amount of waste heat. With current DMLFFC efficiency, approximately two-thirds of the energy potential of the methanol is unused. Half of this lost energy is dissipated in the form of electrochemical efficiencies and heat energy needed to maintain cell stack temperature. The other half of the unused energy is dissipated as waste heat to prevent undesirable increases in stack temperature and power fluctuations. [78, 86] The waste heat generation of the stack is thus roughly equal to its electrical power generation, or 150 W.

The fuel cell stack will be insulated against the Mars environment to maintain its temperature and the waste heat must be actively transferred out of the stack to prevent a heat buildup. Because the EMU will operate in a cold environment, the DMLFFC waste heat can be recycled inside the EMU to provide an auxiliary heat source for the astronaut. A heat exchanger may be used to transfer the waste heat to the liquid cooling/heating ventilation garment. This heat exchanger can be made of aluminum and will conduct heat to feedwater from the LCVG, thus increasing the water temperature and adding heat to the astronaut's body. Having an in-suit active heat source will reduce the need to don and doff thermal insulative overgarments intended to reduce heat loss. This will give astronauts greater mobility in surface activities since these garments are very bulky, reaching up to four inches thick. [23]

DMLFFC reactions will consume methanol and oxygen to produce usable electric power. The amount of the reactants consumed is dependent on the duration of the EVA and the average power produced on the EVA. These two factors can be combined and expressed in terms of energy with units of watt-hours. In general, an average EVA duration of four hours will require 600-800Whr, with an eight hour EVA requiring 1200-1600Whr.

In order to determine the amount of reactants consumed in producing these amounts of power, the baseline of a current DMLFFC developed at Giner, Inc is used as reference. At Giner, with the stack operating conditions of 0.45V/cell, 100 mA/cm^2 and 60°C, a 0.5 M methanol solution has been shown to maximize efficiency. Under these conditions, DMLFFCs will consume methanol at a rate of 1.4×10^{-2} moles per watt-hour and oxygen at a rate of 2.1×10^{-2} moles per watt-hour [57]. In addition, the stack inefficiency known as "cross-over" will consume additional methanol and oxygen at rates approximately 30% of those listed above. Finally, due to imperfect reactant utilization, increased oxygen flow rates oxygen will require quantities on the order of two times those given below.

To maintain maximum reactant utilization and efficiency with a 0.5 M methanol solution while meeting the power requirements of the EMU (including cross-over), EVAs will require the amounts of methanol listed below introduced to the closed loop anode supply evenly over the duration of the EVA.

Table 18: Reactant Consumption Amounts

EVA Duration	Methanol Consumed	O_2 Consumed
4 hr	0.40-0.54 L	0.61-0.81 L
8 hr	0.81-1.08 L	1.22-1.62 L

Note that while the stack operating conditions may vary between those listed above and those used on Mars, the interdependence between the reactant consumption rates and the operating conditions is stable enough that the rates given above would not change radically. The goal here is instead to show that only moderate amounts of methanol and oxygen are required in producing the required power.

In addition to making sure that the fuel cell is reusable and can meet average power needs, it is necessary to make sure that the fuel cell design does not continuously tax the fuel cell at maximum – this creates great stress on the system to be continuously providing peak power. A small, nontoxic battery used in parallel with the fuel cell will help get the cell heated up to start as well as provide peak power requirements so that the fuel cell is not operating at max stress.

A stack of 30 cells with area 9.00 x 5.25 cm^2 is required to achieve the 150W and 18V requirement. System mass and volume was estimated given densities of DuPont's Nafion for PEM, platinum-ruthenium for the anode catalyst and platinum for the cathode catalyst. Aluminum is used for the plates. The total volume is less than 3.5 L and mass is less than 4kg, which are reasonable values for a power system, compared to the current EMU Ag-Zn batteries, which would require 21.8 kg to achieve 150 W for an 8 hour EVA.

Table 19: Current best estimates (CBE) volume and mass for a sample design

(margin = 25%)	CBE volume + margin (L)	CBE mass + margin (kg)
FC dry	0	1.03
methanol	1.25	1.08
water		
-PEM uptake	0.06	0.06
-storage	0.31	0.31
cables, tubes, pumps, storage	1.25	1.25
Total	3.28	3.72

Power options include solar, nuclear, battery, and fuel cells. A fuel cell system was chosen as the best candidate after disqualifying the other options. Solar power requires ~340m^2 to produce 150 Watts on Mars and is highly variable. Nuclear power has many political and flight qualification problems. Battery technology is another practical option and is used in current EMU systems where Ag-Zn mass is not a limitation in micro-gravity, but it is on Mars. Lithium-ion batteries are the best performing battery systems with low weight, high reliability and long lifetime, but flight qualification will be difficult due to the chemical nature of lithium. For the near term, fuel cells, direct methanol liquid feed fuel cells in particular, are chosen as the most practical as the portable power source for the Mars EMU.

3.0 Conclusions from above discussions

3.1 Life Support

Modifications upon the life support component of the EMU for use on the Mars surface include cryogenic oxygen storage, solid amine carbon dioxide removal, a modified dehumidifier, and a regenerable nonventing thermal sink. Further development should focus around refinement of a small scale cryogenic oxygen system, and mass and power reduction in the carbon dioxide, humidity absorption, and thermal system.

3.2 Communications

Through the use of a reconfigurable wireless network using repeater stations, the astronauts will have a dependable communication system within a reasonable range of the base. Emergency locator beacons, with independent navigation capabilities, will provide back up should the primary communication system fail. An advanced communication display, the virtual retinal display, improves audio and video transmissions during EVAs. In the future, satellite networks can provide room for growth and expansion.

3.3 Power

A DMLFFC adequately meets all of the portable power needs of a Mars EMU, and should be used in parallel with a battery for peak loads to avoid overloading and stressing the system. The "waste" heat generated by the fuel cell can be harnessed to perform the functional task of warming the EMU. The use of methanol, a non-toxic easily generated mission resource in small amounts as power makes use of Mars resources and reduces the amount of mass to be transported from Earth.

Future research in designing an electrical power system for an EMU with application on Mars, many of which are funded by the drive to utilize DMLFFCs in the automotive industry, would include developing the following:

- An efficient means of producing methanol from methane at lower temperatures on Mars.
- Improved membranes (JPL & USC) in an effort to improve both reactant utilization and efficiency while reducing cross-over within the DMLFFC.
- Even lighter and cheaper materials for use in the DMLFFC by optimizing the fabrication process of the PEM, MEA and plates.
- A method for maintaining the correct CH_3OH concentration at the anode.

- An improved heat exchanger for transfer of heat from DMLFFC.
- A complete model of the DMLFFC and power distribution system.
- A prototype for field testing under simulated Martian conditions.

3.4 Meeting Functional Requirements

Table 20: System functional requirements checklist and mass tally

EMU System Component	Proposed Solution	Meets Functional Requirements	CBE Mass
Gas Exchange	Cryogenic Oxygen Storage System Back Up Oxygen Tanks Solid Amine Desorbed System	✓ ✓ ✓	4.6 kg 6.4 kg 2.9 kg
Thermal Regulation	Heat Exchanger	✓	7.35 kg
Radiation Protection	No additional mass necessary	✓	--
External Communications System	Reconfigurable Wireless Network	✓	2 kg
Internal Communications System	Virtual Retinal Display	✓	3 kg
Backup Communications System	Hybrid Beacon	✓	0.25 kg
Primary Power System	Direct Methanol Liquid Feed Fuel Cell	✓	3.75 kg
Backup Power System	Lithium-ion battery	✓	0.3 kg
		TOTAL MASS	**30.55 kg**

Combining the system components analyzed above with their current best estimate masses (margin included) puts us only 2.55kg over the allocated mass. This difference will probably be balanced out as the newer technologies improve and CBE margins decrease.

The proposed system components for use in a Mars take the NASA HEDS Reference Mission requirements and also the physical characteristics of Mars into account. The range of technological readiness levels for these components is large, some have been flight-tested or are on their way to be, and some are just making their first commercial debut. Development and prototyping of all of these components to integrate on a Mars EMU will, as with any space mission, take years. However, the time scale for the development and testing of most of the above technologies is in step with the desire to send humans to Mars within the next 10-20 years. In order for humans to reach yet another once-impossible goal, true planetary exploration, research and development for a Mars EMU must begin promptly and proceed unhindered.

4.0 Outreach

The Cornell 1999 HEDS-UP Team committed to two kinds of outreach, through the media and also community service/educational outreach. Due to the cohesiveness of the team and the general interest sparked by the topic of Mars Exploration, both efforts were a great success.

4.1. Media

After our proposal to participate in the 1999 HEDS-UP competition was accepted, the Cornell and local media responded with great interest. The following articles were generated that we are aware of, plus one pending article in the next biannual issue of the Cornell Engineering (Alumni) Magazine.

- The Cornell Daily Sun, 12/4/98, p. 9
- Cornell Chronicle, 12/10/98, p. 1
- The Times-Independent (Moab, Utah), 12/24/98 p. A5
- The Ithaca Journal, 12/24/98

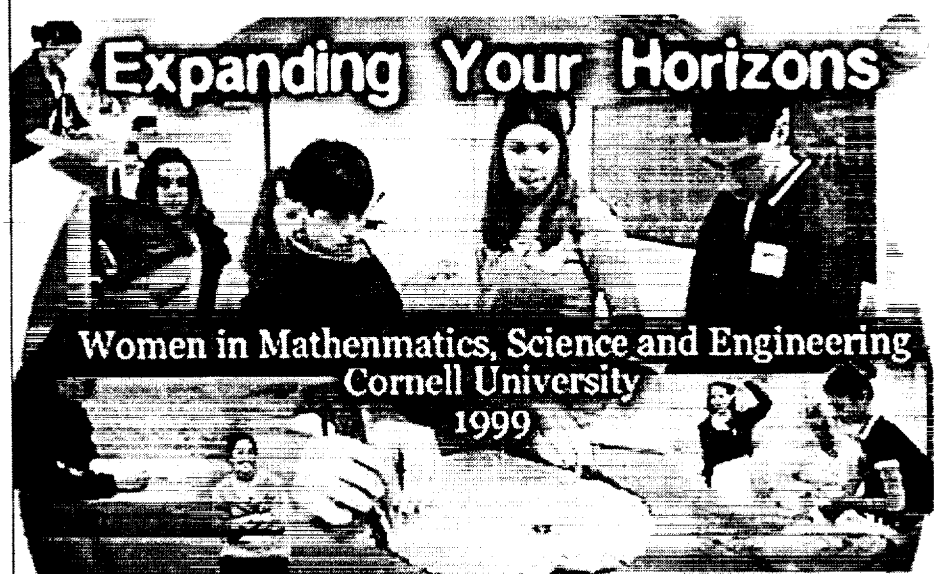
4.2. Educational Outreach: Kaboom! Mars Volcanoes, Expanding Your Horizons, April 10, 1999

(Workshop modified from NASA's Destination: Mars Teacher Activity Packet, printed by the Earth Science and Solar System Exploration Division, Johnson Space Center).

The team hosted a workshop for the Expanding Your Horizions program to help renew the interest of middle school girls in math and science. We hosted two sessions of ten middle-school girls each, plus parents. Each session began with a brief history on Mars and volcanic action before moving quickly into the fun stuff – modeling volcanoes using layers of playdough and eruptions of vinegar and baking soda. Multiple eruptions were "fired" off and the flow mapped with a layer of playdough. Once the mapping was complete, groups traded volcanoes and dissected them by small parts in to try to figure out what the mapping looked like. More than 14 team members came during the day to interact with the girls.

One member of the team was a guest lecturer at Lawrence Middle School in Lawrenceville, New Jersey. The lecture covered a brief history of Mars, the current climatic conditions, and prominent geological features on Mars. The lecture concluded with a small project where groups of students were given a mission objective and had to design an instrument to accomplish the task.

Over the past few months the Cornell HEDS-UP team has enjoyed researching missions and learning about Mars and sharing what we have discovered with others.



Digital Collage Photograph of the Kaboom! Mars Volcanoes Workshop

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MARS SCENARIO-BASED VISIONING: LOGISTICAL OPTIMIZATION OF TRANSPORTATION ARCHITECTURES

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ABSTRACT

The purpose of this conceptual design investigation is to examine transportation forecasts for future human missions to Mars. Scenario-Based Visioning is used to generate possible future demand projections. These scenarios are then coupled with availability, cost, and capacity parameters for indigenously designed Mars Transfer Vehicles (solar electric, nuclear thermal, and chemical propulsion types) and Earth-to-Orbit launch vehicles (current, future, and indigenous) to provide a cost-conscious dual-phase launch manifest to meet such future demand. A simulator named M-SAT (Mars Scenario Analysis Tool) is developed using this method. This simulation is used to examine three specific transportation scenarios to Mars: a limited "flags and footprints" mission, a more ambitious scientific expedition similar to an expanded version of the Design Reference Mission from NASA, and a long-term colonization scenario. Initial results from the simulation indicate that chemical propulsion systems might be the architecture of choice for all three scenarios. With this mind, "what if" analyses were performed which indicated that if nuclear production costs were reduced by 30% for the colonization scenario, then the nuclear architecture would have a lower life cycle cost than the chemical. Results indicate that the most cost-effective solution to the Mars transportation problem is to plan for segmented development, this involves development of one vehicle at one opportunity and derivatives of that vehicle at subsequent opportunities.

NOMENCLATURE

ΔV	velocity increment
AHP	Analytic Hierarchical Process
CER	cost estimating relationship
CHEM	chemical
DDT&E	design, development, testing, & evaluation
DRM	design reference mission
DSM	Design Structure Matrix
EELV	evolved expendable launch vehicle
ETO	earth-to-orbit
GA	genetic algorithm
HEO	highly elliptical orbit
HLLV	heavy lift launch vehicle
HRST	highly reusable space transportation
IMLEO	initial mass in low earth orbit
Isp	specific impulse, s
LCC	life cycle cost
LEO	low Earth orbit
LH2	liquid hydrogen
LOX	liquid oxygen
MER	mass estimating relationship
MR	mass ratio, inbound and outbound
M-SAT	Mars Scenario Analysis Tool
MT	metric ton
MTV	Mars transfer vehicle
NTR	nuclear thermal rocket
SBV	scenario based visioning
SEP	solar electric propulsion
TFU	theoretical first unit
TMI	trans-Mars injection
T/W	thrust-to-weight

I. INTRODUCTION TO THE STUDY

Current thinking on Mars seems limited in examining the links between the depth of available transportation vehicles with a breadth of future scenarios. Analyses such as NASA's Design Reference Mission (DRM) or the Mars Direct Mission can only be used as starting points. In trying to send the initial human mission to Mars, these concepts have not appropriated a long-term philosophy. These visions cannot adequately deal with the inherent problems of transportation systems that are to be used repeatedly in the next millenium. These designs avoid examining the possible synergies between how often a society demands to go to Mars and the transportation methods available to implement that demand.

The inquiry presented here looks into the bimodal shipping arrangement inherent in the Mars transportation market: from Earth-to-orbit (ETO) and from Earth orbit to Mars. A conceptual design method is created that can integrate all aspects of the space transportation infrastructure for going to Mars. Planning space transportation systems for the future in the conceptual design phase requires a method to evaluate how each envisioned future changes the final design. In each imagined future it should be possible to see how cargo requirements actually change the development cycle of the transportation system itself. Specifically, how does cargo demanded affect the payload capability of the "truck" that will be developed to transport that cargo.

II. PROBLEM APPROACH

A new design approach to address the deficiency defined above is based on Mars Scenario-Based Visioning (SBV). SBV is a philosophy that tries to define the future according to various drivers. In essence, visions of the future help drive one to obtain specific scenarios. A process is developed that can utilize these envisioned scenarios, along with availability projections of future launch vehicles and Mars transfer vehicles (MTVs), to determine the cost-conscious combination of such vehicles to meet that scenario requirement.

Large cargo delivery to another planet such as Mars is a problem of transportation logistics. Previous studies indicate that more than 40% of the total cost associated with going to Mars stems from the Earth-to-orbit and TMI phases, with the rest of the cost coming from habitation, operations, and a small percentage for Earth return. This study focuses only on the one-way transportation problem to Mars, specifically up to trans-Mars injection (TMI). The payload at TMI must contain its own equipment for orbit capturing at Mars and for all post-TMI transportation, therefore the costs calculated in this study are only for the transportation segment to TMI.

The timeframe examined in this study spans from 2011 to 2031. Since optimum orbital alignment does not occur for 2.1 years between Mars and Earth the cargo is sent only in those ten launch opportunity years between 2011 and 2031 (2011, 2013/4, 2016, 2018, 2020, 2022, 2024, 2026, 2028, and 2030/1).

With all of these assumptions in mind a process is designed to simulate and optimize a bimodal transportation system to Mars (see Figure 1). At the center of the figure is a new simulation tool called M-SAT (Mars Scenario Analysis Tool), specifically designed for this study. M-SAT is a spreadsheet-based simulation that utilizes inputs of scenario forecasting, ETO vehicle databases, and MTV databases. Internally M-SAT contains modules to calculate the following: vehicle flight rate combinations, vehicle transportation costs, and in-space operations costs. Attached to these modules is a contracted genetic algorithm optimization routine that selects the optimum combination of ETO vehicles and MTVs per year to reduce overall life cycle cost (LCC) of the transportation system.

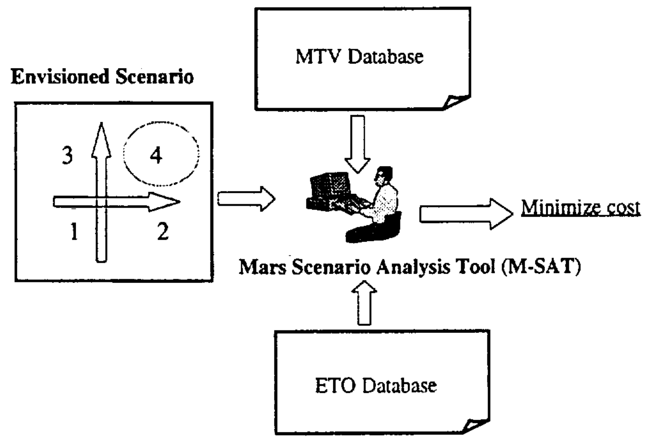


Figure 1: Process Overview Visualized

II.1 Scenario Visioning and Populating the Input Databases

Mission scenarios and available vehicles are what defines the future of Mars transportation systems, therefore these are the inputs to the M-SAT simulation. Envisioning these scenarios and populating the databases is then the first step towards creating a valuable simulation tool.

II.1.1 Scenario Definition and Visioning

The foundation of Scenario-Based Visioning as applied in this examination can be seen in Figure 2. The horizontal axis represents increasing social and political will of going to Mars and the vertical represents reduced transportation costs. In a scenario where the social and political will is low one can imagine a future in which only robotic exploratory missions would be developed. In a future where social and political will for going to Mars is high, and yet high space transportation costs exist, only a limited one-time manned mission might be advanced, called "Flags and Footprints." In another vision of the future where these particular costs have been reduced, missions might increase to point near or slightly above those envisaged by NASA in the Design Reference Mission (DRM). Colonization may be a likely future if both the collective will to go to Mars increases and transportation costs decrease.

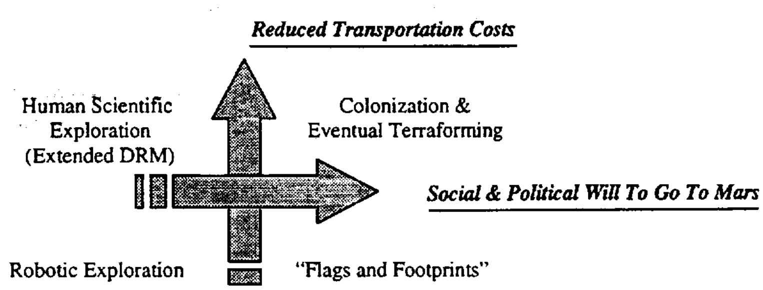


Figure 2: Scenario-Based Visionsing for Going to Mars

Payload requirements in metric tons for each Mars launch opportunity (approximately every 2.1 years) are assigned for each scenario (see Figure 3). The payload requirement per year parameter reflects the fact that for each scenario a certain TMI payload would be demanded. This requirement is essentially a proxy for the societal demand to go to Mars during that year. These payload requirements are the inputs to the simulation from the scenarios.

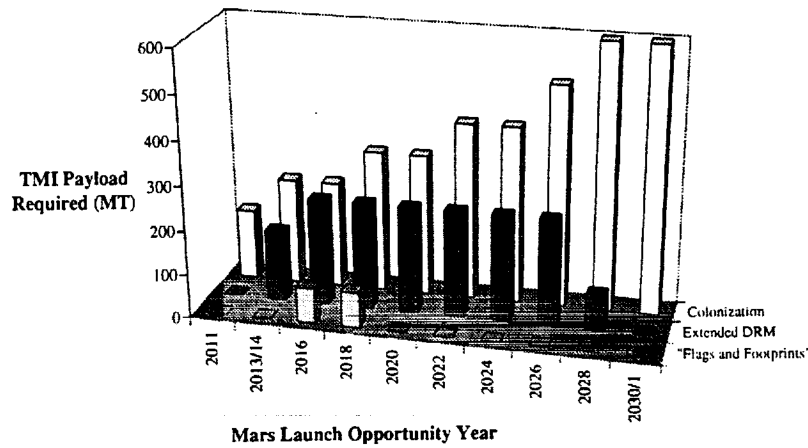


Figure 3: TMI Payload Forecasts for Three "Envisoned" Mars Scenarios

II.1.2 Populating Input Databases

The M-SAT simulation consists of two input databases: one for the ETO vehicles and one for the MTVs. Similar to the scenario definitions, the vehicles used in the databases can define a future scenario. Any vehicles deemed important to a Mars transportation system can be input into the database. Each database contains inputs that describe the vehicle type, availability, reusability, payload capacity, weight, and cost.

Each of the inputs has a different effect on the scenario envisioned. For instance, vehicle type indicates to the simulator whether certain vehicles are derivatives of one another. This is useful in obtaining learning curve effects from production of similar vehicles. Both availability and payload capacity are input for each opportunity to simulate possible performance upgrades and operational improvements in a vehicle, which can be significant over a twenty-year period. Reusability is important to the MTV database because a reusable vehicle will be able to repeat its mission at the next opportunity with only marginal refurbishment. The ETO and MTV databases contain a maximum of eight and three vehicles, respectively. The databases are populated with both existing designs and conceptual designs. The conceptual designs used for the databases are in-house designs.

Creating these in-house designs requires the use of very intensive design processes optimizing for minimum cost. Detailed designs are necessary because specific values are needed in order to estimate the input parameters to the database. An example of this is occurs in vehicle cost estimation, which involves relationships that are dependent on the particular weight of vehicle components. These next two sections will chronicle the designs and design processes in the databases and introduce information that is useful for deciding on the input parameters in each database. These designs are not meant to be a result of this study, designing them is merely part of the process of generating the inputs.

II.1.2.1 Earth-to-Orbit Vehicle Database Definition

The task of the ETO transportation designs is to populate a database of vehicles to be used as inputs to the M-SAT simulation. In order to fully explore the possibilities for ETO transportation, an array of vehicles spanning a wide payload spectrum is needed. By compiling this spectrum the question of what is the "best" way of launching a Mars transfer vehicle can be answered with more confidence. These ETO designs use existing, interim, and future designs to meet the demands of exploring low cost options for a variety of payloads to low Earth orbit or LEO (400 km x 400 km).

The purpose of the database for the ETO team is two-fold. First, it is intended to serve as a test for the trial runs of the scenario vision; second, it is intended for future users who do not have their own vehicle designs. The beauty of the database lies in the fact that it is easily amendable. If a new design is created, it can be placed in the database and accessed as an ETO transportation option. Introduced below are possible launch vehicle candidates come from existing, interim, and future vehicles that can launch more than 10 MT to LEO (see Figure 4).

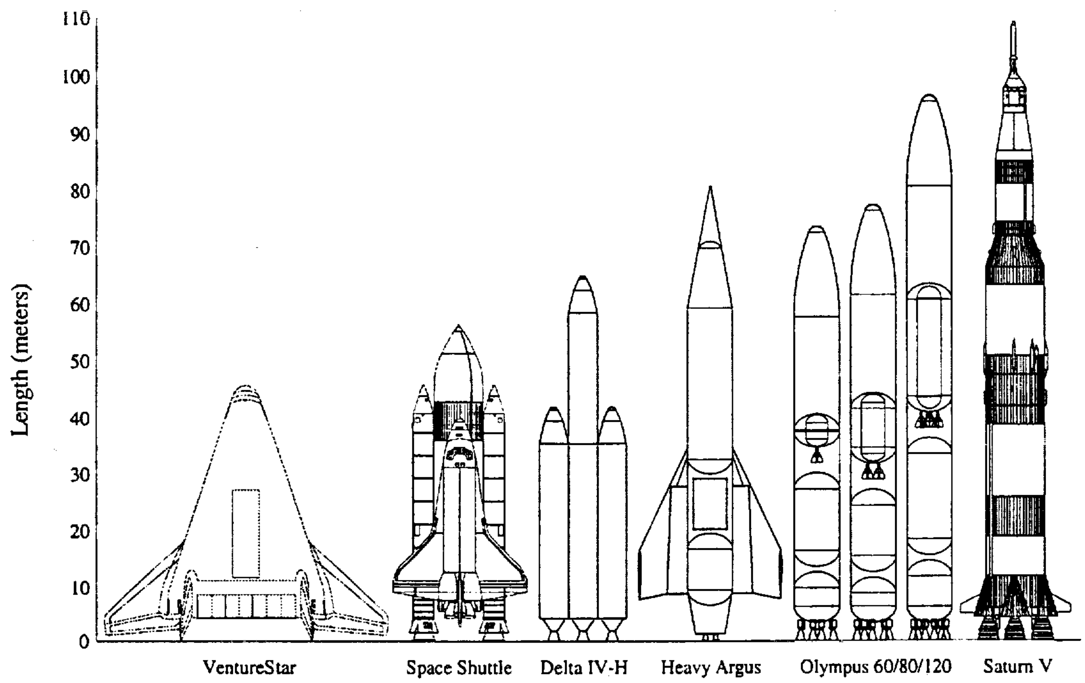


Figure 4: Selected Earth-to-Orbit Vehicle Comparison

- **Existing Vehicles**

The Proton is a standard Russian heavy-lifter, having been used to launch all Russian space station components for the last thirty years. Proton is a four-stage vehicle fueled by hypergolic propellants. Its advantages include flight heritage, relatively low cost, and a surprisingly modern operations scenario. Drawbacks include toxic propellants, high-latitude (lower performance) launch sites, and heavy airframe design.

The latest member of the Ariane family was originally designed to loft the Hermes spaceplane into LEO, and hence retains a significant lift capability. Ariane 5 is a liquid oxygen (LOX) / liquid hydrogen (LH2) core vehicle with solid strap-ons. Among its advantages are a modern design and near-equatorial launch site.

It should be noted that existing vehicles can lift a maximum of 15 MT to LEO, hence more powerful vehicles must be considered. It should also be noted that the Space Shuttle was omitted from the list due to high operations cost and an uncertain future in the timeframe projected.

- **Vehicles Under Development**

The Heavy Lift Variant Evolved Expendable Launch Vehicle (EELV) program seeks to upgrade existing Delta and Atlas vehicles to handle projected commercial and military launch needs in the next decade. The Delta IV and Atlas V will each have a heavy-lift variant capable of lifting around 20 MT to LEO (to replace the Titan IV). The EELV program is in an advanced state of development and these vehicles should enter service in 2001. The EELV variants of Delta and Atlas have superior operational, manufacturing, and performance capabilities and are an excellent candidate for the database.

Lockheed Martin's VentureStar program seeks to drastically lower the cost of ETO transportation through the development of a single-stage-to-orbit, fully-reusable launch vehicle. Projected to be available in 2005, VentureStar is another excellent candidate for the database due to its highly efficient, low-cost, rapid turnaround operations.

- **Conceptual Vehicles**

Heavy Argus is a second-generation reusable launch vehicle designed at the Space Systems Design Lab at the Georgia Institute of Technology, projected to take advantage of advanced technologies that are too immature to be included on VentureStar. Heavy Argus is a 40 MT variant of this vehicle that was originally designed to launch Space Solar Power components. The vehicle is highly reusable, and is projected to be available in or around 2010.

It is clear that none of these vehicles has an ETO capability greater than 40 MT. Hence, as part of this analysis, a Heavy Lift Launch Vehicle (HLLV) design was undertaken and designated Olympus. The philosophy behind the design approach was to look at technology alternatives besides those used on Shuttle or Saturn-V. The Olympus vehicles were designed for 60, 80, and 120 MT payload capabilities.

Design tools used for Olympus included APAS (Aerodynamic Preliminary Analysis System) for aerodynamic analysis, POST (Program to Optimize Simulated Trajectories) for trajectory optimization, MS Excel for weights and sizing, SDRC I-DEAS for visualization, and NAFCOM (NASA Air-Force Cost Model) based cost-estimating relationships (CERs) for weight based costing. These tools were used in conjunction with concurrent engineering philosophies such as Quality Function Deployment and Analytical Hierarchical Processing (AHP). The flow of information between these tools in the Design Structure Matrix (DSM) illustrates the iterative process needed for design convergence (see Figure 5). Each dot represents the passage of information, when information is fed backwards iterations are required. An automated script was constructed as an interface with POST, allowing extremely rapid iterations between the spreadsheet-based weights analysis and the sophisticated trajectory optimization of POST. This allowed optimization of the staging ΔV for minimum dry weight (low cost) for all three vehicles.

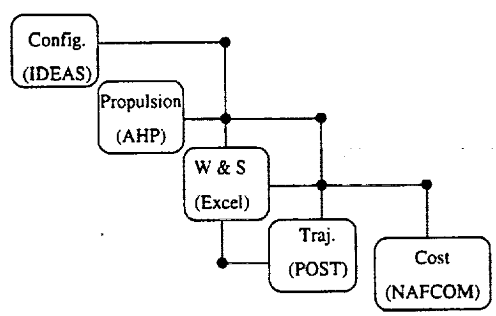


Figure 5: Heavy Lift Launch Vehicle Design Strucuture Matrix (DSM)

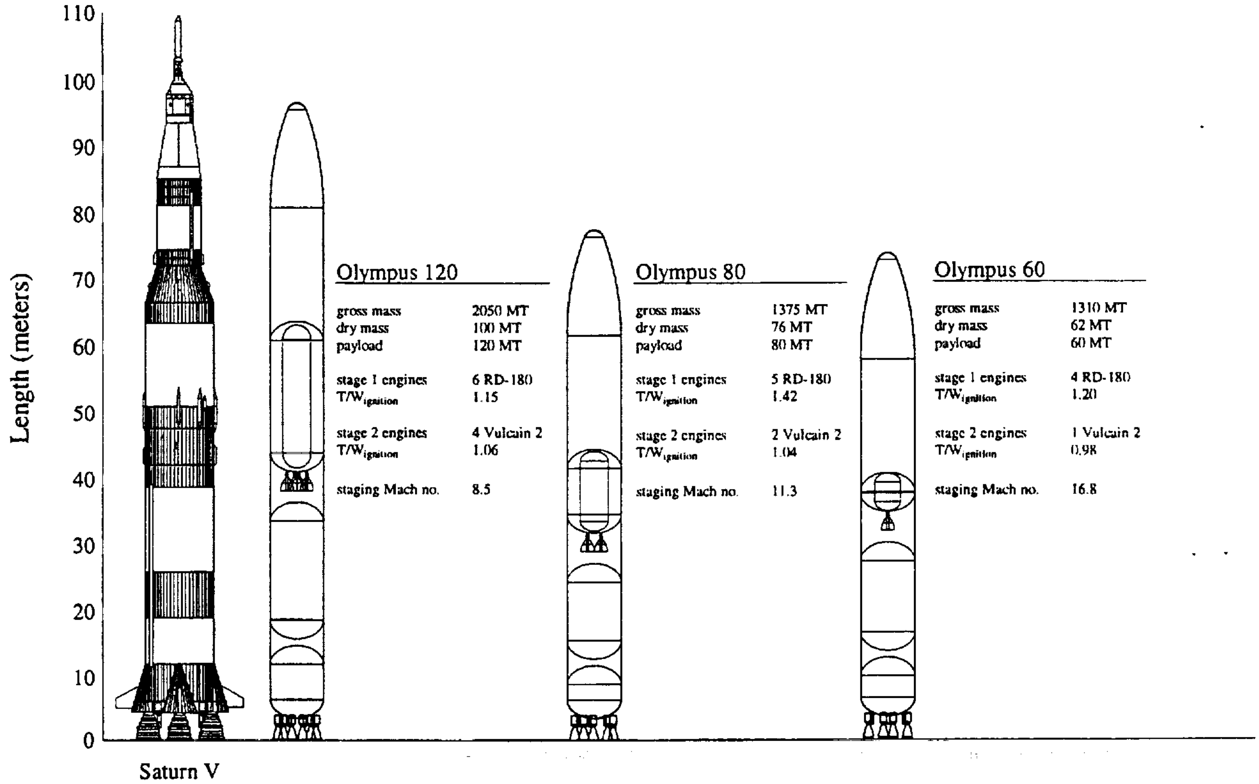


Figure 6: Comparison of In-House Heavy Lift Launch Vehicle Designs

After implementing this process the three designs seen in Figure 6 were obtained. The booster stage utilizes RD-180 engines burning kerosene and liquid oxygen with an upper stage that utilizes LH2/LOX Vulcain 2 engines. These engines were chosen based on cost, specific impulse, thrust, and possibilities for international cooperation.

Another improvement to the Olympus design was to take advantage of new technologies in large, cryogenic, lightweight composite tanks, and structures. The upper-stage tanks also have an innovative design feature called "tank within a tank" wherein the LOX tank is housed inside the LH2 tank. These advanced materials are used for nearly every structural system on the vehicle, and significantly reduce the vehicle's mass compared to older launch vehicles.

Operationally, the Olympus incorporates efficient strategies recently developed for the EELV and VentureStar programs. By designing for operations at the outset, design features that might interfere with an operational strategy can be avoided (such as solid strap-on boosters or parallel staging). The first and second stages are carefully mated at the horizontal integration facility located adjacent to the launch pad. Once raised by the rail mounted strong back the payload is then raised and attached at the launch site (see Figure 7).

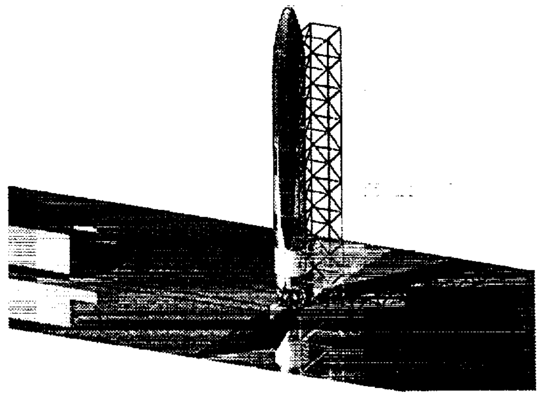


Figure 7: Operational Image of Olympus Heavy Lift Launch Vehicle

II.1.2.2 Mars Transfer Vehicle Database Definition

The purpose of the MTV database was to create a population of vehicles that could fully test the capabilities of the Mars Scenario Analysis Tool (M-SAT) and as reference database for future users. This database was populated by three unique MTVs each having a different propulsion system. The three propulsion systems examined are nuclear thermal rocket (NTR) propulsion, solar electric propulsion (SEP), and LOX/LH2 chemical propulsion. These vehicles were designed to provide new low cost options for Mars missions. Each class of vehicles has five sub-designs, one for each different payload class. The payload classes available in the database are 40MT, 80MT, and 160MT. The MTVs are designed only to perform up to TMI.

All MTV designs are optimized for lowest initial mass in low Earth orbit (IMLEO). Trajectory analysis is done depending on the type of trajectory flown (high or low thrust). Depending on the trajectory the size of the engines can be determined and in turn the mass of the vehicle components. Propulsion analysis and mass estimation were done in spreadsheets. Since these vehicles have never been built, inherent uncertainties exist in the mass-estimating relationships. Therefore, triangular uncertainty distributions are placed on the mass estimators to obtain 90% confidence level mass statements through the use of Monte Carlo simulations.

- **NTR MTV Design**

The NTR MTV is a relatively simple design because its only function is to provide the TMI burn. The MTV uses only one NTR to provide the required thrust. The NTR uses liquid hydrogen that is stored in a foam insulated propellant tank. The NTR was designed with a chamber pressure of 1.0 x 107 Pa and a characteristic velocity of 5500 m/s. The vehicle was sized to be able to provide 4,198 m/s of ΔV. This corresponds to the ΔV requirement to go from a 400km circular Earth orbit to a Mars injection orbit in the year 2022. This is one of the largest ΔV requirements as is seen in Figure 8.

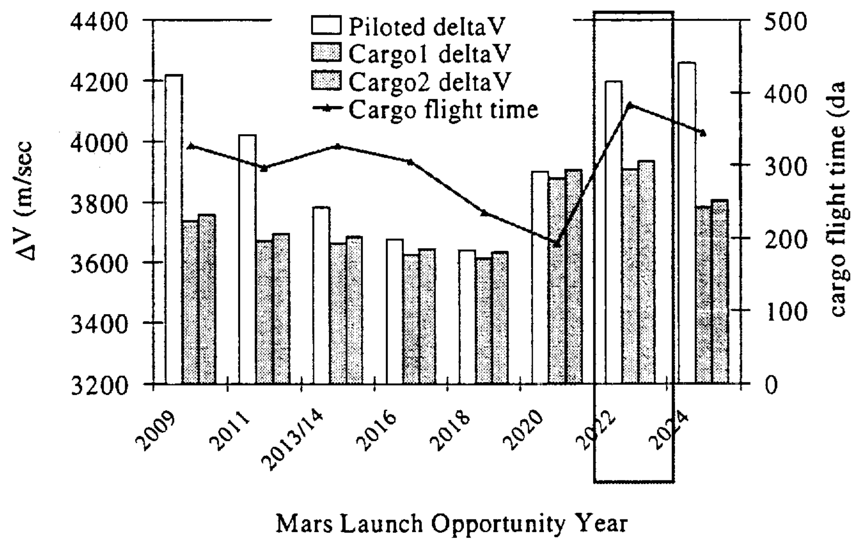


Figure 8: Required ΔV for TMI and Associated Flight Time (Based on DRM Flights)

The mass breakdown for the 80 MT NTR MTV based on the trajectory analysis above is shown in Table 1.

Table 1: Mass Breakdown for 80MT NTR MTV (90% Confidence Values)

No.	Item	Mass (kg)
1.0	LH2 Tank Structure	6,690
2.0	LH2 Tank Insulation	1,640
3.0	Other Structure	2,150
4.0	NTR Nozzle	630
5.0	Nuclear Reactor & Systems	3,310
6.0	Subsystems	5,420
	MTV Dry Mass	19,840
7.0	Payload	80,000
	Mars Arrival Mass	99,840
8.0	TMI Prop Req'd	61,600
9.0	Phase 1 Prop losses	1,910
	Initial Mass in LEO	163,350

The choice of the expansion ratio of the nozzles was an optimization problem. The larger the expansion ratio the higher the specific impulse of the engine, which decreases the amount of propellant used, but increases the nozzle weight. The optimal expansion ratio for minimum IMLEO was found to be 180 (specific impulse = 935 s). After the expansion ratio was determined the remaining engine properties were calculated. The • V of 4,198 m/s plus the additional 3.5% for losses was then used to determine the mass ratio (MR) of the vehicle. This mass ratio (MR) was used to determine the amount of propellant needed by the vehicle. The propellant losses assumed in the weight breakdown structure consisted of boiloff (1.6% of total propellant, sized for 1 month of losses), reserves (1.0% of total propellant), and residuals propellant (0.5% of total propellant). Figure 9 shows a size comparison of the various NTR designs.

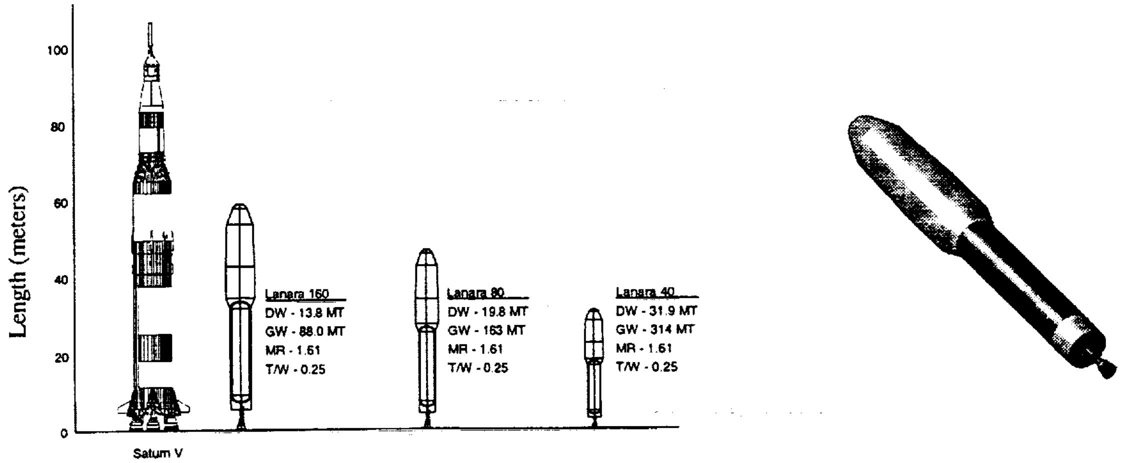
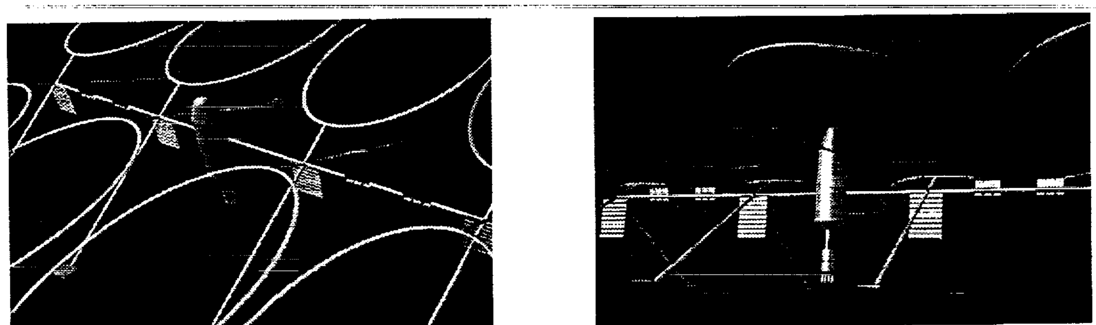


Figure 9: NTR Comparison

- SEP MTV Design

The second Mars transportation system evaluated was a Solar Electric Propulsion (SEP) design (see Figure 10). This architecture uses a SEP orbit transfer vehicle to transport the Mars-bound payload from a LEO to a Highly Elliptical Orbit (HEO) of 71,000 km x 400 km. For human missions the crew is sent out to the orbiting SEP vehicle using a crew taxi that consists of a crew capsule with an attached chemical propulsion stage. After the crew arrives at the SEP vehicle, they transfer to the MTV bound for Mars and then a small chemical kick-stage sends the payload to Mars. The SEP vehicle then returns to LEO for reuse.



a.) SEP Top Angled View **b.) SEP Side View**

Figure 10: SEP Imagery

For trajectory analysis the following parameters were calculated: total ΔV required to go from LEO to HEO using a low thrust trajectory, the time of flight of that trip, the required low thrust ΔV to go from HEO to LEO, and the required chemical kick ΔV needed to go from the HEO orbit to Mars. The crew taxi uses a high thrust chemical engine to go from LEO to HEO, which shortens the trip time and reduces the radiation exposure to the crew. This high thrust ΔV must also be provided by trajectory calculations. Figure 11 below graphically displays the SEP trajectory.

a.) **SEP Spiral Trajectory with Chemical Kick Stage (Top View)**

b.) **SEP Spiral Trajectory with Chemical Kick Stage (Angled View)**

Figure 11: SEP MTV Trajectory

The low thrust ΔV supplied by the trajectory analysis is used to calculate the outgoing mass ratio (MR) for the SEP transfer vehicle, using krypton-fueled ion engines with Isp of 4,000 sec. Specifically, these engines consist of gridded ion thrusters power rated at 500kW with an anticipated lifetime of 12,000 hours.

This MR is used in the weight breakdown statement to calculate the amount of propellant needed for the outbound trip. The inbound MR is used to calculate the inbound propellant needed. The total power needed by the propulsion system is determined by the IMLEO (initial mass in low Earth orbit) of the SEP vehicle. The power to mass ratio used for this determination is 0.025 kW/kg. This power requirement is used to size the inflatable, dense concentrator collectors as well as determine the number of ion engines. Photovoltaic arrays are used in conjunction with the concentrators to harness the necessary solar energy. The remaining items in the SEP vehicle mass breakdown statement were sized using various mass-estimating relationships (MERs) provided by NASA's Mars Orbit Basing and Space Solar Power projects. Table 2 details the mass breakdown structure for the 80 MT SEP.

Table 2: 80 MT SEP MTV Mass Breakdown Statement (90% Confidence Values)

No.	Item	Mass (kg)
1.0	Solar Collectors	14,000
2.0	Body Structure	4,740
3.0	Propulsion	47,670
4.0	Fuel Storage	11,130
5.0	Data Processing	20
6.0	Navigation Sensing/Control	280
7.0	Telecom and Data	80
	Dry Weight	77,920
8.0	Inbound Reserves and Residuals	320
9.0	Outbound Reserves and Residuals	1,250
10.0	Inbound Propellants	6,890
	Elliptical Orbit Departure Mass	86,380
11.0	Payload	80,000
12.0	Chemical Kick Dry Weight	17,550
13.0	Chemical Kick-Stage Propellants	44,630
	Elliptical Orbit Arrival Mass	228,560
14.0	Outbound Propellants	24,100
	IMLEO	252,660

RL-10A4-1 engines are used to power the chemical kick-stage that propels the payload from HEO to Mars. Vulcain 2s were analyzed for the 160 MT and 120 MT payload stages, but this increased the IMLEO of the SEP MTV and the overall weight of the chemical kick-stage propulsion system. RL-10s are also used for the crew taxi's propulsion system. The simple weight breakdown structure for the kick-stage includes total propellant weight, obtained from the stage's mass ratio, tank weight obtained using MERs developed during the NTR MTV design, and unusable propellant.

The crew taxi was designed using an expendable crew capsule. The taxi consists of the capsule and a detachable propulsion stage that contains of all tanks, fuel and hardware necessary to perform the burn to transfer the crew from LEO to HEO. The crew capsule weight of 6500 kg was taken from NASA's DRM. The rest of the weight breakdown structure was determined using the required ΔV of the stage, along with engine properties and various MERs.

The final vehicle designs obtained from above analysis are shown in Figure 12.

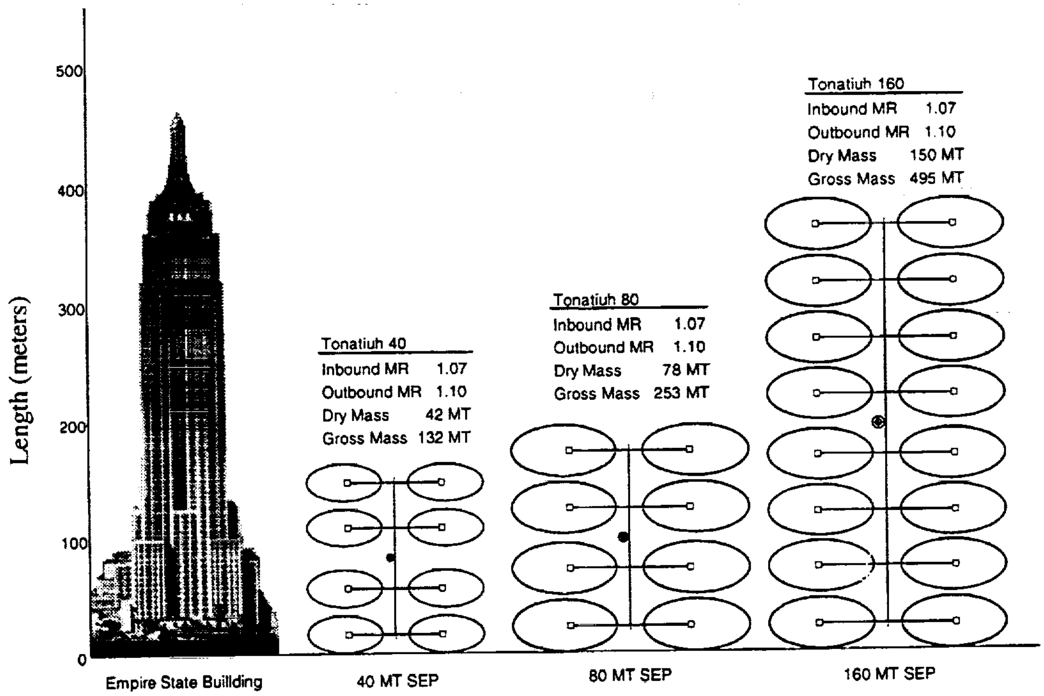


Figure 12: SEP Comparison

- **Chemical MTV Design**

The chemical MTV is designed to perform the same mission as the NTR MTV and similarly designed to provide the same 4,198 m/s of ΔV needed to go from LEO to Mars. The chemical design uses the same mass estimating methods described in the NTR section (see Table 3). The most significant difference between the chemical and NTR design is the propulsion system. The chemical design uses a single LOX/LH2 Vulcain 2 to provide the needed thrust. Since the Vulcain has a much lower Isp than the NTR and because the chemical MTV has to carry LOX, it is much heavy than its NTR counterpart. The final vehicle designs obtained from above analysis are shown in Figure 13.

Table 3: 80MT Chemical MTV Mass Breakdown Statement (90% Confidence Value)

No.	Item	Mass (kg)
1.0	LH2 Tank Structure	2,700
2.0	LH2 Tank Insulation	810
3.0	LOX Tank Structure	1,030
4.0	LOX Tank Insulation	130
5.0	Engine Weight	1,940
6.0	Other Structure	1,480
7.0	Subsystems	5,460
	MTV Dry Mass	13,550
8.0	Payload	80,000
	Mars Arrival Mass	93,550
9.0	Required Propellant	176,360
10.0	Unusable Prop	5,470
	Initial Mass in LEO	275,380

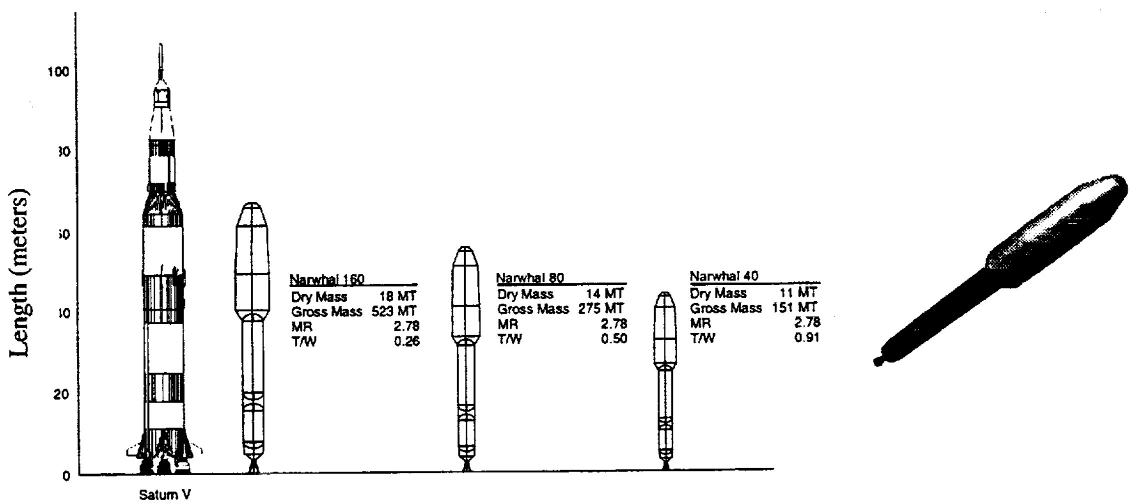


Figure 13: 80 MT Chemical MTV Size Comparison

- **MTV Summary**

Each MTV has its specific attributes and limitations. The nuclear MTV is very efficient, but there are environmental concerns associated with it. A SEP MTV would constitute a large technological advance in that area but would provide an exciting new form of space transportation. The chemical MTV uses very near term technology but uses the least efficient propulsion system and therefore is the most massive. Figure 14 shows IMLEO comparisons for five different payload class MTVs.

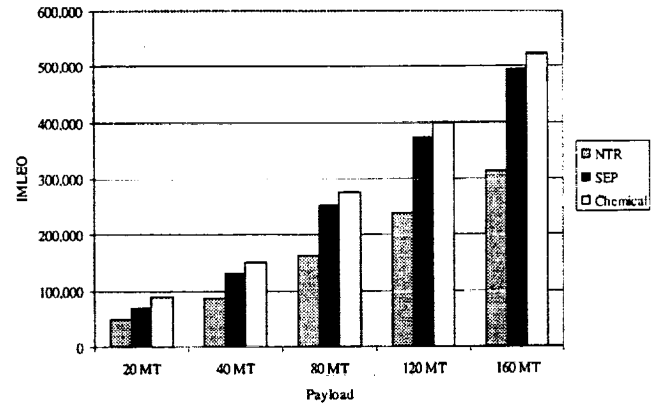


Figure 14: IMLEO versus Payload at TMI for Three Propulsion Arhcitectures (NTR, SEP and Chemical)

II.2 Modules

Now that the inputs have been established in terms of the vehicles and scenarios, the modules in the M-SAT simulation will be discussed. The modules are defined as algorithms that represent the analysis of the space transportation system. The three modules are the vehicle flight rate set generating module, the costing module, and the in-space operations module. The modules are referenced to the inputs in the database and are also cross-referenced to each other.

II.2.1 Vehicle Flight Rate Set Generation Module

The purpose of vehicle flight rate set generation is to reduce the size of the problem. Vehicle flight rate sets are defined as possible combinations of vehicles that can meet the payload constraints. Two example sets are shown in Figure 15.

2011 TMI Payload 80 MT	Set 1	Set 2	Set 3
SEP 160 MT	1	0	0
SEP 80 MT	0	1	0
SEP 40 MT	0	0	2

As the number of vehicles increase and weight of payload increases, so does number of possible sets

MTV set constrains IMLEO

2011 IMLEO 500 MT	Set 1	Set 2	Set 3	....
Delta IV	8	8	8	....
Argus	8	7	6	....
Olympus	1	2	3	....

Figure 15: Example of Vehicle Flight Rate Set Generation

As can be seen from the figure for a given year and a given payload requirement there are only a finite number of vehicles flight rate combinations that deliver the required payload. The module calculates sets first for the MTVs dependent on the TMI payload from the scenario in that year. These sets are usually small since there can only be three MTVs in the database. The ETO vehicle set is dependent on IMLEO of the MTVs, this number is usually high and their can be up to eight ETO vehicles, therefore ETO flight rate sets can be in the tens of thousands for large problems. For problems of this size set generation can take up to an hour. In order to generate the ETO sets one must know IMLEO, but this number changes if a different MTV set is chosen. ETO sets are therefore generated for a range of IMLEO, where the range is based on ranges of MTV IMLEO for each year.

Once the vehicle flight rate sets have been generated they become variables that can be changed in M-SAT. Changing these variables will result in different total LCC of the transportation system.

II.2.2 In-Space Operations Module

To determine the cost and weight penalties of assembly in LEO a module called in-space operations was developed. The cost outputs from this module are based on several factors including number of robots used and number of space platforms (dependent on the number of robots), each of those having fixed and variable costs. A general schematic of a typical in-space operations scenario is given in Figure 16.

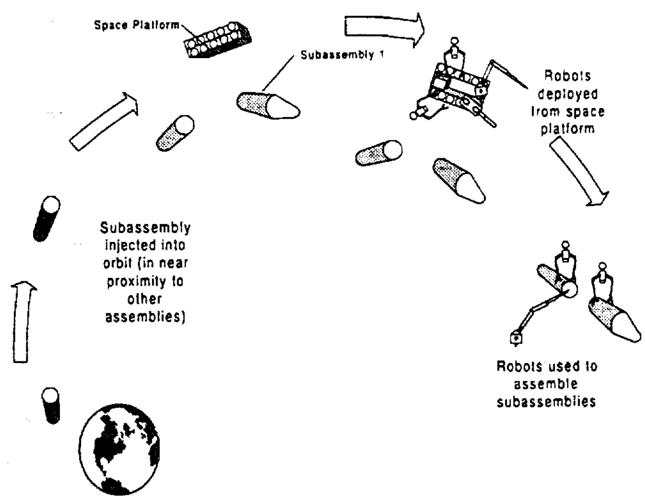


Figure 16. In-Space Assembly of MTV Pieces

The cost of in-space assembly is heavily dependent on the total number of ETO flights. A simple relationship exists between in-space operational complexity and the number of ETO flights. This relationship is the number of assemblies is equal to the number of ETO flights minus one. If there are less than five ETO flights autonomous docking is used for in-space assembly. Robots are used exclusively if there are more than five ETO flights.

The in-space operations module has the capability to use several different types of robots. Each robot is mission specific and designed, developed, tested and evaluated before the beginning of each mission. This non-recurring cost is dispersed evenly over 5 years prior to the initial launch. Learning curves are added into the simulation for the development of the robots.

Space platforms for the robots are also modeled. These space platforms are small "robot space stations" of ten metric tons with lifespans of six years. The robots can keep necessary tools, power, and communication systems on board these platforms. The number of space platforms needed is determined by the total number of robots to be in orbit during assembly. Each can sustain only twenty robots at any given time.

II.2.3 Cost Module

Costs are determined separately for the ETO and MTV transportation segments. Recurring and non-recurring costs are calculated for each MTV. Selected ETO launch vehicles are sunk cost programs whereas others are developed exclusively for these scenarios.

The non-recurring and recurring costs of each type of MTV are determined through the use of cost-estimating relationships or specific costs. The NTR, SEP, and chemical MTV architectures each have varying non-recurring costs (as seen in Table 4 for the 40, 80 and 160 MT payload class vehicles). The non-recurring cost of the NTR is the highest of the three, while SEP has the highest recurring cost. This is due to the fact that the SEP is an architecture made up of many small components.

Table 4: Non-Recurring and Recurring Costs of MTV Architectures for Various Payload Classes (1999$M)

Architecture Type	Payload Capability - Class		
	40 MT	80 MT	160 MT
NTR			
Non-Recurring Cost	$4,291 M	$4,547 M	$5,059 M
Recurring Cost	$829 M	$1,441 M	$2,662 M
SEP			
Non-Recurring Cost	$1,892 M	$2,236 M	$2,911 M
Recurring Cost	$1,368 M	$2,261 M	$4,175 M
CHEM			
Non-Recurring Cost	$1,231 M	$1,323 M	$1,472 M
Recurring Cost	$273 M	$281 M	$293 M

The ETO launch vehicle cost analysis is handled in two ways. For those launch vehicles whose design, development, testing, and evaluation (DDT&E) costs are sunk or already accounted for, such as the VentureStar or Ariane 5, only a constant recurring price per flight is charged to the MTV launch customer. For those vehicles which are developed just to handle these Mars payload missions, such as a new heavy lift launch vehicle (Olympus), the costs for those vehicle developments as well as the recurring cost per flight is included in the transportation cost (see Table 5).

Table 5: Non-Recurring and Recurring Costs of ETO Vehicle Architectures for Various Payload Classes (1999$M)

Architecture Type	Payload Capability - Class		
	60 MT	80 MT	120 MT
Olympus			
Non-Recurring Cost	$2,464 M	$2,813 M	$3,483 M
Recurring Cost	$507 M	$577 M	$713 M

The cost module of the M-SAT simulation accepts the DDT&E (non-recurring) and TFU (theoretical first unit) costs of each transportation vehicle, MTV or ETO. These costs are taken in conjunction with the flight rates of each of the transportation architectures to determine the life cycle transportation costs. Learning curves are included for the DDT&E and TFU costs. The DDT&E is applied starting five years before the construction of the first unit with the DDT&E amount spread out evenly over those five years. A DDT&E learning curve of 85% and a TFU leaning curve of 90% are included in the analysis. The cost module has built-in logic which instructs the module to build the lowest payload class vehicle first if multiple payload class vehicles are built in the same year. With this logic the learning curve effect is applied to the larger payload class vehicle, which is the vehicle with the larger DDT&E cost. With these learning curve effects the cost module is robust enough to handle multiple architectures.

II.3 Optimization Process

Now that the inputs and modules have been introduced the problem statement is redefined in terms of the newly designated parameters. This statement is, "For a given scenario and database find the minimum total life cycle cost by changing the vehicle flight rates of the MTVs and ETO vehicles." In order to find the minimum cost, an optimizer must be incorporated with the databases.

The decision of which type of optimizer to use is facilitated by characterizing the design space. Independent variables in the design space are the vehicle flight rate sets. Because the flight rate sets are determined for each opportunity and for both MTV and ETO vehicles there are twenty independent variables. The design space of the problem is not a continuous function because it is undefined when vehicle flight rat set guesses are not integers. Total LCC is the objective function. The problem is also multi-modal, containing more than one local minimum for the objective function. Finally, the size of the space is defined as possible values of each of the twenty independent variables multiplied by each other. Therefore if there is an average of 8 MTV sets and 6,000 ETO sets for a given scenario the total number of possible combinations is approximately 1 x 10^{50}. The first two characteristics of the problem lead one to choose a stochastic optimizer. The large number of possible flight rate combinations makes a random grid search unreasonable so a 'smart' stochastic search must be done. The stochastic optimizer chosen for this problem is a spreadsheet based contracted genetic algorithm (GA) program. GAs incorporate the idea of survival of the fittest and use binary digits to represent the 'genes' of the independent variables. There are of course problems with GA, the main problem being that GAs do not necessarily generate the optimal solution, but usually approach a near optimal solution.

A schematic of the genetic algorithm integrated with the entire process is shown in Figure 17. Once the database, scenario, and flight rate sets have been created the program is initiated. The inputs to the GA are the upper and lower bounds of each design variable and the value of the objective function, namely total LCC. The optimization begins by first initializing a random 'generation' of independent variables. A generation consists of a specific number (50-100) number of candidate designs. This generation is evaluated and the cost is returned into the optimizer and three 'biological' processes performed. The first is replication of good designs. Next is crossover, where 'parents' are combined to diversify the next 'generation'. Lastly, random mutation occurs, which adds diversity to the design pool.. These new 'fitter' variables are placed back into the simulation and then the process continues. A fourth process called restart is sometimes implemented if the generation becomes inbred. The optimization is stopped when the desired number of iterations is complete, however there is no mathematical proof that the optimized answer has been obtained.

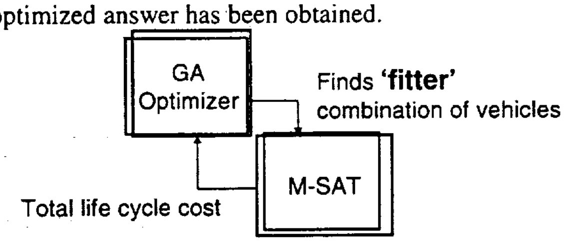


Figure 17: Genetic Algorithm Optimizing for Minimum Cost

III. RESULTS

III.1 Scenario Forecasts

Each of the three mission scenarios is designed to test and evaluate different aspects of the simulation. "Flags and Footprints", which represents a simple exploratory mission to Mars was used as a test case of the simulation's accuracy. This simple scenario was chosen because the optimal answer could be ascertained without running the simulation. For the "Flags and Footprints" mission, all eight launch vehicles slots were used. They were Heavy Argus, the Proton M, EELV, Ariane 5, the Venture Star, and three classes of the Olympus HLLV. The inclusion of all the vehicles gave the simulator the full range of launch vehicle choices needed to determine if it was functioning correctly. This scenario, as well as the others, were each run using three MTV propulsion types. These sets were all SEP, all NTR and all chemical transfer vehicles.

The second scenario analyzed was a DRM based exploratory mission. For this mission, the choice being tested was whether to use HLLVs to place the MTVs in orbit. Ideally, every launch vehicle choice would have been included in this scenario. However, because of the large amounts of tonnage that needed to be placed in orbit, the number of ETO sets generated using all available launch vehicles was too large to allow efficient analysis of the problem. For this reason, the Ariane 5 and the Proton M were not included in the ETO sets. However, the flight rates of the remaining non-HLLVs in the database were increased to allow the simulation the choice of being able to launch the entire payload without using a HLLV.

The third scenario, colonization, was used to determine the characteristics of the simulation's HLLV selection. For this scenario, all three Olympus class vehicles, along with the VentureStar, were included in the ETO sets. The VentureStar's flight rates were limited to two per year. These two flight rates were to allow for the insertion of small amounts of payload left over after the Heavy Lift launches. The main goal of this concept was to determine which classes of HLLVs would be built first and how the payload would be divided between the various classes of Heavy Lifts.

III.2 Results

Each of the results are representative of the simulators ability to evaluate different scenarios and depend greatly on the assumptions made for each simulation. The total number of GA iterations for each case is 10,000 with three restarts. This dependence on the initial assumptions of the launch vehicles and MTVs make these results applicable only to the specific cases discussed here. The main result of this simulation is the optimization process itself. A potential user should be excited about the possibility of placing their own launch and transfer vehicles, with their own assumptions, into the simulation and generating results valid for their particular interests.

The first scenario analyzed was "Flags and Footprints", taking one hour to run through the process. For this scenario it was expected that existing launch vehicles would be used to launch the MTVs. The simulation arrived at the same solution. For the chemical MTV scenario, a total of 13 Heavy Argus, 2 EELVs, and one Proton M were used to launch the payload. This solution approached the optimal, but did not find the true minimum. The true minimum cost would be achieved by having an additional Heavy Argus flight, therefore utilizing all of the vehicle's available flights. This solution is the cheapest because Argus is the cheapest launch vehicle available. The main reason the simulation did not find the exact optimal solution was because these solutions did not yield a guaranteed minimum. However, the simulation does show an important tread, namely no HLLVs were built. This trend is the same for each MTV case. The results for the other MTVs architectures were also as expected. For each type of MTV, the 80MT vehicle was launched in each of the two years.

The DRM-based exploration scenario was analyzed to determine if it was cheaper to build a HLLV for this mission, taking three hours to run though the process. The results of the analysis show that it is more economical to use existing smaller vehicles to complete this mission. The NTR scenario was the only one were no HLLVs were selected. The other two scenarios selected a few HLLVs but showed the general trend of using the smaller vehicles. Given enough optimization time, it is believed these scenarios would converge on a solution containing no HLLVs. Trends were also noticeable for the specific MTV classes chosen. For the NTR and SEP MTVs, the predominate choice was the 80MT class, the chemical MTV predominantly chose the 160MT class. This consistency of choosing the same payload class of MTV for each specific vehicle type was expected because of cost benefits associated with repeatedly producing the same class of vehicle.

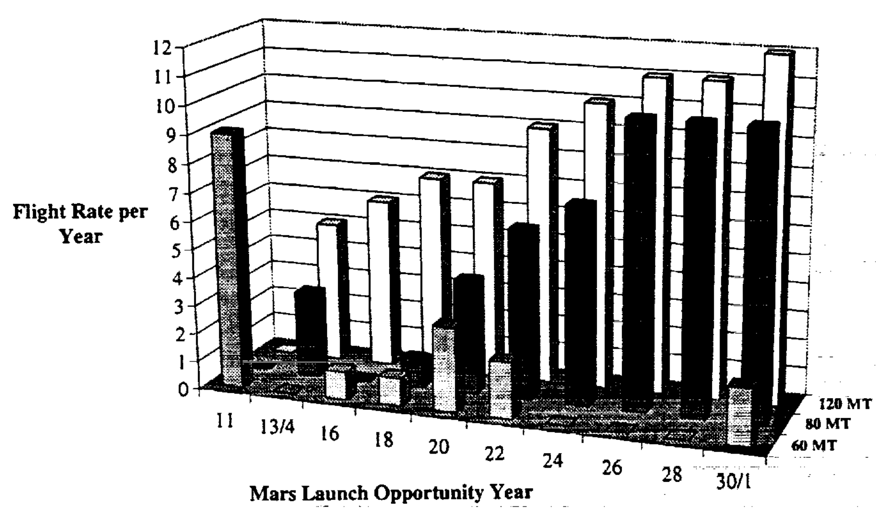


Figure 18: HLLV ETO Traffic Rate for Colonization Scenario of Chemical MTV

The final scenario analyzed was colonization (see Figure 18), taking two hours to ruin through the process. The objective of this scenario was to determine the optimal use of the HLLVs. For each MTV scenario the first HLLV produced was the 60MT version. In the first year of each scenario only 60MT HLLVs were launched. In following years the cost reduction obtained from the learning associated with building the 60MT HLLVs allowed larger vehicles to be built at less cost. As the scenarios proceeded further the trend was to build the larger 120 MT HLLVs. In almost every year the two allowable VentureStar flights were used to transport excess payload that could not be easily integrated with the HLLV flights chosen for that year. Similar trends were noticed with the MTVs. In the first years of the scenarios, the smaller MTVs were chosen. As time progressed, the larger vehicles gave the optimum price for lowest total LCC. This is referred to as segmented development. This makes intuitive sense for a long-term scenario. Segmented development requires one to think long term and develop a vehicle with possible derivatives in mind during the conceptual design phase, developing a family of vehicles rather than just one.

The final cost for each scenario is shown below in Table 6. As can be seen, the chemical MTV was the optimal solution for each case. This result is greatly influenced by the cost assumptions made for chemical vehicle.

Table 6: Scenario Total Costs

MTV Type	Flags & Footprints	DRM Reference	Colonization
Chemical	$3.7 B	$ 19.5 B	$ 59.8 B
Nuclear	$ 6.9 B	$ 30.5 B	$ 69.4 B
Solar Electric	$ 6.5 B	$ 36.0 B	$ 83.9 B

To show the flexibility of M-SAT to various assumptions made by the individual, several cost trade studies were analyzed to see what effect MTV costs would have on the results shown above. These trade studies were coined "what if" studies. They show how M-SAT can be tailored for each individual's vehicle design and cost assumptions. For the extended DRM mission, the question was asked, "What if the NTR MTV costs were reduced by 50%?". The answer is the total cost for the NTR DRM mission would be reduced to $19.4B. Secondly, it was asked, "What if the SEP MTV costs were reduced by 70%?." The answer given by M-SAT was a cost of $19.2B. These same trades were also performed on the colonization mission. For the NTR MTV a cost reduction of 30% yields a new total mission price of $58B. A SEP MTV cost reduction of 55% gives a total colonization mission cost of $59.3B. These results, which are summarized in the Table 7, show that the simulation is inherently non-biased toward any particular vehicle manifest.

Table 7: "What If" Study Results To Beat Chemical MTV

Architecture	Reduction %	Scenario	Cost
NTR	50%	DRM Extended	$ 19.4 B
SEP	70%	DRM Extended	$ 19.2 B
NTR	30%	Colonization	$ 58.0 B
SEP	55%	Colonization	$ 59.3 B

IV. CONCLUSIONS

This study developed the M-SAT simulation that can take launch vehicles, MTVs, and payload demand for a twenty-year period to give the user the optimum manifest of launch vehicles and MTVs for the most cost-conscious solution. Interesting patterns can be seen as far as the development of MTVs or heavy lift launch vehicles for particular scenarios. The simulation is an amalgamation of various modules: ETO launch vehicle's, MTVs, in-space ops model, a cost module, and an optimizer. The power of the simulation lies in the directions it indicates for future transportation architecture developments. One can add substitute vehicles to the database or examine already existing fleets. The simulation is a new tool that can be used to examine the dual phase transportation problem, from Earth-to-orbit and then from Earth orbit to Mars

V. FUTURE STUDIES

Most of the elements of future work involve improving the M-SAT simulation, which can be expanded in both capability and the types of architectures examined. Increasing the speed and ease of use will expand the capabilities of the simulation. The most readily available means of increasing the speed is to reevaluate the logic planning within the code. Improving the vehicle set generation module may also increase speed. Another specific example of possible expansion is in the operations module. Our thinking is to develop a more robust system that can handle different types of assemblies such as combinations of astronauts and the existing robotic assembly. Another concept under study is to create a web-based interface where the user can input the type of scenario they would like to explore and then one hour later results would be e-mailed for review.

VI. OUTREACH

Outreach was accomplished by trying to bring this problem to the attention of the entire School of Aerospace Engineering at Georgia Institute of Technology. Incentives were given for introductory aerospace students in the School to attend the final presentation. Many of the introductory students did attend and were rewarded with an interesting glimpse into what may await them in their future deign careers. In addition, a web presence was developed that not only describes the work performed in this study and eventually allows one to obtain the M-SAT simulation. An overarching philosophy in this study was to allow users to examine their own scenarios within the confines of the simulation. By allowing the general community to obtain the simulation, the communal result will allow both greater access and improvement in the simulation (see http://atlas.cad.gatech.edu/~ksorensen/msbv.html).

VII. ACKNOWEDLGEMENTS

The authors would like to extend their most sincere appreciation to the many students of the Space Systems Design Lab (SSDL) at the Georgia Institute of Technology. Gratitude goes out to Irene Budianto, David Way, Laura Ledsinger, David McCormick, John Bradford, Rebecca Cutri-Kohart, and Tara Poston for their invaluable encouragement and assistance throughout this process of exploration and discovery. We would like to thank the School of Aerospace Engineering at the Georgia Institute of Technology as a whole for their help in providing the support required for such an endeavor. Special consideration is given to Dr. John R. Olds, without whose invaluable experience and guidance this project would not have taken place.

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MADEX: Martian Drilling and Exploration

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Foreword:

The work reported was done by students enrolled in a General Studies course, MET 3350: ROCKETS and STARS: a space trek, during spring 1999. Half of the multi-disciplined class of 26 attended Cherry Creek High School. Mr. David R. Paynter, who teaches biology at Cherry Creek, helped his father in the conduct of the class which met Tuesday and Thursday from 6:55 to 8:35 PM beginning 19 January. A number of outside advisors met the class separately and twice as a group to review results. The students visited the GATES Planetarium to view the Hubble Space Telescope Gallery. The class also visited the Lockheed Martin Astronautics Division at Waterton, Colorado.

The first several weeks were used to understand the HEDS-UP program objectives, the Mars baseline mission and design guidelines, and possible design concepts to satisfy certain objectives. Two groups were formed on 9 February to study, (1) inflatable structures, and (2) human powered applications, e.g., human powered vehicles, or power generation. A third group was formed one week later to search for life using a cliff-walker to house a drill to collect and analyze core samples obtained at various depths below the Mars surface. MADEX evolved after a preliminary design review conducted by the outside experts on 8 April.

Abstract

The concept will take samples and spectrometer data at various depths below the Martian surface. The unit will address three major objectives for the human exploration of Mars. 1) Can humans ultimately inhabit Mars? 2) Is there or has there been life on Mars? 3) What is the history of the Martian planet? Rather than drilling vertically from the surface of the planet and taking samples at predetermined depths, the Martian Drilling and Exploration Unit (MADEX), will be lowered over a cliff to drill horizontally at various depths. The unit will attempt to satisfy the objectives using spectroscopic data taken along the cliff wall and analysis of the core samples taken.

1. Introduction

NASA has proposed that to accomplish the Mars mission goals, it is important that a plentiful source of water be searched for including a water table or a high concentration of water within the polar caps on Mars. By using spectrometers with the capability of recognizing aqueous minerals along a cliff wall, it may be possible to select preferred regions to look for a plentiful source of water. Where this water is, assuming it exists, is where it is most likely that we could find present day life. Thus, core samples taken by the MADEX (Figure 1.1) could then be analyzed for suggestions of life. By analyzing other mineral compositions within the surface we will better understand the planet's history and past atmosphere.

By sending a large drill rig to the planet, a large amount of piping is required. The piping would greatly increase the mass of equipment sent to the planet. Rather than using pipe, it is suggested to bring a rover-type unit to descend a cliff wall to save mass and take data, such as spectrometry and photographs. Past designs of units similar to the MADEX such as the series of Dante units have shown that it is difficult to design a unit that will safely and consistently maneuver itself. Therefore, the MADEX unit will be controlled by a computer system atop the cliff wall. The maneuvering system will consist of two cable spools at 50 m from the edge of the cliff, 60 m apart. By releasing slack on each cable, the MADEX is lowered along the cliff wall. If there is a protrusion on the wall, one cable will release more rapidly thus slowly lowering the MADEX diagonally down the wall and avoiding the obstacle.

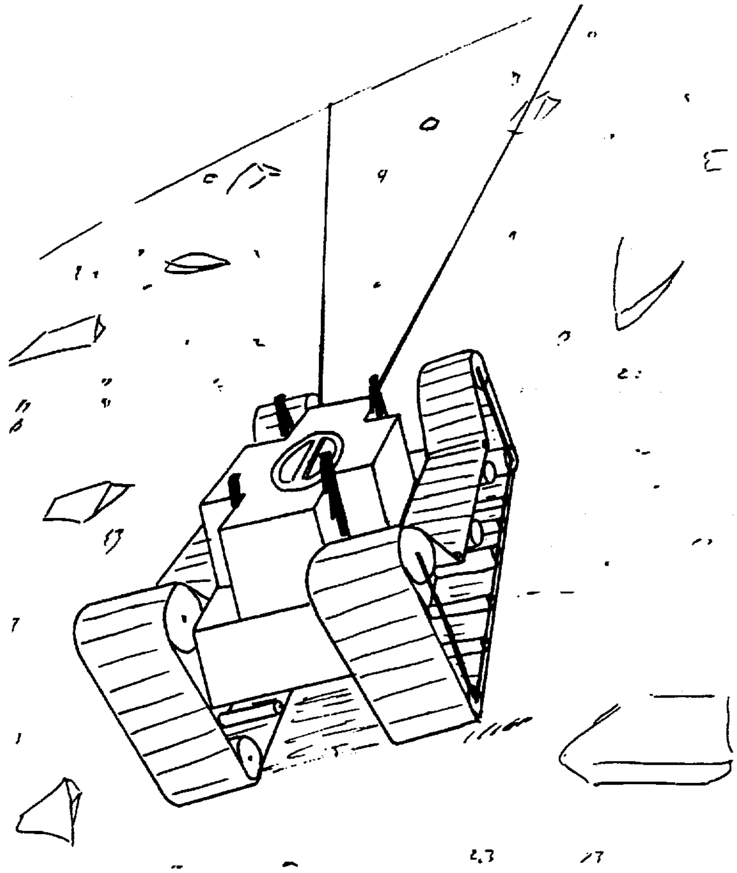


Figure 1.1 MADEX

MADEX will have a pair of treads which will allow the unit to roll down the cliff wall. The tread system will be equipped with a small motor that will control the unit's motion when atop the cliff. The tread system is sturdy, yet provides a somewhat soft ride. The treads will extend below the bottom of the MADEX in order to separate the equipment from the cliff wall by about 20 cm.

As photographs and spectroscopy data are collected, the data will be sent via the communication link to a "base" situated between the two cable control units (CCUs). After every nine stops, the MADEX will transport the core samples to the top of the cliff. Should something happen to the MADEX preventing its return to the base, not all of the data taken is lost.

MADEX is lowered using the 3/8 g gravitational force. The lowering rate is controlled by a friction brake. Friction acting on the cable spools will allow a slow and safe descent of the MADEX and to stop the unit's descent. When stopped, MADEX will lower and position four individually controlled legs against the cliff wall for stability during data collection. The MADEX will drill two samples simultaneously and take spectrometer readings and high resolution photographs of the cliff wall. The digital images provided by these instruments are transported to the base via the communication cable. The MADEX then is lowered five meters and the data collection stage repeats.

Because of the limited knowledge regarding Mars' cliff walls, it is difficult to imagine how rugged they may be. It is therefore necessary for the unit to have the ability to maneuver itself around obstacles in its path. Sonars placed on the front and back of the MADEX will constantly take readings of the obstacles in its path. Thus, a basic map of the cliff wall can be created. Using this map, the computer system situated within the base decides the path to take in order to avoid such obstacles. The CCUs are used to maneuver MADEX around them.

With the CCUs 60m from each other MADEX may descend approximately 250m. As mentioned, the gravitational force is used to descend the cliff wall. The CCUs use friction to brake the descent of MADEX. The unit will make a total of 50 stops along the cliff wall (approximately every 5m) to collect samples and data.

We have located a cliff wall nearby the landing site proposed by NASA. NASA has proposed to land near the region between East and West Candor (Figure 1.2). The cliff wall on the northern end of East Candor is a preferred setting for the MADEX. This cliff works toward the mission's advantage because it is near NASA's landing site and it appears to contain multiple, detailed layers.

MADEX will be stationed and secured near this cliff wall by the human crew. Due to its relatively small size, the entire unit could be transported to the desired location from the habitat by a rover. The base will be set up 50m from the edge of the cliff with the CCUs extended 75m in each direction from the base along the cliff wall. Communications cables are attached between the CCUs and the base, and between the MADEX and the base. Cables from the CCUs are then connected to the MADEX.

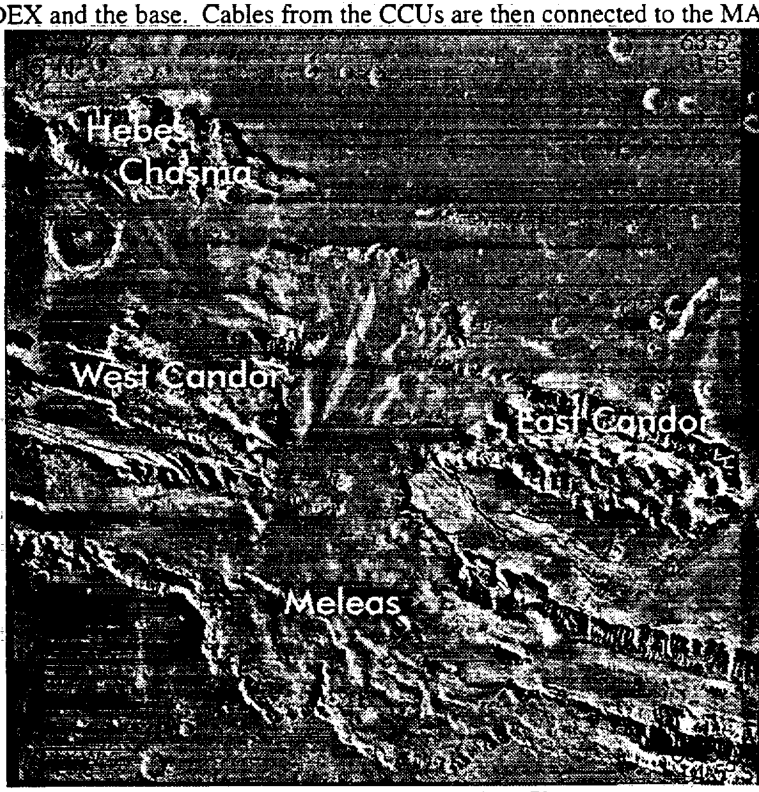


Figure 1.2 Proposed Landing Sites

A test run will be made before the astronauts leave the system. MADEX will descend approximately 20m, take data, and then return the data and samples. If there are problems, they will be assessed and fixed. When the system is functioning properly, it will be programmed to run and the astronauts will leave. They will return and replace the batteries as required with fully charged ones. While there, the system will be checked and inspected to discover any problems, core samples that were returned to the base will be taken to the habitat. This sequence will be repeated until the full 250 meter depth has been achieved. At this point, the MADEX will be returned to the top of the cliff by the CCUs operating on return mode. The unit will be lifted to the top of the cliff. The small motor on the MADEX will then move it the final 50m to the base.

2. MADEX Design

2.1 MADEX Unit

MADEX has been designed to use minimal power to descend the cliff wall and collect data. It allows for gravity to lower the unit. In the event that the unit reaches a ledge on the wall, there is one small motor attached to its tread system so that the unit may continue its descent. This motor is used to move MADEX atop the cliff wall when it moves the 50m distance between the base and the cliff wall.

2.1.1 Interior Layout

Vertically through the center of the MADEX unit will be the drilling cylinder. This cylinder will consist of two main parts: a storage ring and the drilling ring. The storage ring will be the lower 18 cm of the cylinder. It will contain eighteen cylindrical holes placed uniformly around the outer edge of the cylinder. Each hole will have a 3 cm diameter. The upper cylinder will have two holes that match up with the outermost holes on the lower cylinder. Through these holes penetrate the two core sampling drills. The upper cylinder is immobile relative to the MADEX, unlike the lower cylinder which can be rotated about its center axis.

Situated to the front of the drill cylinder will be the Mössbauer Spectrometer and the Miniature Thermal Emission Spectrometer. Because the outermost holes on the cylinder are the only holes that interact outside the MADEX, a digital camera will be situated between the cylinder and the cliff wall. Attached to the front wheels on the treads is a small motor which controls the unit's motion when atop the cliff. One small sonar will be placed at the front, and one at the back of the unit. A small computer which translates commands sent by the base and data to be sent to the base will be placed near the back of the unit. At each of the four corners of the unit will be a small, low output motor that drives its respective support leg.

2.1.2 Exterior Layout

The dimensions of the MADEX will be 120 cm x 120 cm x 100 cm. Much of the volume within these dimensions is empty space, however. For instance, the top of the unit will be cut away to save overall volume of the unit.

On each side of the unit will be a tread system which primarily rolls freely with the motion of the unit, but drives the unit when atop the cliff. Each tread will extend beyond the front and back of the unit by approximately 35 cm. The width of each tread will be 40 cm, approximately 1/3 of the total width of the MADEX. For added support when the unit is stopped for data collection and drilling, the MADEX will be equipped with four legs (one on each corner of the unit). To the front and back, the sonar lenses will be visible through a grate-like shield. The back of the MADEX will be the same as the front except that it will have two eyeholes for connections to each CCU and one port for the communication/power cable. Viewing the top of the unit, one would see the drill cylinder centered. No holes are visible from the top, though two holes (those through which the drills penetrate) are visible from the underside of the unit.

2.1.3 Tread system

The tread system planned for MADEX Unit is somewhat unique (Figure 2.1.3.1). It is different than tread patterns on common tanks where each wheel is on its own frame. MADEX will use only three frames for the wheels. This results in fewer moving parts required. The system will allow for a relatively smooth ride.

As the treads touch a protrusion, the middle wheel pushes against the cliff and the front wheel will rise. As the protrusion reaches the middle wheel, it is raised and the outer wheels are lowered. The front wheel is thus in contact with the wall again sooner than it would without this design. As the unit passes over the protrusion, the treads slowly return to their "relaxed" state. This smoother ride will help ensure that the core samples remain intact and that the electronic equipment on board the MADEX is kept undamaged.

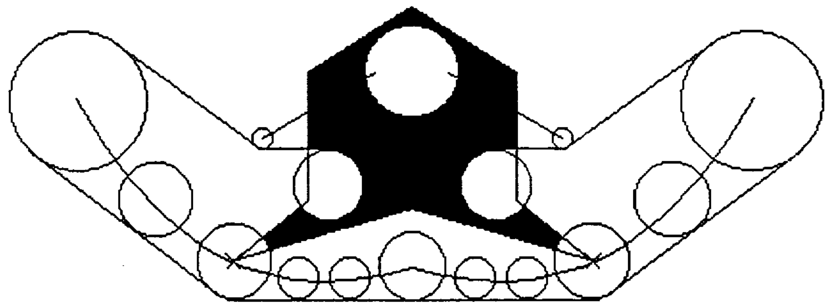


Figure 2.1.3.1 MADEX Tread Design

2.2 Base

The base will be situated at the top of the cliff 50 meters from the edge. It serves as the brain and the storage space for the mission. It stores the power, samples, and data collected.

2.2.1 Computer

The computer that the base will be equipped with will run the entire mission. Stored within the computer are spectrometry data, photographs, detailed records of progress, and maps of the cliff. The computer must tell the CCUs when and how rapidly to release slack or to pull the MADEX back up to the top of the cliff. How quickly to release slack depends upon what obstacles lie ahead in its path. The computer thus must calculate what the best route to avoid such obstacles would be. Once the descent of the unit is halted, the computer must tell the MADEX to take data: first spectroscopy data, then photographs, then core samples.

2.2.2 Battery Storage

The batteries which store the energy necessary to run the MADEX and all of its components will be stored within the base. In order to minimize the weight of the unit itself, thus minimizing the work required to raise it to the top of the cliff, all power necessary to run the MADEX unit itself will be transferred from the batteries to the unit through the communication/power cable.

2.2.3 Sample Storage

After samples are taken from the MADEX to the base, the samples will be removed from the cylinder, packaged safely, labeled, and stored within the base. The system ejects the sample from the cylinder into a small bag made of Mylar. The labels are then labeled with the depth from which the samples were taken and the date of sampling. The sample is then stored in the base until it is retrieved by astronauts.

2.2.4 Mechanical Arm

When the MADEX reaches the top of the cliff after loading the cylinder completely (eighteen total samples), the unit moves to the base to unload the samples (Figure 2.2.1). The base is shaped to funnel the MADEX into the correct location for unloading. Once in this position, a mechanical arm attached to the base grabs the cylinder and pulls it out of the MADEX. Contained within this cylinder are the drills, drill motors, and the samples. Once the cylinder has been removed from the unit, a new cylinder, containing drills and motors is placed into the unit so that more samples can be taken immediately. The MADEX then moves back down the cliff to continue its research. While more samples are being taken, the base removes the samples from the used cylinder.

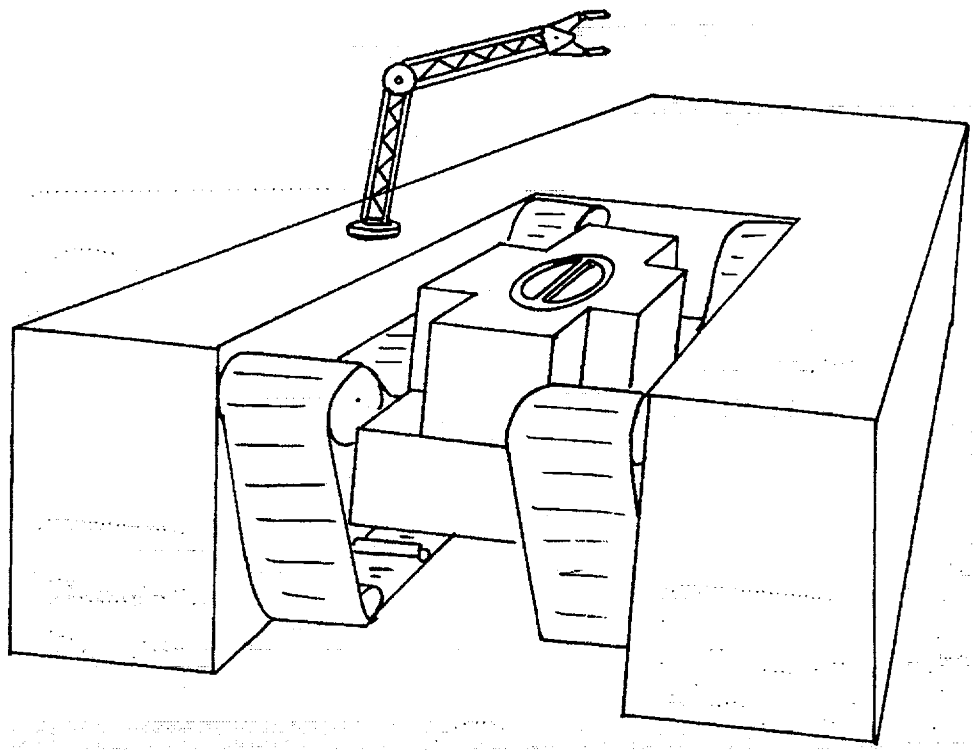


Figure 2.2.1 MADEX Base

2.3 Cable Control Units

Atop the cliff wall and separated by 60 m, two CCUs will be situated. Each unit will have three parts: 1) cable spool, 2) friction brake, and 3) winch. The cable spool must be large enough to hold approximately 350 meters of cable. During the descent stage of the MADEX, the winch will be inoperative. The rate of descent will be controlled entirely by the friction brake. Likewise, during the ascent stage, the brake will be inoperative as the winch pulls the unit to the top of the cliff.

2.4 Communication Devices

The transfer of power and data between the base and MADEX is through a single cable. The spool of extra cable will be stored in the base. Power from the batteries in the base will be delivered to MADEX through this cable, along with instructions as to when to drill, take pictures, and collect spectroscopy data. The sonar readings, digital images, and spectroscopy data taken by the MADEX will be sent to the computer memory bank in the base through this cable as well. Because the map of the cliff made by sonar readings is held within the computer in the base, it must send signals to the CCUs as to when to release slack, when to stop the unit, and when to return it to the top of the cliff.

3. Drills

It is essential that during our stay on Mars we drill core samples. With these samples, it will be possible to return pieces of the Martian planet to Earth for further analysis. The samples that the MADEX will return will range from the surface to 250m below the surface of Mars. This will allow sufficient knowledge of the planet's history and composition for future studies to be designed specifically for certain areas of research. This task is not easy, and our goal was to design a system that would allow for low probability of failure, low mass, minimum power consumption, but high quality samples.

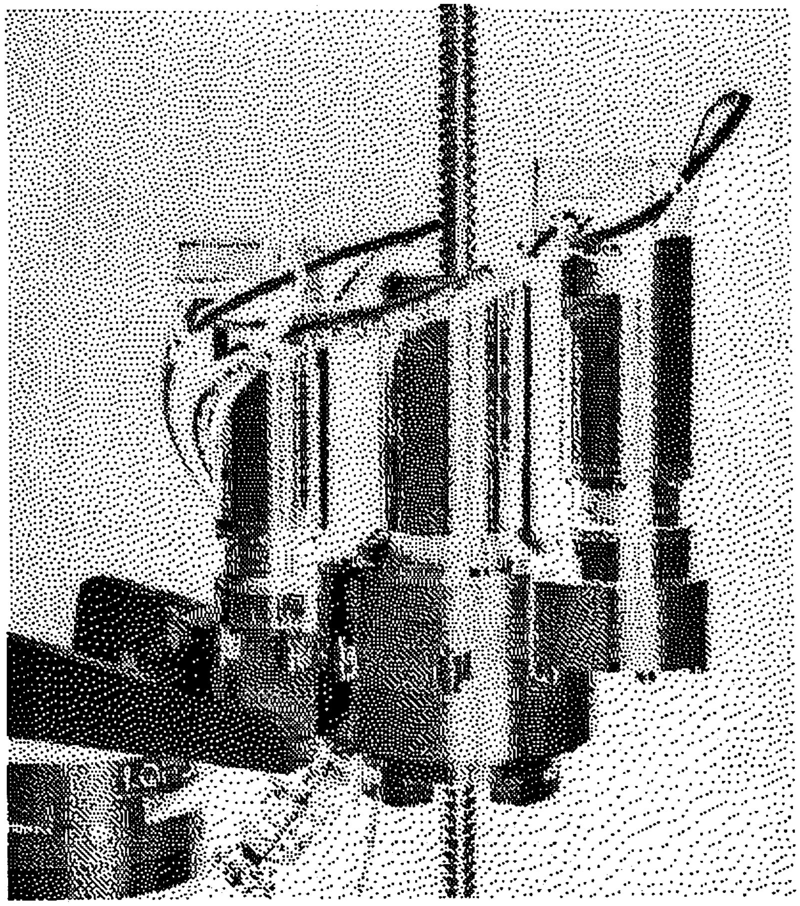


Figure 3.1.1 NASA's Drill

3.1 Sampling Methods

NASA has developed a drill for extra-terrestrial purposes (Figure 3.1.1). This drill weighs about 2.2 kg and runs off only 20 Watts. We have selected this drill for these reasons. The motor has been designed to work in temperatures found on the Martian surface. The drill bits we will use with this drill are diamond bit, and have an inner diameter of about 2.5 cm. The samples will be 15 cm in length. The drill may be attached to a computer system that will "tell" the drill motor how fast to drill depending on what the spectrometer readings are. The motor may drill into the surface from 0.0825 to 0.49 cm/min. This results in the samples taking from 30 to 180 minutes to drill.

At each level that drilling will take place, two samples will be taken. This will be done using two separate drills situated opposite each other within the drill cylinder. Two samples will be taken in case one becomes damaged. This will also allow two separate samples to be analyzed more carefully once returned to Earth.

3.2 Sample Return

As the drills pull the sample into the cylinder, the sample will remain within its respective hole as the drill bit pulls out into the upper portion of the cylinder. With the top portion of the cylinder immobile, the lower portion turns 20 degrees, thus positioning the next two holes beneath the drills. This cycle repeats itself until 18 samples have been taken. At this point, the MADEX returns to the cliff top and moves to the base.

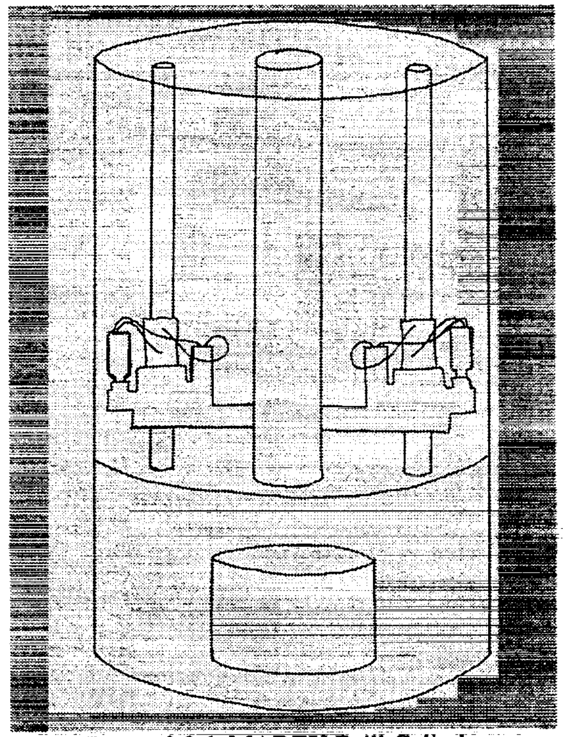


Figure 3.2.1 MADEX Drill Cylinder

Once the MADEX reaches the base, a mechanical arm reaches down and pinches the top, thus "unlocking" the drill cylinder. The cylinder is then removed and placed within the base. An empty cylinder is then placed in the MADEX so that collection may resume. As collection is performed, the base removes, labels, and stores the samples from the first cylinder. When the unit again reaches the top, the cylinders will be switched again.

This system requires two cylinders to be sent. As a precaution, however, a third will be sent. Thus, in case one of the drills is damaged, two cylinders are still available. Thus, in total, six drills must be sent.

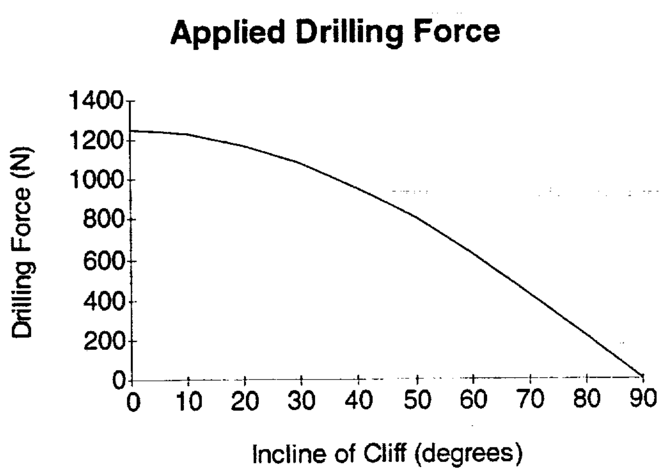


Figure 3.1 Drilling Force Applied at Various Inclines
(Assuming MADEX weighs 750 lbs. on Earth)

4. Spectrometers

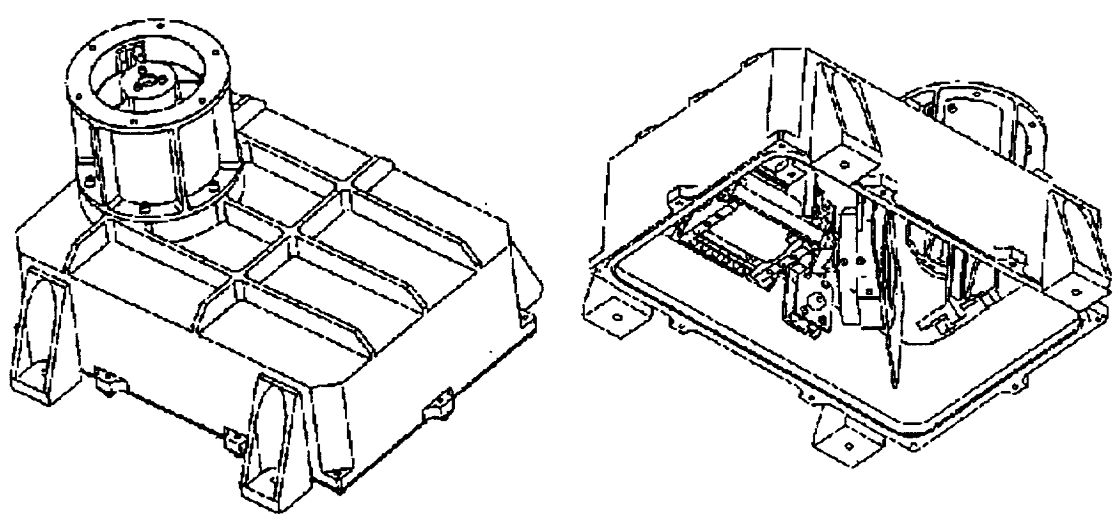
Spectrometers play a vital role for the MADEX. They will assist in uncovering some secrets of Mars by revealing: the proportions of soil bearing elements on Mars, locations of the geologic layers in the soil, previous hydrothermal settings and areas of volatility, and possible water tables.

There are different types of spectrometers with a wide variety of capabilities. One is the Thermal Gas Analyzer (TGA). Using the highest quality laser spectrometer technology, this spectrometer seemed to be the best in its field. However, due to its physical and power requirements, we found that it was impractical for the MADEX. TGA analysis of samples taken by the MADEX is desirable. It may be practical for the TGA to be at the habitat or on Earth rather than at the cliff, itself.

Two other spectrometers that we found to be more practical for MADEX were the Mini-TES and the Mössbauer spectrometer. Both are practical due to size and mass. One of each of these spectrometers will be placed on the MADEX and used at each stop made by the unit.

4.1 Miniature Thermal Emission Spectrometer (Mini-TES)

The Mini-TES, (Figures 4.1.1 and 4.1.2) which is currently being designed by Arizona State University and the Hughes Santa Barbara Research Center, will address geologic and atmospheric science objectives through the study of the mineralogical and physical properties of Martian rocks. The Mini-TES excels at the recognition of aqueous minerals, such as salts that were formed on hydrothermal springs. Salts can provide us with valuable information on the evolution of the atmosphere and its interaction with the surface. The Mini-TES will also search for carbonates, sulfates, phosphates, condensate, hydroxides, oxides, and silicates. In addition to mineralogy, the Mini-TES is able to provide information on thermophysical properties of rocks and soils. It can measure dust aerosol abundance, condensate, gas content, and pressure of the atmospheric boundary layer.



Figures 4.1.1 and 4.1.2 Miniature Thermal Emission Spectrometer

The Mini-TES is designed so that it can examine a specimen as small as 1 mm and determine the mineralogy of the individual grain. The Mini-TES can acquire useful data in 16 seconds in 20 μRad mode or in 120 seconds in 7 μRad mode. The system takes data from wavelengths within the range of 5 to 40μm. This range allows the instruments to penetrate through dust coatings and weather rinds on rocks. The Mini-TES is set up in such a way that the fore-optics, spectrometer, and electronics may be configured in whichever fashion would effectively utilize space within MADEX.

Table 4.1.1 Miniature TES Specifications

Parameter	Mini-TES
Spectral Range	2 – 25 μm
Spectral Resolution	10 and 5 cm^{-1}
Field of View	5 and 20 mrad
Detectors	Uncooled Deuterated Triglycine Sulfate (DTGS) Pyroelectric
Cycle time per measurement	1 and 2 sec
# scans to achieve SNR of 400 at 10 cm^{-1}	15 (on 20 mrad); 110 (on 5 mrad)
Bits per spectral sample	12
Bit Rate	2-6 bits/sec
Size	18 x 20 x 32 cm
Mass	1.9 kg
Power	4.4 W operating; 0.28 W daily average

The Mini-TES has the ability to take data 360° around its position by means of a scope. Because the main focus of data collection by the MADEX is the cliff wall, this scope is not essential. Without the scope, the Mini-TES has a mass of 1.9 kg, and uses 4.4 W of power.

4.2 Mössbauer (MOS)

The Mössbauer specializes in detecting the oxidation states of iron-bearing elements. Oxidation states are the key to determining the ratios and the timeline of the various rock and soil compositions and might lead us to areas of high volatility such as a water table.

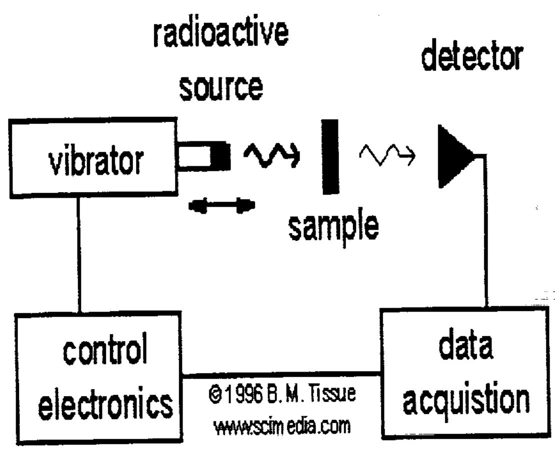


Figure 4.2.1 Mössbauer Spectrometer

The Mössbauer excels in the magnetic study of ferromagnetic samples taken (Table 4.2.2). The Martian soil consists of different levels of oxidative magnetic iron-bearing compounds such as magnetite, maghemite, and pyrrhotite, which bears directly on the formation of different rocks throughout geologic time. This will give us information as to whether Mars has had large river valleys or other types of environmental or chemical weathering. The MADEX is not time limited. Core samples will be taken cautiously and slowly. After each sample is taken, the Mössbauer will take temperature readings and provide information on the magnetic phases and, subsequently, information on the different layers of Martian rock.

Though the Mössbauer excels at recognizing those compounds listed in Table 4.2.2, it is also useful at detecting other mineral groups such as silicates, carbonates, phosphates, and nitrates, which are vital in determining precious geologic information on the layers and the environmental activities.

Table 4.2.2 Mineral Phases and Detection Limits

Mineral Phase	Detection Limit (%)
Hematite (αFe_2O_3)	2.0
Maghemite (γFe_2O_3)	2.0
Magnetite (Fe_3O_4)	2.0
Goethite ($\alpha FeOOOH$)	2.0
Lepidocrocite ($\gamma FeOOOH$)	0.3
Triolite (FeS)	1.0
Siderite ($FeCO_3$)	0.6

The Mössbauer spectrometer is extremely small (250 cm^3) and uses very little power (0.6 W). The Mössbauer is therefore practical for application with the MADEX Unit. Other benefits of the Mössbauer are that no sample preparation is necessary, thanks to the use of back scatter geometry (Figure 4.2.1 and 4.2.4), and it is designed to electronically store the collected data within itself. Despite this advantage, the data will be sent to the base atop the cliff as a safety precaution.

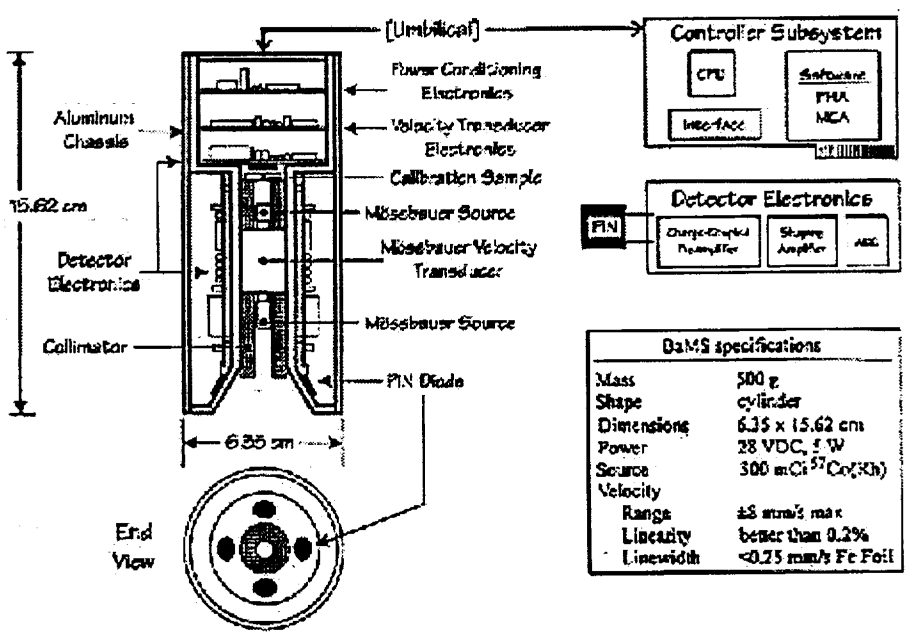


Figure 4.2.3 Mössbauer Spectrometer

Table 4.2.4 Mössbauer Spectrometer

MOS	Mass (kg)	Dimensions (cm)	Op. Power (W)
Electronics	0.15	1.0 x 4.5 x 8.0	0.6
Sensor	0.25	3.1 x 4.5 x 8.0	

5 Power

5.1 Batteries

We have selected batteries made by Northwest Energy Storage (NWES) for use with the MADEX. NWES' Deka batteries have an output of 12 Volts, and 180 amp hours (2.25 kW-hrs). Each battery weighs 168 lbs. and is 10" x 11" x 20.8", and costs $419 from the company. The Deka battery utilizes Gel-Cell technology, which allows the battery to work without damage at temperatures as low as -40° C. The battery's capacity drops by only 2% every month.

By setting up a number of these batteries in parallel we would be able to leave the site unattended for longer periods of time and allowing more time to charge up the replacement batteries.

Table 5.1.1 Battery Specifications

Model	NES GC-8D
Volts	12
Amps Hours	180
Watt Hours	2,250
Dimensions (inches)	20.8 x 11 x 10
Weight (lbs.)	168
Cost	$419

5.2 Human Activity Recharging Trainer (HART)

The crew's health, both mental and physical, is vital to the success of this mission. With this in mind, MADEX will include the HART system. HART will allow the crew to perform the many necessary hours of exercise to maintain muscular strength. In addition, HART will provide power to operate MADEX and its subsystems. HART will also satisfy the goal of using available resources. Without a system similar to HART, the valuable energy expended during the crew's exercises would be wasted.

The need for exercise in a low-g environment is strongly supported by human experience aboard the MIR Space Station. The cosmonauts, who did not do their exercises regularly, required stretchers upon return to Earth, whereas the American Astronaut Shannon Lucid walked away by her own power. She followed her exercise routine daily. This physical activity, likewise, is needed to help astronauts keep their strength on their journey to Mars so they are productive upon arrival. The micro-g during space travel is replaced by a 3.8-g environment on Mars. Human exercise is a key part of the daily life on Mars. HART simply uses the energy for a meaningful and desired purpose.

5.2.1 Health Benefits

HART offers both mental and physical health benefits to the crew. The system will train both the upper and lower body, by means of pedals and a pair of reciprocating handlebars in the form of an exercise bicycle. An alternator and flywheel assembly will provide resistance for the astronauts.

The mental health of the astronaut will also benefit from HART. The astronauts will tend to feel that they are more in control of the fate of the mission. Without this motivation, they may become overwhelmed by work and ignore their physical needs. The trainer will force them to use the equipment in order to power one of the critical pieces of scientific equipment. Exercise has also been proven to relieve stress. The surface of Mars will be an extremely high stress location, and being able to get away from this situation without feeling guilty will be invaluable to the future astronauts.

5.2.2 Power Supply Benefits

The HART system will deliver the power that the MADEX requires to operate. The human body can generate approximately .5 hp for up to an hour (Figure 5.2.2.1). If a crew of six astronauts each exercises for 60 minutes every day, a total of 2.237 kW-hrs will be produced each day. This will approximately charge one battery to completion. However, losses are inherent with a charging device. We are approximating a 20% efficiency. This would mean that one battery would be fully charged in about five days.

5.2.3 Design

HART will use a basic stationary bicycle set up with some minor adjustments. The pedals and arm booms will turn a gear which, in turn, will spin an alternator (Figure 5.2.3.1) used to charge batteries. The battery will have a small LED which will indicate when the battery is fully-charged. At this point, a new battery is connected to the alternator allowing for more exercise and stored power.

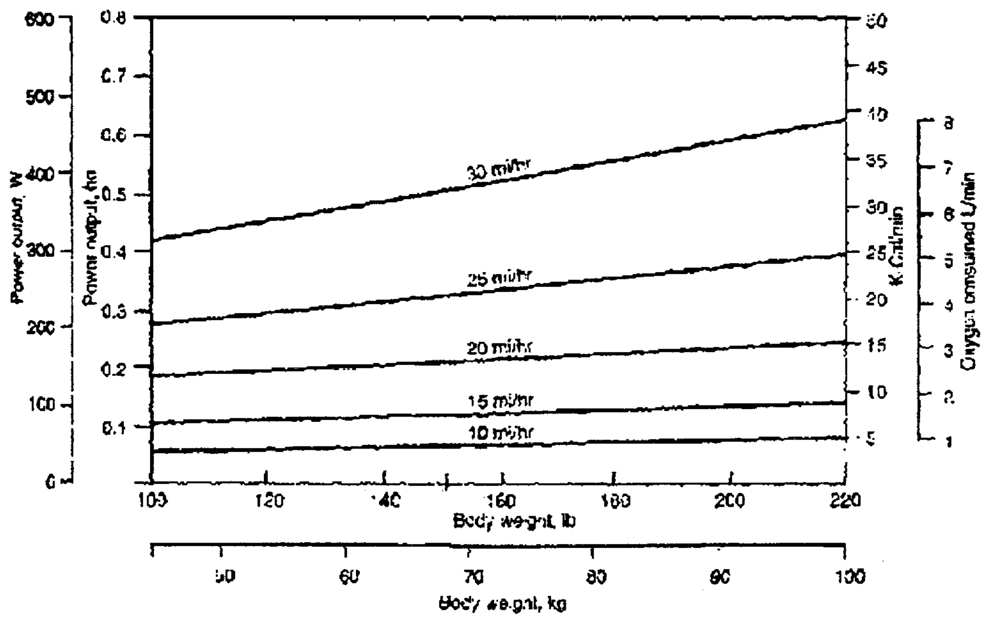


Figure 5.2.2.1 Human Output

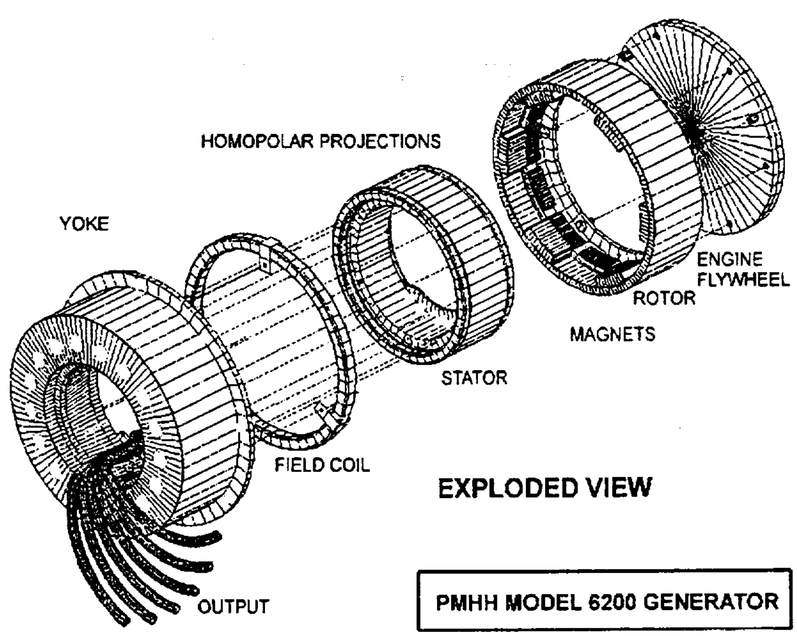


Figure 5.2.3.1 HART's Alternator

6 Results

Once the MADEX returns to the top of the cliff wall, the core samples will be returned to Earth. With the ability to use equipment unavailable or impractical to send to Mars we will be able to further analyze the samples. In comparing data taken on the surface with experimentation of the samples on Earth, we could create a timeline of Mars' history. We would know what the Martian planet is composed of. We would know where to find a supply of water. If we are extremely lucky, we will find fossils within the samples proving that life has existed on Mars. If not so lucky, we will know how deep to look for possible fossils. We will know much more about Mars than we know today.

7 Conclusion

MADEX is a design which we believe has great potential. While in its conceptual stage there are still a number of issues to be explored and resolved. Primarily stability while on the cliff face with regard to overturning due to wind and the lack of normal force desired to overcome the drill force reactions. We believe that the former has been sufficiently addressed by the use of outriggers previously mentioned. The latter case could be resolved by the use of pitons fired into the rock wall and then anchored to for drilling, or perhaps an auger system that screws itself into softer mediums. In either case once the drilling at the particular level is completed the anchor is released or unscrewed and the MADEX is moved to the new location.

Additional areas of further research are listed in *Figure A*. Any and all of our proposals will be fully tested and modified when NASA adopts the design.

Table 7.1 MADEX Specifications

Part; quantity	Mass (per) (kg)	Power Required (per) (W)	Volume (cm^3)
Drill; 2	2.2	20	TBD
Drill Bit; 2	TBD	N/A	N/A
Drill Cylinder; 1	TBD	Negotiable	~24,000
Mössbauer; 1	0.4	0.6	147.6
Mini-TES; 1	1.9	4.4	11,520
Computer; 1	TBD	TBD	TBD
Sonar; 2	TBD	Negotiable	TBD
Tread Motor; 2	TBD	TBD	TBD
Leg Motor; 4	TBD	TBD	TBD
Sum of Known Parts	6.7	45	35,667.6

Future Studies

1. Exploration of Different Locations

By exploring more locations (*Figure A*) with the MADEX, we will create a more general timeline for Mars. We will then know approximately where to find different layers at various positions around the Martian planet.

Figure A Landing Sites For Future MADEX Missions

Site	Location	Landform	Reason for Interest	Problems With Site
Chasma Boreale	85 N, 110 W	Canyon in polar regions; between the residual ice cap and layered Martian terrain	Conducting climatic studies; further analysis of ice caps	High latitude may cause difficulty in utilizing human interaction
Ganges Chasma	10 S, 45 W	Canyon in Valles Marineris	Study debris from large landslide and more analysis of cliff layers	N/A
West Candor region (primary site selected for SIMM manned mission)	6 S, 75 W	Canyon in Valles Marineris	Further analysis of layering on Mars	Difficult terrain; the proposed site is 50 km from canyon wall

2. Materials

Studies exploring what materials best suit the MADEX unit are necessary. Cost, weight, and coefficient of expansion are all important factors that must be considered. The MADEX frame, treads, lubricants, and cables.

3. Testing

Once the first MADEX unit is created, tests must be run to ensure the reliability of the unit. First the MADEX must be tested in normal Earth conditions followed by fine tuning of communications and storage use. Then the unit must be tested in simulated Mars conditions, such as in Antarctica. This must be followed by further fine tunings. One the unit proves to work efficiently, it is ready for use on Mars.

4. Backup System Check

It must be ensured that if something goes wrong when the unit is put to use on Mars, any necessary repairs may be done by astronauts present on the surface or automatically by systems present in the base.

5. Exploring the possibility of a portable MADEX unit

Further development of the MADEX may lead to the possibility of a system where one cliff is fully explored before the unit is moved to another cliff on Mars. Should this be possible, a better understanding of the entire planet's composition and history could be achieved.

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Interactive Design Environment

Tools for Facilitating Communication and Collaboration Among Universities on Projects Related to a Mars Mission

University of California, Berkeley

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1.0 Introduction

The HEDS-UP program is comprised of student groups from many different universities across the United States working independently on various aspects of the grand objective – a manned mission to Mars. The inherent value of the program is in the nature of the students working in it. Students offer a different perspective on an existing project. Their contribution is in bringing the off the wall ideas to the table, among others. Students are unbounded by tradition and precedents in methodology. This enables them to approach the problem from a unique angle. They have the potential to bring fresh ideas and new dimensions to the overall project, thus contributing something original rather than mimicking existing projects. With proper facilitation the HEDS-UP program can become an evolutionary dynamic environment in which ideas are proposed and tested under pressure and those with sufficient merit survive. Moreover, the incredibly cheap price of student labour gives the HEDS-UP program enormous potential to provide a substantial and lasting contribution to the Mars mission.

The potential value of the projects completed by the HEDS-UP universities is limited by the geographical and academic separation of the universities, the short term nature of the projects, and insufficient input from NASA. If communication exists between the universities at all, it is minimal and limited to the conference. The projects are limited by the school term and the turn over rate of the participants is exceedingly high with an influx of new students each semester. This means that much of the work from previous semesters is lost as it is improperly passed on, incompletely understood, and consequently disregarded. There is no consistent method employed across the universities for storing the information and making it accessible to others in the field. Moreover the projects suffer from a dislocation from NASA itself. The insufficient feedback and inadequate resources for the projects limit their technical content. If a means of overcoming these limiting factors is found, the Mars mission project could then fully take advantage of the enormous pool of talent that currently exists within the HEDS-UP program.

The combined projects offer the very real possibility of contributing to a mission to Mars without simply replicating what NASA is already fully equipped to do. Our design proposal set about to solve these existing problems so that the HEDS UP program can flourish. Better communication between university projects would lead to proposals that can be better integrated into an overall mission design. They would be able to consider a broader context and work within those constraints. Communication is of particular use to students working on mutually dependent or similar projects. The exchange of ideas rather than facts would greatly further the conception of design projects.

It is important to keep the work of previous semesters. This is not to suggest that it will all be correct or even useful, but it provides a basis which further work can develop and give a greater understanding of the mission constraints. In fact, learning from past mistakes could be one of the most valuable assets to come from the compilation of such a body of work. This past work could provide an even greater foundation if it were broader, and therefore a compilation and organisation of all the projects proposed throughout the HEDS UP program would be immensely useful for creating a more complete picture of all the intricacies and implications of a mission to Mars.

When the projects are fully linked to NASA they can become technically sound and, as such, gain validity as serious design projects. They contain ideas that need a solid grounding in data in order to be considered and also give a more immediate indication as to their feasibility. An easy channel of communication would enable the exchange of data between project designers and those at NASA that have the pertinent information. This is especially useful as the Mars mission is highly specific and the number of experts in related fields is acutely limited.

Problem Statement: Our aim was to develop a tool that overcame the barriers of geography and limited communication to provide an interactive design environment which fostered academic communication and continued project development in order to further a mission to Mars.

2.0 Methodology

The organization of UC Berkeley's class into separate but concurrent design projects was highly similar to that of the HEDS-UP program and suffered from the same limitations. The Interactive Design Environment was created to meet the needs of the class and aid communication between the groups, compile past work in an accessible and useful manner, and provide a bridge between interdependent groups. Additionally it was designed to be efficient and easy to use with specific tools to meet different needs such as chat capabilities, a method of posting ideas, and storage and retrieval functions. Moreover, specific applications, namely MarsHOT (Habitat Optimization Tool) and CAPS (Computer-Aided Power Simulator), were integrated into the project to allow easy and direct comparison of projects in the fields of habitat and power, respectively. In conjunction, these tools create an interactive forum for the exchange and development of concepts with applications to analyze design proposals in a consistent manner

Direct parallels can be drawn between the structure of the HEDS-UP program and UC Berkeley's class. HEDS-UP comprises a collection of universities working on separate projects. Our class mirrored this on a smaller scale, as it was divided into several project groups working on differing but related aspects of the Mars mission and, as such, suffered from many of the same limitations faced by the HEDS-UP program. The IDE was designed around the class with the intent of overcoming those problems and also with the foresight that it could be scaled up to cover the entire HEDS-UP program. This would enable easy communication not only among related groups at one university, but also among groups at different universities.

The IDE is based on Internet technologies, therefore scaling it to service larger groups located disparately is not even necessary -- it is already capable of doing so. It cannot, however, scale indefinitely. At some point a system must develop whereby each university or research center will have its own IDE, or perhaps multiple IDEs, each with its own database, and access to all IDE databases through a common set of protocols.

2.1 IDE Methodology

Easy and efficient information exchange is vital to scientific advancement. The IDE was created with this in mind. The proposed design is one that provides an interactive web site that houses a suite of tools for data storage and retrieval and for collaboration and communication on many different levels. Additionally, specific tools created to aid design project work are built into the IDE to increase its functionality. The site and tools therein are easy to use and require minimal time to learn. This is a general design principle which any good software project should follow, and it is especially important in the case of tools such as those the IDE provides, because easy access facilitates better communication.

No special software or hardware is required to use the IDE beyond the minimal requirements for accessing the World Wide Web. Custom software is especially cumbersome when wide deployment, as is the intent of the IDE, is involved. There are multiple concerns, the foremost being the requirement that the software be made available for multiple platforms (e.g. PC, Mac, UNIX). By necessity this adds complexity to the software in both development and maintenance, and drastically increases the amount of time required for each.

The IDE's design is simple and utilitarian. Fancy graphics, sounds, and other bells and whistles, while nice, tend to increase complexity of design and add greater load to the system without adding much functionality. Such features were to be used sparingly, and only if they added considerable functionality. This has the additional benefit of enabling the IDE to be accessed by users with limited access to the Internet, such as older software or limited terminal capabilities.

The design was developed with the needs of students in mind. There already exist products which mimic many of the functions of the IDE, but most of these products are targeted at commercial users, and thus assume greater capabilities of both the server and client computers than can reasonably expected from students with limited means. These off-the-shelf products also do not tend to integrate well, meaning that the users must learn multiple interfaces, and must often manually transfer data from one tool to another.

Multiple design elements have been created that facilitate communication between participating universities or other concerned parties and aid development of their design projects. These are similar in both application and implication. The *chat system* provides immediate text-based communication between individuals. The ideas proposed can be posted, with optional complimentary graphics, for further comment on a *virtual whiteboard*. The refinement of the designs requires a foundation. This is provided by a *digital library* which houses all contributions made, both published and not, to a research or design effort. These tools are supported by a software infrastructure built around a database subsystem. They provide both communication and collaboration abilities for the user through a customized web server. The IDE also has a simple and consistent interface that meets the user needs, as detailed above, while presenting a solution to the problem faced.

2.2 CAPS and MarsHOT Methodology

In addition to the communication and collaboration tools discussed previously, specific design tools have been developed that demonstrate the usefulness of the IDE for specific design projects. These two applications, MarsHOT (Mars Habitat Optimization Tool) and CAPS (Computer-Aided Power Simulator), demonstrate the use of the IDE as a forum for comparison between specific projects. As such they can provide a consistent basis for determining an idea's feasibility as part of a design project and highlight areas for further refinement and development. Most importantly, the applications themselves can be further refined and improved upon as the problems facing the Mars mission continue to be identified and defined.

Along with the communication and exchange of ideas, design teams require a consistent format for making comparisons between proposed and existing designs. Without such comparison ability it is a purely subjective decision as to whether they are feasible. Two such tools have been developed and integrated into the IDE in order to provide such a yardstick for comparison. MarsHOT and CAPS are specific applications that provide mathematical analyses of design missions based on specified parameters and independent variables. There was tandem research and development of these tools, because in order to make a comparison, one needs a standard with which to compare. The control variables, equations, and overall concepts for these tools were determined by research conducted in the fields of power and habitat design.

These communication tools can be used on a local all the way though to a global level. They were developed to support research groups and their utility is not limited to a single university. Their integration into the IDE assists research and design efforts when the participants are in disparate geographical locations. These tools are the text-based chat system and the virtual whiteboard.

3.0 Results

3.1 Chat System

The chat system allows participants to meet online to collaborate in real-time. Any participant in the chat system may create a channel (or "chat room") which other participants can join, subject to the access restrictions the creator of the channel specifies. All users present in a chat room receive all messages directed to the room, but users may also direct private messages to a specified person or persons. This facilitates open as well as confidential communication.

A special function of a chat room's integration in the IDE is the ability of participants in a chat room to select documents from the IDE's digital library to present to others in the room. The chat system is integrated with the database functionality of the IDE, and because it is accessible using a standard web browser, it can easily provide the ability for multiple users to view the same document at the same time and share hyperlinks to external sites on the World Wide Web. The participants also have the option of having the chat system record their chat session into the database for future reference by themselves or others.

3.2 Virtual Whiteboard

The chat system is, however, inherently limited by both its immediate nature and its confinement to the printed word. The virtual whiteboard was designed to overcome the necessity of real-time participation by all involved. It allows contributors to post ideas for further perusal at a later date. Additionally, it provides the ability to present not just text, but graphical representations of ideas, through what have been dubbed "virtual cocktail napkins," in recognition of the many great ideas born on said medium. These small drawings are composed and submitted at the same time to compliment the textual explanation of an idea.

The virtual whiteboard facilitates discussion between participants who infrequently or never meet face-to-face, but unlike the chat system, it is for extended discussions occurring over a period of hours or days. Any registered IDE user may create a whiteboard or participate on any active whiteboard. The whiteboard's functionality is made available through the World Wide Web, therefore any person with web access may view the whiteboards in the database. Access controls are also provided so that the whiteboard's creator may restrict the ability to view or contribute to the whiteboard should this be desired.

A virtual whiteboard, shown in figure 1, appears very much like a real whiteboard, with bulleted lists of ideas and scattered sketches, but underneath lies a powerful threaded messaging system. Participants can associate hyperlinks with their entries on the whiteboard, referencing entries within the IDE database, such as documents from the digital library, transcripts of past chat sessions, archives of mailing lists, or even other whiteboards. Other whiteboard participants can reply to an idea posted on the whiteboard with a comment of their own, causing a hyperlink to appear below the idea.

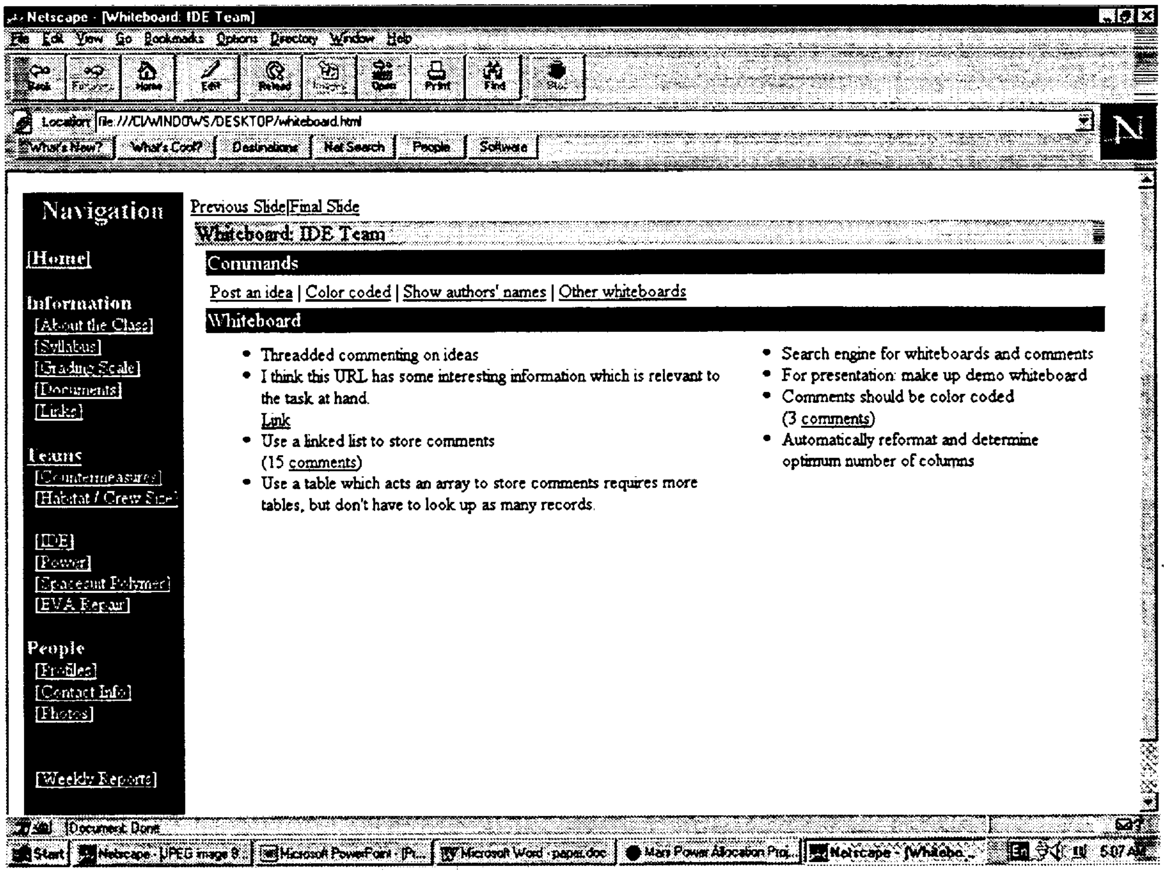

Figure 1. Screenshot of Virtual Whiteboard.

The functionality of the virtual cocktail napkins is provided by a small application (applet) which is executed within the user's web browser. The applet is written in the platform-independent Java programming language, meaning that any computer user with a standard modern web browser would be able to use this tool.

3.3 Digital Library

There is a recent trend moving away from the traditional distribution of paper documents toward their distribution in digital form. As such, it is becoming clear that traditional libraries are no longer the most efficient mechanism for the archival and dissemination of large volumes of information. When the documents a researcher is interested in are available in digital form, the time involved in manual retrieval is eliminated by instant electronic retrieval at the researcher's desktop.

The digital library, shown in figure 2, includes not only published material but all contributions made to a project, in a variety of media. These contributions could include transcripts of chat sessions, completed whiteboards, and additional postulations. The initial, most basic form of the usable digital library consisted of a collection of documents on computer media and a user interface for document retrieval. Extensions to the functionality of this library include categorized listings of documents, keyword searching, and hypertext cross-referencing of footnotes and endnotes.

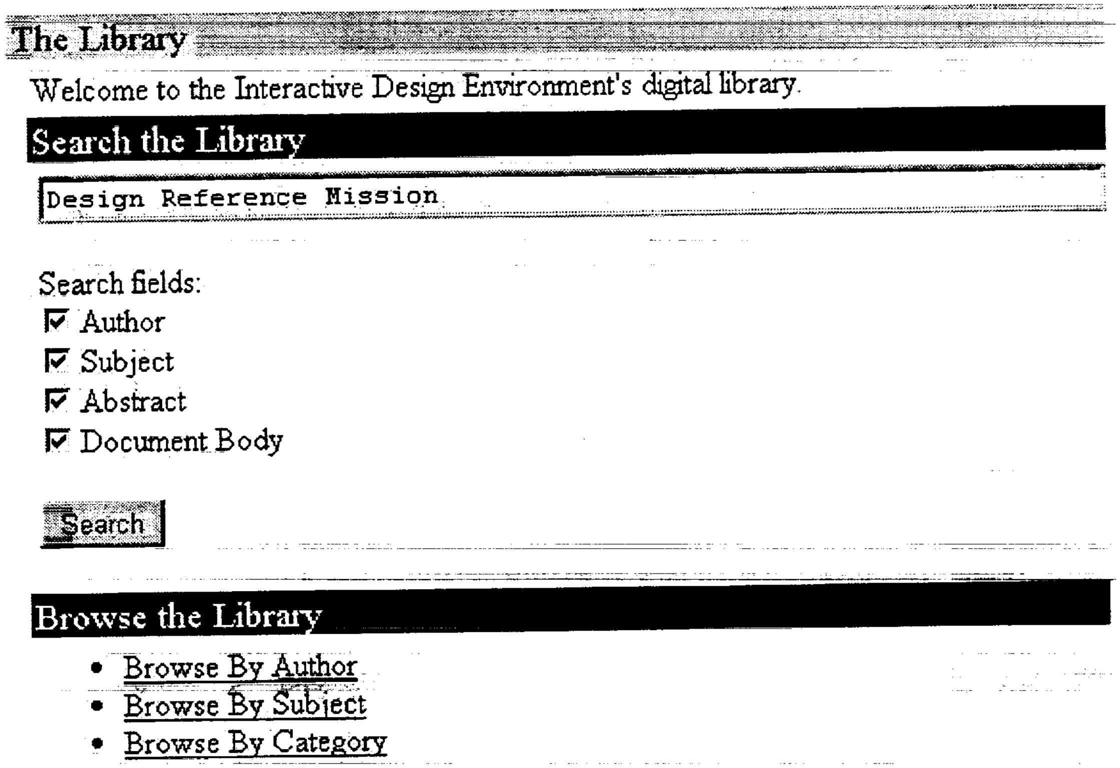

Figure 2. Screenshot of Digital Library.

The IDE digital library includes these extensions, and goes one step further to extend the degree of interaction. Registered IDE users are able to annotate documents and to provide hypertext links to other related documents in the library or external to the library (e.g. World Wide Web addresses). Documents within the library can also be referenced from an IDE chat session or virtual whiteboard.

Access to the library is through a web-based interface. The actual library itself is stored as a Structured Query Language (SQL) database using the database backend of the IDE maintainer's choice, as the IDE's modular nature provides a layer of abstraction between database accessories (i.e. the document library interface) and the communication between the IDE and the database backend.

The interface provides the user with the option to browse the library by author, by category, by date of document publication or inclusion in the library, or to search by any of the aforementioned listings or by keywords within the documents. From there, based on the user's selection, a list of matching documents is presented along with the option to narrow or redefine the search.

Upon selecting a document, the user is presented with the best possible presentation of the document. If the document is in HyperText Markup Language (HTML), it is immediately presented in that form, but if it is available only in a format or formats which cannot normally be displayed by a vanilla web-browser, such as PostScript (PS) or Portable Document Format (PDF), a hypertext link to download the PS or PDF version of the document is presented, along with a short explanatory paragraph about the document's format and possibly an abstract or other short description of the document if available. It is also possible that a document is present in multiple formats, e.g. the original document in PDF format, and an HTML version created by a PDF-to-HTML translation program. In these cases, an IDE user's configured preferences determine whether the HTML document, or a list of document formats to select from, is presented.

Documents presented in HTML format provide the reader with the benefits of hyperlinks to referenced sources and annotations, as mentioned earlier. When presenting documents in formats not renderable by web browsers, the digital library provides any annotations or hyperlinks associated with the document as part of the web page providing the download link to the document file. For registered users of the IDE several hyperlinks to digital library functions for annotating the document and creating hyperlinks to other relevant documents are included in the document presentation. If the user is currently engaged in a chat system conversation, the ability to present the document to other chat participants is also available.

3.4 Supporting Infrastructure

While IDE's user interface provides a convenient way for users to interact with the system, it cannot function without sophisticated support software to provide low-level functionality. The most immediate support facility is the HTTP module, which is in essence specialized web server software designed to provide a common interface for the other components, or modules, of the IDE. The other facility provided is the database module, which provides uniform methods for storing and retrieving information in the IDE's SQL database.

The HTTP module is designed to be compliant with the HTTP/1.1 draft specifications[cite], though it does not implement some portions of the specification's functionality which are not applicable to the IDE system. The module provides the interface between the user's web browser and the other functional modules and handles much of the necessary text parsing and composition related to the HTTP protocol.

The database module provides an interface for modules to read from and write to the IDE's database. Most of the work required to access the database backend is done within the database module, which has the benefits of both simplifying the writing of other modules and making the database module a replaceable component. There are numerous third-party SQL database implementations, each one having its own advantages and disadvantages, so it is desirable for changing the SQL implementation to be as easy as possible. Database access abstracted through the database module, so it is only necessary to modify this module when the implementation is changed, instead of every module which accesses the database.

3.5 MarsHOT

Mars Habitat Optimization Tool, or MarsHOT is a specific application of the Interactive Design Environment. It was designed as a tool for determining the most suitable habitat design out of the hundreds that currently exist. While there are many different proposals, the designs have certain features in common that enable direct comparison between existing and proposed designs. MarsHOT is written in Excel, making it simple to use, and it additionally may be downloaded from the class web site. As such, it is easily accessible to a multitude of users and provides a consistent basis for comparison of habitat designs. Moreover, the program is designed such that it can be improved upon by its users. Its principal limiting factor is the data supplied to it. As the design proposals are refined using MarsHOT, they can contribute to its further particularization and enhance its utility.

A prominent feature of all the designs is the great attention paid to the mass of the habitat. Mass is a premium commodity in the Mars mission due to the enormous amounts of fuel needed to transport the payload safely to the planet. Therefore, mass is used as the optimization criterion against which the proposals are measured in the program. For the purposes of this program, the habitat was divided and analyzed based on the following subsystem: habitat structure, crew accommodations, consumables, CELSS, communications and information systems, medical equipment, rover, airlock/ports, radiation shielding, power generators and science equipment. Also, due to the basic nature of the analysis, the program does not make a distinction between interplanetary and surface habitats.

Design Reference Mission

19 Independent Variables	
Length of Mission (Days)	879
Number of Crew Members	6
Height Per Deck (m)	2.32
Minimum Volume Per Crew Member (m^3)	90.00
Diameter of Habitat (m)	7.50
Aspect Ratio on End Caps	0.50
Material Used for Habitat Design	6061-T6 Aluminum
Ultimate Strength of Material (Pa)	3.103E+08
Density of Material (kg/m^3)	2712.64
Pressure Ports	4
Rover Type	Minerva (Unmanned)
Recycling Efficiency for Water	0.9
Water Buffer	100
Recycling Efficiency for Oxygen	0.8
Oxygen Buffer	50
Shielding Type	Storm
Acceptable Radiation Dose	23.00
Power Supply	DRM
Internal Pressure (Pa)	6.70E+04
Factor of Safety	3.00

Fixed Portions of Habitat	
Exercise Facility (kgs)	770
Communication/Information Systems (kgs)	213
EVA Airlock System (kgs)	3000
Doorlock (kgs)	1370
Scientific Lab (kgs)	1770
Medical Supplies (kgs)	700

Ramifications of Variables	
Material Thickness (m)	0.0024
Exterior Surface Area of Habitat (m^2)	149.27
Mass of Unreinforced Exterior (kg)	983.56
Mass of Reinfroced Exteror (kg)	1081.91
Structural Mass of Empty Hab (kg)	**1622.87**
Number of Decks	5
Total Minimum Volume	540
Radius	3.75
Consumables	**17976.12**
Other Water (kg)	79110
Potable Water (kg)	14767.2
Total Water Used on Mission(kg)	93877.2
Total Amount of Water per day (kg)	106.8
Total Water Needed (kg)	9487.72
Food (kg)	7383.6
Oxygen (kg)	5274
Total Amount of Oxygen per day (kg)	6
Total Oxygen Needed (kg)	1104.8
Crew Accomodations	**552**

The inputs into MarsHOT are 19 independent variables, length of mission; number of crew members; height per deck; minimum volume per crew member; diameter of habitat; aspect ratio on end caps; material used for habitat design; ultimate strength of material; density of material; pressure ports; rover type; recycling efficiency for water; water buffer; recycling efficiency for oxygen; oxygen buffer; shielding type; acceptable radiation does; power supply; internal pressure; and factor of safety; which parameterize the design. The relationships between these variables and the mass of each subsystem have been derived by researching established habitat designs. The NASA Design Reference Mission, Zubrin's Mars Direct, and the Stanford International Mars Mission were the principal designs used. In its simplest function, MarsHOT can take all 19 variables as input and calculate the mass of the designed habitat. From these, small changes may be made to see their effect on the overall mass. As a more powerful use, the program may take a number of the variables as input along with specified mass and solve for the optimum configuration of the remaining variables using Excel's "Solver" function.

3.5.1 Subsystems

In order to calculate the total mass of a habitat it is divided into 10 subsystems. Relationships between the mass of each of the subsystems and the independent variables were derived by examining pre-existing, accepted designs.

Habitat structure

The structure of the habitat, defined as the exterior and interior structure and their supports, is a significant portion of the overall mass. We assume it to be separate from the shielding. However it is difficult to fully incorporate the structure into the MarsHOT application. This is because the interior structure of the habitat is subject to many environmental and architectural constraints that can not be easily represented numerically. Thus the assumption is made that the interior structure will have a mass equivalent to 50% of the exterior structure mass. This estimate is consistent with the habitat design of the NASA Design Reference Mission.[i] Also, of the many materials that can be used to create both the pressure vessel and the supporting structure of the habitat, the data for three known materials, aluminum 6Al/4V; titanium 22-19-T8; aluminum 6061-T6, was entered. Each safety factor of the three was chosen to represent the use of a reliable material in difficult environmental conditions[ii].

The structural mass is therefore calculated as follows. The external surface area is determined from the radius, height, and ellipsoid aspect ratio, and the material thickness is determined according to the formula for a thin-walled pressure vessel, given the material properties, factor of safety, and internal pressure defined by the user. The external area is then multiplied by the material thickness by the density of the material to give the mass for the un-reinforced shell. The mass of reinforcements is taken to be 10% of this calculated value. As stated above, the mass of the internal structure is taken to be 50% of the mass of the reinforced external structure. The total structural mass, therefore, is the external area times the material thickness times the material density times 165%. In summary:

$m'_s = pAt$

$m_r = (0.01)(m'_s)$

$m_{is} = 0.50(m_r + m'_s)$

$m_s = m'_s + m_r + m_{is} = 1.65\ m'_s$.

Crew Accommodations

The crew accommodations are those items or systems that uphold the standard of living for the astronauts. The crew accommodations specified in this report are neither necessary to the immediate survival of the astronauts, nor do they directly contribute to the scientific goals of the mission. The overall mass estimate for the entire crew accommodations, based on the Stanford International Mars Mission with a crew of six, is 1320 kg. Of that, the mass of the exercise facility remains fixed at 770 kg while the remaining mass varies linearly with crew size. The resulting equation for accommodations is
$m = 770\text{kg} + 92N$ when N is the number of crew members. The additional 92 kg for each crew member includes the galley/wardroom, crew quarters, and personal hygiene facilities.

Consumables

Consumables, comprising the food, water, and oxygen that the astronauts will use during the mission, depends on the number of crew members and the length of the mission. Therefore, in the calculations, the parameter used was mass per person per day. However, a factor that greatly affects the mass of consumables is the efficiency with which the habitat is able to recycle them. Each of the major manned-missions to Mars has established their own estimate for the mass of the consumables. By averaging the recommendations it was determined that each person uses 2.8kg H_2O, 1.0kg O_2, and 1.4kg food per day. Additionally, hygiene requirements add 15kg/day per person.

Life Support System

The life support system includes an air revitalisation subsystem, water purification subsystem, and waste management subsystem. Each of these subsystems must be capable of supporting the crew and yet not be too heavy. This balance depends on the recycling efficiency of each subsystem. The NASA design reference mission specifies a mass of 4661 kg for the life support system to support a crew of six people. The Stanford mission and the Mars Direct mission both allocate a mass of 3000 for the life support system. Using these total masses as guidelines, a total mass penalty of 750 kg per crew member was determined for air, water and waste subsystems. Although there are many other aspects of the mission that will influence the life support system, the relationship was simplified to a linear one between the mass of the life support subsystems and the crew size.

$m_{eclss} = 750N$

Communication and Information System

The communication and information systems are the critical interfaces between the crew on Mars and Mission Control on Earth, as well as between the crew and various computer controlled systems of the habitat. The overall mass of the combined system is estimated at a constant 0.3 metric tonnes.

3.6 CAPS

There are many different permutations of power generation systems, each of which is limited in output by different parameters of safety, cost, longevity, mass, volume and necessary redundancy. CAPS was created to efficiently analyze the feasibility of each proposed power generation system for function on Mars. CAPS, shown in figure 4, is a user-friendly and effective application, written in Delphi, a windows object based programming language, that enables the user to set the usage and supply restraints of a power system and run a balance to determine necessary power supply component. The consistent format of the analysis allows for easy comparison of existing and proposed missions. The utility of CAPS is that it can be refined and adapted as technology develops. It provides an analysis based on the information provided, and thus can improve as this data becomes more specific and accurate. The program is an application of logic, which can be used as a tool for further research by other students.

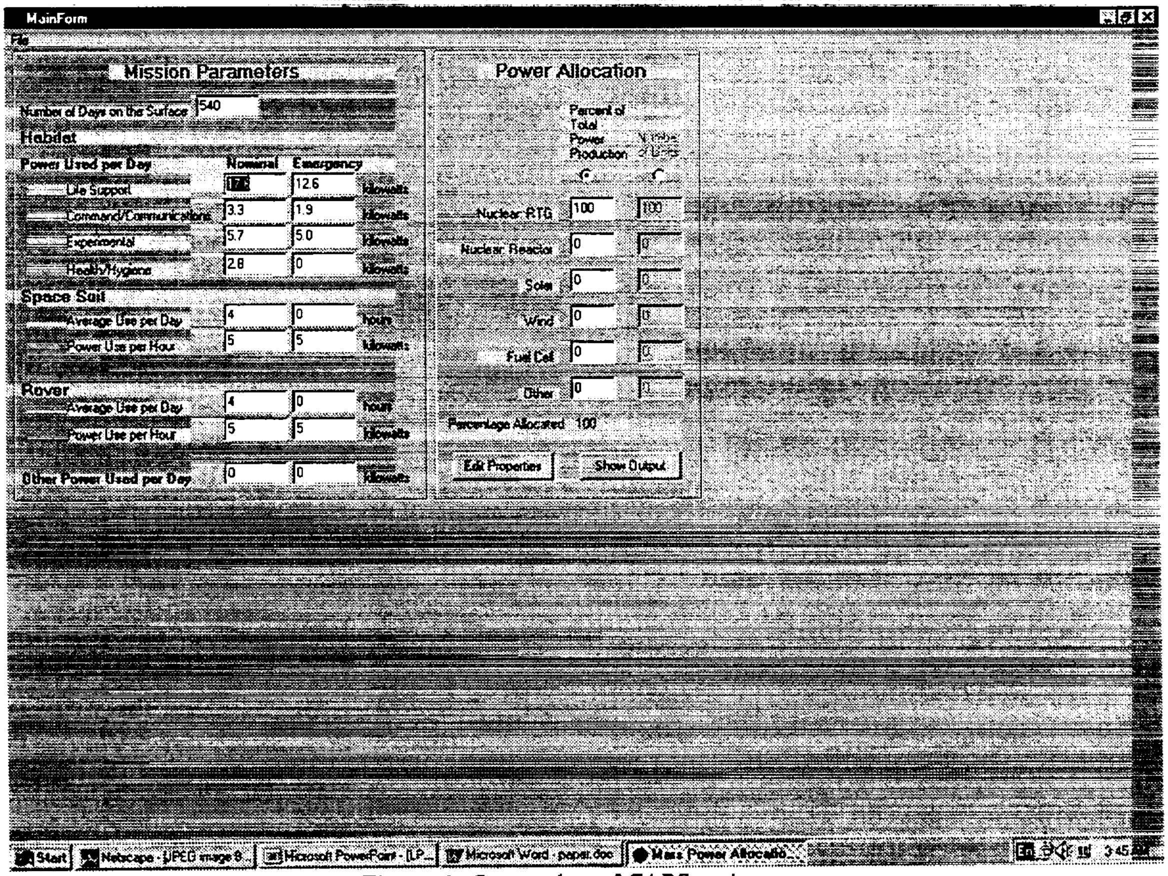

Figure 4. Screenshot of CAPS main screen.

A power generation system is constrained by the many mission parameters of the power generators and consumers. The nature of the Mars mission is such that humans need to live and work in a hostile environment. Therefore, power is needed to compensate for every aspect for this and provide a local inhabitable environment, namely the habitat. This means that there are an enormous number of power consumers. For simplicity's sake they have been grouped into five principal categories: life support systems, scientific equipment, rovers, space suits and communication equipment. The specifications of the mission are confined by the parameters of the consumers, which consist of average power use per day, peak consumption at any given time, and minimum emergency power needs. These values are different for every power consumer and are incorporated into the program as default values. The defaults for both consumers and generators were determined by the extensive research conducted over the course of the project.

Similarly, the power generators have constraints that the CAPS program must take into account. There are certain basic needs that any power source must meet: it needs to be brought from Earth, function safely on the planet, and optimize the cost/benefit ratios. The transport of the generator to Mars greatly limits its mass and volume, because at the heaviest extremes the payload could not be physically supported by the rockets and above certain values the power generators are no longer fiscally viable. Longevity and

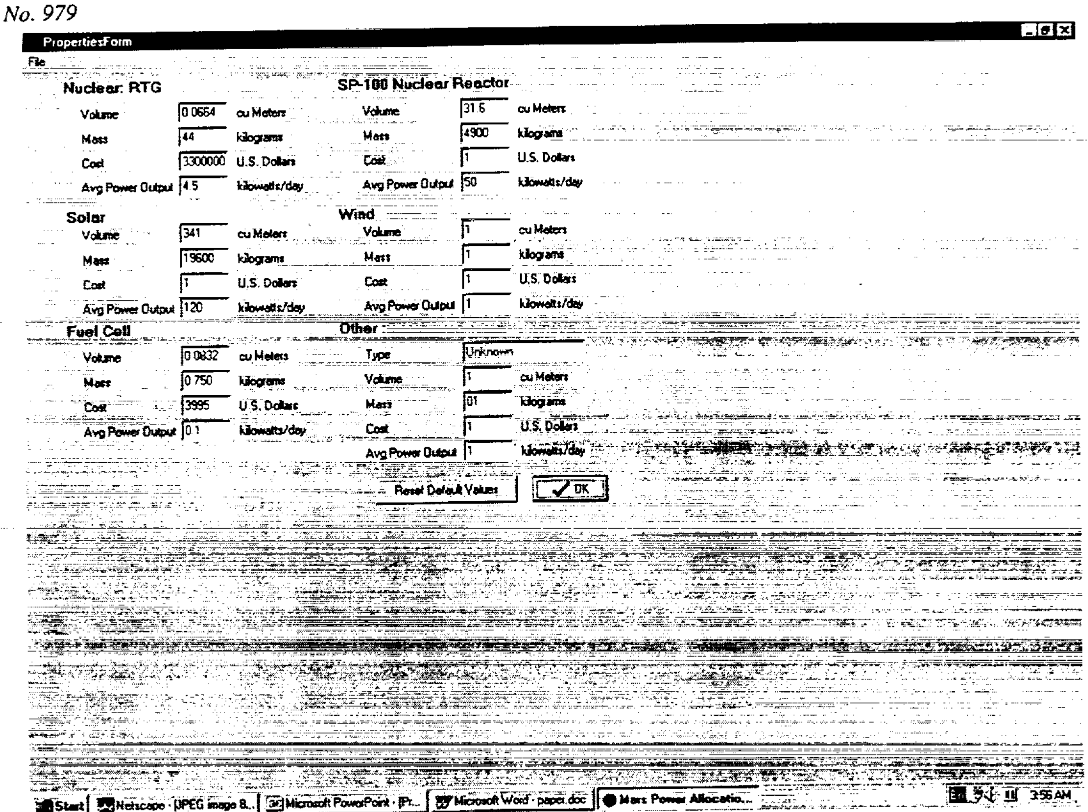

Figure 5. Screenshot of CAPS, Production parameters form.

reliability of the generator are also vital factors in its functioning on Mars. The habitat will be completely isolated from Earth and thus there is no possibility for replacement or aided repair of any generator. The power system as a whole must be able to function safely to preserve the well being of the crew and it must be able to meet the peak requirements of the power consumer as well as satisfy the baseline life support system requirements with at least 50% redundancy. As yet, there is no ideal power source that meets the needs of power generation as well as meeting the transport limitations. Therefore, known power sources were researched and evaluated for a Mars mission using the criteria discussed previously. The power sources researched were nuclear, solar, wind, and fuel cells, and their data provided the technical framework for the application – a basis upon which the calculations can be made.

The CAPS interface allows users to easily change the parameters of power allocation for the mission and redefine the properties of the power generators. Specifically, a user can change the average use of each consumer and any and all of the parameters of each power generator such as efficiency, cost, and power output. The parameters for each power generation system are changed on an secondary screen shown in figure 5. Given these constraints CAPS will calculate the total mass, cost and volume of the system of each generator, as well as cost/power, mass/power and volume/power ratios for each of components. The basic ratios provide a consistent basis for comparison of possible missions.

The user can also specify the components of the system by setting the number of units of each generator to bring, or by setting a percentage distribution. The analysis provided, shown in figure 6, will not only display the ratios discussed, but also determine whether each of the power suppliers would be able to accommodate the entire emergency power requirement. If any of these is in excess or is insufficient, the analysis will show by how much, and also calculate, in the event of an inadequate power supply, how many supplemental units would need to be brought to cover the emergency requirements.

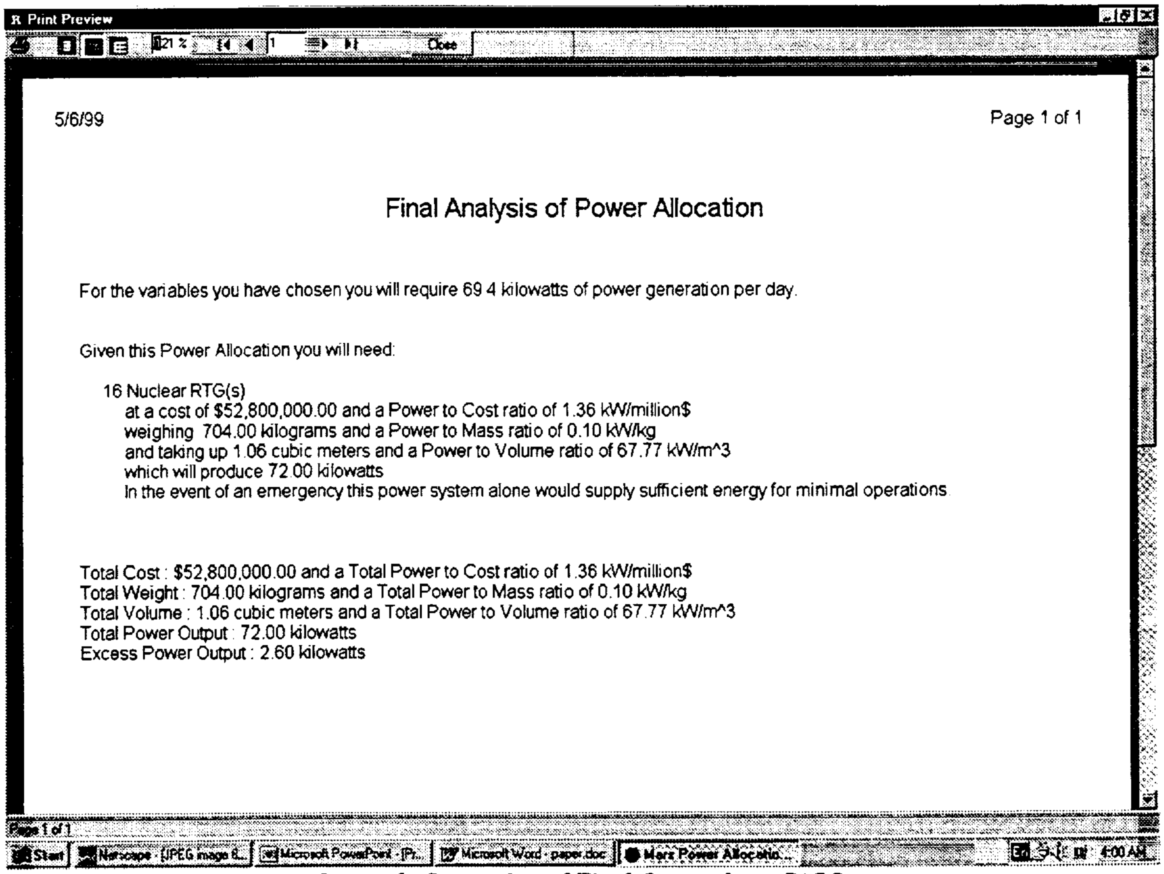

Figure 6. Screenshot of Final Output from CAPS.

CAPS is a dynamic tool, limited by the current understanding of the mission parameters. As these parameters and their constraints become more clearly defined, CAPS can be developed and refined. It currently provides an analysis based on the research conducted thus far. As the design project for the Mars mission develop through the contributions of those students involved, the program will adapt to provide a more detailed and comprehensive analysis of proposed power generation systems. The program is an application of logic, which can be used as a tool for further research by other students. It is an invitation to others to extend and particularize it. It is not complete because it presumes only to be a beginning.

4.0 Conclusion

Due to the composition of the HEDS-UP program, specifically being made up of dynamic, enthusiastic, and creative minds, the potential exists for production of design projects that will significantly aid in NASA's mission to send a human to Mars. The usefulness of the IDE to overcome barriers such as geographic dislocation amongst design partners and transmission of knowledge from one semester's project to the next has been demonstrated in UC Berkeley's "To Mars By 2012" class. The use of the IDE, easily scaled from its current single university implementation to a HEDS-UP wide system for communication, collaboration and design would propel HEDS towards that goal.

The IDE as it is currently designed is a very useful tool for enabling communication and collaboration among research groups and researchers. However, for the most part, it is only a design. The most immediate goal for the IDE project is to complete implementation of the components already designed. While this is in progress, the design of the next version of the IDE will occur in parallel.

The first major item for consideration in the next version is how it can be made better suited for use by larger groups of participants. Facilities must be designed to allow designation of administrative authority to local leaders (i.e. Berkeley-based users of the IDE should be under control of someone at Berkeley, while users based at NASA Ames should be under control of someone at Ames, and so on).

Considerations for distributing the IDE into multiple semi-autonomous servers scattered among the participating universities and research centers should also be made. This will allow for redundancy, ensuring that each center can access its own database should network problems temporarily disrupt communication with the other sites, and increase scalability by distributing processor load and consumption of storage space across multiple computers.

Finally, the current version of the IDE, once fully implemented, should be tested with actual users and observations should be made on how well the IDE improves their productivity. Comments from the users on how to improve the IDE should be solicited as well. Modifications should be made to the designers of the next version to take these data and comments into account. This applies especially to the design tools MarsHOT and CAPS which are available to any user or interested party to examine, critique, or modify. Consider this an invitation to the members of HEDS to begin to develop an organizational paradigm focused on interaction and collaboration with the aim of making HEDS a forum for university input directly into the design of a NASA led, manned mission to Mars.

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[i] Design Reference Mission Version 2.0
[ii] Machinery's Handbook

CONCEPTUAL DESIGN OF A MARTIAN POWER GENERATING SYSTEM UTILIZING SOLAR AND WIND ENERGY

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Abstract

An all-solar manned mission to Mars must overdesign the photovoltaic array in order to handle dust storm conditions. Wind energy extraction is proven terrestrial technology which can offset the dust storm (and night-time) reductions. A multi-phase project is underway to assess the feasibility and drive the development of wind energy extraction systems for Mars. This project has specifically addressed the design of a Darrieus-style Vertical Axis Wind Turbine (VAWT). The project assumed that wind energy extraction would be a secondary production system to the photovoltaic array. Energy production of 300 kW-hr per Martian day is required for this application. The wind turbine is designed by iteratively stepping through the following tasks:

1. Choose a blade shape;
2. Calculate the aerodynamic loads (primarily to estimate performance);
3. Design the guy cables;
4. Design the blades;
5. Design the tower; and
6. Choose support equipment.

The resulting system was estimated at 944 kg. Based on the feasibility assessment mentioned above, a wind speed of 28 m/s or higher must be seen for at least an hour each day. This wind speed is in the realm of possibility as the expected slope winds on Mars will likely be this high or higher. In order to meet this feasibility, the following design trends were seen: low pre-tension guy wires; ultralight blades; and thin lightweight towers. This work also found that if 25 to 35 m/s winds are available for at least one hour during a Martian day (during a dust storm), then wind energy extraction can be expected to be at least as mass-efficient as solar arrays (during a dust storm). Significant issues such as structural dynamics, thermal expansion/contraction, fatigue, blade struts, deployability, and maintainability were not considered at this time.

Introduction

The objective of the project was to produce the conceptual design of a power generating system for Mars, which combines solar and wind energy. The design of the wind power generating system was of primary concern. The wind and solar power generating systems are excellent methods for the utilization of the available natural resources on Mars. However, these energy sources are highly variable. The production of energy through solar arrays is dependent on the availability of sunlight. Similarly, the production of energy through wind turbines is dependent on favorable wind conditions. With an atmospheric density 1/75 of the Earth, Mars would at first appear to be an unlikely candidate for wind energy. However, the extraction potential of power from the wind is a function of velocity cubed and only proportional to density. Therefore, high winds can make-up for low density. Fortunately, some models suggest that Mars is subjected to regular high velocity winds in some locations. Additionally, these winds are expected to operate at night and during dust storms (times when solar energy is ineffective). A study of the wind energy option has been initiated and is currently underway with input from a variety of organizations including the University of Houston, NASA-JSC, Sandia National Laboratories, the Texas Space Grant Consortium, and ETM, Inc. (James, et. al., 1999). The overall project includes several phases:

(1) an initial assessment of the solar and wind resources,
(2) an assessment of the energy needs for various applications,
(3) a conceptual design of a traditional Horizontal Axis Wind Turbine (HAWT),
(4) a conceptual design of a traditional Vertical Axis Wind Turbine (VAWT),
(5) development of novel construction and wind turbine design concepts,
(6) a feasibility assessment of power generation using a solar/wind hybrid system, and
(7) the specific development of Martian as well as terrestrial systems.

The immediate objective of this work is to provide mission planners with sufficient information to consider the inclusion of wind energy in Mars mission planning and to target precursor mission objectives. The design work presented in this report primarily addresses phase four of the above list. However, aspects of the other topics listed above will be discussed as they relate to primary objectives of this work. The final design will be used to drive feasibility studies of the solar/wind production concept as well as to drive the development of novel construction/design concepts that will further enhance the feasibility of *in-situ* Martian power-production systems.

Supporting Information

Initial Assessment of Solar and Wind Resources

Martian Solar Energy Resources

Solar power will make an essential contribution to the success of Martian missions. Solar power is abundant, cheap, and does not involve safety concerns. More importantly, proven technology for its application in space missions already exists. The use of solar arrays made up of photovoltaic cells is the most ideal solar power generating system for Mars since photovoltaic cells do not depend on a single point light source. In spite of all these positives, Mars is farther from Sun, has an atmosphere, prone to seasonal dust storms and dust accumulation, and Mars' orbital eccentricity make solar power a highly variable source. These factors must be carefully considered in designing a solar power generating system for Mars.

The incident radiation on Mars (irradiation), S is given by

$$S = \mu S_0 \left(\frac{\bar{r}}{r} \right)^2 \qquad (1)$$

where μ is the cosine of the solar zenith angle, $S_0 = 590$ W/m^2 is the solar irradiance at Mars' mean distance $\bar{r}$ from Sun (Haberle, et. al. 1993). This solar irradiance is subject to high seasonal variations due to Mars' high orbital eccentricity relative to that of earth. The maximum available irradiance at perihelion (point of orbit nearest to Sun) is 717 W/m^2, whereas at aphelion (point of orbit farthest from Sun) the maximum available irradiance drops down to 493 W/m^2 (Haberle, et. al. 1993). The daily average insolation $\bar{S}$ is described by the following equation:

$$\bar{S} = \frac{S_0}{\pi}\left(\frac{\bar{r}}{r}\right)^2 \left(\cos\delta \cos\theta \sin H + H \sin\delta \sin\theta\right) \qquad (2)$$

where θ and δ are latitude and solar declination and H is the half day length. According to Haberle et al the least variation is observed at low latitudes of northern hemisphere and most variation is observed at high latitudes of southern hemisphere (Haberle, et. al., 1993). The amount of solar radiation available is also impacted by the Martian atmosphere as it allows radiation only with wavelengths greater than 200 nm to pass through. The dust particles suspended in the atmosphere scatter the incident radiation and cause a major degradation in the availability of solar power. Figure 1 illustrates the effects listed above. The Viking 2 lander data was used in this example (James, et. al., 1998). The upper left hand plot provides the estimated solar insolation calculated above the Viking 2 lander site. The effects of orbital eccentricity and solar zenith angle variations are seen. Although, this calculation did not depend on measured data, the drop-out regions correspond to times during which the Viking 2 lander was not providing data. The data represents approximately 1.3 Martian years. The upper right hand plot provides the optical depth data as measured by the Viking 2 lander. The higher optical depth values denote times of a dust laden atmosphere. The normalized net irradiance function estimates the solar energy reaching the surface. The instantaneous power, which would have been produced by an 1850 m^2 photovoltaic array with 20% efficiency, is provided in the lower right hand plot.

In addition to the atmospheric conditions impacting the availability of sunlight for use in photovoltaic cells, other factors also effect the performance of solar arrays. The solar cells give the best output at temperature ranges of 150 K to 200 K (Haberle, et. al., 1993). Large solar arrays must be able to withstand wind loads under high winds and wind-blown dust can cause abrasion on the surface of the cells. Dust accumulation resulting from dust storms is one of the major concerns in the use of solar arrays. The performance of the solar cells will undergo significant decline if several monolayers of dust are deposited on the surface. The estimated decline for a two year period is 77 percent (Landis, 1997). Therefore, adequate provisions for dust removal are necessary to harness maximum results from the use of solar arrays.

Martian Wind Energy Resources

Wind results from the motions of the air in the atmosphere which is caused by the variable heating of air from the sun. The wind speed at the surface is zero, it increases with height rapidly when close to the surface, and the rate of increase declines with greater height. The variation of the wind speed with height can be estimated using a power exponent function

$$V(z) = V_r\left(\frac{z}{z_r}\right)^\alpha \qquad (3)$$

where z is height above the surface, V_r is the wind speed at the reference height z_r above surface, V(z) is the wind speed at height z, and α is an exponent which depends on the roughness of the terrain. Therefore, the above equation can be used for estimation of the mean wind velocity at a certain height, if the mean wind velocity at a reference height is known (Walker and Jenkins, 1997). This relationship is demonstrated in Figure 2.

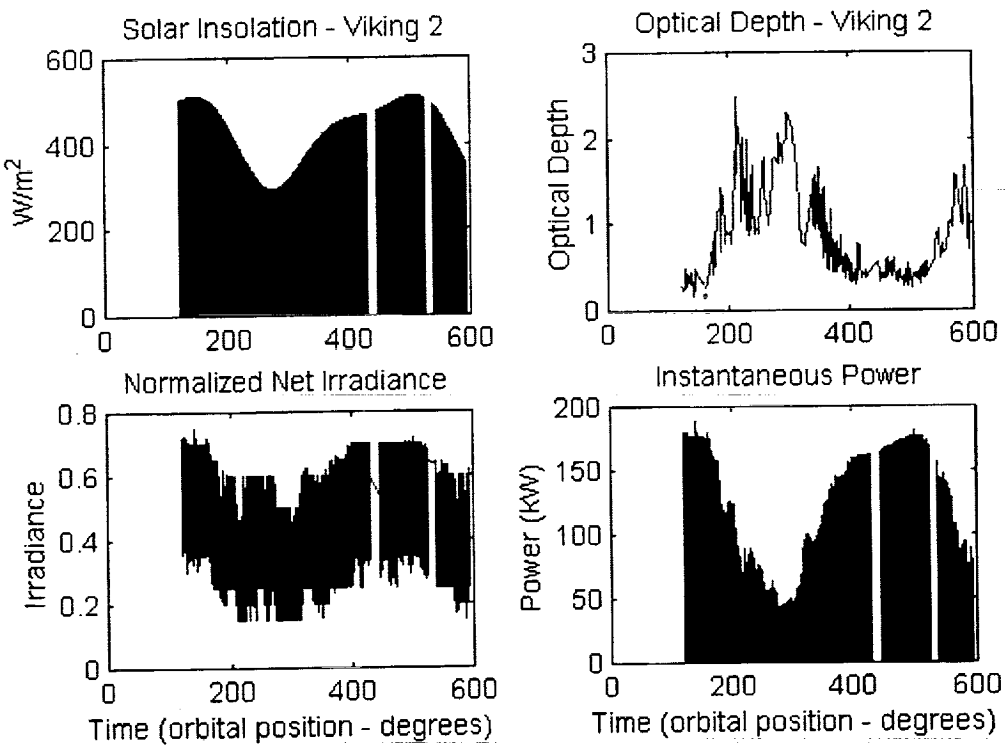

Figure 1. Variation in solar power at Viking 2 Lander Site (Source: James, et. al., 1998)

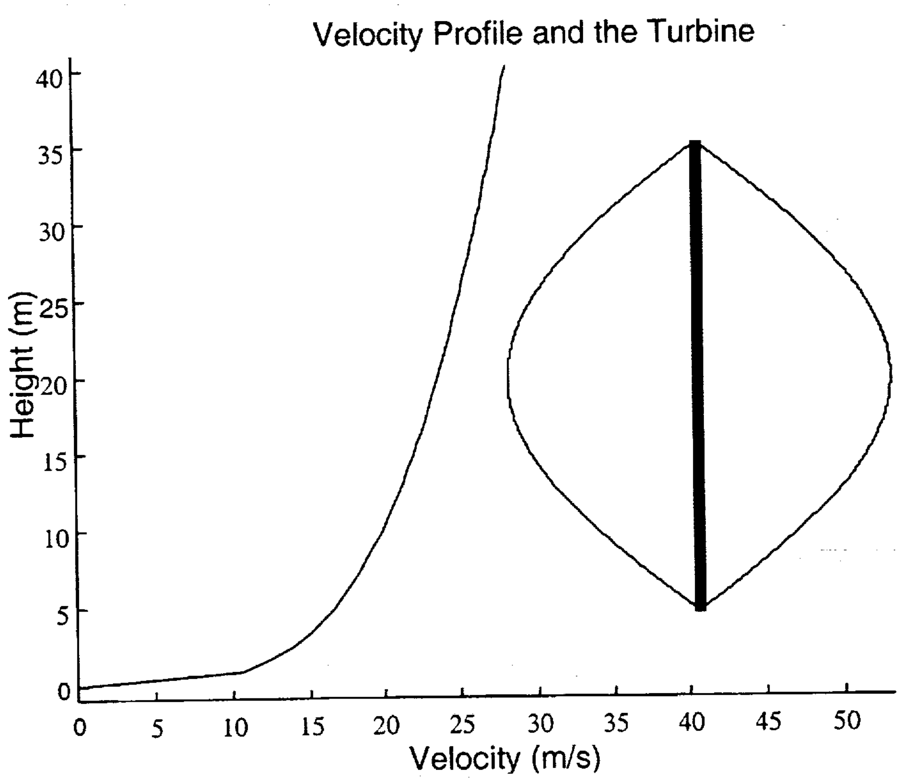

Figure 2. Wind velocity profile compared to local height of the wind turbine.

The wind velocity profile in Figure 2 is typical of the wind velocity profile expected during Martian slope wind conditions. The winds are expected to be nightly conditions which result on the slopes of the large Martian shield volcanoes as well as other regions of low angle yet long distance slopes (Magalhaes and Gierasch, 1982). These winds are predicted to have velocities of 25 to 33 m/s at 25 meters above the surface. These winds have been analytically reproduced using data generated from measurements made by the Viking 1 lander (James, et. al., 1998). There are also suggestions that large diameter craters may produce an increase in wind velocity above the ambient flow (Haslach, 1989). Note that Figure 2 also includes (for comparison) a schematic of a 25 meter tall VAWT positioned five meters off of the ground.

In order to design a wind turbine we must also focus on two fundamental concepts – energy, and power, or energy per unit time. The kinetic energy in a flow of air through a unit area perpendicular to the wind direction is ½ mV^2. The power associated with this flow is

$$P = \tfrac{1}{2} \rho A V^3 \quad (4)$$

Where, ρ is the density of air (Martian air density is 1/75 of that on earth), A is the area, V is the wind speed, and P is the power produced. The above equation can be used to calculate the power density for a given wind speed. It must be pointed out that the wind speed on Mars is equivalent to much lower wind speeds on earth. Figure 3 displays the relationship between the Martian and terrestrial wind velocities.

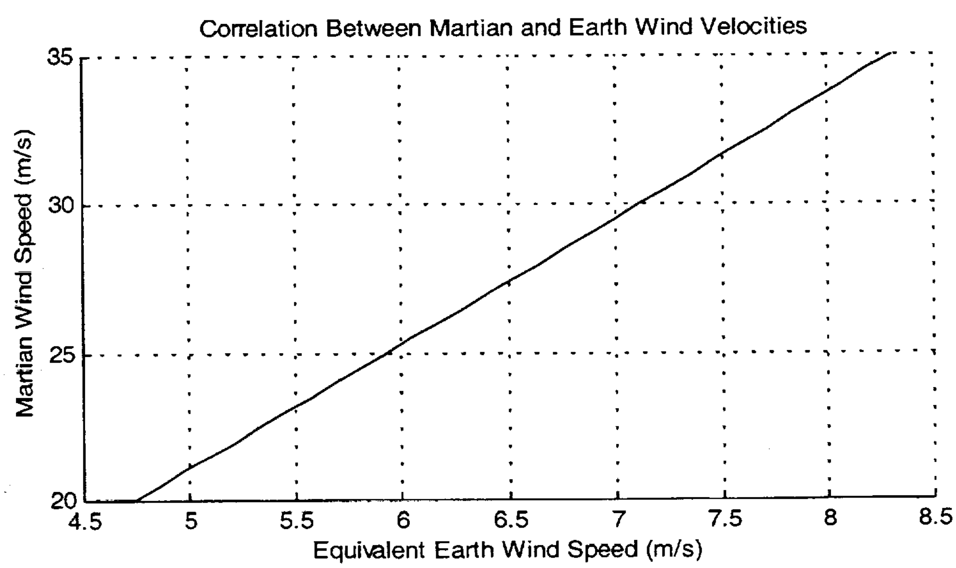

Figure 3. Relationship between the Earth and Mars wind velocities.

Only a portion of the total available energy can be converted to useful energy by a wind turbine. The power available to a wind turbine is equal to the change in kinetic energy of the air as it passes through the rotor. The fraction of energy extracted by the wind turbine from the total available energy is called the coefficient of performance C_p, given by

$$C_p = \tfrac{1}{2} (1 - b^2)(1 + b) \quad (5)$$

Where, b is the ratio of upstream and downstream wind speeds (refer to Figure 4).

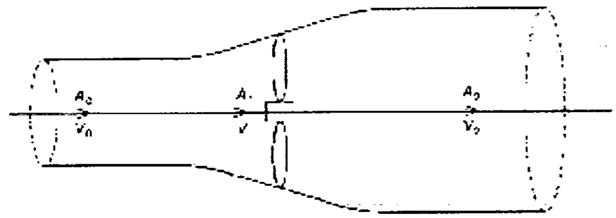

Figure 4. Typical air flow through a wind turbine (Source: Walker and Jenkins, 1997)

$$b = V_2/V_0 \tag{6}$$

Differentiation of C_p with respect to b shows that this coefficient is maximum when b=1/3, giving $C_p \approx 60$ %

The above value of coefficient of performance is known as the Betz limit. Hence, the real power output of a wind turbine is given by

$$P = C_p\,(\tfrac{1}{2}\,\rho A V^3) \tag{7}$$

Modern designs of wind turbines operate at C_p values of about 0.4. The above equation does not actually produce a true cubic relationship between the power output and wind speed because C_p varies with the wind speed. The above equation can also be used to determine the swept area of a wind turbine for a particular wind speed at a given coefficient of power.

Asessment of the Energy Needs

Wind Energy Applications

Some critical objectives of initial manned Mars missions will be to establish a human habitat, power life support systems, enable science and exploration activities, and produce propellant. The achievement of these objectives is dependent on the ability to generate sufficient power to meet the energy needs of the systems and processes involved. The type and design of a power generating system is interrelated with specific mission scenario considered. However, the following three energy needs are assumed for a Mars mission: baseline life support, science/exploration activities (such as rover operations or drilling), and ascent vehicle propellant production. The relative requirements and timing of these needs will determine the niche wind energy will fill. As such, the following niches for wind energy generation in the manned Mars mission planning and implementation are assumed:

1. as a tertiary power supply in a primarily nuclear mission to enhance the safety and reliability as well as limiting abort-to-orbit scenarios;
2. as a secondary power supply in an all-solar mission to lessen the effects of dust-storm power reductions;
3. as a primary power supply in an early Martian settlement with rudimentary *in-situ* construction capabilities;
4. as a mobile power supply option to enhance and/or enable long-distance rover operations; and
5. as a cooperative power supply to enable non-nuclear unmanned precursor mission of extended surface duration.

This project will consider only scenario number 2 listed above.

Energy Needs for an All-Solar Mission

The current estimates of energy needs for an all-solar mission call for an energy budget of 17 kW continuous energy during the day and 9 kW continuous energy during the night for clear conditions (George, 1999). This includes 1 kW continuous energy during the day for rover operations. During dust storm conditions, the daytime utilization needs drop to 16 kW continuous, as rover operations will be curtailed. Hence, the baseline energy needs for an initial outpost are assumed to be 16 + 9 = 25 kW continuous during daylight hours (assuming no energy storage losses). Hence, if a Martian day is assumed to be 12 hrs per day, the daily energy needs are 25 x 12 = 300 kW-hr. Therefore, for 600 Martian days, total baseline energy requirement is 600 x 300 / 1000 = 180 MW-hr.

However, due to losses during dust-storms (radiation reaching the array may drop to 15% of clear condition values), an all-solar mission must utilize a solar array eight times larger than needed for the baseline requirements during clear conditions. Given this requirement, the daily solar power produced during clear conditions is 8 x 300 = 2400 kW-hr. Assuming that dust storms could operate for up to 150 Martian days, the total energy production over 600 Martian days with such a production system would be: (450 x 2400 + 150 x 300) / 1000 = 1,125 MW-hr.

Additionally, daily rover operation requirements during a clear 12 hour day equals 12 kW-hr. Over the course of 450 clear Martian days, the total rover energy requirement is 5 MW-hr.

Also, (Baker and Zubrin, 1990) and (Zubrin, et. al. 1991) propose that 107 tons of methane/oxygen propellant (for ascent and Earth-return) can be produced on Mars from 5.7 tons of hydrogen brought from earth and carbon dioxide from Martian atmosphere. The energy needs for this activity are 370 MW-hr over the 600 day mission. There is expected to be sufficient excess energy production during clear conditions to meet these needs.

Hence total energy needs for the entire 600 day mission is 180 + 5 + 370 = 555 MW-hr. Therefore, the excess energy production is 1,125 – 555 = 570 MW-hr. The utility of wind energy production systems in an all-solar mission would be to allow the reduction of mass (and therefore cost) of the solar arrays needed to meet dust storm conditions.

Design of a Horizontal Wind Turbine

Wind turbine designs can be categorized into two different groups – turbines that depend on aerodynamic lift and turbines that utilize aerodynamic drag. For the same swept area the power produced by lift type turbines far exceeds the power generated by drag type turbines. Therefore, due to size constraints on the wind turbine design for Mars, the lift type turbines are preferable. Some of the other important features of the two types that need to be considered in a Martian wind turbine design are listed below:

Drag Type Turbines:

- mainly low speed devices driven by drag forces acting on the rotor
- move slower than the wind, and their motion reduces power extraction
- torque at the rotor shaft is relatively high
- examples include traditional windmills and pumping devices

Lift Type Turbines:

- mainly high speed devices driven by lift forces on the blades
- linear speed of blades is generally faster than the wind speed
- torque on the rotor shaft is relatively lower
- examples include modern electrical power producing turbines

(Walker and Jenkins, 1997).

The wind turbines can be further classified into horizontal axis and vertical axis machines. The horizontal axis or propeller type turbines are more abundant and this technology is highly developed. The previous work by (Ferrell, et. al., 1998) considered an 18 meter diameter Horizontal Axis Wind Turbine (HAWT) which would produce 2.5 kW in a wind speed of 13 m/s. Alternatively, a 30 meter diameter turbine in a 25 m/s wind would produce 28 kW. This design effort suggested that wind turbines with sizes approaching large utility scale terrestrial wind turbines would be required. However, the chord lengths would be three times the values for similar turbines on Earth. Likewise, the thickness to chord ratio could be expected to be 1.5 times that of terrestrial turbines. Also, the power output (and imposed torque values) would be 1/10 the values seen on terrestrial turbines of a similar size.

Design Approach and Results

Design of a Vertical Axis Wind Turbine (VAWT)

This project will design a Darrieus type Vertical Axis Wind Turbine (VAWT) for energy production on Mars. There are some specific advantages associated with the vertical-axis machines, especially the Darrieus type, making the design concept potentially more suitable for a Martian application:

- their symmetry about the vertical axis allows them to operate independent of the wind direction, so a yawning mechanism is not required
- heavy gearboxes and generators may be situated at ground level permitting easier maintenance, a low support platform, and easier deployment
- the device is shaped such that centrifugal loads are balanced by pre tension forces in the blades, thus avoiding bending moments; thus, the turbine shaft carries axial and torque loads only
- the blades do not suffer fatigue stresses from gravitational forces during rotation

(Walker and Jenkins, 1997).

Additionally, the Darrieus VAWT may be much more amenable to a deployable installation.

In spite of the many advantages associated with the Darrieus type vertical axis wind turbine, some of the accompanying drawbacks listed below cannot be overlooked because

- they are not self-starting
- the torque fluctuates during each revolution as the blades move into and out of the wind
- speed regulation at high wind speeds can be difficult.

The next section discusses the steps used in the turbine design.

Typical Darrieus VAWT Configuration

The rotor subsystem consists of two curved blades whose ends are attached to fixed upper and lower hubs. The hubs are attached to the rotor column or the tower. The blades are symmetrical in cross-section and they have a troposkien shape that results in a minimal internal bending stress. The rotor height is usually 15% to 30% larger than the rotor diameter. The power train subsystem consists of mechanical and electrical equipment to convert the rotor's mechanical power into electrical power. The essential power train components are a turbine shaft, a gearbox to enhance speed, a generator drive shaft, a rotor brake, and an electrical generator. All these components are located close to the ground thereby permitting convenient maintenance and deployment. The proximity of the components to the ground also leads to a low support platform. The support structure subsystem consists of a support stand, three structural cables, and the upper and lower rotor bearings. The support cables connect the top of the tower to the ground anchors at an angle of elevation ranging in between 30 to 40 degrees thereby providing stability to the tower by restraining the movement of its center of the mass (Spera, 94). The cable tension results in a downward thrust load on upper rotor bearing which is ultimately transferred to the system foundation through the rotor column, lower rotor bearing, and the support stand. The height of the support stand is such that it provides sufficient ground clearance for the rotating blades.

Wind Turbine Design Process

Overview

The wind turbine design process utilized in this work is based on a series of performance analysis steps covering the major aspects of the wind turbine design. These steps include defining the blade shape, calculating the aerodynamic loads (primarily used to estimate conversion efficiency), design of the guy cables, calculating the axial blade loads, tower design, and estimation of support system mass. As with most design exercises, these steps were performed iteratively. However, the process and the results will be discussed sequentially in this section.

Blade Shape

Darrieus-style VAWTs typically utilize blades shaped to minimize bending stesses (Eggars, 1991). The shape of the blade excluding the effects of gravity is:

$$R = \cos(y / \sqrt{R}_{CE}) \tag{8}$$

where R is the blade radius, y is the height along the tower, and R_{CE} is the radius of curvature at the equator.

Figure 5 shows a typical Darrieus-style VAWT with two blades shaped using the above equation.

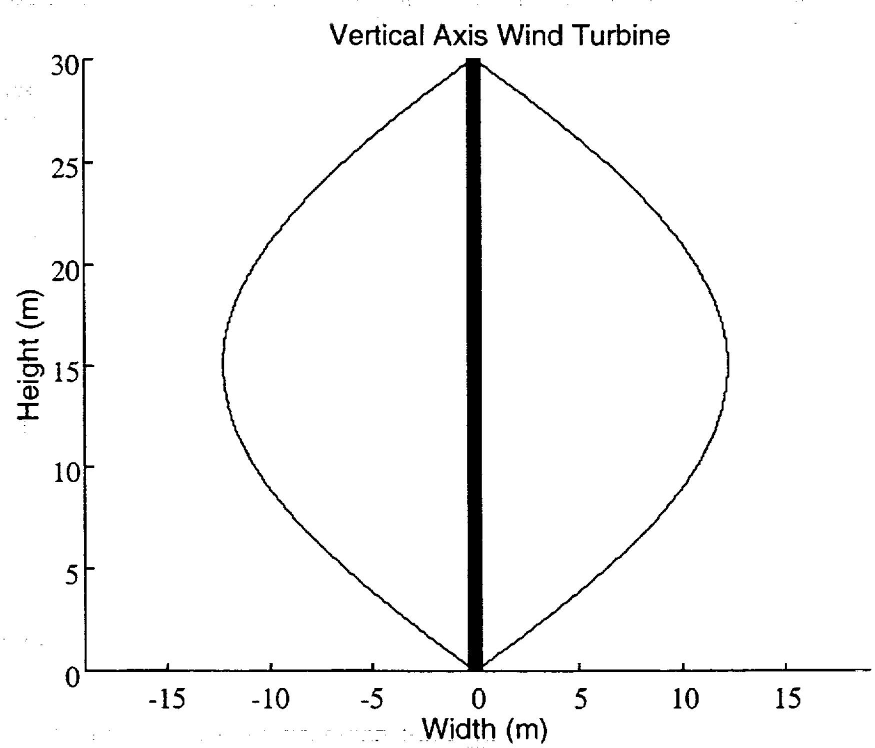

Figure 5. Shape of the Darrieus-type wind turbine.

Aerodynamic Loads

The aerodynamic loads and performance were calculated in this work using the double-multiple-streamtube model as presented by (Paraschivoiu, 1982). Multi-streamtube modeling considers the volume represented by the revolution of the rotor as a series of adjacent, aerodynamically independent stream tubes. The torque and normal forces generated by aerodynamic lift and drag forces are estimated and used to calculate overall turbine performance. It should be noted that the procedure was limited to small set of airfoils designed for terrestrial conditions. Also, the lift and drag coefficients that were calculated for these airfoils may not be completely accurate for Martian conditions. And finally, the double multiple streamtube model is most appropropriate for low solidity turbines (solidity is the ratio of blade chord to total swept area). This may have caused some inaccuracies in the results as the final design had a relatively high solidity.

The performance of the wind turbine was determined using SLICEIT computer code as developed and provided by Sandia National Laboratories (Berg and Rumsey, 1991 and Berg, 1992). The computer program is in FORTRAN consisting of a main program and several text files which store airfoil and input data fields. The program allows a model to be created based on existing data by evaluating number of blade sections and interpolating as a function of the Reynolds number and the angle of attack. The program also uses Martian parameters such as the viscosity, and atmospheric and gravitational data to create a model. Several different airfoil families were included and evaluated by trail runs with the program SLICEIT. These airfoils had been specifically designed for terrestrial vertical axis wind turbines. These include the SNLA 0021/50, SNLA 0018/50, and the SNLA 0118/50. These airfoils were the first to be examined using the SLICEIT.

The calculations were made at 5 mph increments from 20 to a maximum of 100 mph wind speed. The program takes into account the effect of the boundary layer on the velocity profile by a user input of the empirical wind shear exponent. This was ignored in preliminary test cases but was added in later cases. A single airfoil type was used for the entire blade length on initial trial runs. However, the most efficient earth-based wind turbines use two or three different airfoils types along the length of the blade. After unimpressive results, a multiple airfoil type model was quickly adopted. The SNLA 21/50 and the SNLA 0018/50, located along the equatorial region and near the tower respectively, produced the largest power outputs. A chord length to radius ratio as a required input was determined through experimentation and correlating earth based wind turbine parameters. This ratio was also allowed to vary along the blade. The power extraction is obviously higher with three blades as opposed to two, but the added power output is often not worth the weight gain associated with the extra blade.

The final design included the two airfoil shapes (SNLA 0018/50 near the tower and SNLA 21/50 at the equator) and three chord lengths (3.2 m, 2.8m, and 1.8 m) along the blade. The turbine was chosen to rotate at 75 rpm. The turbine height was 30.5 meters and had a diameter of 19 meters. The turbine had a maximum coefficient of power of .59 and produced 14.1 kW in a 25 m/s wind.

Guy Cable Design

The proper design of guy cables is complicated and highly dependent on tower design, structural dynamics, static loads, and temperature cycling. However, (Sullivan,1979) provides some guidelines for an initial guy cable design. The main purpose of this system is to offset aerodynamic loads. Since these loads are an order-of-magnitude less than the corresponding loads on a similar-sized turbine on Earth, the suggested cable tension values were also decreased by an order-of-magnitude. It should be noted that this analysis did not include the critical issues of structural dynamics or thermal changes.

The final guy cables were chosen as aluminum (although no detailed materials selection process was performed). For this size turbine, (Sullivan, 1979) suggested 15,000 lbs of pretension. However, it was pointed out that the suggested values were conservative and successful applications existed with ½ of the suggested tension. Hence, 7,500 lbs was chosen as the appropriate value of tension for a terrestrial turbine. Since the aerodynamic loads on Mars are expected to be 1/10 of the terrestrial values, 750 lbs was chosen as the pretension value. Three cables were chosen with a 30 degree angle. A .004 meter radius was chosen. The resulting mass of the cable system was assumed to be 24 kg. An additional 50 kg was added for anchoring systems such as augers.

Axial Load on Blade

The calculations of the axial loads on the blades are examined as presented by (Eggars,1991). The initial calculation of axial loads on the turbine blades can be made without considering the aerodynamic forces normal to the turbine blade as the inertial forces tend to dominate. Equation 9 was used to estimate axial loads:

$$F = mgy + \frac{1}{2}m\Omega^2\left[R_E^2 - R^2\right] + mR_E R_{CE}\Omega^2. \qquad (9)$$

where m is the mass per unit length of the blade, g is the acceleration due to gravity, Ω is the rotation rate, and R is the radius at height y, R_E is the radius at the equator, and R_{CE} is the Radius of curvature at the equator. Investigations of the effects of gravitational and inertial forces acting on the blades reveal that the effects of gravity on the blade are also modest in comparison to the inertial forces. To completely examine the effects of gravity and centrifugal loading, the mass distribution needs to be known. A force to mass per unit length ratio was assumed constant based on a requirement that the local stresses on the beam element are constant everywhere. This assumption allows a simplified model to be developed for the wind turbine to assess the effects of gravity and inertia. It should be noted that axial loads only were used to design the blade (material properties and wall thickness). Aerodynamic loads were not considered at this time. Structural dynamics issues were not addressed either.

It was quickly determined that an ultralight blade design was required in order to approach mass feasibility. Hence, the final design resulted in a thickness of .0005 m and a material density of 1100 kg/m^3. This material could be met with a variety of synthetic materials including nylon, rubber, neoprene, or polystyrene. It should be noted that such a design would invariably need to be inflated and supported with an internal frame or external strut. The next design iteration would necessarily include these structures. The mass of the blades was therefore 284 kg.

Tower Design

The parameters needed to define the tower include the optimal tower diameter , its wall thickness, the material, and the imposed axial load. The relationships, which take into, account the critical buckling loads, compressive stresses, and shear stresses, are found in (Sullivan, 1979 and Beer and Johnston, 1981). Neither, structural dynamics nor thermal changes were considered in the design. The final design was a .3 m diameter aluminum tube with a .002 inch wall thickness. Approaching mass feasibility was the primary driving factor in this design. The final design was 435 kg.

Total Mass

An additional 150 kg was added for support equipment such as the generator, gearbox, bearings, and brakes. Therefore, the total mass of the system was found to be 944 kg upon summing the estimated masses from the above sections.

Feasibility Verification

The feasibility of the initial design was evaluated using the energy to mass ratio of solar cells during dust storm conditions. The resulting solar array size for the scenario listed in the previous section is expected to be six to seven thousand square meters (George, 1999). The mass of such a system is estimated to be 11 to 13 metric tons. One half of this mass results from the solar cells and the support structure. Assuming a 12 metric ton system, this would suggest six metric tons of solar array. The remaining six metric tons would be dedicated to fuel cells and power conditioning equipment. Therefore, energy output for these arrays per unit mass during dust storms = 300/6 = 50 kW-hr per metric ton. For comparison, on a clear day the energy output per unit mass = 2400/6 = 400 kW-hr per metric ton. Hence the following relationship will be used to define feasibility of the initial design:

$$M_W \text{ (metric ton)} \leq (1/50)\, E_W \text{ (kW-hr)} = .02\, E_W \qquad (9)$$

where M_W is the mass of the wind turbine, and E_W is the energy produced in one day. The energy produced by the wind turbine would be estimated as the integrated value of power produced over a Martian day. A typical approximation is to define this energy based on the maximum wind speed the turbine sees for at least one hour in a given Martian day. Figure 6 shows the variation in the minimum sustained wind speed of 28 m/s needed to assure feasibility of the turbine design described above. It should be noted that this value falls in the range of the expected slope winds on the planet.

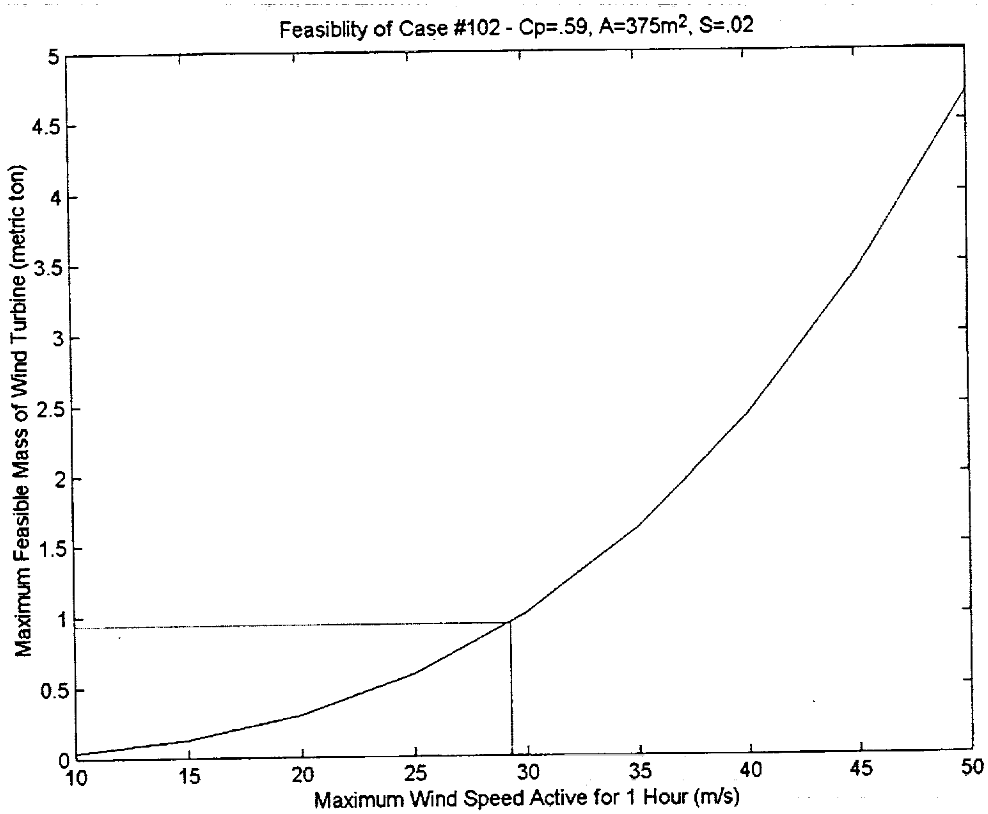

Figure 6. The mass feasibility curve for wind turbine performance characteristics determined from Test Case 102.

Feasibility of the system will include other aspects such as reliability, maintainability, packaged volume, and deployability as other design constraints. Although, these issues were not studied in detail in this work, some tantalizing directions to explore can be gleaned from the effort. Figure 7 illustrates a packaging deployability concept that can be considered given the design presented above.

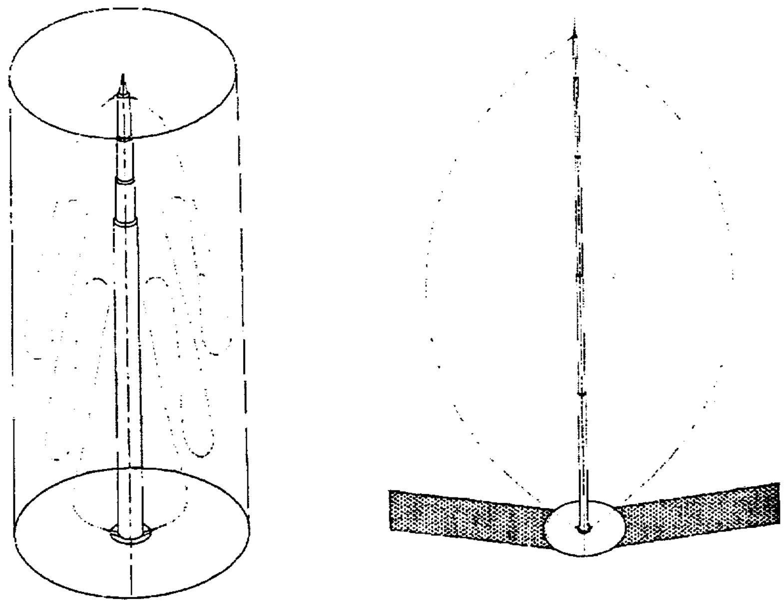

Figure 7: A conceptual design for a vertical axis wind turbine with inflatable blades and a deployable tower.

Conclusions and Recommendations

This project utilized modified tools drawn from the state-of-the-art in vertical-axis wind turbine design to study the feasibility of a wind power generation on Mars. The following findings were significant:

1. Wind energy generation is at least as mass-efficient as solar arrays in dust storm conditions;
2. The above statement is dependant upon the presence of regular high-winds which are sustained for at least an hour each night (especially during dust storm conditions);
3. Feasibility of wind energy generation on Mars will be enhanced by reducing the mass (or utilizing in-situ materials) in the structural systems;
4. This work suggests ultralight blade construction techniques, small guy cable tensions, and thin towers as viable design trends; and
5. Terrestrial design tools are useful in designing for Mars but care must be taken as the initial assumptions in the code development may be invalid.

The following recommendations may be made:

1. Ultralight designs should be developed for Mars including dedicated airfoil shapes;
2. Mission planning studies should explore the solar/wind energy option;
3. Options to perform wind energy mapping and system verification should be considered;
4. Ultralight terrestrial designs should be developed to gain operational experience and to fund development of the Martian systems; and
5. Modified terrestrial design codes and experienced wind energy technologists should be used as resources.

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1.0 Abstract

Current Mars reference missions specify mission lengths that exceed present day limits on long-duration space flight. An increase in the length of on-site times over previous manned missions allows for greater scientific and exploratory flexibility. With this flexibility comes new opportunities for increasing our understanding of the solar system in which we live. To take advantage of these opportunities requires the development of new long-range transportation systems. Such systems must be able to operate in adverse environments with minimal investment in infrastructure while providing a high scientific return on investment.

An aerial research vehicle provides a great deal of mobility in selecting specific sites to explore. Recent developments in the space community have recognized the need to explore beyond the limited scope offered by a stationary base camp or even a land rover. An airborne research vehicle provides the mobility to explore areas of interest that lie at great distances from the base site. Long distance missions requires the crew to have a great deal of discretion in selecting destinations based on information not available before the mission. Airships have the flexibility to travel to sites without prior knowledge of the site's conditions. The selection of an airship as the primary mode of transportation maximizes the crew's ability to act on "local decisions" which in turn increase the mission's scientific return.

2.0 Reference Mission

As a starting point, the class was given the NASA Mars Reference Mission. This document is available from the following URL: http://www-sn.jsc.nasa.gov/marsref/contents.html.

3.0 Design Constraints

The Mars Reference Mission provided a base set of design criteria from which our other criteria were created. Project requirements for distance and total trip time also played a large part in the determination of the final vehicle configuration. MARV must be capable of safely transporting at least three people and the requisite life support and scientific equipment to virtually any point on Mars within a pre-determined amount of time. The base camp, located in Western Daedalia Planum, will serve as an operational base for all mission sorties and at times as a communications relay for contact with Earth.

4.0 Why a Lighter-Than-Air Vehicle?

Several possible designs were originally considered. Three designs were explored in depth, an airplane, a ballistic vehicle, and a lighter-than-air vehicle. From these three a final configuration was selected and developed further.

4.1 Airplane

Several variations of the Mars plane configuration were studied, all of which entailed very large wings spans and many structural difficulties. A length of 65m and a wingspan of 155m posed the greatest challenges to the airplane concept. Stowing the vehicle inside the descent lander's 19m long payload shroud requires the wings and the fuselage to fold at multiple points, thus making assembly on Mars difficult and uncertain. Finally through the requirements, the plane has to be capable of landing at unimproved landing sites. With such a large wing span, the possibility of landing at any unimproved sites is very uncertain and difficult at best. The combination of these problems led to the search for a better solution.

4.2 Ballistic

Sub-orbital flight was originally considered as a viable option because of the high velocities and short trip times (~1 hour) inherent in ballistic vehicles. Driving the design were the requirements for antipodal range and in-situ resource utilization (ISRU). Carbon Dioxide and Liquid Oxygen were chosen as the vehicle's propellant combination since both CO_2 and LOX can be extracted from the Martian Atmosphere. Round trip antipodal flight on Mars requires at least 11.4 km/s ΔV. Using CO_2/LOX, with an Isp of 300 sec, the vehicle can only produce a maximum of 6.9 km/s ΔV. Extracting the necessary 622 mt of propellant at a rate of 21 mt/sol requires 829 mt of equipment and 25 MW of power. Manufacturing sufficient quantities of propellant would have necessitated more time, power, and support equipment than were available within the scope of the Mars Reference Mission and the Design Constraints.

4.3 Lighter-Than-Air

A lighter-than-air vehicle with a volume of 2.3 million cubic meters would be capable of lifting the required structure and payload. Landing on unimproved terrain can be accomplished, and final approach can be done visually due to the slow speed. The vehicle has the advantage of high definition mapping and exploring any sights of interest while en-route. The proposed non-rigid design can be stored compactly for transportation to Mars. Solar generated electric power could provide for both propulsion needs during flight and equipment power on-site. The power produced and thus the speed varies throughout the day. The system was planned around 12.5 meters per second average velocity.

4.4 Conclusion

Of the three configurations evaluated the lighter-than-air vehicle was deemed most viable. Both the airplane and ballistic vehicles suffered logistics problems resulting from structural complexity or resource availability. A lighter-than-air vehicle, while very large, is able to circumvent these obstacles through its inherent simplicity of design, construction, and assembly.

5.0 Missions

Sortie mission sites, as well as the base site, have been determined as part of our mission requirements. MARV and the crew are required to land, perform scientific experiments at site, gather scientific samples from site and return to base camp. Western Daedalia Planum, base site, is located at 19°S, 144°W. The site is located at an altitude of 3 km. The terrain is a relatively smooth, flat plain. The required sites are listed in the table below in the order of execution.

Load Source	Affected Component	Load Quantity
Gondola and Propulsion	Envelope	35000 N
Tail Fins	Envelope	4500 N
Hydrogen Pressure	Envelope	
Thrust	Envelope and Propeller Pylon	2100 N
Aerodynamic Loading - Fins	Envelope and Fins	4800 N
Aerodynamic Bending Moment	Envelope	157,000 N-m
Cabin Pressure	Pressure vessel	50,000 Pa
Payload	Internal Structure of Pressure Vessel	24000 N
Hoop Stress	Pressure vessel	33 MPa
Longitudinal Stress	Pressure vessel	466000 N

Table 1

All sortie sites are out of line of site with base camp and located at different altitudes, in different terrain and experience varying weather conditions. The vehicle must be able to land at each sortie site within a 1km radius, except for the base site. Since the base site will be improved the vehicle must be able to land within 50 meters of the landing area. In order to visit each site in relatively mild weather with longer hours of daylight, the northern sorties are visited first. In accordance with the "Human Exploration of Mars : The Reference Mission of the NASA Mars Exploration Study Team", expected landing on Mars is July 2014, using the fast-transit mission profile. In July of 2014 Mars will be experienceing a northern summer, southern winter. A travel time table is presented below.

Sortie	Mission Description	Site Name	First Leg (sols)	Mission Execution Time (sols)	Returning Leg (sol)	Total Length of Sortie (sols)
1	Medium Range Mission	Olympus Rupes	2.2	30.0	5.7	37.9
2	Polar Mission	North Polar Region	8.3	30.0	17.4	55.7
3	Triangle Mission	Maja Valles (leg 1)	7.6	13.0		51.4
		Connecting (leg 2)	9.3			
		Candor (leg 3)	0.0	13.0	8.4	
4	Short Range Mission	Upper Mangala	1.1	30.0	1.1	32.2
5	Antipodal Mission	Sinus Sabaecus	12.8	28.0	11.5	52.4

Table 2

There was an important wind direction variation found in Martian Atmospheric Profiles. Easterly winds are prominent in the southern hemisphere and westerly winds are prominent in the northern hemisphere. Viking lander missions measured for a wind variation of 3- 7 m/s. Seventy-five percent of the maximum winds sustained during the Viking lander mission was used to generalize the amount of head and tail winds that MARV would experience. Travel time varies due to the expected winds. Travel distance varies from a direct route due to high-elevation terrain features.

6.0 Science

6.1 Scientific Objectives

The goal of the each sortie is to collect information about Mars that will allow the completion of the following scientific objectives:

- Gain an understanding of the current state of the Martian environment to determine the possibility of human habitation
- Study Martian geology, meteorology, and seismology to determine what the Martian environment may have been like in the past as well as what it may be like in the future
- Search for evidence of extinct or extant life that may aid the understanding of how life began on Earth

6.2 Olympus Rupes

This site is on the side of Olympus Mons. The surface is composed of smooth lava flows of basaltic material from three different ages. Geologic and atmospheric equipment will be needed to perform experiments and collect samples from the surface, below the surface and the atmosphere. It does not appear that this site is of biological significance, so no exobiological equipment will be needed on this sortie.

6.3 North Polar Region

This site is on layered polar deposits of Amazonian-age materials. It is believed that the layering is due to variations in the ice-dust mixture. This suggests that the climate in this area has been this way for a long time. An important goal is to measure the water and $C0_2$ content of the ice. Seismology, geology and atmospheric sciences will be studied here. Since there is ice present at this site, exobiological experiments will be performed on certain samples.

6.4 Maja Valles and Candor

This sortie is knows as "The Triangle Mission" because two sites will be visited and the path from base camp to Maja Valles to Candor and back to base camp makes a triangle. Maja Valles lies in an outflow channel. From this site information about Martian morphology, outflow dynamics, stratigraphy can be collected. Important samples that can be collected are sediments from channels and fan-deltas and lake deposit. Ancient crustal material, crater ejecta, and exposed strata will also be studied at this site. As with most of the other sites, atmospheric samples will be collected and studied. There is also remote chance that ground ice may exist at this site. If so, it will be studied and exobiological experiments will be performed.

The second site on the sortie is Candor. It is believed that Candor was a locus of ground water discharge on Mars. Because of this, there could be microbial minerals precipitated by iron bacteria located at the heads of channels, sapping sites, and in lake sediments. One of the main objectives at this site is to collect samples of iron ore from red and black ground patches to examine the morphology of iron and magnesium oxides that may have been precipitated by ancient iron bacteria.

6.5 Upper Mangala Valles

Most of the scientific work done here will include taking samples of Hespian-age and Nochian-age materials. Also, nearby crater ejecta will be collected and studied. Meteorological experiments will also be performed to study the atmosphere. No equipment for studying exobiology will be taken to this site, as this area is not believed to have any significance as far as life is concerned. However, exobiological experiments can still be performed at base-camp on samples that have been collected.

6.6 Sinus Sabaeus Northeast

This site is located on a smooth plain known as the Plateau Sequence. Of major interest on this site is a mound known as "White Rock" which is in a crater on the plain. It is believed that this rock made up of playa deposits composed of chemically precipitated evaporite minerals. These types of formations are important to study because they have the potential to for preservation of organisms and biomolecules. Also of interest are channels that flow into the crater, which resemble terrestrial dendritic drainage systems.

6.7 Scientific Equipment

In order to satisfy the scientific objectives, EVAs will be performed on a regular basis to collect samples and perform experiments. Most of the samples collected will be in the form of rocks, dust, or samples obtained from the subsurface by drilling. The vehicle's laboratory will be able to perform tests to determine the age and composition of the samples, as well as detect water or any volatiles that may be present in the sample. Meteorological experiments will be performed to determine properties of the air, such as aerosol content, wind speed, pressure, and temperature. Exobiological equipment will be taken on some of the sorties that appear to have biological significance. This equipment will allow for the detection of organic materials and the determination of whether they are of Martian origin or the result of contamination from terrestrial origins. To perform these experiments, the following list of equipment will be needed.

Name	Description	Mass (kg)	Power (kw)	Volume (m³)	Use
Mars Geophysics Package	Determines local magnetic and gravitational fields and detects water and volatiles	25	.01	.02	Field
Marsnet	Seismological stations that measure long-term seismic activity	25	.01	.02	Field
Geological Field Package	Hand tools for use on EVA, sample containers, and documentaion tools	335	.2	.55	Field
Differential Scanning Calorimeter	Identifies minerals and volatiles	20	.04	.03	Field
10-meter Dril Rig	Used for obtaining samples below the surface	260	5.5	10	Field
Thermal/Evolved Gas Analyzer	Analyzes gasses released from the soil	2	.014	.0014	Lab
Multispectral Imager	Close range imaging	35	.024	.16	Field
Binocular Microscope	Preliminary sample examination and evaluation	5	.02	.01	Lab
Petrographic Microscope	More intensive petrographic analysis of samples	20	.04	.04	Lab
X-ray Fluorescence Spectrometer	Mineralogical analysis	3	.01	.02	Lab
X-ray Diffractometer	Elemental anaylsis	5	.015	.015	Lab
Mossbauer Spectrometer	Analyzes iron oxides and dust particles containing iron.	3	.01	.01	Lab
Mass Spectrometer	Determines absolute ages of rocks	50	.1	1	Lab

Table 3 Geological Equipment

Name	Description	Mass (kg)	Power (kw)	Volume (m³)	Use
Surface Atmospheric Package	Measures temperature, pressure, wind velocity, and aerosol content	5	.05	.02	Field
Aerosol Volatile Sniffer	Collects aerosol particles in order to analyze volatiles	15	.05	.1	Field
Ionospheric Sounder	Measures the ion compostion of the upper atmosphere	50	.14	.3	Field
Meteorological Balloons	Determines wind speed, cloud height, pressure, temperature and humidity	50	.05	.1	Field
Aerosol Laser Ranger	Measures the height and content of clouds	40	.3	.1	Lab

Table 4 Meteorological Equipment

Name	Description	Mass (kg)	Power (kw)	Volume (m³)	Use
Incubator	Used for incubating petri dishes for exobiological experiments	3	.03	.01	Lab
Neutron Spectrometer	Anaylsis and detection of organics	6	.006	.00015	Lab
Specific Electrode Analyzer	Analysis of solutes that may be of biological significance	1	.002	.008	Lab
Soil Oxidant Survey	Equipment used to analyze the oxidants in the Martian soil	1	.005	.003	Lab
IR Laser Spectrometer	Study trace gasses in the atmosphere and soil which may contain biological activity	5	.01	.03	Lab
Optical Microscope	High resolution optical microscope	3	.02	.002	Lab
Biological Apparatus	Petri dishes, glass spreaders, and other biological equipment	30	0	.08	Lab

Table 5 Exobiological Equipment

7.0 Vehicle Description

The non-rigid airship will have an overall length of 344 meters and a height of 114 meters. The Kevlar/Mylar envelope will have a total gas volume of 2.3 million cubic meters and a surface area of 122,000 square meters. All lift will be generated by filling the envelope with hydrogen gas. Varying the volume of

Martian atmosphere in each of the ballonets will provide trim control for both altitude and pitch. The ballonets will also be used to maintain an envelope gage pressure between 45 Pa and 200 Pa.

Primary yaw, pitch, and roll control will be provided by the rear mounted fins. The fins will be composed of non-rigid cylinders filled with hydrogen gas for compact storage and increased buoyancy. The neutrally buoyant fins will eliminate the extra weight of conventional rigid fins. As with the envelope, all inflatable structures will be computer monitored for leaks and pressure loss.

Section	Mass (kg)	Power (kW)
Communications, Navigation, and Electronics	150	7.5
Flight Propulsion System	1350	50
Science Equipment	1500	7
Life Support and Personnel	2200	2.5
Power Generation System	11200	60
Vehicle Structure	120000	<1

Table 6

8.0 Structures

8.1 Structural Requirements

All structural components must have non-negative margins of safety, and be able to accommodate touchdown velocities of 1 m/s lateral and 1 m/s vertical. In addition, all safety-critical mechanisms shall have redundant sensing and actuation.

The following factors of safety must be incorporated

- Secondary structure: 1.5
- Primary structure: 2.0
- Pressurized tanks: 3.0
- Pressure lines: 4.0

8.2 Load Sources and Quantities

The major loads on the vehicle are generated from the lifting gas. The internal pressure of the gas creates the greatest amount of stress, which is in the hoop direction. The longitudinal stress of 17 MPa is about half the amount of the stress in the hoop direction. The lifting gas imparts a bending moment of 976,000 N-m. This moment is a result of the super-pressure created, which causes the ends of the envelope to bend downward thereby, creating stress on the top of the envelope. All other loads are listed below.

Load Source	Affected Component	Load Quantity
Gondola and Propulsion	Envelope	35000 N
Tail Fins	Envelope	4500 N
Hydrogen Pressure	Envelope	200 Pa
Thrust	Envelope and Propeller Pylon	2100 N
Aerodynamic Loading - Fins	Envelope and Fins	4800 N
Aerodynamic Bending Moment	Envelope	157,000 N-m
Cabin Pressure	Pressure vessel	50,000 Pa
Payload	Internal Structure of Pressure Vessel	24000 N
Hoop Stress	Envelope	33 MPa
Longitudinal Stress	Envelope	466000 N

Table 7

8.3 Envelope Design

The 0.07mm thick Kevlar outer hull will provide the strength necessary to withstand the maximum hoop stress of 33.6 MPa. Kevlar was chosen because it is twice as strong as Nylon-66 and 50% stronger than E-glass. Kevlar also has a high tear resistance. The inner hull will consist of 0.012mm thick Mylar, which will be used to contain the lifting gas. Mylar was chosen because of its low permeability to hydrogen. Ballonets constructed of Mylar will occupy the lower half of the envelope and provide trim control.

8.4 Pressure Vessel Design

8.4.1 Assumptions

The analysis for the minimum wall thickness was based on the hoop stress of a cylinder with hemispherical endcaps. A variety of materials, internal pressures, and radii were initially studied. The internal volume and pressure were later determined by Life Support and Human Factors to be 100 m^3 and 50 kPa, respectively. A factor of safety of 3 was used in accordance with the structural requirements.

8.4.2 Results

The dimensions will be as shown in Figure 1, with an internal cylinder radius of 1.8m. The material will be Kevlar-90, and the wall thickness will be 3mm except for the rear endcaps which will be 5mm. All loads will be transmitted to the internal structure, not to the pressure vessel walls. The resulting minimum margin of safety is 3.25

8.5 Pressure Vessel Internal Structure Design

The internal structure will consist of a main support beam with hanging rings. All internal loads will be transmitted by simply-supported beams that connect to the rings. All doors and windows will be framed so that all loads can transmit to the rings. The Kevlar-90 pressure vessel will be attached to the outside of the rings.

8.6 Airlock Design

The airlock will be attached to the back of the pressure vessel. The rear cap and cylindrical section of the airlock are the same dimensions as the pressure vessel itself. The airlock will accommodate at least two astronauts with substantial scientific loads. The door between the pressure vessel and the airlock will open towards the pressure vessel so that pressure assisted sealing is achieved when the airlock is depressurized. The door to the outside must be pulled inward before it will open outward so that it achieves pressure assisted sealing when the airlock is pressurized. The door to the outside will act as a staircase to provide access to the surface, but the design is still TBD.

8.7 Cockpit Windshield Design

The window will have a field of view of 60 x 90 degrees (vertical by horizontal), and will be constructed of Polycarbonate plastic. The stress was calculated by assuming a 1.8 m sphere of polycarbonate plastic subjected to the internal pressure of the cabin. The resulting thickness was 6 mm, which yields a margin of safety of 7. The reason for the high margin of safety was the concern over the approximations made during analysis.

8.8 Figures

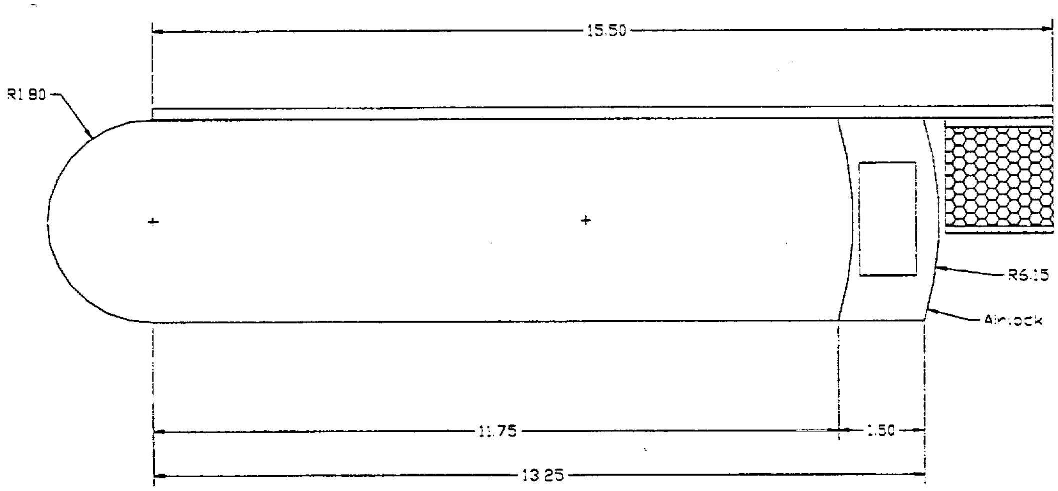

Figure 1 Pressure Vessel External Dimensions

All dimensions in meters

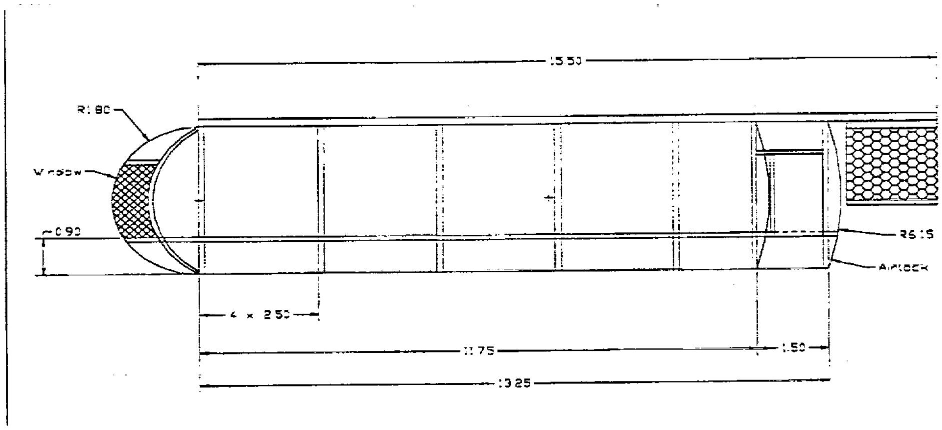

Figure 2 Pressure Vessel Structural Diagram

All Dimensions in meters

9.0 Power, Propulsion, and Thermal

9.1 Power

The power subsystem can be broken down into several parts, which include a primary power source, power distribution mechanisms, power regulation and control, and energy storage. The two main sections are those of primary power and energy storage, for which a solar array system and battery configuration, respectively, were chosen.

Power system trade studies were based primarily on weight and volume considerations. Photovoltaic arrays proved to be the most economical and efficient method of power generation. Energy storage needs will be provided by nickel-hydrogen batteries.

The primary power generation and power storage systems were sized according to the power needs of the vehicle. These included 60 kW required inflight during daylight hours, 18 kW required for onsite daylight operations, 8 kW required over 10.6 hours for night operation, which amounts to storage capabilities of 84.9 kW-hours, and a 10% contingency plan, which amounted to an additional 6.8 kW.

The primary power generation system, consisting of photovoltaic arrays, will be mounted on the vehicle's outer structure in order to take advantage of the envelope area. Thus, it will be a non-tracking system. Blocking diodes can be used to prevent battery discharging in cases where sections are shadowed at a given time. The design was limited by the following parameters:

	Ultra Flex Arrays
Efficiency, %	22
Degradation, %/yr	3.75
Density, kg/m^2	1.75
Power Output Density, W/m^2	66.8
Lifetime, yrs	5
Peol, W/m^2	55.2
Power Output, kW	66.8
Area, m^2	1210

Table 8

This assumption was based on the premise that further development over the next 15 years will lead to higher efficiency solar cells. Furthermore, this is also taking into consideration the factor that the operating temperature on Mars is much lower than that of an Earth-based satellite. The Ultra Flex Array Design was chosen as the anticipated array design because it provides the type of flexibility that will be needed to mount the solar array on the blimp's envelope. The solar flux is approximately 22% of that received in Earth orbit because of atmospheric losses and Mars' greater distance from the Sun. Furthermore, dust storm conditions characteristic of the Martian environment drop this percentage to roughly 6.5%, or approximately 86.9 W/m^2. Power at the beginning of life, Pbol, is a function of cell efficiency and solar flux. Power required by the solar array, Psa, is a function of the power necessary to conduct day and night operations. Power required at the end of life, Peol, is a function of lifetime degradation and Pbol. Lifetime of the solar array system was estimated at one year for the primary purpose of attempting to reduce the overall mass of the system.

The power storage system will consist of nickel-hydrogen (NiH_2) batteries. The required 84.9 kW-hours of energy will be stored in batteries consisting of 17 cells rated at an 81 A-h capacity. Battery lifetime was assumed to be five years. A short battery life was chosen with the expectation that they will be replaced as they wear out. Furthermore, because depth of discharge (DoD) is a function of cycle life, as lifetimes increased, DoD decreases significantly. Thus, a shorter lifetime is more efficient.

NiH_2 Battery	Operating Conditions
Daylight Duration, hrs	14
Eclipse Duration, hrs	10.6
Bus Current, A	100
Charging Power, kW	6.8
Depth of Discharge, V	0.7
Discharge Voltage, V	1.25
Charge Voltage, V	1.4
Rating, A-h	81

Table 9

10.0 Crew Systems

10.1 Cabin Conditions

Pressure of the crew cabin will be maintained at 50 kPa for the duration of the mission in order to have zero pre-breathe time for EVAs. The percentage of oxygen maintained in the cabin is 45 %, this value allows the cabin to operate at equivalent sea level conditions. The cabin conditions are given in Table 10.

Atmosphere Parameter	Cabin Value
Total Pressure	50 kPa
ppOxygen	22 kPa
ppNitrogen	27.5 kPa
ppCarbon Dioxide	0.4 kPa
Temperature	18.3 - 26.7 C
Relative Humidity	25 - 70 %

Table 10

This pressure is governed by the requirement for daily EVAs on the mission. A cabin pressure of 50 kPa allows for daily EVAs without decompression or pre-breathing . The suit pressure is kept at 30 kPa to allow for mobility and dexterity.

10.2 Air Revitilization

Air in the crew cabin must be monitored and maintained to ensure crew survival. For this design, particles such as dust and micro-organisms are removed from the air using High Efficiency Particulate Arrestance (HEPA) filters. Carbon dioxide is then removed using Solid Amine Water Desorption. The carbon dioxide collected from the cabin air is reduced using both the Sabatier process and carbon dioxide electrolysis. Carbon dioxide electrolysis will be used to generate oxygen for the mission. Additional carbon dioixde can be obtained from the Martian atmosphere in order to produce more oxygen.

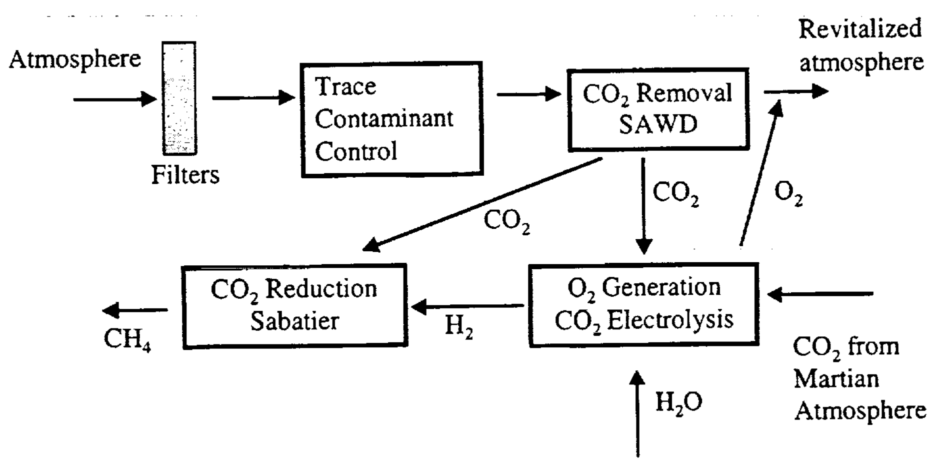

Figure 3

10.3 Air Maintenance

Air quality must be maintained closely in a small environment such as a crew cabin. Contaminants, temperature, humidity and components of the air must be carefully monitored. Particulate levels will be monitored using a "non-dispersive infrared" (NDIR) technique with a Gas Chromatograph/Mass Spectrometer (GC/MS). As mentioned before, High Efficiency Particulate Arrestance (HEPA) filters will be used to remove dust and micro-organisms. A Temperature and Humidity Controller (THC), which is a condensing heat exchanger, is used to maintain comfortable temperature levels in the cabin. Partial pressures of oxygen, carbon dioxide, carbon monoxide, nitrogen, and water vapor will be monitored and maintained on an "as needed" basis. Ionization fire detection devices will be placed in air ducts with the ability to detect smoke particles 0.3 microns or larger in less than 5 seconds. In larger spaces flame detectors will be used to monitor flicker rate in the UV and infrared bands.

10.4 Airlock and EVA Operation

The airlock on the vehicle is able to accommodate two fully suited crew members and EVA equipment. The airlock is capable of supporting both crew members with oxygen, power and water during EVA suit donning and checkout. The airlock can also function as a hyperbaric chamber in order to treat decompression sickness. The EVA suit pressure is maintained at 29.6 kPa with 100 % oxygen. The EVA suits should be self carrying, with a Personal Life Support System (PLSS) to monitor health, life support and power system status. EVAs will require no pre-breathe time due to the lowered cabin pressure. Each EVA has the capacity to last for 6 hours with 15 minutes for donning, 15 minutes for checkout and 30 minutes of reserves.

10.5 Water

The human body requires approximately 2.5 kg/man-day of consumed water in order to survive. The water design loads provided for this mission are 3.0 kg/man-sol potable water and 6.0 kg/man-sol hygiene water. These will be provided to the crew through a closed water loop consisting of a Multifiltration process for potable water recovery and Vapor Compression Distillation System (VCDS) for hygiene and urine water recovery. Potable water will be recovered from the humidity control system condensate and water transfer from the hygiene water reservoir. Hygiene water will be supplied from recovered hygiene water and urine water.

10.6 Food

The crew will be provided with approximately 0.62 kg/man-sol (dry weight) of food. Food packaging weighs approximately 0.45 kg/man-sol. Dry beverage powder mixes, freeze dried, irradiated, rehydratable, and thermostabalized foods will be provided to the crew during their mission. Three meals will be allowed per sol with repeat of meals after 6 sols.

10.7 Radiation Protection

The established radiation exposure limits for low Earth orbit are defined in the table below. These limits are used since there have been no well defined limits for a Martian mission. Also required is an exposure limit of 3 REM body dose exposure limit in a maximum length mission including a Class IV solar flare in the worst case mission location. From this table, it can be seen that the blood forming organs (BFO) have the most stringent exposure limits.

Exposure Duration	BFO (cSv)	Eye (cSv)	Skin (cSv)
Daily	0.2	0.3	600
30 days	25	100	150
90 days	35	52	105
Annual	50	200	300
Career	100 - 400	400	600

Table 11

The protection provided by the Martian atmosphere is summarized in the table below. Atmospheric protection on the surface of Mars varies directly with the surface pressure. Therefore, the amount of protection varies with altitude and also as the atmosphere density changes seasonally.

Altitude (km)	Protection, g/cm ^{2} CO_3	
	Low-density Model	High-density Model
0	16	23
4	11	16
8	7	11
12	3	8

Table 12 Simonsen, L.C., Nealy, J.E., 1993

The equivalent doses to the blood forming organs (BFO) as a function of altitude for both galactic cosmic rays and solar flare events are summarized in the graphs below. From these graphs, it is concluded that sufficient radiation protection is provided by the Martian atmosphere alone.

11.0 Avionics

11.1 Navigation System

The primary vehicle navigation system will be inertial based navigation, with position and velocity updates provided by a terrain contour navigation (TCN) system and a sun tracking system. An air data system is also present to provide dissimilar redundancy.

The TCN system provides positional resets by comparing the terrain profile, as measured by laser altimeters, against the terrain profile as stored in a database in the vicinity of the estimated position provided by the inertial system. The TCN system can provide positional accuracy of up to 50 m with present technology. These position estimates may also be differentiated to provide velocity estimates.

The sun tracking system provides attitude information based on the measured location of the sun in the sky and its known position based on location and time estimates.

A breakdown of the navigational sensor system is given below.

Component	Accuracy	Mass (kg)	Power (W)	Volume (cc)	MTBF (hrs)	Levels
Inertial Sensor	-	1.4	30	1600	50,000	2
- Gyros	0.01 °/hr	-	-	-	-	-
- Accelerometers	50μg	-	-	-	-	-
Laser Altimeter	15 cm	2.0	50	6000	25,000	3
Sun Sensor	.008°	2.5	15	1200	40,000	2
Air Data System	-	5.5	20	8000	12,000	2
- Pitot Probe	0.1 m/s	-	-	-	-	-
- Barometric Alt	6 m	-	-	-	-	-
- Temp. Probe	1 K	-	-	-	-	-
Totals	-	24.8	280	39600	-	-

Table 13 Navigational Sensors

In order to meet the requirement of landing within 1 km of an unimproved landing site with these sensors, a positional reset rate of 0.752/hour is required, and the overall system reliability is 99%.

Below is a schematic representation of the navigation system. It shows how the strap down inertial navigation system works with the position and attitude updates. These updates are optimally combined with the inertial estimates in order to produce an overall position estimate.

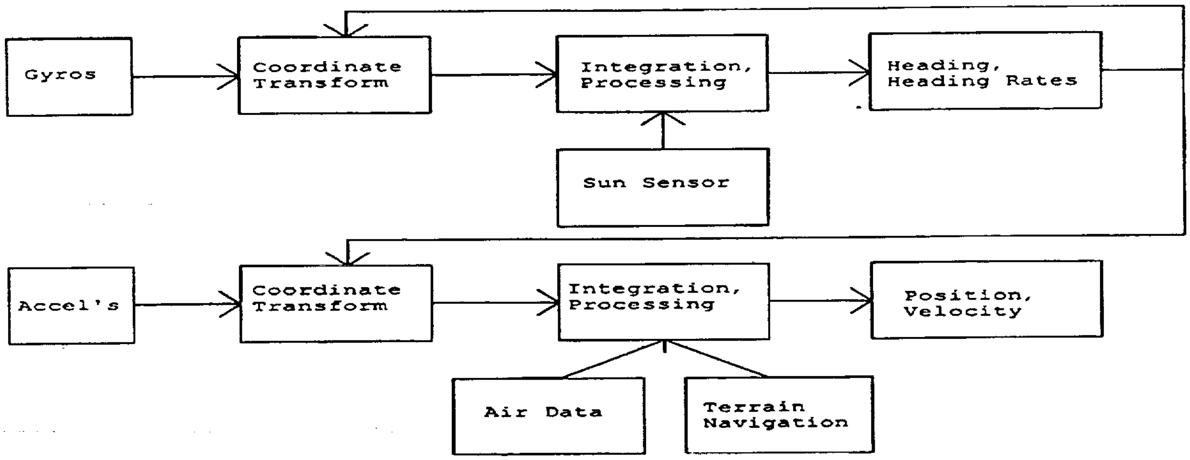

Figure 4 Navigation System Diagram

11.2 Flight Control System

The vehicle is controlled in flight by a Fly-by-Light system. This type of control system is similar in structure to a more conventional Fly-by-Wire system, but replaces electrical wires by fiber optic cables, thereby saving weight and increasing reliability.

Present fiber optic technology has proven a typical MTBF of 30,000 hours. In order to meet the reliability requirement of 99%, 5 levels of redundancy in flight control is needed. This gives an overall power requirement of 50 W and a mass of 85 kg.

In flight attitude control is achieved primarily by use of aerodynamic control surfaces, whose movements are commanded by the flight control system. Secondary sources of attitude control are provided by differential power output from the engines and differential inflation of the fore and aft ballonet, for yaw and pitch control, respectively. Active roll control is not required, because there is no need to rotate the craft about its roll axis, but in the event of perturbing forces the aerodynamic control surfaces combined with the overall system stability may be used to damp out the resulting motions.

Reaction control while on station performing the scientific mission, or while anchored for night is primarily passive in nature. Here the vehicle is simply allowed to turn into the wind, as would a weathervane. In a more violent dust storm situation, it may be required to use some combination of aerodynamic controls and the secondary controls as previously discussed, for which there is some reserve power.

A diagram of the flight control system is given below.

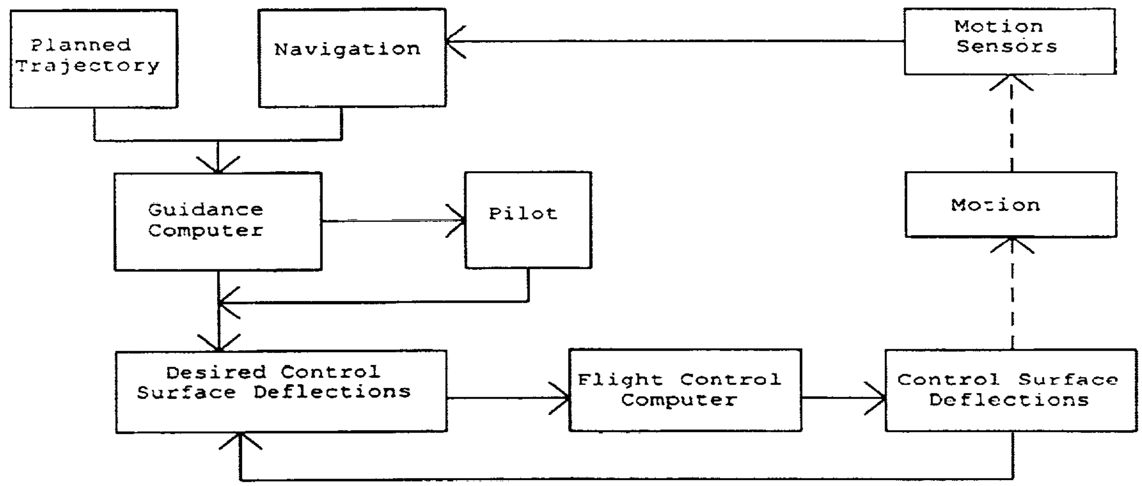

Figure 5 Flight Control System Diagram

In the event of any deviation from the planned trajectory, as determined by the navigation system, the guidance computer determines the necessary course corrections to bring the craft back onto its nominal path. The flight control computer then implements these corrections by issuing commands to the control units. The resulting motions are then detected by the navigation system thereby repeating the process until the craft is back on its planned track. As required, it is also possible for the craft to be capable of active human control to modify its trajectory.

11.3 Communications

The communications system is required to provide two-way communications incorporating high-rate telemetry, voice, and high-definition video simultaneously to three EVA suits, the EVA Rover Vehicle, Earth Deep Space Network (DSN) Stations, and Base Camp whenever possible. In addition to this, the vehicle must also be capable of receiving, storing, and forwarding data to a Low Mars Orbiting (LMO) satellite for non-real-time communications to base camp.

To satisfy these requirements, there are three antenna systems required. The first is an off-the-shelf dipole system that will provide the ability to communicate with the three EVA suits as well as with the rover vehicle to relay the high-definition video, voice, and telemetry required. The second antenna system is a five-meter parabolic dish that is used for communications back to Earth, when in view. This system makes use of the DSN's k-band capabilities while also including an encoding scheme of Reed-Solomon and 1/4 convolution to improve the data integrity during transmission. The third system is a two-meter parabolic dish that is used to communicate to base camp when in line of sight or to the LMO satellite for non-real-time communications.

	Earth	LMO Satellite/Base
Receive/Transmit Frequency (GHz)	25.8 / 23.7	4.6 /5.1
Receive Data Rate (Mbps)	127.5	108.1
Transmit Data Rate (Mbps)	138.5	100.0
Bit Error Rate (bps)	10^{-3}	10^{-9}
Power Required (W)	5000	50

Table 14

In addition to these nominal operating systems for communications, there is an Omni antenna for emergency transmissions from the vehicle to the LMO satellite or to base camp when in a line of sight.

The total power required for the communications system is 7kW when all the available links are being utilized simultaneously. During emergency transmissions, only 200 W is required. The total mass of the entire system is 80 kg.

12.0 Cost Analysis

MARV's cost breakdown was based on the Spacecraft/Vehicle Level Cost Model (SVLCM) developed by NASA's Johnson Space Flight Center. The SVLCM is a simplified cost model that provides cost estimates for the development and production of spacecraft, launch vehicle stages, engines and scientific instruments. SVLCM is a top-level implementation of the NASA/Air Force Cost Model (NAFCOMM). (http://www.jsc.nasa.gov/bu2/guidelines.html)

The input for the SLVCM requires the user to know what type of spacecraft (manned, launch vehicle, etc.), dry weight of the spacecraft, quantity that your going to produce, and the learning curve percentage. As a rough approximation, MARV's dry weight is 30mt, the quantity is two, and we assumed a learning curve percentage of eighty-five percent.

MARV's largest cost was that of the Development phase accounting for 82% of the total project with an estimated cost of 5.2 billion ($FY99). The Production phase second, accounting for fourteen percent at an estimated cost of 890 million ($FY99). Including Mission Operations, which was calculated by the Mission Operations Cost Model (MOCM), was four percent of MARV's cost budget, 276 million ($FY99). The total program cost approximates to 6.1 billion ($FY99).
(Figure 9)

Spacecraft/Vehide Level Cost Model	
Vehicle Dry Weight (kg)	30000
Quantity	2
Learning Curve (%)	85
Validity range (kg)	231 - 69638
Number of Data Points	8

Table 15

SVLCM Results	
Development ($BFY99	5.2
Production($BFY99)	0.89
Total($BFY99)	6.0

Table 16

Mission Operations Cost Model	
Investment ($BFY99)	6.0
Mission Type:	Manned
Average Annual MODA ($MFY99)	276
Total MODA ($BFY99)	2.2

Table 17

Every aerospace program includes in their cost analysis an allocated yearly cost through the duration of their program. Annual cost allocation is based on two parameters: cost fraction, and the peakedness factor. The cost fraction represents the total cost spent when fifty percent of the project time is complete. For MARV, it was estimated to be 0.5. The second parameter is the peakedness factor, which helps determine the maximum annual cost. For MARV, the peakedness factor was 1.0 which assumes that the development activity rises rapidly, peaks, and then falls rapidly. The project timeline for MARV is eight years, starting in 1999 and ending in 2007. (Figure 10) The projected cost throughout the program takes on a chevron form with small cost required to begin the project, large costs in the middle and small costs towards the end of the program. (Figure 10)

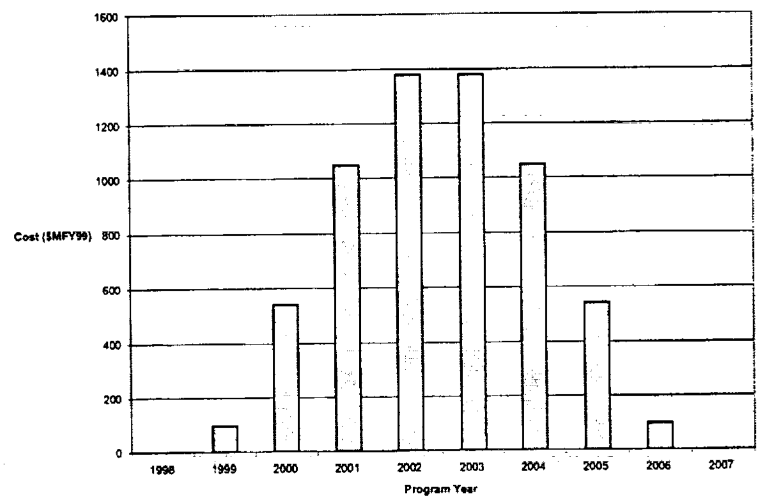

Figure 6 Cost Allocation

13.0 Conclusions/Recommendations

Analysis of the Lighter-Than-Air concept indicates that it is both practical and feasible It should , however be noted that further, more detailed, studies are required before this design can be brought to implementation. Areas needing additional consideration include storage and deployment, maneuverability, solar array technology, envelope maintenance, ground handling, and Martian weather forecasting systems.

An Astronaut Assistant Rover for Martian Surface Exploration

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Abstract

Lunar exploration, recent field tests, and even on-orbit operations suggest the need for a robotic assistant for an astronaut during extravehicular activity (EVA) tasks. The focus of this paper is the design of a 300-kg, 2 cubic meter, semi-autonomous robotic rover to assist astronauts during Mars surface exploration. General uses of this rover include remote teleoperated control, local EVA astronaut control, and autonomous control. Rover size, speed, sample capacity, scientific payload and dexterous fidelity were based on known Martian environmental parameters, established National Aeronautics and Space Administration (NASA) standards, the NASA Mars Exploration Reference Mission, and lessons learned from lunar and on-orbit sorties. An assumed protocol of a geological, two astronaut EVA performed during daylight hours with a maximum duration of four hours dictated the following design requirements: (1) autonomously follow the EVA team over astronaut traversable Martian terrain for four hours; (2) retrieve, catalog, and carry 12 kg of samples; (3) carry tools and minimal in-field scientific equipment; (4) provide contingency life support; (5) compile and store a detailed map of surrounding terrain and estimate current position with respect to base camp; (6) provide supplemental communications systems; and (7) carry and support the use of a 7 degree-of-freedom dexterous manipulator.

Introduction

As the National Aeronautics and Space Administration (NASA) turns toward interplanetary exploration[1], it is obvious that further innovations in and improvements to astronaut tools, transports, procedures, and life support systems must be pursued. Lunar exploration, recent field tests, and even on-orbit operations suggest the need for a robotic assistant to the astronaut during extravehicular activity (EVA) tasks. A robotic rover assistant would greatly reduce astronaut fatigue and increase productivity by performing time consuming, fatiguing, and repetitive dexterous tasks, which are made more difficult when working against the pressurized suit necessary for life support in any extraterrestrial environment. The assistant would also relieve the astronaut of carrying tools, scientific instruments, and samples. In addition, the rover could carry backup life support gear and supplemental communications systems, increasing the safety of EVAs. The focus of this paper is the summary of a semester long graduate project of the design of a robotic rover to assist EVA astronauts during Mars surface exploration. The overall design objectives of the EVA assistant are covered as well as a description of the subsystems. The ways in which the rover assists the EVA crewmember are also discussed. A detailed description on the design can be found in "Design of an Astronaut Assistant Rover for Martian Surface Exploration."[2]

Design approach

The design of this rover utilized an iterative approach, with three groups initially considering three preliminary missions. Analysis of the resulting designs led to formation of mission assumptions, design scenarios, and design requirements for the final vehicle. This approach was selected to rapidly narrow down mission requirements, to consider widely varying missions, and to provide a reasoned basis for the selection of requirements of the final rover.

The mission requirements for the preliminary designs were:

- Support two astronauts on four hour EVAs
- A four hundred day useful life-time
- Rover capable of astronaut-traversable terrain -0.3 m obstacles with a maximum astronaut speed of 4 kilometers per hour (kph)
- Carry EVA tools and contingency life support

The three designs differed in the geological packages they supported and their ability to carry astronauts.

The smallest vehicle, the single arm assistant shown in Figure 1, carried a single dexterous arm for obtaining geological samples, carried 75 kilogram (kg) of samples out of a total mass of 760 kg, required 960 watts (W) average power, and did not have the ability to carry the astronauts.

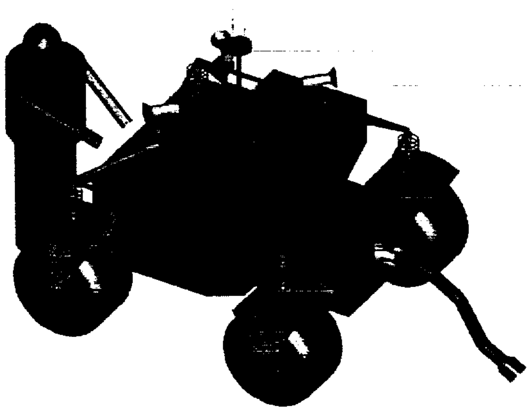

Figure 1: Single arm assistant

The second vehicle, or dual arm assistant shown in Figure 2, carried a pair of dexterous arms for geological sampling, carried 370 kg of samples out of a total 700 kg mass, used 1 kW average power, provided a degree of modularity by trading battery packs for sample storage containers, and could carry a single EVA subject in a contingency.

Figure 2: Dual arm assistant

The final vehicle, the large manipulator assistant shown Figure 3, supported a large positioning manipulator that positioned a pair of dexterous arms, had over 1,700 kg in mass, used over 4 kW of power on average, and was designed to carry the astronauts to and from the exploration site.

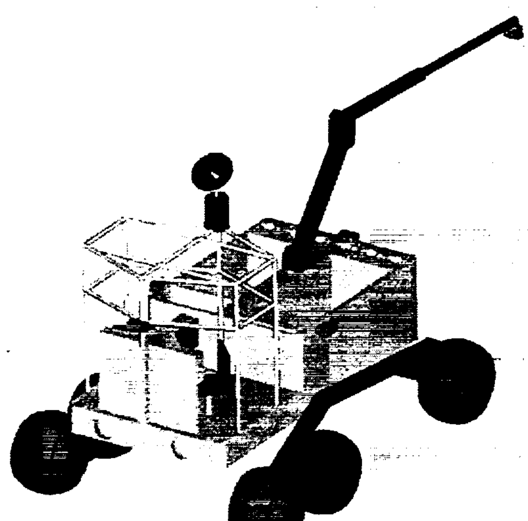

Figure 3: Large manipulator assistant

Based on the resulting designs, assumptions were made for the mission, mission objectives were prioritized, design scenarios were selected, and the design requirements were finalized.

Assumptions

In order to establish specific design requirements, reasonable assumptions must be made regarding EVA protocol, working environment, desired functionality, size and weight of the vehicle. In this design, there are three general uses of the rover: remote teleoperated control, local astronaut control, and autonomous control. It is assumed that EVAs occur only during daylight hours, are conducted by two astronauts, have a maximum duration of four hours, and do not exceed a 6-km radius from the base camp. It is assumed that satellite images of the Martian terrain are available to the crew for EVA site selection and area mapping with resolution worthy to be used by the rover during autonomous EVAs. It is further assumed that the rover is operated in the Martian temperate zones and is not required to traverse terrain that can not be traversed by a suited astronaut. For purposes of this design, the largest obstacle an astronaut on Mars can cross is 0.45 meters and the maximum terrain slope that can be traversed is 20 degrees. Further, the maximum speed an astronaut can maintain over level unbroken ground is assumed to be 4.8 kph, which drove an assumed rover maximum speed of 8 kph. It is also assumed that one component of a human Mars mission is to deploy small, permanent, scientific packages, or Deployable Instrument Packages (DIPs), on the surface of Mars. It is assumed that these self-contained packages may contain instruments such as barometers, anemometers, thermometers, or seismometers with a mass of 20 kg and dimensions of 40 centimeters (cm) by 40 cm by 20 cm. The rover also carries a 7 degree-of-freedom (DOF) dexterous manipulator for geological sampling, which is provided as a pre-integrated payload but depends upon the vehicle to provide power and intelligence. Finally, the size and mass of the rover is such that it meets functional requirements for scientific and carrying capabilities and yet is limited to accommodate small storage space, both on the transit spacecraft and at the base camp. A target vehicle mass and size of 300 kg and 2 cubic meters are set with the consideration that, in the event of a tumble, the astronaut is capable of righting the vehicle.

Design Philosophy

To assist in facilitating the iteration process, a design philosophy including prioritized objectives was established. Due to the harshness of the Martian environment, limited parts and resources available, and the rigorous performance demands on the rover, durability of parts and systems is considered the first priority. Second, in order to minimize launch costs, propulsion, and power requirements, and to allow astronaut manipulation in case of emergency, the vehicle is required to be lightweight. Third, the rover and its capabilities are tightly integrated into the daily tasks of astronauts working on Mars. Because the rover is used in an environment far from Earth nearly everyday for an extended period of time, it must be easily repaired and maintained. The design of all subsystems is such that the EVA Assistant is able to perform back-to-back sorties with minimal refurbishment and can receive any necessary servicing easily during "stand down" periods. This servicing is done in a pressurized volume at base camp and can be done in parallel to the servicing of the space suits between sorties. All subsystem trade studies and component decisions were made based on mission demands, functional requirements, and this priority outline.

Design Scenarios

To adequately determine design and functional requirements, it is necessary to consider specifically how the vehicle is used. The following are samples of detailed design scenarios used to illustrate the range of rover applications as well as depict expected worst-case situations for various aspects of the rover. Table 1 summarizes the requirements and rover configurations for each scenario

Scenario 1

The rover is either teleoperated from base camp or preprogrammed using satellite maps to travel to a potential EVA site. Because there is no astronaut accompanying the vehicle, it may travel at maximum speed to the site. Taking a similar route to and from the site, it is estimated that the rover will travel 1.5 hours one way (three hours round trip). Of that three hours, one hour of travel is spent traversing +/- 3 degree slope terrain (30 minutes at +3 degrees, 30 minutes at -3 degrees), and a minimum of 20 minutes are spent on +/- 20 degree sloped terrain (10 minutes at +20 degrees, 10 minutes at -20 degrees) at a maximum speed of 4.8 km/hr. The remainder of the travel time is spent traversing level ground. Once the rover reaches the site, functions such as mapping, full video survey, and sample retrieval are employed as needed. The dexterous arm is expected to be in use for 60 minutes and may retrieve pre-designated or directed samples through teleoperation, use an already deployed DIP to perform scientific experiments, or a variety of other tasks. Human factors equipment is not necessary for this mission and is therefore removed to save mass and subsequently lower power requirements. Throughout the scenario, however, navigation, mapping, and video/telemetry communications are used.

Scenario 2

Following Martian landing, the rover may be required to perform contingency operations within the immediate area of the base camp while the astronauts are adjusting to Martian gravity. It is assumed that the site for base camp is pre-selected for large open spaces, so the terrain can be considered mild and unthreatening. Rover locomotion requirements amount to an estimated 30 minutes at maximum speed on zero grade terrain. The rover is fully configured, excluding human factors equipment (tools, contingency life support and other interfaces). The rover must carry minimal tools and necessary end effectors for use by the dexterous arm. Full video is required to adequately maneuver to the site and manipulate the dexterous arm. The estimated time required for arm operations is 3 hours at average power.

Scenario 3

Many tasks, such as deployment of DIPs or beacons to remote sites, can be accomplished through teleoperation or autonomously by the rover, eliminating the need for a human EVA excursion. The terrain stipulations are much like scenario 1 in that the parameters of maximum speed, slope encounters during travel, and time of traverse remain the same. However, for this scenario, the rover has all unnecessary systems and interchangeable modules removed and carries only 20 kg of DIPs out to the site. At the site, the payload is deployed, resulting in an estimated 30 minutes of arm operations. It is assumed that the DIP starts operations independent of the rover or with minimum interaction. When the DIP is delivered to the site, the rover then returns to base camp free of payload. Communications, navigation, and mapping capabilities are required throughout the scenario.

Scenario 4

One of the greatest benefits of the rover is its ability to act as a geological/technical assistant during human EVAs. The rover travels at 4.8 kph for two hours including one hour on a +/- 3-degree slope and 20 minutes on a +/- 20-degree slope. The residual two hours of the sortie are spent at 1.6 kph for one hour at on level grade and one hour at rest. The rover is configured for full human factors support, prepared with a full science payload, powered for 30 minutes of arm operations and equipped to carry 12 kg of collected samples, of which up to 10 kg can be collected by the arm and up to 2 kg personally by the astronaut. The guidance, navigation, and control (GNC) system keeps current position estimated at all times to enable "fetch" and "lead home" commands, as well as terrain mapping. This scenario is by far the most demanding on all of the subsystems and can be considered a worst-case estimate of the performance requirements for many subsystems. Tasks such as core sampling, payload deployment, and extensive, widespread sampling may be designated through colored markers (see Markers) left by the astronaut during the sortie. After the astronaut returns to base, the rover may be sent to complete the designated tasks by teleoperation or autonomously.

Scenario 5

The final scenario is one in which a DIP is not fully automated or independent of the rover, and requires human assistance during deployment. The same terrain and traverse profile of scenario 4 applies to this mission. During the one hour at rest, the dexterous arm will perform approximately 30 minutes of operation. The rover carries contingency life support and necessary tooling. No scientific payload (excluding the DIP) or sample storage and support are required. The DIP contributes 20 kg of mass on the journey to the site.

Design Requirements

The following design requirements were developed based on the assumptions and scenarios outlined above.

Terrain

- Maximum speed of 8 kph over flat Martian terrain.
- Forward and lateral operation capabilities on a 20-degree maximum slope.
- Obstacle clearance of 45 cm, comparable to allowed Martian suit mobility.
- Accompany astronaut on 4-hour sortie with battery capacity for a total of 8 hours, or quick change out of 4-hour battery block, each day, six days of the week for four hundred days.

Payload

- Retrieve, label, catalogue, and carry 12 kg of collected samples from the EVA site back to base camp.
- Carry and support one 7 DOF dexterous manipulator to assist astronaut with difficult and fatiguing tasks.

Table 1: Scenario summary

Scenario	Control	Terrain/ Locomotion Requirements	Arm Operations	Rover Configuration
1	Teleoperation/ Autonomous	3 hours at maximum power • 20 min: +/- 20 degree grade (4.8 kph) • 1 hr: +/- 3 degree grade (4.8 kph) • 1 hr 40 min: 0 degree grade (8 kph)	1 hr	2 kg samples No human factors
2	Teleoperation/ Autonomous	30 min: 0 degrees (8 kph)	3 hrs	Min tools Min science No human factors
3	Teleoperation/ Autonomous	3 hours at maximum power • 20 min: +/- 20 degree grade (4.8 kph) • 1 hr: +/- 3 degree grade (4.8 kph) • 1 hr 40 min: 0 degree grade (8 kph)	30 min	Min science DIP (20 kg) No human factors
4	Supervisory	2 hours • 20 min: +/- 20 degree grade (4.8 kph) • 1 hr: +/- 3 degree grade (4.8 kph) • 40 min, 0 degree grade (4.8 kph) 1 hr: 0 degree grade (6 kph) 1 hr: rest	30 min	12 kg samples Full science Full human factors
5	Supervisory	2 hours • 20 min: +/- 20 degree grade (4.8 kph) • 1 hr: +/- 3 degree grade (4.8 kph) • 40 min, 0 degree grade (4.8 kph) 1 hr: 0 degree grade (6 kph) 1 hr: rest	30 min	Min science DIP (20 kg) Full human factors

- Carry astronaut hand tools necessary to meet mission objectives.
- Provide two hours of contingency life support for two astronauts (it is assumed that the astronauts will never exceed a two hour traverse by foot- approximately 6 km -from base camp).
- Carry instrumentation to support minimal in-field scientific testing of atmosphere, soil and geological patterns.

Autonomy

- Basic obstacle avoidance capabilities contingent to following the astronaut through the rocky Martian terrain.
- Track two astronauts at all times, employ safety measures to avoid contacting the astronaut at any time, and relay video to base camp for additional safety.
- Maintain a current position estimate with respect to base camp at all times to enable "fetch" (rover returns to base, acquires necessary items, and returns to the field site autonomously) and "lead home" (guide astronauts back to base camp) commands.

Rover design

The final rover design, shown in Figure 5 and Figure 6, incorporates a rocker-bogie suspension, a dexterous arm placed on the front, all EVA related tools and equipment on the rear, and a centrally located arch support to elevate cameras and antennas. The vehicle body is 1.5-m long, 1.0-m wide, and 0.25-m high, with a 0.5-m ground clearance.

The vehicle's mass budget is shown in Table 2 and, depending on the scenario, ranges from a little over 220 kg to almost 300 kg. This is within the vehicle design requirements and shows the heavy dependency upon the specific scenario. The power requirements are also heavily dependent upon the scenario, and are shown in Table 3.

Subsystems

An overview and brief summary of each of the rover subsystems is given in the following section. A more detailed account of these subsystem designs, including trade studies and optimization, can be found in "Design of an Astronaut Assistant Rover for Martian Surface Exploration."[2]

Science Payload

The Science Payload supports Martian optical terrain studies, geological surveys and sample collection, seismology and meteorology data collection, and minimal in-field sample testing.

Optical terrain studies are carried out using a panospheric, omni, and stereo camera; a 24X telescope; an infrared camera; and a small camera mounted on the manipulator arm. The panospheric camera uses a hyperbolic lens, enabling a full 360-degree view of the surrounding terrain. The stereo camera is a pan and tilt unit (PTU) that may be controlled either from base camp during teleoperation mode or by the EVA astronaut through a remote control pendant (see Manual Control of the Assistant). The telescope is essential during EVAs to allow astronauts to evaluate a sight removed by some distance before executing a long and fatiguing traverse. The infrared camera provides data about Martian landscape temperatures and potential geologic activity. Output from the small camera mounted on the manipulator arm can be accessed by the EVA astronaut or base camp to inspect more closely rock formations not accessible to the astronaut either because of height, accessibility, or distance.

The manipulator arm is a 7 DOF pre-integrated package that is 114 cm long, weighs 18 kg, and has an average power requirement of 10 amps (A) and 28 volts (V) and reaches peak power consumption at 21 A at 8 V. The arm is able to produce a tip force of 111 newtons (N). Figure 4 shows the 7 DOF manipulator arm without end effectors.

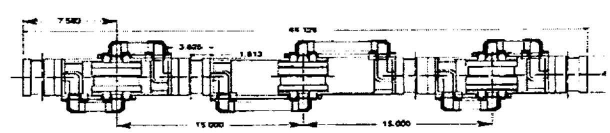

Figure 4: Manipulator arm

The arm autonomously swaps and stores interchangeable end effectors and utilizes a drive unit capable of activating grasping or rotary end effectors similar to the dexterous manipulators of the *Ranger*[3] neutral buoyancy vehicle used in the Space Systems Laboratory at the University of Maryland, College Park. One such end effector is the scooper/grasper that is able to retrieve 27 cubic cm of soil or larger rock samples. An impulse jackhammer is also included to break up rocks too big to retrieve or carry back to base

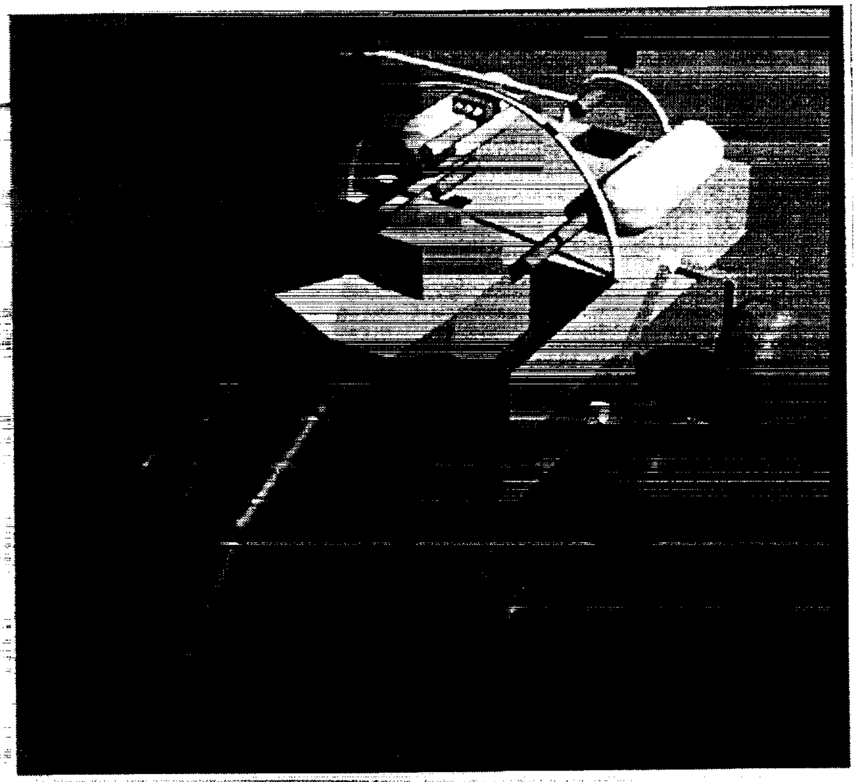

Figure 5: Fully configured EVA assistant

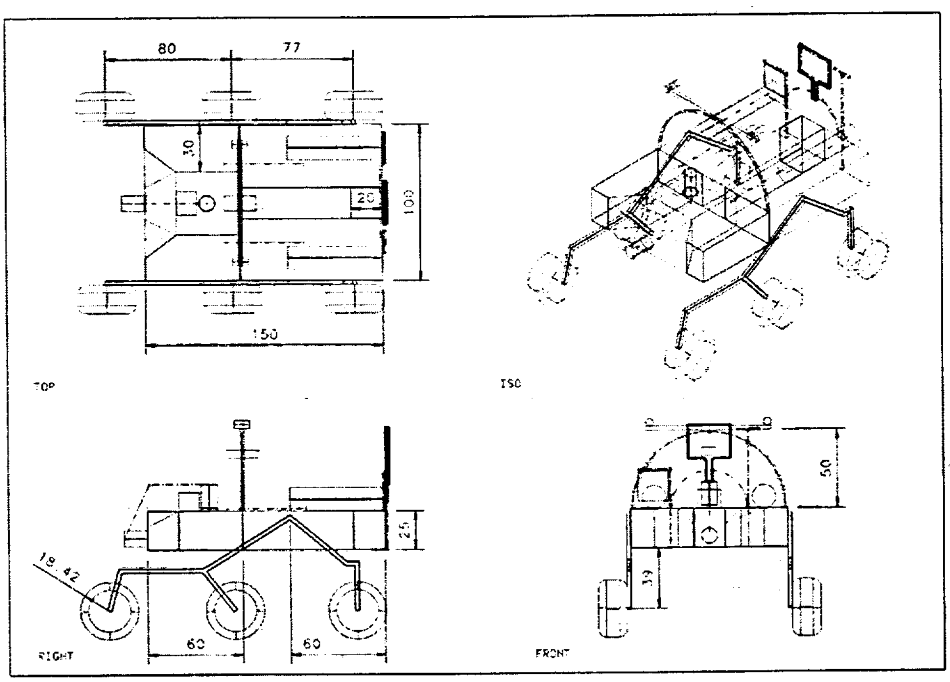

Figure 6: Dimensions of the EVA assistant

Table 2: Mass budget (all values in kg)

Subsystem	Scenario 1	Scenario 2	Scenario 3	Scenario 4	Scenario 5
Human factors	4	4	4	49	49
GNC	17	17	17	17	17
Communications	7	7	7	7	7
Science payload	44	29	49	54	49
Power	61	61	61	61	61
Thermal	1	1	1	1	1
Structure	30	30	30	30	30
Suspension	70	70	70	70	70
Total	**234**	**220**	**239**	**289**	**284**

Table 3: Power budget (all values in kg)

Subsystem	Scenario 1	Scenario 2	Scenario 3	Scenario 4	Scenario 5
Battery mass	**54**	**35**	**54**	**39**	**33**

so a sample may be taken. The arm is capable of deploying a core driller, which can take samples 3 cm in diameter and 0.5 meters (m) long. Astronauts designate sites for sample retrieval, DIP deployment, and arm operations through the use of marker flags, on-site remote control, voice commands, teleoperation, or preprogrammed autonomous control. All samples retrieved by the arm are placed and sealed in a bag on site and digitally marked and encoded with a bar code that can be linked to position, video, and other scientific data taken at the site of retrieval and placed in the main sample stowage bins. Samples retrieved by the astronaut receive the same encoding specific to the sample but are placed by the astronaut in a smaller sample stowage bin at the back of the vehicle out of manipulator arm workspace.

A 28X-180X microscope is provided when required by the EVA protocol to conduct in-field testing of samples. Adjustments to magnification, focus and other features is done via remote control. Optical feedback is given to the astronaut via video link.

All images collected, either through vehicle or arm mounted cameras, telescope, or microscope, are stored digitally on the rover and at base camp to minimize data loss.

Guidance, Navigation, and Control

The objective of this subsystem is simply to determine where the rover is, where it needs to go, and how to get there. The rover can perform autonomous and teleoperated navigation and mapping. Navigation is divided into two parts: global and local. Global navigation is going from base camp to the area of interest and back. It is composed of astronaut following (rover follows behind or beside the astronaut at a safe distance); autonomous navigation (final destination or path is specified and the rover must autonomously navigate); and teleoperated navigation (human operator drives the rover through an interface at the base or on-site). Local navigation is the process of obstacle avoidance that is performed by the rover and may be supplemented by a teleoperator. Mapping is composed of general terrain topography, and site and sample location recording.

The rover operates in several different modes — autonomous, semi-autonomous, teleoperated, manual, and voice command — each requiring different levels of functionality and performance. During *autonomous control mode*, the rover uses an on-board map from satellite images and referenced to the base camp to execute a mission to a pre-programmed site, perform necessary operations and return to base. *Semi-autonomous mode*, also referred to as "Follow Me" mode, allows the rover to follow the astronaut over Martian terrain. The rover is controlled from the base camp during *teleoperation mode* via a radio command link using the stereo camera PTU or omni cameras on the vehicle. The astronaut on site uses the control pendant to position the rover precisely or perform specific tasks during *manual control mode* (see Manual Control of the Assistant). Finally, *voice control mode* allows the astronaut hands free command of the rover to perform basic autonomous tasks. A list of voice commands is given below:

Rover Movement Commands

CREEP	Move forward slowly
GO	Move forward at moderate speed
RUN	Move forward at high speed
BACK	Move backward at moderate speed
SLOWER	Decrease speed until given STEADY
FASTER	Speed up until given STEADY
STEADY	Maintain current speed
TURN RT/LT	Low angle
SHARP RT/LT	Sharp angle
EASY	Ease the turn until given HOLD
HARDER	Sharpen turn until given HOLD
HOLD	Hold current turning angle
CENTER	Center the wheels
HALT	Stop all operations

Manipulator Commands

CORE	Take a core sample
SCOOP	Scoop up soil
PUSH X (Y)	Move end effector slightly in positive X (Y) direction
PULL X (Y)	Move end effector slightly in the negative X (Y) direction

Command hierarchy:
(1) Pendant Control
(2) Voice Control
(3) Base Control
(4) On-Board Control

This prioritization is made under the assumption that the astronauts at the scene will have the fullest understanding of the rover and operations status, and that the astronaut using pendant control will have the highest awareness with respect to the rover, the environment, the task at hand, and the other astronaut. It is, however, worthy of mention that the HALT voice command from any source overrides all control hierarchy and previously commanded tasks.

The rover must be capable of position determination, mapping and obstacle detection, path planning, and obstacle avoidance. Position determination is accomplished by using triangulated pulsing radio frequency (RF) beacons and an inertial navigation system (INS). The RF beacon system uses a concept similar to the Global Positioning System does on Earth by using land based beacons on Mars which can be expanded as necessary by adding more beacons. The RF system provides a position fix on the rover and astronauts with approximately 1-meter accuracy. The INS records lateral and rotational accelerations in three axes for an inertial record of the rover's movement relative to a pre-calibrated reference point. These systems are necessary for accurate mapping as well as assisting astronaut return to base in the event that the unfamiliar Martian terrain and lack of significant landmarks disorient the astronauts.

A tri-stereo camera system, laser striping, as well as the aforementioned omni camera are used for obstacle detection. Three stereo cameras mounted on a horizontal mast allow the construction of a three-dimensional map of the surrounding terrain complete with depth estimates to determine obstacles large enough to impede rover mobility. A laser striper may be used to augment this system during the day, and should a night sortie become necessary, provides all obstacle detection. The omni camera is used to detect astronauts and markers that may be located virtually anywhere around the vehicle. The PTU is the main source for teleoperated obstacle avoidance by providing flexible visual input to the operator and relying on the operator's judgment and driving skill to safely maneuver the rover. Regardless of what system is used, strict heed is paid to allow a minimum of 2 meters between the astronaut and the moving rover. The astronaut may approach the vehicle when it is stationary; however, all arm operations will cease if the astronaut ventures into the arm workspace and safety zone as detected by the laser striper and omni-view camera

<u>Communications</u>
The objective of this subsystem is to provide a method of transmitting control commands to the rover; receiving telemetry regarding the vehicle's state, health and data collected; and facilitating communications, voice, and video links. RF communication links are assigned and budgeted for each of these uses depending on data to be transmitted and received, frequency of use, and power required. Various RF links are required to complete the rover's communications system. Because astronauts must be able to command the vehicle from within the field or from the base, a commanding link, which the rover can receive, interpret, and forward to the appropriate internal controllers is required. The rover must also act as a relay station for the astronaut voice links in the event the signal transmitted by the astronaut is not strong enough to be received at the base. Telemetry for reporting the status of the rover and measured science data, along with video information, are combined into one data stream. This data is

transmitted back to the base camp at all times for recording and support from astronauts at the base camp. As can be seen, two way communication links are established between all participating members and potential command sources of the EVA team: (1) *base-to-rover command link* to allow an astronaut at base camp to remotely pilot the vehicle, position the pan and tilt cameras, and maneuver the robot arm; (2) *astronaut-to-rover command link* which transmits signals from a pendant (see Manual Control of the Assistant) allowing the EVA astronaut to send commands to the rover and to receive video information from the rover; (3) *voice links* that allows the rover to act as a relay station for voice communications with the base; (4) *astronaut-to-rover voice/video link* transmits video information from a camera mounted on the astronaut's helmet and then incorporates it into the video/voice/telemetry data stream for transmission to base; (5) *rover-to-base video/voice/telemetry link* transmits high rate video, voice, and telemetry data continuously back to base; (6 & 7) *base-to-rover voice link* is used for backup communications when the base voice data must be transmitted to the rover for retransmission to the EVA astronauts via the *rover-to-astronaut voice link*; (8) *video link* transmits video information taken from the variety of cameras; (9) *astronaut-to-rover voice/video link* transmits the EVA astronaut voice and video collected from the small video camera on the suit's helmet; (10) *rover-to-base video/voice/telemetry link* used to transmit video, voice, and telemetry to the base; (11) *rover-to-astronaut video/voice/telemetry link* is received by the pendant (see Manual Control of the Assistant) so that the EVA astronaut can see what the rover cameras are seeing; and (12) *telemetry link* used to transmit information on the health and status of its systems, information from the science instruments, and mapping and obstacle avoidance back to base. A detailed listing of channels, data rates and frequencies used for each link is available in the document "Design of an Astronaut Assistant Rover for Martian Surface Exploration."[2] The scope of this paper requires only the summary statement that all above mentioned team members are connected through necessary voice, video, or telemetry links to support all of the scenarios given in the introduction.

Locomotion and Suspension

The objective of this subsystem is to provide a reliable, stable, maneuverable suspension and locomotion platform capable of meeting the specified speed and lifetime requirements while minimizing sinkage, mass, power and required maintenance. Upon consideration of previous planetary vehicles, such as the Lunar Rover and Sojourner, the Martian terrain, approximate size and mass of the vehicle, and the imposed design and functional requirements, a six-wheel, five bar linkage Rocker-Bogey suspension is chosen. This design offers maximum obstacle clearance, allows the use of Ackerman steering, and increases rover agility within a reasonable mass budget. Two axles run laterally through the body of the rover aft of the centerline. Each axle meets at a differential that can exert a force to maintain the pitch of the vehicle for stabilization. Figure 7 shows the suspension of the vehicle and a surmountable obstacle. Figure 8 shows the vehicle clearing an obstacle of maximum size with vehicle body pitch control.

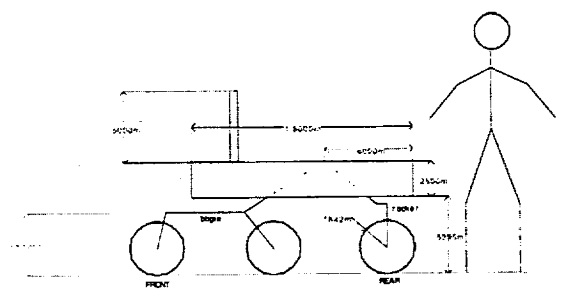

Figure 7: Side view of suspension with obstacle

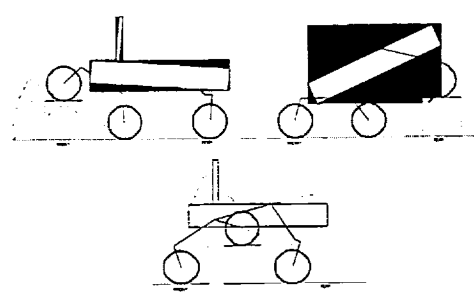

Figure 8: Rocker-bogey suspension clears the worst-case obstacle

Wheel dimensions are determined to minimize sinkage and maximize obstacle clearance. Rocker-bogey suspension studies have reported obstacle clearance of twice the wheel diameter, therefore a diameter of 36.38 cm and a width of half the diameter, or 18.19 cm, is chosen . The wheel exteriors are made from a wire mesh material similar to that used on Boeing's *Rover 1* to reduce mass and increase durability. The two forward and two aft wheels are each equipped with independent steering and motor capabilities to increase maneuverability and drive efficiency.

Power

The power subsystem provides the power required of all vehicle subsystems, exceeding worst-case estimates. It is also easy to integrate, maintain, recharge, and support on the Martian surface. Individual subsystem power requirements are dwarfed when compared to the power needs of the locomotion and suspension system. Therefore, an assumed maximum power requirement of 1.5 - 2 kW is based on worst-case design scenarios with respect to the locomotion and suspension system. Trade studies suggested Nickel Metal Hydride (NiMH) batteries be used because they meet the required power needs with minimal mass, volume, and support equipment. Batteries are swapped out individually, or in a block, allowing minimal down time and variable power capabilities. They may be used in any orientation, which is desirable when determining component placement on the vehicle, and recharge within a reasonable time period (less than six hours and 60% within 15 minutes) with a lifetime suitable to the rover needs. The low working and storage temperatures required by NiMH batteries also makes them a positive choice.

Two buses manage and distribute power to the subsystems. The large bus includes the arm and wheel motor controllers and maintains an operating voltage of 110 V. All other electronics (GNC, communications, etc.) receive power from a second bus at 28 V.

Thermal Control

The objective of this subsystem is to maintain elements of the rover within operating temperature limits during all mission phases. Thermal dissipation, material selection, exterior and interior temperature predictions, as well as thermal control are addressed.

Subsystem components fall nicely into three groups that warrant separate and unique thermal requirements. The first group consisted of external surfaces such as electronic or battery boxes, human interfaces and task boards, etc. According to NASA-STD-3000[4] any surface an astronaut may touch for more than 30 seconds must be between -120 and 113 °C. Therefore external interfaces and components are made with gold-coated aluminum. Extreme Martian temperatures are beyond optimal battery ranges and therefore warrant active thermal control. Batteries are contained in a Warm Battery Box (WBB) and electronics are contained within a Warm Electronics Box (WEB). Each box is made of gold-coated aluminum insulated with aerogel and is thermally monitored by a thermostat with thermistor control sensors controlled by the GNC computer. Both the WBB and WEB have exit ports for cables that allow heat dissipation, and the WEB uses other components as heat sinks to moderate temperature within the box. Motors and some science equipment are heated by low current flow through coils or strategically placed strip heaters

Structures

The objective of this subsystem is to support rover body mass, subsystem components and astronaut tools and interfaces to NASA-STD-3000[4] standards, withstand beam deflections and shear forces encountered during mission operations and accommodate design requirements.

A materials trade study, conducted according to the design priorities listed in the introduction, suggests a frame made of hollow, square beralcast alloy (aluminum-beryllium) tubing because of its high strength-to-weight ratio- meeting both the lightweight and durable design requirements. Structural analysis of frame loading and shear forces determined tube dimensions and subsequent mass estimate of 30 kg.

Following material selection, component placement dominated physical design of the rover. Human interfaces, such as astronaut sample storage, microscope, other interactive science payloads, the tool holder, and contingency life support are placed at the rear of the vehicle out of the workspace of the dexterous manipulator. Contingency life support is placed outside of the NASA-STD-3000[4] workspace suggestions, however, given the contingent nature of their use, this breach was deemed inconsequential. Batteries make up a large portion of the vehicle mass and are therefore placed across the center-aft portion of the vehicle for stability and quick changeout from the back. Cameras are mounted on the vehicle mast to avoid astronaut interference and enhance field of view. All end effectors, sample storage and packaging, and other scientific payload associated with arm operations are located at the front of the vehicle to minimize arm/astronaut interaction. The WEB is nestled under the scientific payload as it is necessary for all missions and may be removed for repairs from the underside of the vehicle as necessary. This design of component placement allows configuration changes due to vehicle use, as discussed in the Design Scenarios section, to minimally impact the vehicle's center of gravity, stability and performance.

Human Factors

Because the impetus for the creation of the EVA Assistant is to design a vehicle tailored to the needs of the astronauts during Martian EVAs, human interfaces and astronaut-rover interaction drove many of the design decisions. The following sections describe specific considerations made in the design of the vehicle and its interfaces to cater to direct human use.

Size of the EVA Assistant

Any interface that the crew nominally touches is kept between 117 cm and 158 cm from the ground, and a minimum of 13 cm and no more than 58 cm from the center of the heel.[4] The back of the EVA Assistant, which is the only section up to which the astronaut will walk, is at least 86 cm wide to allow for clearance of the spacesuit and accompanying portable life support system.

Astronaut Detection and Avoidance

As will be discussed later, the color white is used to designate a traverse hazard. Therefore, the white color of the astronaut's suit will be detected as something the EVA assistant will avoid, eliminating the potential of the EVA Assistant running into or over the crew.

Manual Control of the Assistant

Assistant Mobility

During normal operations, an astronaut controls the assistant's mobility by voice commands or a RF linked control pendant (see Figure 9). The pendant is shaped like a T and is approximately 20 cm wide by 30 cm long. So that the astronaut can do other work while holding the pendant, the pendant can be attached to the front of the astronaut's suit.

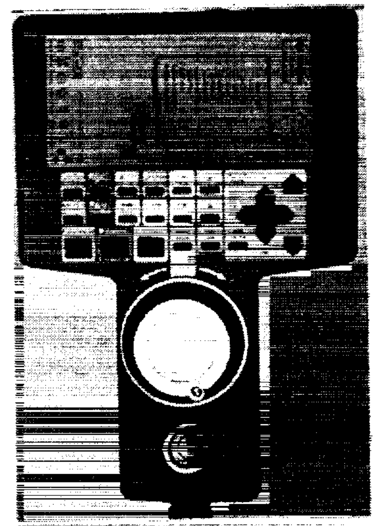

Figure 9: Pendant

The assistant's mobility can be controlled in two modes from this pendant. The first is *semi-autonomous control* using the 9-cm by 18-cm flat panel touch screen. The astronaut can select one of the assistant's video images to be displayed and, by selecting a point in that image, command the assistant to move to that point. A cross arrow keypad can also be used to move a cursor to the desired position on the video screen.

The second mode is *commanded direct drive*. In this mode, the astronaut uses +/- buttons to adjust the speed of the assistant and a direction dial to steer. For safety, if the assistant loses the RF link with the pendant or detects that it is too close to the pendant, the assistant halts all motion. A red kill switch is on the pendant at the bottom, and when pressed, halts the assistant.

In the case of RF link failure, a hard-wired cable can be used to attach the pendant to the assistant. If for any reason this cable becomes disconnected (rover moves faster than the crewmember, etc.), all assistant motion will halt.

If the on board computer system fails, the hard-wired cable can still be used with the pendant. Contingency software on the pendant allows direct connection to the assistant's motor drivers and sensors, allowing the astronaut to drive the assistant manually. In this mode, the astronaut uses the +/- buttons to adjust the speed of the motors and the direction dial to directly move the wheel actuators to adjust the direction of the wheels.

If all other systems fail, the wheels can be set to spin freely. This allows the astronaut to "push" the assistant for a short distance to get it into a safe spot and configuration.

Arm Control

The standard operation of the arm is conducted through voice commands and in this mode, the arm is fully autonomous. If more precise control is necessary, the arm may be controlled using end-point supervisory control in two modes from the pendant. For the first mode, the cross arrow keypad or the touch screen can be used to move a cursor to the desired end point position on the video screen. The second mode, direct end point motion, can be attained using the touch screen to select the arm frame of reference and the cross arrow keypad to move that frame.

Contingency Operations

If the arm stops working in any of the standard modes, there is a button on the pendant to stow the arm. If all command authority is lost, disconnecting the power to the arm allows the arm joints to be manually backdriven.

Emergency Life Support

Emergency life support is provided for two astronauts for two hours each. In case of an emergency, there are two oxygen bottles mounted on the back of the rover. A hose is connected to the bottle and then to the astronaut's suit. The bottles provide free-flow oxygen at 41 mega Pascals. There are no provisions for water cooling of the suit. The bottles can be removed in the case of EVA assistant failure allowing the crewmember to carry the bottle with them back to base camp.

Mobility Aids

Mobility aids provide stability during walking traverses over rough ground, through rock rubble areas, and both up and down sloped traverses. They are also useful as body support members during rest periods. The mobility staff is used for downhill traverses, particularly rocky ones and the modified ice ax is used for flat surfaces and for uphill traversing. Figure 10 shows a modified ice ax being used during a field test in Flagstaff, Arizona area.[5]

Figure 10: Modified ice ax[3]

Markers

Markers are used by the crewmember to let the EVA assistant (either in autonomous mode or while being teleoperated) know what to do at a particular site. Three colored markers, each with a different shape on it, are used to designate a scientific site[7] and a white marker with a black cross denotes that the site is a hazard and should be avoided. These colors were selected because they are on opposite sides of the color wheel from the red color of the Martian surface and atmosphere. The type of scientific data that is to be collected at each site is designated by the shade of color of the marker and the shape that it contains. The color is used by the rover to discriminate what type of site it is and the combination of the shape and color is used by the crewmember at base to determine what type of scientific activities to conduct at the site. The markers are actually cubes that fit over a 1 m pole that is driven into the ground, allowing the marker to be seen from four sides.

A green marker with a black triangle in the center, as shown in Figure 11, designates that the crew members want complete imagery of site, including partial panorama, stereo imaging for topographic setting, close-up imagery, and some sort of science data.

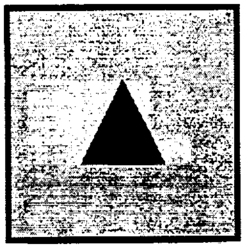

Figure 11: Complete imagery marker

A yellow marker with a black circle, as shown Figure 12, designates that the crewmembers want close-up imagery only.

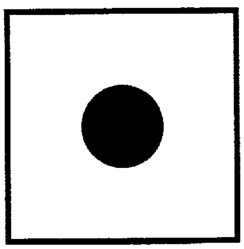

Figure 12: Close-up imagery marker

A blue marker with a square, as shown in Figure 13, designates that this is a site at which the rover should collect samples.

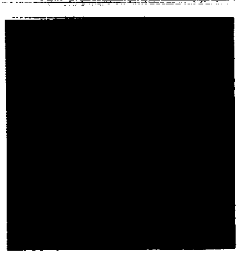

Figure 13: Sample collection marker

A white marker with an "X", as shown in Figure 14, is used to designate that this site is a potential traverse hazard (i.e., steep slope, ravine, and trough). As discussed earlier, the white color of the astronaut's suit also designates them as a traverse hazard.

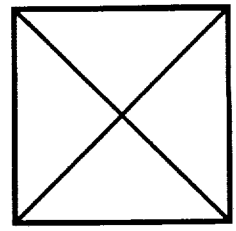

Figure 14: Potential traverse hazard marker

Four of each type of marker and poles are carried and the total mass of markers is 1 kg.

Tools

The tools are designed such that gripping the tool is not necessary to control it. This is done through the use of straps on the handles of the tool. A wrench, similar to what is flown on Space Shuttle and International Space Station missions, shown in Figure 15, is carried on the rover for contingency operations such as releasing and stowing the arm.

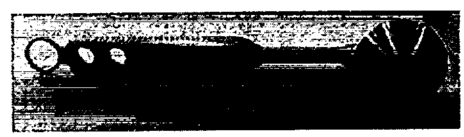

Figure 15: Wrench

A rock hammer is carried that is similar to the Heavier Weight Hammer flown on Apollo 14 – 17, shown in Figure 16.

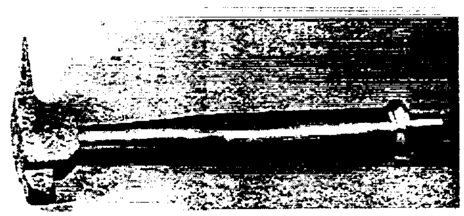

Figure 16: Rock hammer[6] (NASA photo S71-22471)

A scoop device, similar to the Large Adjustable-Angle Scoop flown on Apollo 16 and 17, shown in Figure 17, is needed for digging trenches and sampling particulate material.[6]

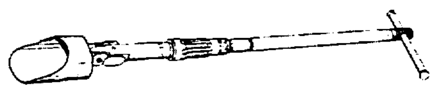

Figure 17: Scoop device[6]

The long handle tongs, similar to the 32-inch Tongs used on Apollo as shown in Figure 18, are used to pick up 6-10 cm diameter rocks.[7]

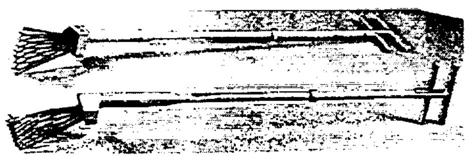

Figure 18: Long handled tongs[6] (NASA photo S71-22469)

A 10X-power hand magnifying glass similar to that used by geologists on the Earth is also carried.

A tool holder is mounted on the EVA assistant so crewmembers can select the proper tool and carry it to the site, rather than carry the whole tool kit. This is similar to the procedure that was used in the later Apollo missions and suggested by field tests here on Earth.[3]

Conclusions

The 300 kg, 2 cubic meter, semi-autonomous robotic astronaut assistant detailed in this paper helps to reduce astronaut fatigue and increases productivity by performing time consuming, fatiguing and repetitive dexterous tasks. The rover operates in teleoperated, astronaut assisted, and autonomous EVA modes. It relieves the astronaut by retrieving, cataloging and carrying collected samples; carrying tools and in-field scientific equipment; and carrying a 7 DOF dexterous manipulator. Safety of the EVA sortie is increased because the rover can compile and store a detailed map of surrounding terrain; estimate the current position with respect to base camp; provide a redundant communications system; and carry emergency life support. As NASA begins to think about human exploration of Mars[1], the astronaut assistant detailed here is a logical complement to any Martian surface exploration.

Acknowledgments

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Acronyms

A	ampere
cm	centimeter
DIP	deployable instrument packages
DOF	degree of freedom
EVA	extravehicular activity
GNC	guidance, navigation, and control
INS	inertial navigation system
kg	kilogram
kph	kilometers per hour
m	meter
N	newton
NASA	National Aeronautics and Space Administration
NiMH	nickel metal hydride
PTU	pan and tilt unit
RF	radio frequency
V	volt
W	watt
WBB	warm battery box
WEB	warm electronics box

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The Exploration of Mars: Crew Surface Activities[1]

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Abstract

Surface activities of the first Mars mission crew, as suggested in phase I of the NASA HEDS reference mission, are discussed in this paper. The HEDS reference mission calls for a two phased approach. In phase I, humans supported by robotic systems will explore the Martian surface, collect and analyze geologic, geophysical, and meteorological data, search for potential permanent base sites, and conduct technology verification experiments. In phase II, a Mars base site will be selected, and the building of a permanent human base will be initiated. In this report two complementary architectures are portrayed. First, a permanent base for 3-6 people consisting of an ISRU unit, two nuclear power systems, a green house, and inflatable habitats and laboratories, built inside adobe structures. Second, a reusable, and resupplyable methane propelled very long range type traverse vehicle capable of collecting and analyzing data, and repairing and deploying scientific payloads during its planned 150 days 4800 km traverse. The very long range traverse vehicle will carry smaller rovers, crawlers, blimps, and an air drill capable of quickly reaching depths beyond 100m. The report presents a global vision of human activities on the surface of Mars at a programmatic level. It consists of several vignettes called "concept architectures" We speculate that these activities will facilitate a phase I Mars exploration architecture.

1. Introduction

With the ongoing construction of the International Space Station, NASA and space agencies around the world are seeking a vision for humanity's next step at the space frontier. It is quite possible that a human mission to Mars might provide a nucleus to align the efforts of the agencies for space activities in the new millennium.

A mission to Mars has many goals, including the study of comparative evolution of Earth and Mars, the assessment of how Mars has changed, and most importantly, determining whether our planet is faced with a similar fate. The answers to these questions hold not only scientific value but when viewed in the larger social context could change the way we view ourselves. The recent discovery of meteorites that might have bacterial fossils gave the initial spark and helped galvanize the need to send explorers and scientists to the Martian surface. By sending these explorers to the Martian surface we may finally be able to answer one of humankind's biggest questions: "Are we the exception or the rule?"

The exploration of Mars has begun in earnest with robotic missions, currently either in orbit, en route, or in production. These missions will establish the skeleton infrastructure that will be needed before human activities can commence. These missions have also begun to provide a picture of the planet that was previously unavailable. Data being returned suggests that Mars is a planet with dynamic geophysical processes and could have harbored life. Geologists are eager to go there and conduct conclusive experiments to find out if life ever evolved there, if certain forms still exist and also if the planet might be able support humanity's ambition to extend a branch of civilization to that planet.

The next step will be to use the gathered data to select potential landing sites for the first human missions. We envision these human missions to be the precursors to the establishment of a permanent settlement on Mars. We imagine that this could happen in the next fifty to seventy five years.

A human mission to Mars is being studied by NASA and other space agencies. NASA has identified an opportunity for such a mission in the year 2012. Alternative mission architectures, including the nature and duration of activities of the crew during a surface stay of approximately 600 days are currently being discussed by the agency. The USC Mars Exploration Team's preliminary findings of a candidate expedition crew surface mission are presented in this paper.

2. Mission Objectives

Our focus is on the activities of a first human mission to Mars, but we assume that the ultimate goal is a phased, long term human exploration of the planet. Our primary concern is in conducting activities and investigations that will lay the foundation for, enable, or enhance the next several human exploration missions.
The goal is to sustain a six person crew on the surface of Mars for approximately 600 days, and return them safely to Earth. During the surface stay, the crew will perform tasks that are considered impractical or impossible for robotic missions. The proposed crew activities include:

- Demonstrating equipment and techniques to sustain human life on the surface of Mars (e.g., in situ resource use, agricultural experiments, life support technology verification, demonstration of recycling strategies)
- Exploring the feasibility of human settlement on Mars
- Gathering and analyzing data for selection of potential permanent base.
- Providing infrastructure elements for future missions
- Searching for permafrost, ice, or water
- Searching for evidence of past life

The mission system architecture elements for the human exploration of Mars can be divided into two areas of focus; habitation and exploration. Each of these areas is important to determine the feasibility of a human presence and sustainability on Mars. The habitation elements will focus on the development of structures and systems to sustain a human presence, while the exploration elements will examine the Mars environment for natural resources and a permanent base camp for future missions.

The habitation base camp architecture will focus on the survivability and productivity of the human species in a foreign and hostile environment. The crew will experiment and develop habitable structures using materials from Earth and Mars that are capable of providing safe and comfortable working and living spaces. Different types of building technologies and techniques including inflatables, expandable domes and indigenous material structures will be built and tested for safe, long term occupancy. Agricultural experiments will also be carried out to study the effects of the Martian environment on plants. These experiments will be developed in a controlled environment to study the feasibility of human self-sustainability on Mars.

The rover exploration architecture will focus on expanding our knowledge of the Martian environment while searching for natural resources and future permanent base camp locations. Most of the exploration will be performed using the very long range traverse vehicle (VLTV). This vehicle will be capable of traversing the Martian landscape with a crew of three for six months minimum, while providing a comfortable living and working environment. The vehicle will be supplied by cargo landers that will place supply caches at five to seven locations along the VLTV traverse route. Each of the re-supply landers will provide human necessities along with scientific and monitoring equipment to be used for the duration of the traverse. Experiments will be performed to analyze the Martian soil using the Deep Driller which will be capable of drilling down to 100 meters in 3-5 days. Safety is a major issue for the VLTV crew. For this reason an emergency rover concept for rescuing the VLTV crew in an emergency is being studied. This vehicle is capable of carrying a crew of four, (three from the VLTV, one from the base) for up to 3 weeks to return the VLTV crew to the base camp.

A communication system will be implemented to support ground and remote communications for the crew. This system will contain high-gain directional antennas, areostationary relay satellites, and local transmitter/receivers for line-of-sight communications.

Robust power systems are critical to the survival and sustainability of the Martian crew. Different types of power systems deployed will include nuclear systems for the base camp, methane/fuel cell technology for the VLTV, and fuel cell utilization for the emergency rover. In situ resource utilization (ISRU) will be used extensively throughout the architecture. Methane production from the Martian atmosphere is but one example of ISRU that can be implemented for power and propulsion systems.

Tele-operated rovers, both aerial and land based, will be used extensively by the Martian crew to explore terrain that is too dangerous for the human crew, and to study proposed traverse routes. These craft may be operated by the crew from the safety of the habitats/VLTV using haptic systems and virtual reality displays to provide real time data input and acquisition. The mini-rovers can also gather material samples to support the scientific experiments being studied throughout the mission from both the base camp and VLTV.

Whether supporting the base habitation camp, or the VLTV traverse, each system element has a significant impact on the feasibility and success of the human Mars expedition. The following sections in this report will describe in more detail each of these elements, and the significance each has on the proposed mission architecture.

3. Candidate Mission Profile

The first human mission to Mars consists of three phases of operation:

- Precursor Missions
- Cargo Missions
- Crewed Mission

Precursor missions include robotic science and technology verification payloads designed to help us better understand and predict the nature of Mars and its resources. Some examples of robotic mission objectives are to: survey the Martian surface for optimum landing and geo-bio site locations; return data on the exact nature of the Martian radiation environment, the planetary protection provided; obtain meteorological data on Martian weather patterns; and return samples of Martian soil to Earth for agriculture and construction testing. Specific details about these missions are provided in Appendix A.

The second phase of the overall architecture is the multi-staged cargo mission. The cargo mission will provide the human crew with a fully fueled Earth Return Vehicle (ERV) and Mars Ascent Vehicle (MAV) before they leave Earth. Cargo missions will deploy the power plant, ISRU facility, consumable supplies, VLTV, and Earth manufactured construction materials on the surface before they land.

The third phase of the mission architecture is the human mission. The crewed mission objectives are to determine the feasibility of humans living and sustaining themselves on Mars, and to explore the geological, geophysical, meteorological, and biological history of Mars during a long range scientific expedition of the planet's surface.

4. Base Camp

4.1 Site Selection

Using the NASA reference mission as transportation baseline, six crew are landed at a safely accessible site in the low latitude region surrounding the Valles Marineris Canyon system and the Tharsis volcanic region. The site selected was Chasma Perrotin. It was selected because of it being in a safe and temperate equatorial region of Mars, and because of its rich variety of terrain and features in a compact area for geological and geophysical explorations. The site also offers the best choice from a trajectory alignment point of view for landing, orbital support, and ascent and departure.

4.2 Base Camp Architecture

The Mars outpost base camp is designed to test the feasibility of humans sustaining themselves on Mars. The base camp architecture fuses Earth and Mars based resources to allow the crew to grow food, create habitats and a greenhouse, produce their own fuel, H2O and O2, and protect themselves from the hazards of the Martian environment. The base camp consists of the crew lander module, central Adobe habitat, agriculture-life science module, MAV, ISRU facility (in addition to the MAV ISRU plant), 2 nuclear reactors, and redundant lander module filled with an emergency 600 day consumable (food, water, air) supply. The Lander module serves as the Command and Control Center of base camp, provides additional living space, a

Solar Particle Event (SPE) storm shelter, and radiation monitoring-testing laboratory space (See Fig. 1). The life science module serves as the camp greenhouse and houses/controls the base camp bio-regenerative CELSS experiment. Additional infrastructuredevelopment over the course of the mission includes a simple road network around base camp, adobe storage houses, adobe lift-off shield, and landing/liftoff pads for the MAV and future landers. With redundant power sources, living spaces, and transportation systems, the base camp tests the potential of ISRU but does not force the crew to rely on it in the case of mission failure. The crew and robotic devices landed in the cargo missions will construct the base camp over the course of the 600-day mission.

4.3 Adobe Habitat Architecture and Construction

The central habitat of base camp as well as the related infrastructure including exposed platforms, roads, aprons, shields, wind breaks and other protected areas and utility channels are envisioned as Adobe structures made with in situ Martian regolith. The Adobe shell/exterior of the habitat houses an inflatable membrane that functions as a self-contained pressure vessel for the crew to live in. Adobe material was chosen to expand on the ISRU for habitat construction and environmental protection objective of the base camp mission. Martian Regolith is a free resource available to us in unlimited quantities on the surface of Mars. The Adobe exterior serves as an excellent Galactic Cosmic Radiation (GCR), Solar Particle Event (SPE), and Ultra Violet (UV) radiation shield, a thermal insulation layer, and provides complete dust storm and micro meteoritic (MM) impact protection. Although the radiation environment of Mars is not expected to be as severe(lower annual dosage) as the moon or interplanetary space, two years of exposure to constant unprotected levels on the surface can pose a considerable radiation risk

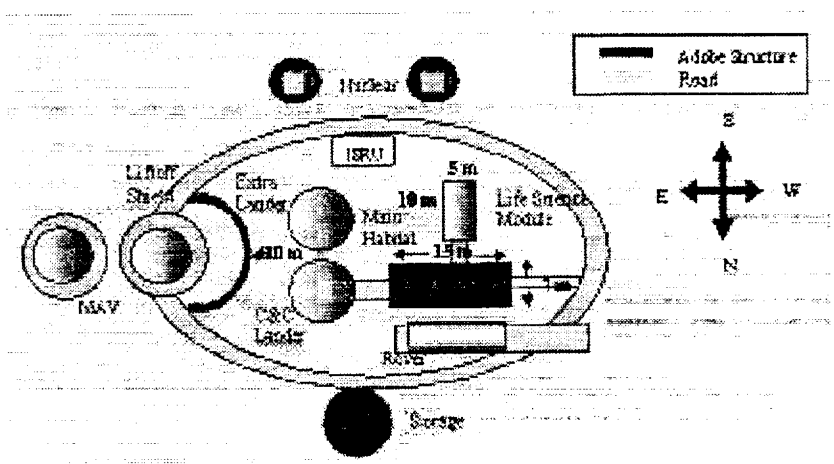

Figure 1: Base Camp Site Plan

to the human crew. Using Adobe would dramatically reduce this radiation risk and reduce the overall mission cost in terms of materials development and payload that needs to be sent to Mars [Simonsen 1997]. Adobe construction and use plays a vital role in this base camp mission architecture and in the eventual development and self-sufficiency of humans on the surface of Mars. The Adobe exterior serves as a shell to house the pressurized membrane; not as the structural support to mold and contain the 9 psi environment within the habitat. To compare it with the reference Transhab technology for perspective, using an Adobe shell to house the inflatable membrane dramatically reduces the overall mass of a Transhab type of structure on the surface of Mars. The layers for GCR radiation and MMOD impact protection would no longer be needed, and the insulation layer thickness would be greatly reduced. Water shielding and most typical types (Al) of radiation shielding material are useful against UV and SPE radiation, but have minor reduction against highly energetic galactic cosmic rays. Metal radiation shields can even endanger the crew due to the harmful secondary electron production that results when impacted by GCR's. Galactic Cosmic rays can only be stopped by a large, dense spatial distribution of material; for example Martian Regolith.

The Adobe habitat construction relies on the regolith drill/pump, ISRU brick baker, plastic superadobe bags to hold soil, an inflatable half cylinder membrane (smaller radius of curvature on the bottom), and a Plexiglass sheet supply on the surface. The brick baker is a simple device, which pours Martian regolith into a mild compression mold, bakes it in a small furnace, and produces a brick in the shape of the mold. Molds used in the base camp construction include quarter and half arches, flat square panes, and rectangular bricks. Heat for the furnace is provided by the nuclear reactor. Experiments will also be carried out using solar concentrators to bake the bricks into the desired shapes. The habitat wall structure is created using coil and rectangular bags filled with the free supply of regolith. Each layer is formed directly on top of the other layer and hardened due to compression of the layers above it. A plastic brace is used to support the construction of the walls. Specific spaces (arches) are built into the wall structure for the placement power and gas lines, Plexiglas windows, and airlock junctions. The roof of the habitat is constructed using the "leaning arch" technique. Arches are laid from both ends of the structure in an inclined position gaining support from the previous arch until both sides meet. Quarter and half arches are placed on a movable hard plastic arch track supported by struts and moved on rollers. As each inclined arch is laid in place, the arch track is moved to the next position and the process is repeated from both sides until the roof is complete. Crew involvement in the construction of the habitat includes measuring and triangulation of the site, placement of the empty bags on each successive layer, connecting their open end to the regolith pump, and operating/guiding the servo-operated three pulley system to place the arch fragments on the track. The 1000 kg capacity crane on the VLTV can also be used as a construction aid if needed. The three-pulley system raises the arch segments vertically, horizontally, and laterally over the arch track, with a crew member on a platform guiding each arch segment into place. The pulley is operated by a servo located on the ground. The inflatable, half-cylinder, polymer membrane provides the pressurized environment of the habitat. The inflatable has one main space qualified airlock, and two smaller internal airlocks. It is fitted with built in plumbing and power connections accessed from inside and out for easy checkout and assembly. After the habitat exterior is constructed, the crew will lay down the base floor matting, and then bring in the inflatable to match up its airlocks with the wall openings accordingly (significant margin is built into the wall openings to provide easy match up with the membrane airlock locations). Gas and power line connections are made between the membrane and ISRU facility by the crew. The membrane is then inflated with oxygen and required buffer gases at a regulated 9 psi supply pressure from the ISRU facility. After some initial inflation, the crew will affix the base of the membrane to the Martian surface to line up the permanent airlock junctions. The membrane inflation is completed when the internal pressure reaches 9 psi. Two inflatable tunnel airlocks (IT) are attached to the habitat, with one joining the lander to the habitat (IT 1), and the other joining the habitat to the agriculture-life science (LS) module (IT 2). Each IT has a seal at each end. The inner seal of IT 1 is connected to the habitat and opened to pressurize it and the habitat to 9 psi. The outer seal is then opened and the crew can walk freely out of the lander and into the hab. Due to the automated construction of the habitat, wall construction is expected to take 2 weeks, roof construction 2 weeks, and membrane deployment and integration 1.5 weeks. A margin of two weeks is added to account for any delays. During the construction period, the crew will use the lander coupled with the VLTV to provide additional living space. Internal supplies, furniture, and equipment will be brought in after inflation. Some room partitions are built into the inflatable design, but shelves and hard walls need to be assembled after inflation. The habitat module is divided into two parts: the living/recreation area and the work area. The work area has science laboratories, a workshop, exercise room and a clinic. Equipment in the science laboratories focuses on analyzing the radiation and geological environment of Mars. Laboratory capabilities and equipment includes stereo microscopes, SEM, TEM, EDS machine, X-Ray diffraction, chemical analyzers, small furnace, wet-chem lab, and computer stations for analysis. Radiation monitoring and testing equipment will be housed in the C&C lander module. The clinic will be used to provide crew checkups and to monitor the effects of the low gravity environment on the human body. Some primary care is possible including minor surgery and dentistry The living space has three bedrooms (with two beds per room), toilet / washroom, meeting area, eating area, and pantry to store additional food and supplies. The habitat can house all six crew members, but will be occupied for the majority of the mission, by the base camp crew of three. The ECS system of the habitat sends waste water, food, and CO2 to the LS module where it is processed and recycled as part of the bio-regenerative CELSS experiment. Additional backup recycling units are also found within the habitat and lander modules in case of LS module failure.

4.4 Life Science Module Architecture and Construction

The life science (LS) module is a 9 psi pressurized air environment. It is connected to the main habitat via an internal IT airlock. The outer structure of the LS module is a prefabricated polycarbonate honeycomb structure with square shaped openings. The honeycomb is folded out by the crew and fixed into place (in relation to the main habitat) by hammering stakes through tabs at the base of the honeycomb into the ground. Additional Martian soil bags can be added to the base if needed. The LS module will experiment with growing plants in natural and artificial light; therefore, part of the LS module will be transparent and part opaque. 60% of the honeycomb will be filled with Plexiglas squares treated with a Cerous 3 polymethacrylate (CPMA) UV [Yen 1979] resistant coating, and the remaining 40% filled with solar cells mounted on Martian bricks. The crew will be responsible for slotting these pieces into place as part of the construction phase. The inner structure of the LS module is a bi-layered polyethylene inflatable (transparent) similar to the habitat membrane with only one minor airlock. Polyethylene and Adobe materials were chosen for their GCR radiation shielding properties [Wilson 1997]. Polycarbonate and CMPA were chosen for their UV radiation resistance properties.

The inner membrane environment is equivalent to the habitat / lander environment. The second layer of the inflatable is a dense layer of Martian CO2 pumped in from outside. This layer enhances the greenhouse heating and provides additional GCR radiation protection. The greenhouse heating of the LS module is used (in conjunction with electric power heaters) to heat the main habitat using fans and cross flow heat exchanger lines to re-circulate the warm air. After the bi-layer inflation is complete, IT 2 is connected to the LS and main habitat module. When the LS module is pressurized and heated, the seals of the tunnel will be opened allowing full access between all modules. The biomass recycling/production chamber is then brought into the complex to bring the CELSS online. LS module research labs include Martian soil growth experiments with (non)genetically engineered plants in natural and artificial light, hydroponics growth laboratory, and an Extremophile laboratory where plants are genetically engineered to thrive in a low temperature and pressure CO2 environment for eventual growth on the surface. CO2, O2, and H2O recycling units will also be placed in the LS module to accompany the biomass recycling units. The LS module functions as the chemi/bio-regenerative CELSS experiment. Waste water is recycled using filtration for urine, baking of feces, and Zeolite Sieves to recapture water vapor produced by perspiration and plant transpiration. Plant photosynthesis and carbon molecular sieves located in the LS and habitat modules remove the excess CO2. The waste CO2 is then returned to the ISRU facility to be re-entered into the Sabatier reaction for H2O, O2, and CH4 production.

4.5 Site Development

The Mars Base camp development continues beyond the main adobe habitat and LS module construction, over the course of the 600 day stay on the surface. Adobe construction techniques will continue with the base camp crew working with camp robotic devices to develop a simple elliptical road network around base camp. The road network will connect the main airlocks,

habitat ISRU facility, storage locations, and the MAV liftoff pad. The road network development consists of clearing paths and using solar light concentrators (parabolic lenses) to bake the surface of the path. The crew will build Adobe storage houses to store equipment, supplies, and surface mobility vehicles and construct an adobe liftoff shield to protect base camp from debris brought up during liftoff and future landings.

4.6 Communications

High data rate communication between Mars and Earth is accomplished at Ka-band directly from the surface of Mars. This approach allows high rate (20 Mbps) communications by exploiting the mission's nuclear power source, and the presence of astronauts on the surface allows the construction of a relatively large antenna compared to what might be feasible on a Mars-orbiting spacecraft. In addition, unlike a satellite system, the ground-based system can be maintained or repaired by astronauts if necessary. This high rate ground system, however, provides communications for less than half of the Martian day because of line-of-sight obstructions.

The other key elements of the communications plan are a pair of areostationary satellites. These satellites provide a moderate rate (100 kbps each) X-band link between Mars and Earth, and also provide a communications relay between astronauts at the surface base station and those on long-term on exploration expeditions. The use of two areostationary satellites provides redundancy in case of failure. Other backup options were considered, such as the use of surface relay stations, but such stations would be complex, time consuming to assemble, and difficult to power and maintain.

By separating the satellites 19 degrees or more, we can maintain constant contact with Earth (a single satellite would experience daily communication losses of up to 71 minutes per sol due to eclipses when Mars blocks the line-of-sight between the satellite and Earth). The satellites could also provide aerial photographs which could provide advance warning of dust storms. Launching the satellites well in advance of the crewed mission would provide a communication link for preceding robotic missions. Because the satellites are supporting modest data rates compared to the surface-based Ka-band link, the satellites can be powered using solar arrays, avoiding the political objections to additional launches of nuclear power sources.

Planetary geometry has significant impact on communications between Mars and Earth. If the data rate is adjusted to maintain a constant bit-error probability, then the sustainable bit rate varies by approximately a factor of 20 over the mission duration. In addition, communication is disrupted for a brief period from solar scintillation effects when the Sun-Earth-Mars angle becomes sufficiently small. Providing a reliable link during such events might be accomplished by the use of an additional satellite in Solar orbit at about 1 AU (perhaps at the Earth-Sun libration point) which would provide a communications path that didn't require a small Sun-Earth-Mars angle. This solution is deemed unjustifiably expensive.
Under any communications scenario, the value of the Earth-Mars link is significantly augmented by exploiting data compression and buffering. For example, such technologies could allow daily transmission of brief high-definition television (HDTV) clips which could significantly increase public involvement. Further development of optical communications technology could also significantly enhance data return from a human mission to Mars, and provide mass and power savings [Hemmati97, Hall90]. We have not selected such a system as our baseline because, while the development of optical deep space optical links would significantly enhance the mission, communications needs could be supported with currently available technology.

4.7 Power System

An initial source of large-scale power for the base camp will come from the same power source used to fuel the ISRU production of methane and oxygen for the Earth Return Vehicle. A 4 ton, 100kW nuclear reactor manufactured on Earth can provide a continuous source of power for 7 - 10 years which is more than enough power for the base operations of the first human mission. Although compact in size, this unit will require at least 12 tons of shielding in order to be medically safe for the crew. The reactor will most likely be located away from the base camp, preferably in an ancient impact crater whose walls would provide shielding.

Transmission of electrical power will require the use of high voltage power lines buried at least one meter below the soil. Power can be distributed to batteries and electric motors that power the habitat, agricultural, and science modules. This will be the beginning of a power infrastructure whose electric distribution network can be expanded to meet the needs of a growing base camp.

Solar power arrays as a primary source of power for the base camp will be infeasible for the first mission since it would require approximately 25,000 kg of material from Earth to manufacture solar panels that are capable of providing the same power output as the reactor. Smaller solar arrays can be used for the generation of power at a much smaller scale that can be used for various sub-systems. One consideration could be kinetic flywheels that whose electric motors are powered by solar energy. These 2-3m flywheels can then be used to generate electric power. This power source will not be considered as a source of power for the mission, but can be a technology demonstration of future power systems that can be utilized on the surface.

The reference mission also calls for the recycling of water from organic wastes. These same waste products also naturally release methane, which can be used as a source of fuel for space heating and cooking. The process, known as pyrolysis of biomass, occurs when organic wastes are placed under high temperature and pressure to decompose organic material. During this process, several gases are released, including hydrogen, which can be used in fuel cells.

4.8 ISRU/ISRP Technology & Application

ISRU (In-Situ Resource Utilization) and ISRP (In-Situ Resource Processing) are two very important concepts to be used to make human presence on Mars possible. The concept would be to take a raw resource, such as the atmosphere or soil, and then using a sequence of processes, create useful commodities such as Oxygen from the atmosphere or Iron from the soil. In general, there is agreement that ISRU can significantly reduce the cost of exploration, especially for extended duration missions. If the concept ISRU is applied as part of the initial mission design for a human Mars mission, it will provide substantial mass savings and reduce mission risk because the amount of consumables that will need to be taken. This will in turn, have considerable impact on the overall mission size. The ability to automate the production of propellant, consumables and other materials from available resources from Mars will allow a higher level of mission feasibility and reliability. For example, the overall mission robustness of certain surface systems would be increased. This would be because caches of consumables, such as surface vehicle fuels, and Mars Accent Vehicle fuels can be maintained and kept at peak levels.

The sources for ISRU on Mars will come from two primary sources, the Martian atmosphere and soil. The atmosphere of Mars is much thinner than that of Earth, with a surface pressure averaging 1/100th that at the surface of the Earth. Surface temperatures range from -133° C at the winter pole to 23° C on the dayside during summer. The soil and atmosphere composition is shown in the following chart and table and is only based upon previous Mars Mission data.

Mars atmospheric composition (%): (CO2) Carbon Dioxide, 95.32; (N2) Nitrogen, 2.7; (Ar) Argon, 1.6; (O2) Oxygen, 0.13; (CO) Carbon Monoxide, 0.08; (H20) Water, 0.0325; (NO) Nitrogen Oxide, 0.1547; (Ne) Neon, 0.00039; (Kr) Krypton, 0.0000464; (Xe) Xenon, 0.0000124.

Future missions and exploration will expand this list of resources, and could provide even more options for future mission planning. One such option would be the significant presence of water that could be easily obtained and utilized. This resource would be the most valuable to any type of human presence. Until such future resources are determined and verified, mining and then processing the Martian atmosphere and soil for known substances, will provide several key materials for Manned mission success. Future mining and processing of the surface materials listed here will require equipment, which will probably not be included in early Mars missions. With processing "Air mining" could yield Water, Oxygen, Nitrogen, Carbon Monoxide, Methane, and Ammonia, while "surface mining" could yield Water, Sulfur, Iron, Titanium, Aluminum, magnesium, ceramics, glass, and other building materials

For the first series of Manned Mars Missions, the primary needs from ISRU will be the manufacture of consumables. Such manufacturing will be Oxygen and propellant in the form of Methane (CH_4). Other important resources that can be extracted from the Mars atmosphere are buffer gases. Nitrogen and Argon make up a significant volume percent of the Martian atmosphere. When they are separated from the predominantly carbon dioxide atmosphere, these gases have a variety of applications ranging from their use as carrier and sweep gases for scientific instruments to buffer gas for human life support. Other applications include using compressed gases to deploy inflatable structures and drive pneumatic tools. Such tooling could be designed to be powered by several different modes of power input (electric, combustion, or gas pressure) using dual-use techniques.

The process for the production of Oxygen and propellant will make use of the chemical processes as shown in the figure. The production of Oxygen will either be by the electrolysis of Water, or the extraction from the Martian atmosphere using a Solid Oxide Electrolysis Cell, which involves the direct dissociation of carbon di oxide into Carbon Monoxide and Oxygen gas. The other, known as "Sabatier-Electrolysis", by combining hydrogen with Martian carbon dioxide in the presence of a nickel or ruthenium catalyst yields methane and water. The methane is stored and the water electrolyzed to produce hydrogen and oxygen. The oxygen is stored, and the hydrogen reacted with Martian carbon dioxide to produce more methane and water. There are also several other approaches and that could be used to create a form of a Martian produced energy economy. Each approach has it's own advantages and disadvantages. Currently the Solid Oxide Electrolysis Cell technology has been a proven process on Earth, and at this time it is currently going to be flown and tested on the Mars 2001 lander. Methane/Oxygen combustion has about 80 percent of hydrogen/oxygen's specific impulse, yet it is easier to store than hydrogen. Thus it is a more attractive choice for the Mars Ascent Vehicle fuel since the long time period that will be spent on the Martian surface. Another process to make Water and Carbon Monoxide is from a reverse water gas shift process as well as an output from the Solid Oxide Electrolysis. The Carbon Monoxide that is processed in these reactions could also be used as a fuel, but since it has low specific-impulse (30 percent of hydrogen/oxygen) and high burning temperatures, it would be better suited for other operations such as in ground and surface equipment than rocket engine propellant. A potential system to make use of all of these systems in a reaction would be as shown in the figure to the figure to the right. Oxygen would be produced by this system along with Carbon Monoxide. Other types of ISRU methods that could be used on the first human Mars missions would involve a more low tech approach. They would entail using the Martian soil for construction purposes. The manufacture of bricks or other building construction materials as described in an earlier section.

4.9 System and Equipment Reuse

Much of the systems and equipment mass that has been used on previous Manned Missions (Apollo), and much of the same mass represented in the designs of future projected human Mars missions all have used a similar design approach. This approach has the systems equipment only performing one function and then becoming inert mass with only a limited role as a structural function. For instance using an Apollo example, the LM had a decent engine, stage hardware, system equipment and structure. On this mission it was only used once for the primary function of landing safely on the Lunar surface. After it performed that function it became just inert structure, only providing a base or platform for the return launch and some of the surface operations. The current philosophy is to approach a human Mars mission in the same manner. The design philosophy has expanded somewhat by trying to make use of ISRU techniques for the production of propellants and consumables, but with equipment and mass that we bring to the surface we are still using the previous philosophy. This gives us a tremendous opportunity. With the ISRU philosophy we are trying to make some changes in the design to obtain mission advantages and benefits by using some very raw materials that we hope to find there. However, we seem to not consider much of the "resources" that we have brought to the surface with us and then are not using after their one time use. If we used the same type of creative design methods and philosophy as we do for ISRU and apply it to the design of our own equipment. Lets call this philosophy DRU (Design Resource Utilization). If we applied DRU, to the design of for instance the same type of example as the Apollo mission. The LM could possibly have used the same stage to land and then take off again after shedding some of the landing structure. This would have been a mass savings of a considerable margin, especially if you backtrack it all the way back the initial launch. There would have been some trade-offs that would have to be considered but it would have been an option. More options would have been with for instance with structural components on the LM. They were only used once for the landing and then had little secondary use. If DRU was applied you could come up with a number of design scenarios. Such as if you designed a structural member on the leg of the LM to then be used for a structural member on the lunar rover giving it a second life. This would have reduced the mass that you would have had to bring as primary equipment. Another example would be to use part of the foot pad, if it was designed with this in mind, to detach and then connect to a structural member from another part of the vehicle, and thus creating a shovel, one of the most basic of tools. This number of feasible dual-use designs is significant, and is a function of time of construction. The Apollo type mission is limited, because of the time scales involved. But for a human Mars mission, with a much longer duration time, the complexity of systems and number of potential dual-use systems would be enormous an only limited by our creativity. Such designs could also be applied to secondary or back up systems, increasing mission safety and reducing risk. The type and number of systems that could be devised, and the potential benefits in safety, mass savings, mission reliability, it would be a favorable to apply DRU techniques for the future planning of Mars and other human missions.

4.10 Astronaut Surface Mobility Systems

The objective of the Astronaut Surface Mobility System (ASMS) is to provide transportation of one or two space suited astronauts (one pilot and one passenger) within short range (5 km radius of operation) from base camp or main rover in a

short and effective manner. The system will be simple, lightweight, and quite frugal in power consumption. Because the astronauts will spend only a short period of time to conduct a mission within this short range, the system will be totally unpressurized. In this short mission duration, the EVA spacesuits will be expected to handle all of the hazards of the Mars environment.

The following are the candidates for ASMS:

1. Mars Stilts - Due to rough and dusty terrain on Mars, it may prove difficult for astronauts to walk around and conduct geological exploration in certain interesting regions. The lightweight, collapsible, telescopic stilts that operate using compressed gas will occupy only a very small footprint on the surface. They may be carried along with the geology backpack kit and deployed as needed. On difficult terrain, it may be possible for the astronaut to take much larger strides wearing this system (and hence cover larger exploration areas) and because astronauts will be at a higher elevation from the surface, they could also appreciate the surface topography from a better vantage point.
2. Jet Pack - The acceleration due to gravity on Mars's surface is about 40% of that on Earth. A jet pack with a simple, cold gas pressure vessel gas tank will provide a good lifting force on Mars. Astronauts will be able to travel relatively fast in an emergency situation. The gas tank can be re-pressurized at the base camp or main rover with the abundant CO_2 from Mars's atmosphere (See Fig. 2).

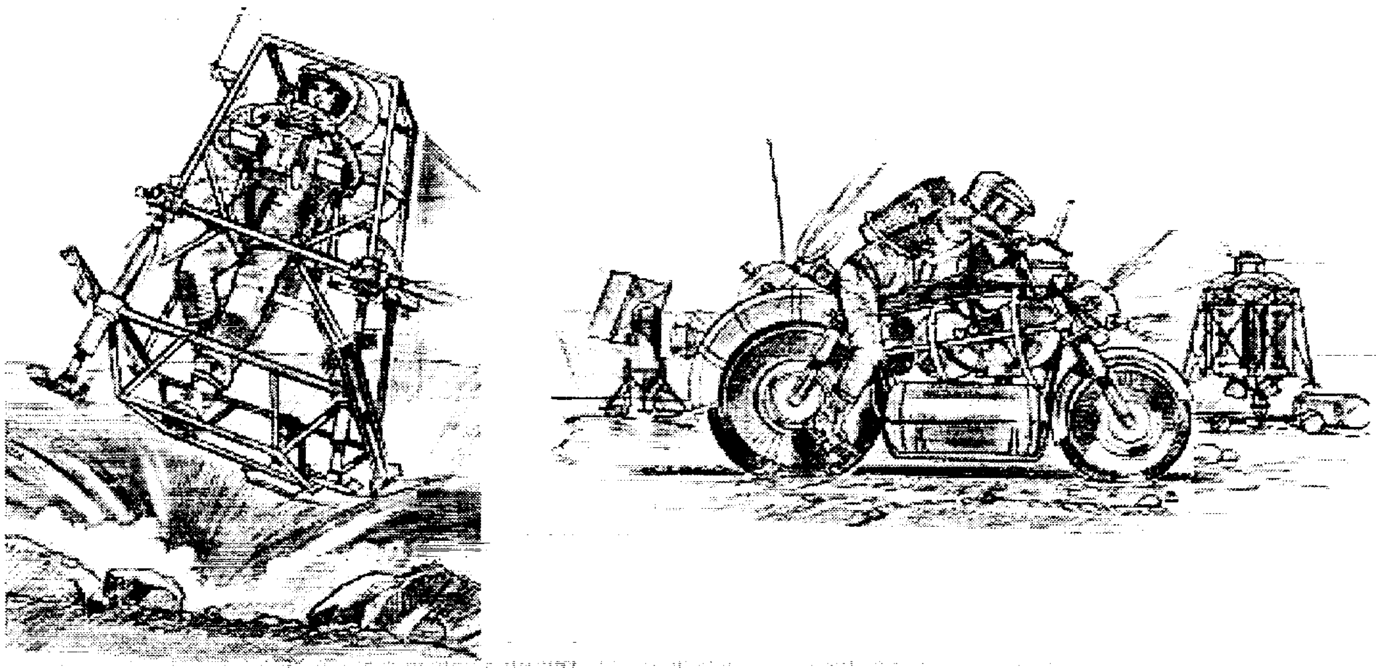

Figure 2: Surface Mobility Concepts

3. Hang Glider - Astronauts could ride the hang glider after they are lifted up by the jet pack. They will remain in the air for a while before approaching surface again and then they can re-fire the jet pack to lift themselves to the air again. However, since the atmospheric density is only one hundredth that of Earth's at sea level, very large wingspans/surface areas that need to be effectively deployed during this mission still need more investigation.
4. Blimp - An aerodynamically shaped blimp with large aspect ratio and a small engine could fly an astronaut pretty fast with little fuel. The blimp could be equipped with solar arrays to generate some power. The engine could be a regular gasoline/methane engine with Oxygen/CO2. The Methane could be from in situ resource utilization. The radiation heat generated during operation will keep the engine warm enough to operate and the water by product could be collected and used.
5. Hot Air Balloon - The hot air balloon could be easily deployed by heating up the gas in Mars atmosphere. It could attach solar arrays and turbines, which run by solar power to help generate lift and thrust.
6. Cable Cars - The temporary cable car could be use to transport astronauts just like at the ski resort. It could run by man power or solar power. This system could transport crew and cargo between chasms, gorges and large gullies that are hard to get across using conventional means.
7. Parachute - In Mars canyon surveying, particularly in the shallow sloped hills, Astronauts could use controlled parachutes/parafoils to speed down hill with their feet only 2-3 feet above the surface. The parachute will slow them down and they will be able to control their movement in the air.
8. Ski - In some parts of Mars canyon and surface, the soil may be mainly composed of sand and dust accumulation due to the vigorous aeolic activity presenting a very smooth topography akin to sand dunes and quick sand pits. Astronauts could use skis and gravity to move around in this environment.
9. Small Helicopter/Gyrocopter - A small, lightweight helicopter/gyrocopter with counter-rotating turbines to diminish residual angular momentum, small engine, and open structure could be an alternative to the blimp. These piloted vehicles could hover over areas of interest and fly in and out of tight spaces such as would be encountered in canyon and valley floor exploration.
10. Two Wheeled Rover - Due to the rough terrain on Mars, four wheeled vehicles would experience a lot of vibration because it would be difficult to maneuver through the rocky terrain. A two wheeled rover or motorcycle could easily maneuver itself over this kind of terrain and hold its track to the smoother surface (See Fig. 2).

5. The Very Long Range Traverse Vehicle

5.1 Landing Sites & Traverse

The landing site selected for this mission was chosen for its closeness to equator, and for its proximity to the geologically interesting regions of Valles Marineris and the Tharsis regions. As stated earlier, the main objective of the mission is to explore geological and geophysical features for determining suitable areas for future permanent bases. The landing site selected, Chasma Perrotin, which is located in a safe distance for eastward approaches from the tall volcanoes of the Tharsis region, fulfilled this main criteria. Additionally, all the Martian region types are easily within reach of this site. These regions are defined as Equatorial Plains (EP), Northern Plains (NP), Lava Flow (LF), Channel Terrain (CT).

For effectively exploring these vast regions, a very long range traverse vehicle (VLTV) concept (See Fig. 3) was developed. Comparable to NASA's earlier designs, the VLTV was designed to be able to operate independently of a base, to sustain a crew of three for durations of more than 150 days, over traverses of more than 5000 km. The VLTV, therefore, includes a well-equipped laboratory for detailed sample analyses, a separate workshop area for repairs and preparation of scientific payloads that is soft landed from an equatorial orbiting cargo bus, and a number of systems for exploring and sampling the immediate areas around the VLTV during its planned 3-5 day stops, including blimps, small unmanned rovers, and crawlers, and a drilling system (See Fig. 4).

The VLTV is a large (13m long with a diameter of 5m) and heavy vehicle (25-30 MT). Therefore, detailed visual and sounding information regarding the traverse path giving surface structure, and surface and subsurface conditions are required. A blimp sized at 10x20 m deployed and some <1m^3 stowed would be able to carry a payload of 10 kg, including an air pressure operated harpoon for sample collection, camera, sounder (1-1000 MHz), and an IR spectrometer. It would be used both for sample and data gathering, as well as for flying ahead of the VLTV for checking and assuring stability of the unimproved terrain along its planned track. For the detailed planning of the VLTV traverse, MGS high resolution images (down to 1.5m) in combination with sounding information from the Mars Express will be utilized. The VLTV will also carry Athena derived rovers, with instrumentation similar to that of the blimp. However, instead of the harpoon, the rover will have robotic arms for scooping and picking up surface material. Crawlers will also be included on the VLTV. Supported by an "active" umbilical, that serves as structural tether, power cord and communications link, these vehicles will be designed for controlled descent down steep slopes and cliffs of the canyon system for reaching otherwise hard to get to rock surfaces. Instruments will be similar to those on the rover.

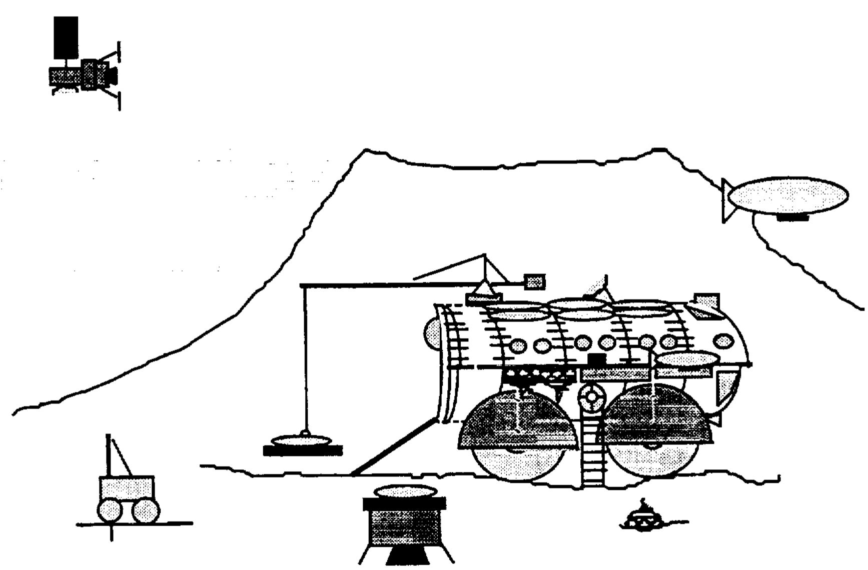

Figure 3: VLTV and Related Support Systems

The VLTV will carry basic geological field tools such as a rock drill for coring down to 15 cm, rock hammer, chisel, rake, shovel, scoops, tongs, long extension handle, sample scale, sample collection bags, and sealable containers, binocular microscope. These tools will be used by the VLTV crew during EVA.

An extensive suite of geophysical instruments will also be available to the VLTV crew. Some of these instruments are, electromagnetic sounder for detecting water and volatiles, and for measuring the variation in the dielectric constant and bulk densities of the soil. Frequencies 1 to 1000 MHz; Active seismic experiments for determining the structure of the upper Martian crust (2 km depth) through the use of geophones and possible detonation of explosives; Traverse gravimeter for determining gravity variation over the Martian surface; Electrical properties experiment for determining subsurface structure and water down to 1 - 2 km; Profiling magnetometer for measuring local variations in the Martian magnetic field.

The VLTV internal laboratory would be fitted to make it possible for the rover crew to analyze the collected samples for possibly changing or modifying a planned traverse without going back to a base. The laboratory should therefore include equipment such as mass Spectrometers for determining molecular level particles, optical, electron, and atomic force microscope for high magnification sample analysis, crystal growth experiments, X-ray spectrometer for accurate elemental analysis, neutron spectrometer, analysis and detection of organic materials, IR Laser Spectrometer for detecting trace gases indicative of biological activity, UV laser spectrometer for detecting Nicotinamide Adenine Dinucleotide (NADH) which plays a central role in the oxidative metabolism process, centrifuge spinning up to 30,000rpm for molecular level separation, possibly a vacuum chamber, and an oven with mass spectrometer.

In addition to these systems carried on the VLTV, a logistics lander in an equatorial orbit will place geophysical /meteorological monitoring packages at selected spots along the traverse. This package may include a magnetometer for measuring magnetic field strength and direction; passive seismometer for measuring seismic events; heat flow probes for measuring heat flow in the near Martian interior; meteorology sensors for measuring temperature, atmospheric pressure, wind velocity, humidity and atmospheric opacity and dust transport, mass 100 kg. The VLTV work shop area will used for testing and setting up these packages. The actual deployment of the instruments will require some EVA. It is assumed that the mass of this package will be around 100 kg. Micro and nano technologies are expected to reduce this mass estimate by 2006, which

Figure 4: View From the VLTV Cockpit

is the defined technology cut-off time for this mission. The package will be left by the VLTV crew and set to transmit data back to base camp, and possibly directly to Earth via an orbiter. The lander will also be able to carry fuel tanks, water, and oxygen making it possible to extend a traverse, if needed.

Based on preliminary findings, a geologically diverse traverse covering the areas of Tithonia Chasma (CT), Ius Chasma (CT), Noctis Fossae (CT), Noctis Labyrinthus (CT), Tharsis Montes (LF), Pavonis Mons (LF), Ascraeus Mons (LF), Fortuna Fosse (EP), Tharsis Tholus (LF), Echus Chasma (CT), and Hebos Mensa is proposed (See Fig 5). Later high resolution image, spectral, sounding, and altimeter data may change the route of this suggested traverse.

The VLTV will first head for the Tithonia Chasma/Ius Chasma region. In some places the Tithonia Chasma is about 6 km deep. Overlapping landslide lobes cover the canyon floor and scarps that bound a rift valley within the canyon. On the south canyon wall, distinct bright and dark horizontal stripes can be seen. New imagery indicate possible layering of nearly the

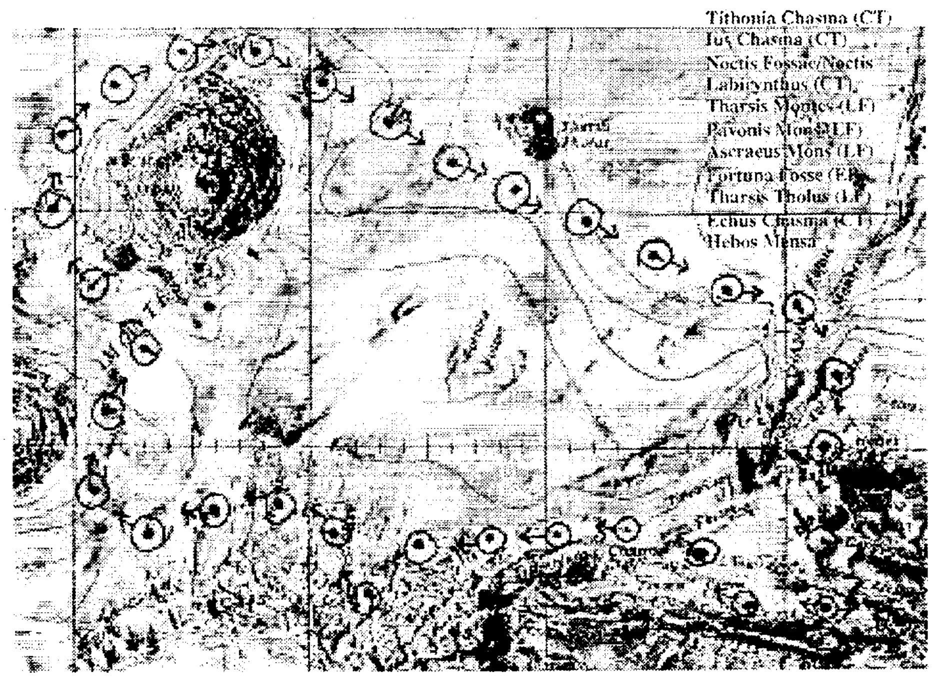

Figure 5: VLTV Traverse Route

entire depth of the canyon. This type of extensive layering has not been seen before in Valles Marineris. It calls into question common views about the upper crust of Mars, that there is a deep layer of rubble underlying most of the Martian surface, and points to possibly a much more complex early history for the planet. Landslide deposits and windblown drifts are other typical features of the Tithonia Chasma.

Next the VLTV moves into the areas of Noctis Labyrinthus and Noctis Fossae. This appears to be very difficult terrain, and more high resolution images are required before the final route can be set. The origin of the Valles Marineris by faulting is very apparent in these regions. Most canyons have a classic "graben" form. Other canyons are more irregular in form and have rough floor terrains, possibly the consequence of landsliding and the puzzling process of pit formation. In some areas it seems as if surface materials have sifted downward into large subsurface cavities. Noctis Labyrinthus is near the crest of a several thousand km updoming of the Martian crust.

After having explored canyons and cliffs, the VLTV heads towards the large shield volcanoes of Pavonis Mons and Ascraeus Mons in the Lava Flow region of the Tharsis Montes. The VLTV will first drive towards the east side of Pavonis Mons, then turn north west, a route between Pavonis Mons and Ascraeus Mons, then turn north and east to encircle Ascraeus Mons.

Pavonis Mons with a base diameter of some 500 km features a single circular caldera, possibly the result of the last eruption having eliminated all trace of earlier event collapses. On the west side of the caldera small white features, interpreted as dust clouds generated by strong downslope winds, can be seen. The Ascraeus Mons (base diameter 300 km) caldera, stretching to 11 km above the surrounding plains, shows a high number of event collapses.

The VLTV leaves the Tharsis Tholis region and heads over the Equatorial Plains in the Fortuna Fossae area to the south of Tharsis Tholis. This is a smaller volcano with a base diameter of 150 km, and height of 8 km. Approaching the volcano, the indented western flanks can be seen. Similar indentations appears on the east flank. These indentations might have been caused by the center of the volcano collapsing when the lava supply drained away.

From Tharsis Tholus, the traverse continues over equatorial plain type of terrain east towards Echus Chasma. The initial plan calls for a crossing of the chasma. However, more accurate altimeter data, and high resolution images are required before making that decision. The same type of data is required for selecting the route from the Echus Chasma, through Hebos Mensa back to base camp at Chasma Perrotin.

If the first traverse is completed successfully, and time permits, another traverse might also be attempted eastwards starting at day 350-400. Such a traverse will be discussed in a later paper.

5.2 Power and Drivetrain Systems for the VLTV

For any mobile mission architecture to be successful, propulsion systems for mobility must be able to provide sufficient power and reliability for an extended stay on the surface of Mars. The various surface mission elements require a significant amount of work in terms of horsepower (drilling for surface water, traction for a 30 - ton surface VLTV, digging of soil, etc.) and the question of power becomes more apparent. Nuclear power for the base camp architecture is a definite first choice but does not meet the needs for the mobile mission presented in this paper. The amount of shielding required and the risks of moving a reactor over unimproved terrain in a trailer quickly leads to the need for an alternate source of power. Since methane/oxygen ISRU production is already providing the propellant for the ERV proposed in the NASA reference mission, internal combustion of methane and oxygen in piston engines provides an attractive option for surface mobility. The greater power density of combustion engines provides for enhanced mobility, which enables a year-round, cost-effective mobile exploration program on the surface of Mars.

5.2.1 Current and Future Alternative Fuels Technology

Methane combustion and hydrogen fuels cells have not received much consideration in past space exploration mission for obvious reasons. But for a manned mission to Mars, these technologies become increasingly feasible as ISRU methane and

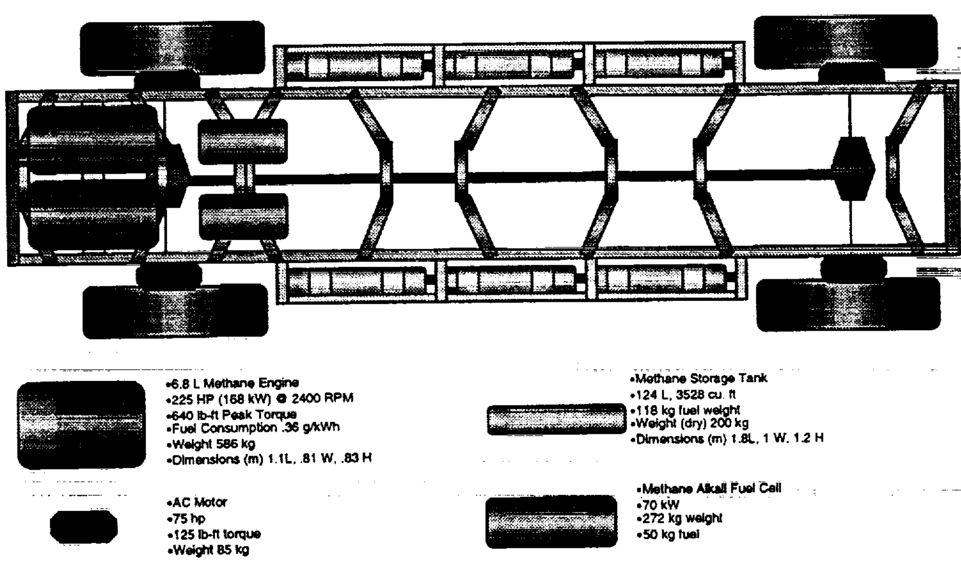

Figure 6: VLTV Drive Train and Power System

oxygen production on the surface becomes practical. On Earth, the amount of research dollars pouring into alternative fuels technology for transportation is being driven by the onslaught of deregulation in the natural gas and electric industry. State politicians wishing to provide alternative electricity suppliers with a fair chance in the new competitive US energy market while addressing the needs of environmental lobbyists are encouraging the advancement of green power generating technologies through legislation and government subsidies. For example, California State Senate Bill 90, enacted on October 12, 1997 placed the $540 million of the renewable energies program, the aim of which is to subsidize the cost of alternative fuel research. Advances in natural gas (methane) and hydrogen fuel cells for vehicle transportation are offering viable alternatives to enable surface mobility of crewed vehicles on Mars.

5.2.2 Methane Combustion on Mars

The current reference mission calls for the on-site production of methane and oxygen to fuel the Earth Return Vehicle. Six tons of liquid hydrogen brought from Earth will react with the carbon dioxide rich atmosphere of Mars to produce water and methane (methanation). Electrolysis will strip the hydrogen atoms from the water molecules to be re-used in the chemical process. The result in 10 months of production will be 108 tons of methane-oxygen propellant. Assuming that the ERV will require 96 tons for the trip back home, this leaves 12 tons for the VLTV [Exploration99, p. 47].

The VLTV's primary power source will be two 6.8L dedicated methane engines, one operational and the other standby (See Fig. 6). An internal combustion of 10% methane, 70% oxygen, and 20% carbon dioxide (replaces nitrogen as expander gas in Earth-based combustion engines) will provide 225 HP (168 kW) of power per unit. A turbocharger will be needed to compress the Martian atmosphere prior to induction into the engine cylinders given that the outside atmospheric pressure is only 1% that of Earth. Assuming the VLTV travels an average speed of 24 km/hr, this engine will consume approximately .37 kg of methane/oxygen fuel for every kilometer traveled. Given that one liter of methane/oxygen fuel weighs .95 kg and the engine is able to push the VLTV 2.55 km per liter of fuel consumed, a 4800 km trip will require approximately two tons of fuel. Liquid methane and oxygen produced by the ISRU plant can be transferred to storage tanks on-board the VLTV. The fuel storage capacity with 6 methane tanks and two oxygen tanks can give the VLTV a range (before refueling) of 1800 km. The following drawing shows the VLTV chassis with the engine and storage tank configuration.

5.2.3 Regenerative Hydrogen Fuel Cells and Electric Motors

Since the methane engine will provide electric power through a belt driven alternator when the VLTV is in transit, the need for power arises during the times when the VLTV is at rest. Also, a back up propulsion system will be needed in the event of engine failure. An alkali cell, fueled by 50 kg of liquid methane can provide 70 kW of power at 70% efficiency. Two of these power plants with a combined weight of 544 kg would provide sufficient electric generating capability to run independent AC motors on each of the wheels, enough power would remain for on-board VLTV systems. The alkali cell also produces water vapor as part of the reaction, which can either be condensed for crew consumption or stripped of hydrogen to be used in the cell (remaining oxygen to be pumped into the oxygen tanks).

One major drawback of this technology is that the potassium hydroxide electrolyte reacts with carbon dioxide to form potassium carbonate, which not only gradually degrades the electrolyte, but also precipitates out and clogs up the pores of the electrodes. Since carbon dioxide is abundant in Mars's atmosphere, this is a major problem, though one European company (Zevco) claims to have overcome this limitation, and is producing alkali fuel cells for vehicles on Earth [Fuel00]. If this technical problem has already been solved today, given the rapid advances in fuel cell technology, the alkali cell will be a practical power source in five years time.

5.2.4 Risks/Benefits of Methane Technology for Mars Exploration

Methane combustion engines and fuel cells provide an efficient, cost-effective source of power. The health concerns of the power systems are non-existent when compared to nuclear power as an alternative to powering a mobile system. Methane combustion engines have been tested and are currently in use in Arctic environments whose operating conditions differ from that of Mars only slightly. These engines can be tested in a vacuum chamber simulating the atmospheric pressure of Mars in order to verify the feasibility of this power source.

Any vehicle powered by internal combustion will contain numerous moving parts. Lubricants and other fluids will need to be developed that can be effective on the Mars surface environment. Spare parts such as hoses, filters, and drive belts will have to be brought along also. The power systems presented here are also entirely dependent on the success of ISRU methane/oxygen production, as is the successful return to Earth. Given 5 years of further research, the size, weight, and power output of methane combustion engines and fuel cells will be more than sufficient to meet the power needs of any mobile mission architecture for the surface exploration of Mars.

5.3 Science On Mars

Does Mars have the necessary geophysical, geological, and meteorological characteristics and resources to support humanity's ambition to settle there? Several scientific experiments may be done by the first crew to arrive and move about the Mars surface. The crew may be instrumental in setting up experiments and monitoring activity and evolving and changing out science payloads during the course of their 619 day stay. These include exobiology experiments to detect if life could have or continues to exist there, a variety of geological and geophysical experiments covering areas such as plate tectonics, volcanism and mineralogy, and magnetotelluric experiments to find out more about the dynamics of the planetary core and present activity.

Science payloads include equipment to detect and monitor water content in the atmosphere and on the surface as well as probes to look for it in subsurface strata (the deep drill is explained in a separate section); Biological and palentological payloads to explore for signs of life, past and present; Stations to continually monitor aeolic activity, dust transport mechanisms and seasonal changes over long durations, possibly in the order of 15-20 Martian years; Seismic stations operating over similar periods, enabling us to build up better models of tectonics or other local phenomena; and Solar studies and interaction with Mars environment including insulation, radiation and effects in the thin CO2 atmosphere. Potentially, one might also consider placing more exotic payloads such as gravity wave detectors, and very long base interferometry systems on the Martian surface.

Some of the unique physical features on Mars lend themselves to conducting unusual experiments. One such experiment is the Long Term Mars Atmospheric Profiler (MAP). The MAP could make it possible to profile the atmospheric environment of Mars over the long term, from surface level all the way to the top of a 28 km high volcano such as Olympus Mons, the tallest volcano in the solar system.

The MAP suggests a conceptual method for gathering data on atmospheric characteristics and environmental interactions through the use of a very long "active" tether. This tether has built in nanotechnology sensors that can detect and monitor variables including pressure, temperature, wind direction and speed, moisture, vibration, magnetic measurements, dust, and Solar radiation.

These nanotechnology sensors connected to a fiber optic data cable are integrated with the tether. The tether system may be deployed by a spacecraft lander on the top of a volcano, or by using a tow missile from the surface. Once deployed, the sensors will relay data to the lander/tow missile launcher. From there the data may be transmitted to a Mars orbiting satellite. Over a period of several years, it might be possible to build up a model of the Mars atmospheric profile.

5.4 Mars Drilling Operations

The Mars Deep Driller. (See Fig 7) is a drilling system that will initially reach depths of 100 meters using an air dust drilling method (schematic shown below). With this method, compressed air (CO_2) will travel down the shaft of the drill rod, blowing the dust and debris out of the wellbore. The compressed air also serves as a lubricant. Most types of Terrestrial deep drilling systems use water saturated mud as lubricant instead.

The deep drill system consists of compressor components and drill components. The proposed Compressor system has the following specifications: 2500 CFM (cubic feet per minute) average while filling, 2500 cm^3 tank capable of holding liquid CO_2, valves will release gaseous CO_2 at 100 CFM, heaters to keep tank/valves above -60° F, and a 30 HP (22.4 kW) pump. The drill system can be characterized as follows. It includes drill rods and bits, a air/dust deflection plumbing & wellbore chute, and a drill holding structure with momentum wheel & hammering system. Overall dimensions stowed are 3 meters long, 3 meters wide, and 3 meters high, and overall dimensions deployed are 3 meters long, 3 meters wide, and 7 meters high. Power Requirements: 50 kW minimum continuous, 22.4 kW while roving. Mass excluding drill rods for the system is 1300 kg, and the mass of the required liquid CO_2 150 kg. The system is capable of drilling to depths of 3 km.

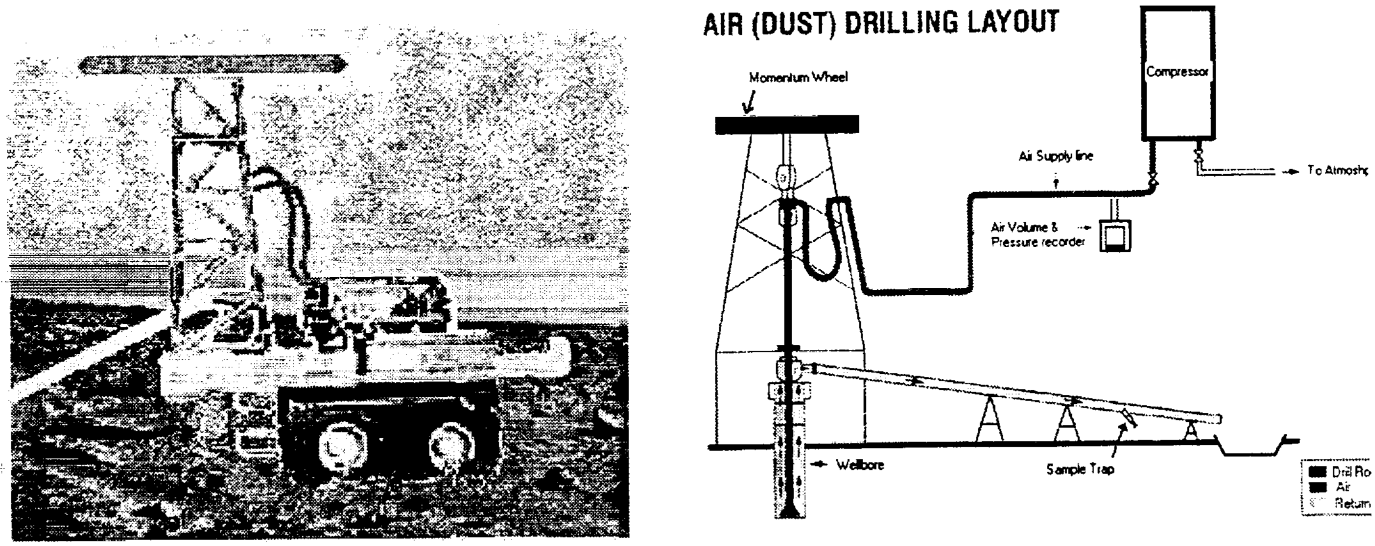

Figure 7: The Mars Deep Driller

Since the probing depth of the Deep Driller will only be 100 meters for the first roving mission, the best spots to drill will be at the low lying terrains for finding water or ice. The bottom of any river channel such as the bottom of Valles Marineris, or any dried up lake bed will be a prime spot. However, it will also be useful to drill to smaller depths of only 10 meters along the route to map out the characteristics of the soil over the entire general regions. Using software to monitor and record the composition of the dust in the various areas will aid further in determining the best possible sites to drill.

The Deep Driller is mounted on a 4-wheeled trailer for maximum portability. The drill can be used in conjunction with the VLTV, or at various sites around the habitat area.

The Deep Driller will have its trailer connected to the VLTV for towing along the route. When the VLTV has stopped at a drilling site, the drill can be automatically disconnected from the VLTV (except for the power and data cables) and guided to a drill spot at maximum 500 feet away. There will be a connecting power/data line to the drill to perform the initial set-up sequence.

There are two different scenarios in which the Deep Driller will operate. The first is at the base camp. Before and after roving missions, there will be ample time to perform drilling operations. The second is along with the VLTV while exploring the surface. When the VLTV begins its operation, the compressor will also be running to fill the CO_2 tank. Power is limited with the Driller; therefore it will drill in bursts. Once there is enough CO_2, a momentum wheel will power up to assist with the torque required for drilling. Once the drill rig is operating, EVA may be required to assist the robotics in adding the drill rods at which time core samples can also be gathered for analysis within the VLTV. All information will be recorded including the amount of ice and the mineral composition at the depth it was drilled. Drill rates of up to 12.5 meters per hour are achievable. This is much better than most conventional drill methods used currently on Earth.

Continuous air drilling can get to substantial depths in a short period of time. The limitation for the first mission is the weight and volume of the drill rods. The thin air on the surface also requires more time to fill up the compressor. Future missions will continue to bring additional drill rods increasing probe depths to 3 km, depending on subsurface layer content. For the first few missions, it will take about 30 days to pump enough CO_2 to drill for 12 hrs (up to 150m). The Driller will be working intermittently, taking up to five days to deplete the 30-day supply. The system is electrically driven which works well given the VLTV and base camp will be providing electricity. However it restricts the distance at which the Driller can be from its electrical power source. A total autonomous system would be the most desirable, however complexity becomes inevitable. EVA may therefore be required if component mechanisms becomes stuck or jammed.

5.5 Emergency/Rescue System

Crew Safety is of the highest priority in human space missions. Therefore, the Emergency Rescue System is an integral part of this mission plan and architecture. There are various alternatives to emergency or rescue operations, and each one of them are specific to possible failure modes and effects that we envision. Our Mission Plan focuses on the VLTV for terrain exploration. This VLTV would cover long distances and any one traverse would take 150 days. On such long duration traverse crew safety would be essential in the event that the VLTV systems fail. For such failures we have a rescue rover that would be positioned at Base Camp and would set on the rescue mission when summoned. This rover might take as long as 6 days to get to the VLTV and depending on the type of emergency this may not always be a viable solution. Another alternative is to have a "rocket hopper" which would be a sub orbital vehicle stationed at the base camp. The rocket would be loaded with the necessary supplies to sustain the crew and then launched to the location of the VLTV. Landing would take place by parachute and the ISRU station would supply the fuel. This would be a one-time use vehicle and would provide immediate assistance to the stranded VLTV crew until the rescue team gets there.

6. Further Studies

All of the sections above require more detailed investigation. Also, these concept architectures could be better coordinated with NASA studies in progress. In particular, experiments need to conducted in the following areas.

ISRU Structures: The USC Mars team would like to build and test ISRU structures for extraterrestrial infrastructure development and derive metrics as well as study human robot interaction and synergy and develop tools for improving the efficiency and rate of build up activity. Land and facilities are available to construct and test simulations of extraterrestrial habitats and infrastructure.

The VLTV: More research on similar vehicles and their capabilities on Earth need assessment. It may be possible to adapt existing systems and mechanisms for the Mars missions. Simulations need be conducted on Earth that prove the validity of using large vehicles in harsh conditions over unimproved terrain. It is possible to imagine that a full up simulation may be undertaken on the lunar surface in advance of the Mars Surface Expedition.

Interplanetary communications: The potential of using NASA reference mission elements for enhancing both local and interplanetary communications needs further study. For example, it may be possible to use the Transhab or the Earth Return Vehicle (ERV) communications platforms to augment the Mars orbital communications infrastructure. Also, during predicted communication blackout periods, it may be possible to use other spacecraft operating in the inner solar system as missions of opportunity at the time to relay data.

Multipurpose Systems and Equipment Reuse: This whole area needs detailed investigation so that uses can be built in very early in the design of systems rather than seize opportunities as and when they occur. It should be possible to coordinate this approach with the NASA reference mission activity and explore alternative uses for expendables.

7. Conclusion

A human mission to Mars in the new millennium could be the next major program for the space agencies of the world as the construction of the International Space Station is completed. International in scope, employing a highly synergetic human - robot complementary architecture, the first crew would set about exploring Mars to find out if life ever existed there and if humanity can settle there. An aggressive first mission architecture employing a base camp and long range traverse vehicle may provide all the data needed for future long term settlement missions. Such a mission would provide a coherent vision and nucleus for humanity's next step at the space frontier.

Such an interplanetary mission requires extended and complex preparation. End to end simulations of the mission on Earth, in orbit and the moon are proposed as a way to evolve a robust mission architecture. As humans are perceived to be the most fragile part of this system chain, particular attention needs to be paid to sustain their psyche and health and overall well being.

The human Mars missions offer the opportunity for mankind to continue exploration and ask questions that are open-ended in our quest to understand the world and universe that we live in. The technical issues are not the hardest ones to answer as they are usually black and white. It is the policy questions that will be the gray areas that we will have to provide the answers to. The development of our space faring capabilities are a natural extension of the seafaring development that took place prior to the great exploration period and will allow us to continue to grow in the future. We have to be willing to take risks in the quest for knowledge, information and opportunities for economic improvement.

8. Acknowledgments

This project was done in the AE599 Space Exploration Architectures Concepts Synthesis Studio, a 3 unit course offered in the new Masters Program in Astronautics, Department of Aerospace Engineering, University of Southern California, in the Spring semester of 1999. These conceptual ideas were created employing a unique architectural approach to complex engineering concepts generation. Thirteen graduate and post graduate participants from the industry as well as the schools of Architecture and Engineering were exposed to poignant scientific and engineering concepts in space exploration, and lectures by visiting experts. They were instructed in rapid concept generation and visualization methods. The resulting ideas were presented to a panel of experts, who reviewed their work and provided feedback. This paper is the combined result of that activity over a period of fifteen weeks which included forty five hours of meetings that were equally split among instruction in complex systems concepts generation, expert lectures in space exploration concepts and space systems, and debate and discussion. Thanks are due the members of the expert panel and visiting lecturers who included Norm Haynes, Carl Ruoff, Mark Adler, Sylvia Miller, Richard Zurek from the Mars Exploration Office at JPL, J.D. Burke from California Institute of Technology, John Connolly of NASA JSC, Harvey Willenberg from Boeing, Bob Walquist from TRW, Dan Sullivan from Hughes Space and Communications, Richard Kaplan, Jerry Mendel, Don Shemanksy, R.F. Brodsky, P. Lissaman, M. Gruntman, from USC Aerospace and Goetz Schierle and Marc Shiler from USC Architecture and L. Friedman from the Planetary Society, R. Ridenoure from SpaceDev Inc, Nader Khalili of CalEarth Institute, architect Van Der Schyff, John Spencer of Space Tourism Society, M. Thangavelu M.D. from WHO, David Schrunk M.D. from the Science of Laws Institute and Marshall Burns from Ennex Corp

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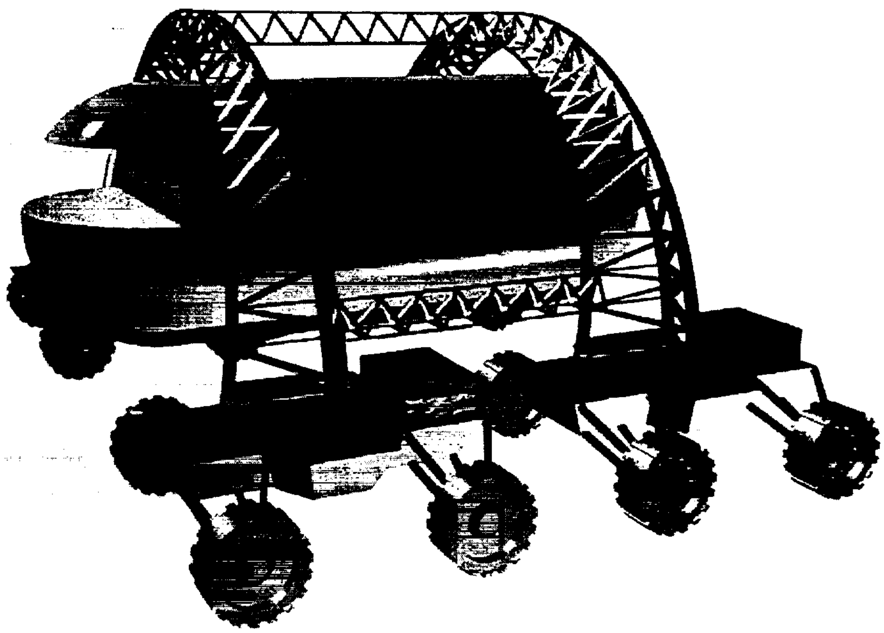
Conceptual Design of A
Mars Surface Transportation System (MSTS)



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1999

EXECUTIVE SUMMARY

We have proposed a design for a Mars Surface Transportation System. The design will support multi-range and multi-purpose scientific/exploratory activities for extended periods. Several assumptions were made before developing a design:

1. This system is to be deployed early in a series of piloted landings on the planet surface.
2. A Mars surface base has already been established.
3. A transport system to and from Mars already exists.
4. The capacity to transport this proposed system exists within the current transport design.
5. Facilities exist at this base for the supply of fuel and other consumables.
6. Medical facilities are a component of the main base.
7. The surface conditions of Mars are known and are accurate.

It was decided that the transportation system design should support a crew of two for up to four weeks away from the primary base. In order to support multiple mission requirements, the system is modular and multi-configurable. The main structural aspects of the design are:

1. An inflatable habitat module.
2. Independently powered and remotely controllable wheel trucks to allow multiple configurations and ease of system assembly.
3. Parabolic space trusses for high structural stability with low overall system mass.

In addition to these design aspects, new and existing concepts for control systems, power, radiation protection, and crew safety have been incorporated into the transportation system design.

1.0 INTRODUCTION

This report presents a proposed design for a Mars Surface Transportation system, abbreviated MSTS. This system is designed around the principle of maximum flexibility while maintaining minimal weight and minimal power requirements. It was the intent of the design team to explore new approaches to ground transportation rather than optimize previous approaches.

The system presented in this report consists of an inflatable habitat/laboratory module, multiple electrically powered trucks, and a supporting space truss. The proposed series of configurations of these components is by no means meant to be comprehensive. The primary function of this system is to allow the development of new configurations as needed for mission requirements not yet identified. Furthermore, additional components and newly developed components of this system can be transported to Mars as they are needed or become available, as the case may be.

Also contained within this report is a discussion of radiation and shielding considerations. Although a detailed discussion of this subject is beyond the scope of this report, the design team believes consideration of these issues is critical for a viable design proposal. Specifically, radiation exposure impacts on shielding requirements, and shielding requirements impact directly on mass, range, and duration away from a more adequately shielded home base.

Finally, this report concludes with a discussion of subject areas in which the design team was unable to complete a thorough evaluation of because of time limitations. Suggestions are also made for further research by the scientific community to clarify issues that prevent a definitive design at this time.

The authors of this proposal encourage feedback from interested parties that may lead to improvement of the design of this system.

2.0 BACKGROUND

2.1 DESIGN HERITAGE

During the late 1960s, the Boeing Company received a contract to build rovers for the Apollo 15, 16, and 17 missions. Engineers developed a simple lightweight rover that could be stowed on the exterior of the Lunar Excursion Module (LEM). These vehicles weighed 464 lbs. and could manage a payload of crew, portable life support systems, communications equipment, scientific equipment, photographic gear and lunar samples totaling as much as 1600 lbs. The lunar roving vehicle, or LRV, was powered by two 36-volt batteries driving four ¼ horsepower electric motors located at each wheel and had an operating range of 57 miles. However, the LRV was restricted to a radius of 6 miles from the LEM due to the limitations of the astronaut's portable life support systems. Figure 2.1 shows the LRV on the lunar surface.

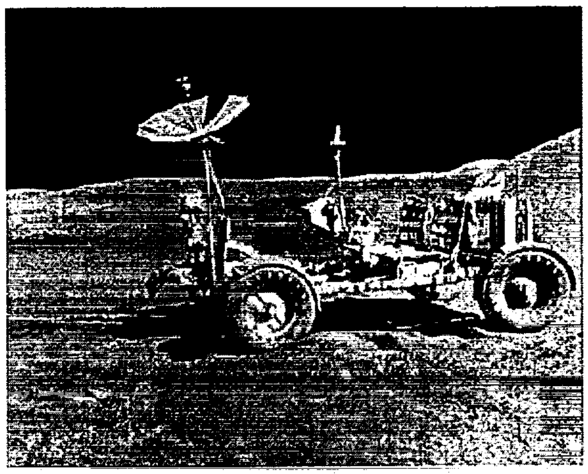

Fig. 2.1 Lunar Roving Vehicle (LRV)

On July 4, 1997 the Mars Pathfinder lander deployed the Sojourner rover which explored the Mars terrain in the vicinity of the lander. Sojourner was a semi-autonomous vehicle, which received command signals from Earth with two way transmission times on the order of 20 to 30 minutes. Much of the control philosophy the J.P.L. engineers incorporated into the rover design will be applied to the Mars Transportation System concept. Navigation systems included inertial measurement units, stereo cameras and other hardware that will be discussed in subsequent sections. Sojourner had a six-wheel rocker bogey suspension system with four corner steering and was built without conventional spring-dampers to increase the traction from the steel cleated wheels.

2.2 CURRENT AND PRIOR TECHNOLOGIES

Technologies that currently exist or that are under development have been considered for incorporation in the design of this transportation system. Emphasis will be placed on modular multi-function lightweight technologies.

2.3 EXTRATERRESTRIAL GROUND CRAFT

Several innovative technologies have been developed in the area of extraterrestrial mobile surface equipment. The Apollo moon rover, for example, provided surface transport for astronauts on the lunar surface. The lunar surface, however, is probably more uniform with fewer rocks than the Martian surface. In addition, robotic rovers have been developed to explore the Martian surface, Sojourner being the most recent.

2.4 INFLATABLE HABITAT TECHNOLOGIES

Various systems have been under development by NASA and the private sector to provide inflatable habitats for either Lunar or Martian missions. In addition, spacesuit manufacturers and NASA have considerable experience with the performance of various fabric composites in the space environment.

2.5 ANALOGUE ENVIRONMENTS ON EARTH

Currently, academic and national research organizations have development capabilities similar to those outlined in this proposal for the purpose of scientific study of the Arctic and Antarctic. Their experiences should be reviewed since many of the problems encountered may be similar in nature to those of the Martian surface. These may include food and water storage, mobility considerations, terrain difficulty, infrequent servicing capability, etc.

2.6 MINIATURIZED LABORATORY TECHNOLOGIES

Severe weight restrictions and the need for maximum analytic capacity will require the use of highly miniaturized laboratory equipment. Previous micro-laboratory technology from missions such as Viking and Pathfinder should be examined. Current capabilities of nano-electronics should also be reviewed for their potential role in this system.

2.7 POWER TECHNOLOGIES

Current technologies in low weight and non-combustion systems should be examined. In particular space station solutions should be reviewed, including both Mir and the International Space Station. Fuel cell, battery, solar, and nuclear systems should be examined.

2.8 DRIVE-TRAIN AND MECHANICAL TECHNOLOGIES

Many designs already exist for vehicles capable of negotiating irregular terrain. The commercial automotive industry, recreational vehicles (all-terrain, tracked, etc.) and the military have considerable experience in designing vehicles capable of travelling over sandy, rocky and uneven ground. Based on photographic images returned from probes such as Viking and Pathfinder suggest the Martian surface is irregular, sandy and strewn with boulders of varying size.

3.0 PROBLEM STATEMENT

3.1.1 GENERAL REQUIREMENT

Design a transport system for the Martian surface. This system should support multiple capabilities, including mission support, scientific exploration and analysis, and non-scientific mission objectives. Specific requirements and specifications are detailed below. This system will provide an increase in habitability, an increase in safety for the crew, and allow expanded surface exploration. Ideally, this system should entail simple and reliable deployment, a minimal of maintenance, a high degree of ongoing reliability, and maximum flexibility in purpose modification.

3.1.2 MISSION SUPPORT

This system must support the immediate and subsequent needs of an early expeditionary-piloted mission to the planet surface. Earliest of these needs will include the local transportation in the vicinity of the primary base. The system should include a pressurized mobile habitat that can serve multiple purposes. These purposes include, but are not limited to 1) expansion of existing habitat volume, 2) provision of a pressured emergency medical transport capable of retrieving ill or injured personnel from off-base locations, 3) provision of an on-site medical facility, 4) a back up habitat for personnel in the event of a failure or partial failure of the primary habitat/life support

3.1.3 SCIENTIFIC REQUIREMENTS

This mobile surface transport must provide the capability for a crew of two to travel extensively across the planet surface, providing life support for extended periods on the order of four weeks at a time, and allowing advanced on-site analysis. These analysis will include 1) biological analysis over extended and varied terrain, allowing for sample collection and analysis, 2) geological analysis, including the ability to collect samples over the large expanses required to obtain accurate geological mapping of the surface. This system should have the capacity to eventually include a drilling apparatus to satisfy core sampling and seismic/electromagnetic exploration needs. Finally, in support of atmospheric and geophysics sciences, this system should provide a means to transport, deploy, and service monitoring equipment to distant sites.

3.1.4 NON-SCIENTIFIC OBJECTIVES

A third mission objective that this system should support is the exploration of the Martian surface for exploitable resources. The discovery of materials on the planet surface that could be used for construction or support of mission elements would significantly reduce the cost of subsequent missions. Such materials could include water, gases, fuels, and construction materials analogous to concrete. In addition, discovery of any resources that may be exploitable for profit would also serve to underwrite the high cost of future missions.

3.2 ASSUMPTIONS

This design proposal makes the following assumptions:

1. This system is to be deployed early in a series of piloted landings on the planet surface.
2. A Mars surface base has already been established.
3. A transport system to and from Mars already exists.
4. The capacity to transport this proposed system exists within the current transport design.
5. Facilities exist at this base for the supply of fuel and other consumables.
6. Medical facilities are a component of the main base.
7. The surface conditions of Mars are known and are accurate.

3.3 CONSTRAINTS

The proposed design must satisfy the following constraints:

1. Transport weight must be kept to a minimum of 5000kg per transport.
2. The system must be multifunctional.
3. There must be a high degree of interchangeability between components.
4. The system must support a crew of two for up to four weeks at a time away from the primary base.
5. The system must provide the capability to cover a 500km radius of the planet surface.
6. The system must minimize exposure of the astronauts to the external surface environment.
7. Internal configuration must allow variable configuration to support specific tasks.

4.0 PRELIMINARY CONSIDERATIONS OF MARS TRANSPORTATION SYSTEM AND REJECTED DESIGNS

Research in the early stages of the project resulted in a modular concept for a surface transport vehicle. Precise systems were yet undetermined, however, life support, propulsion, and science were relegated into independent interconnected modules. Theoretically, mission parameters would dictate the need for a science module or a life-support habitat for long duration objectives, which could be removed from the system without compromising the operation of the remaining components. The modules linked together as a *train* would be pulled by a manned or unmanned pressurized rover. Figure 4.1 is a sketch of the initial design model.

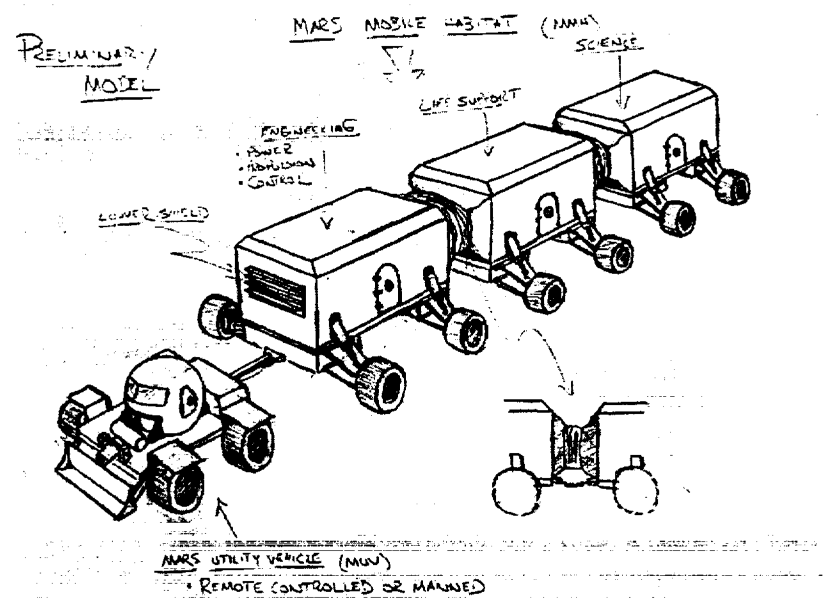

Fig. 4.1 Modular surface transport vehicle concept.

The large hard-shell exteriors of the modules would make Earth-to-Mars transport of the necessary materials for construction impractical. Also, the system does not allow for flexibility of individual components due to their rigid design.

A minimum weight large volume module could satisfy the design constraint and an inflatable habitat proved to be the ideal solution. Relatively little assembly would be required to bring the surface transport system into operation.

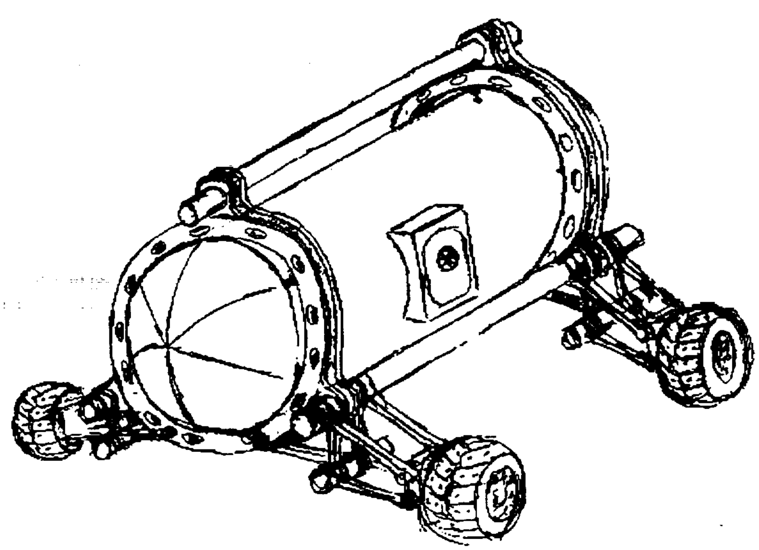

Fig. 4.2 Transport using an inflatable compartment.

Figure 4.2 is a sketch of the redesigned module using the inflatable compartment concept. The design incorporated a conventional independent suspension system and could be self-propelled if necessary. The inflatable body represented a significant reduction in weight and volume for transport to mars.

The final design chosen by the team involved the addition of a supporting truss system to suspend the inflatable module. This design had further advantages over the design illustrated in figure 4.2. The truss system allowed the inflatable module to be lowered to the planet surface, and also allowed for the option to carry other payloads. In addition, the powered wheel assemblies could be detached and used for other purposes. The specifics of this design will be discussed in the next section, and the various advantages will be explored.

5.0 PROPOSED MARS TRANSPORTATION SYSTEM DESIGN

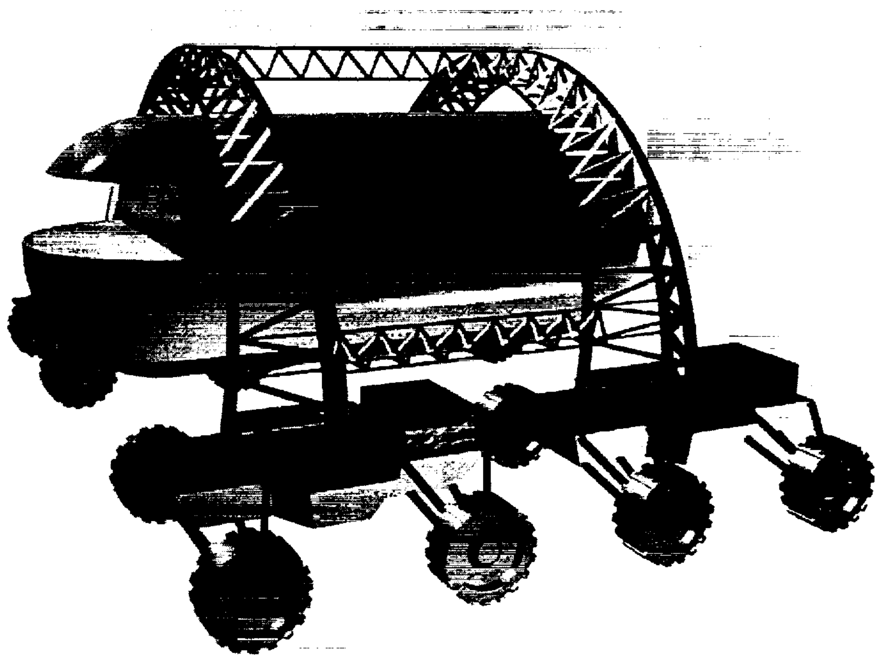

Fig. 5.1. Proposed Mars Surface Transportation System assembly.

5.1.1 INFLATABLE MODULE

5.1.1.1 DESIGN

The habitable module will be an inflatable structure made of kevlar reinforced materials. This concept will utilize technology already under development for the Transhab module currently planned for a Mars mission and possible inclusion on the International Space Station. Extensive analysis of inflatable habitat structures has been performed by the Center for Engineering Infrastructure and Sciences in Space at Colorado State University (see references). The advantages of an inflatable module include low-mass and low storage volume for transport to the Martian surface.

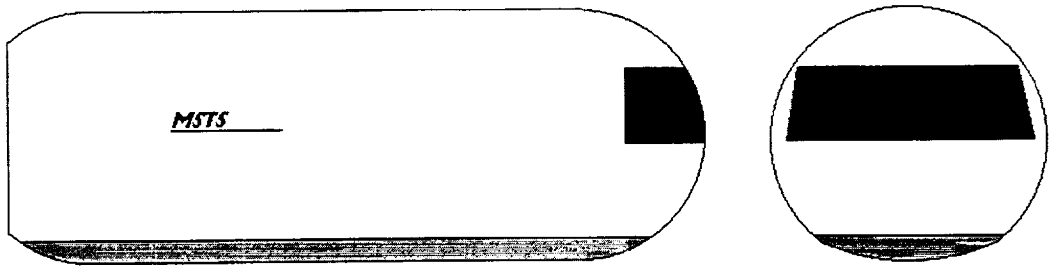

Fig. 5.2. Inflatable laboratory module (IHM), external view

5.1.1.2 INTERNAL CONFIGURATION

The inflatable habitat module (IHM) is designed to allow maximum flexibility in mission support. The design allows for variable internal configurations based on rack-mounted interchangeable equipment modules. The crew can configure the IHM while the module is attached to the main landing craft, allowing the opportunity to test and modify the configuration as needed.

Sleeping chambers will likely be placed above the equipment module to minimize exposure to spallation radiation (Spallation of cosmic radiation occurs when incident particles contact shielding materials and produce a cascade of secondary particles. This problem is discussed in detail later in this report.) In order to shield against periodic solar particle radiation, safe areas could be located under the equipment racks where radiation protection is maximal.

Oxygen, food and water storage will be internal to allow for access and maintenance. Further stores could be configured externally on the system as need. Waste storage will also be required in the module. All of these compounds could be utilized to increase shielding when required. There is no capacity to recycle fluids or waste in this module as recycling capabilities represent excessive equipment and energy requirements for this system. Waste and scrubbed CO2 will be stored for treatment or disposal at the home base.

The underside of this module is envisioned to be constructed of a hard metal or composite material derived from the exterior surface of the landing craft. This concept is expanded upon below. The floor of the module will likely have wiring and gas lines worked into the sub-floor, simplifying connectivity of equipment within the module.

Egress from and entry to this module will be facilitated by a folding ramp on the posterior aspect of the IHM at the airlock entry port. This system will be similar to that found on smaller commercial aircraft.

Power requirements will be supplied by fuel cells and batteries contained in the module. Fuel cells are discussed in detail later in this report.

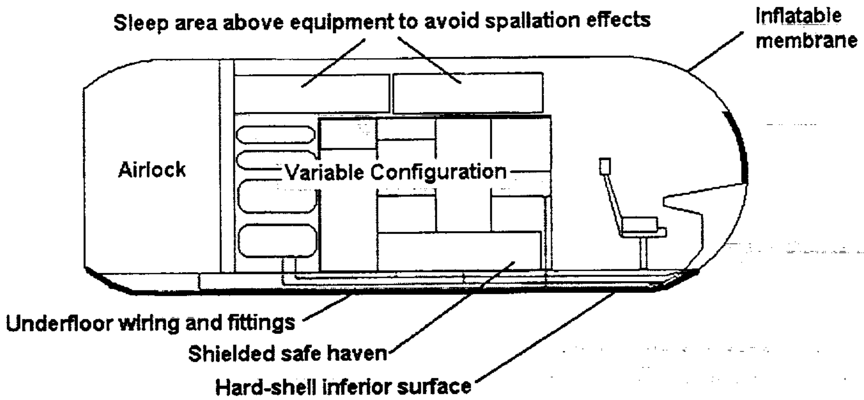

Fig. 5.3. IHM internal configuration

5.1.1.3 AIRLOCK

An airlock is a mandatory feature of the IHM. As an onsite laboratory, there will be a requirement for frequent access to and from the Martian surface. The concept of a suitport has been around for sometime. Although it has not been used in spacecraft prior to the present time, it is the most efficient means yet devised to conserve precious atmospheric gases when astronauts exit and enter a space vehicle. The concept involves a suit mounted on the exterior of a craft. The astronaut enters the posterior of the suit through a portal, which is then closed behind him/her. The suit then separates from the wall of the craft and minimal atmosphere is lost. In addition, the introduction of contaminants would be minimized by such a system. This concept is illustrated in figure 5.3. This concept seems highly appropriate for the IHM as frequent excursions to the planet surface are expected to be the norm.

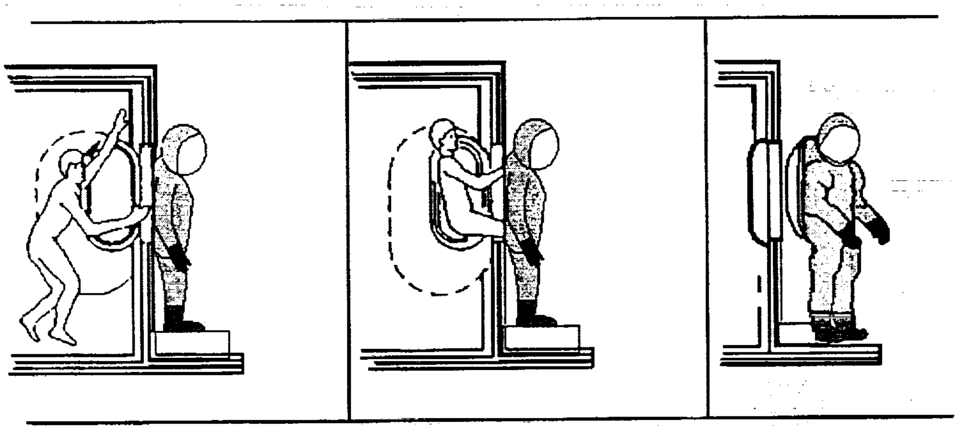

Fig. 5.4. Suitport concept

Combining the suitport concept with a standard airlock creates a system that is both highly safe and highly efficient. For routine use, the airlock would remain unpressurized throughout the egress/ingress process. The airlock, in general, will serve as a failsafe against malfunctions of either the port seal or the suit itself. In case of a decompression or catastrophic failure of the suit/suitport, the airlock could be rapidly pressurized with no external loss of atmosphere. In addition, the airlock provides a storage facility for the suits that is protected from the damaging effects of the external environment. Finally, the airlock could be pressurized for routine or unexpected maintenance of the suits and suitport as required.

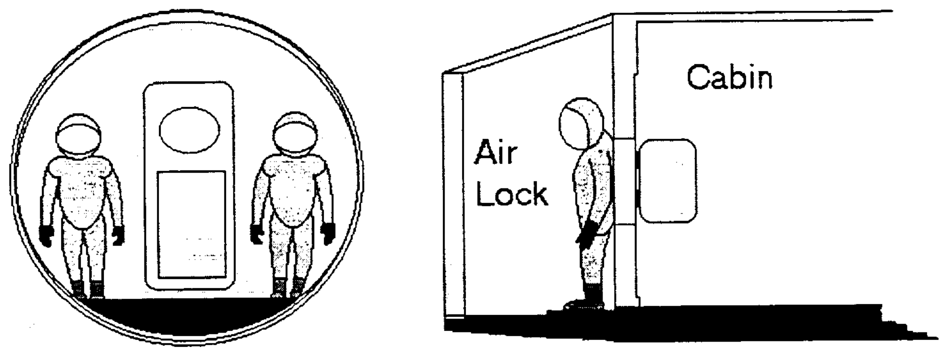

Fig. 5.5. Airlock and suitport combination.

5.1.1.4 IHM DEPLOYMENT

The IHM will be deployed remotely from the exterior surface of the landing module, with a panel of the exterior surface of the landing craft forming the underside of the IHM. In its uninflated state, the IHM will occupy minimal volume on the trip from the Earth to the planet surface. Upon landing, the IHM will fold down from the side of the landing craft, probably onto support legs that deploy on the underside of the IHM. The IHM will be contiguous with the internal cabin environment via the airlock, and inflation will occur through the process of pressurizing the IHM to normal atmospheric pressure.

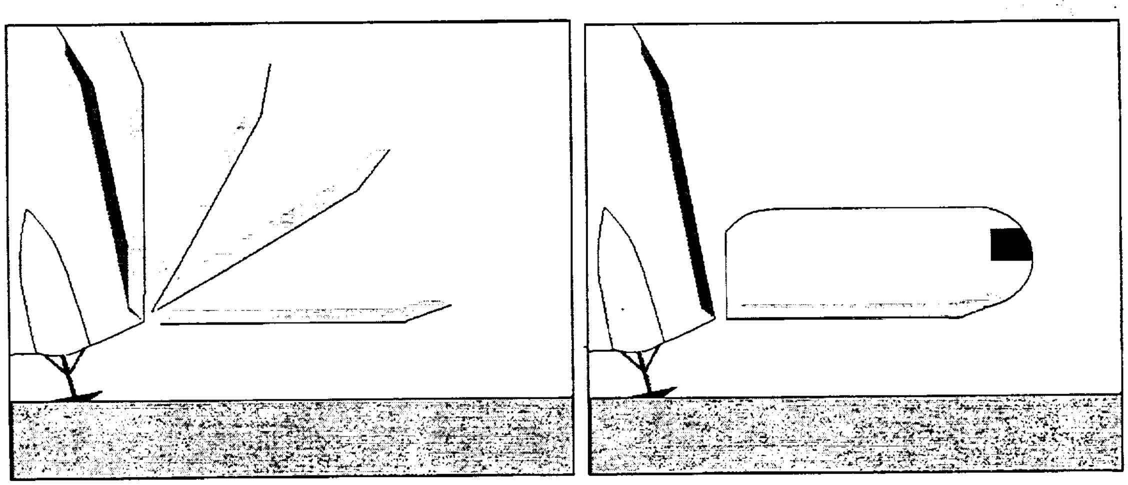

Figure 5.6. Schematic illustrating deployment of ILM from side of landing craft.

5.1.1.5 CONFIGURATIONS

The IHM may remain in this deployed position, connected to the main craft indefinitely. In this configuration, the IHM may serve as extended living or laboratory space to supplement the capacity of the primary base. When the remaining components of the Mars Transportation System (MSTS) are functional, the IHM can serve as the body of a mobile craft that can be ferried around the planet surface as needed. If the mission plan requires a prolonged manned or unmanned facility to be placed for extended periods at a remote location, the IHM can be lowered onto the planet surface and left in place indefinitely.

5.2 WHEEL TRUCKS

The independently powered wheel truck is illustrated below in figure 5.6. Both the configuration with an optional driver seat (rover configuration) and the basic configuration are illustrated.

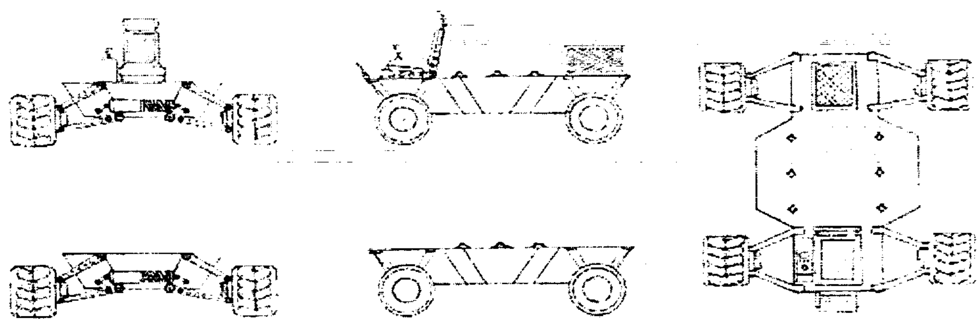

Figure 5.7. Wheel truck with (upper) and without (lower) optional rover seat.

5.2.1 FUNCTION

Two four-wheeled trucks on each hoop frame will provide the Mars Transportation System with its mobility. The wheel trucks are independently functional and individually powered. They will attach at the bases of each hoop frame. By adding the capability of remote control to each of the trucks, the entire system becomes capable of assuming multiple configurations. A winch will be mounted on each wheel truck.

5.2.2 POWER

Each truck will contain a hydrogen/oxygen regenerative fuel cell (RFC). The power systems for the wheel trucks are discussed in Section 6.0.

5.2.3 SUSPENSION

The suspension system for the wheel trucks has not yet been chosen. Suspension systems of existing rovers and all-terrain vehicles will be looked at so that an acceptable system can be developed for our transportation system.

5.2.4 USES

The advantage of having removable, individually powered, independently controllable wheel trucks is that they can serve many purposes. Just a few of the possible functions of the wheel trucks beyond habitat mobility are outlined here.

1. <u>Transportation System Assembly</u> – The wheel trucks can be pre-programmed to deploy from the landing craft, then begin auto assembly of the transportation system before astronauts take over the operation.

2. <u>Equipment Transport for Scientific Tasks</u> – During stationary periods of scientific research, the wheel trucks can be disconnected and used to move research equipment and supplies for the scientists. The trucks would be operated remotely or by an astronaut sitting or standing on the wheel truck.

3. <u>Mobile Power Generators</u> – A single truck could be used as a mobile power generator for the science equipment during research phases. The equipment would simply be connected to the truck through an adapter on the truck.

4. Mobile Crane – When combined with the parabolic space trusses, with the habitat removed, the trucks will form a mobile crane for lifting large objects. This function would be useful at the mission site when structures are being built and moved. The crane function will also be very important to the initial assembly of the transportation system, and to reconfigurations.

5. Retrieving Payloads – Payloads that land long and out of range from the main base could be retrieved by having several trucks working in unison to lift and move the payload across the planet surface.

5.3 PARABOLIC SPACE TRUSS

The ultimate purpose of the space truss is not to enable a specific configuration; but to deliver a discrete system of tension and compression elements to serve as a set of basic structural building components (building blocks). These elements may be configured to conform to any arbitrary mission requirements within the connectivity restrictions of the nodes of the individual elements.

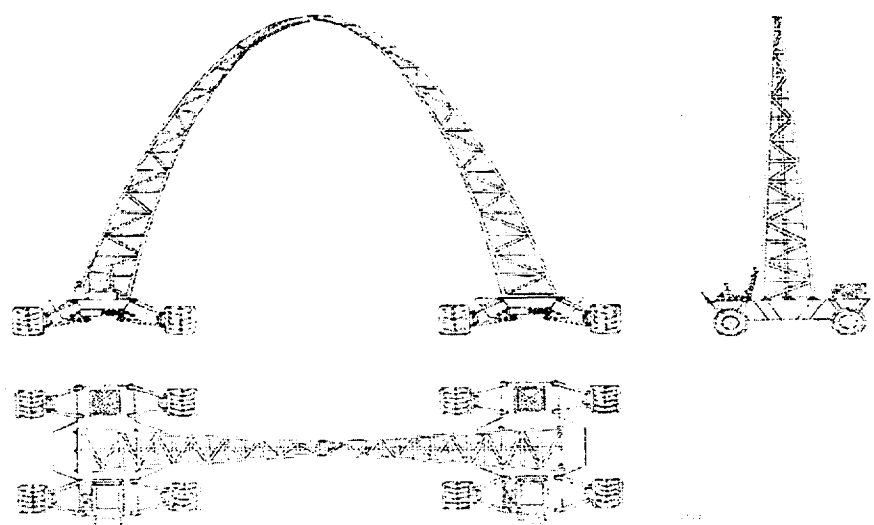

Fig. 5.8. Parabolic space truss, three views.

5.3.1 TRUSS MEMBERS

The exact number of truss cascade and cross members has not been determined at the time of writing of this report. An absolute height for the space truss in full (long-range) assembly must be agreed upon first so that the geometry of the parabola can be determined. The full assembly configuration has a center point at the intersection of the base line (y-axis) parallel to the motor truck wheel axis, and the symmetric centerline (z-axis). This will separate the parabola into two distinct half sections from a front view.

The cross members will vary in length from the vertex to the base of the space parabola. All members will be circular, thin walled tubes with a common radius. The material of each member will be homogeneous through out the global system and must satisfy the environmental conditions of Mars, primarily a broad range of thermal loading, as well as meet the constraint for launch weight from Earth.

5.3.2 CONNECTIVITY

The connection couplings at each node of each member are critical to the fulfillment of the design goal for versatility. The arced members must have connection joints that are strategically positioned along the arc length with connection lines intersecting the radial origin of the member's geometric arc, originating from each connection joint.

In order to promote the "Lego" design aspect, the coupling mechanisms of the connection joints must be designed as simply as possible so that a minimum number tools is required. Each linear truss member will most likely have a standard threading at each end. Concentric coupling fittings should be available to allow two or more linear members to be connected end-to-end. T-couplings could facilitate orthogonal connection of linear elements. The threaded coupling fittings will allow the individual members to be assembled as a frame structure. A separate connection mechanism must be considered for the full assembly so that connections of linear members to the arced members can achieve a *pinned* connection characteristic versus a *rigid* characteristic obtained by the coupling fittings.

5.3.3 DELIVERY PACKAGE

The tension members, defining the (triangular) outer skeletal system, are comprised of three series of arced members that are connected in cascade. The arced members will all have the same distinct contour corresponding to the camber of the aero-brake shield used for entry to the Martian atmosphere. The linear cross members may be packaged in the cylindrical wall of the delivery lander, aligned parallel to the longitudinal axis of the cylindrical shell.

5.3.4 ASSEMBLY

The assembly of the truss components into a usable structure is a manual task to be accomplished by the available Mars surface crew. Assembly instructions for the three primary configurations described below will be provided upon the completion of the final structural design and static analysis of each structural form corresponding to the long-range, intermediate-range, and short-range configurations respectively. All of the tools required for assembly have not yet determined however, an adjustable torque-wrench with a contoured rack of contact teeth will definitely be required.

5.3.5 FUNCTIONS

The truss members will possess sufficient strength and connectivity to support a diverse number of demands that may arise. The three primary configurations for vehicle support are for long-range, intermediate range, and short-range assemblies.

5.3.5.1 LONG-RANGE CONFIGURATION

The long-range configuration will embody the full assembly of the space truss.
The purpose of the truss elements in the full assembly configuration is to provide a parabolic space truss structure capable of sustaining large suspension loads. The long-range configuration consists of two or three fully assembled parabolic space truss structures connected in series by longitudinal connection beams. The longitudinal spacing between the individual truss structures is undetermined at this time because it is a function of the IHM module length. The parabolic structures will be able to suspend inflatable IHM modules where the longitudinal axes of the IHM modules are coincident and concentric with the longitudinal axis of the three connected parabolic trusses. The long-range structural configuration will also be used to support the loading of ballast and water contained in cylindrical shells on the top of the structure as a radiation shield.

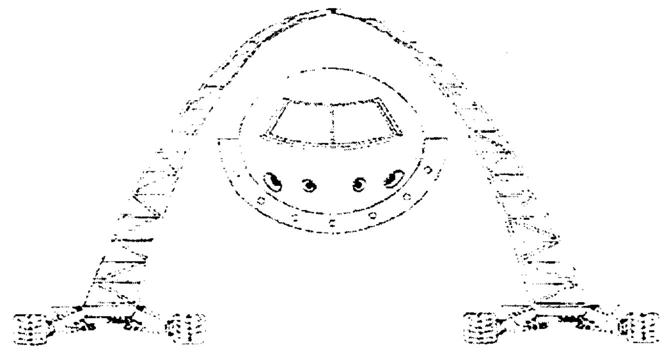

Figure 5.9. Suspended configuration.

5.3.5.2 INTERMEDIATE RANGE CONFIGURATION

This configuration consists of a partial system assembly of selected members to support full weight of a single IHM module and contents. The partial truss assembly will serve as an under truss to rest and secure the IHM between four motor trucks configured in parallel.

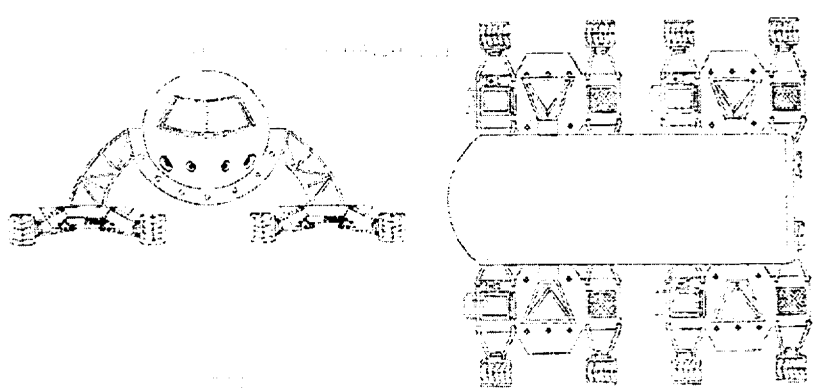

Figure 5.10. Cradled configuration.

5.3.5.3 SHORT-RANGE CONFIGURATION

The system allows for individual operation of the motor truck elements. Pictured below are three views of the motor truck configured with an external seat. This setup allows for an astronaut to use the wheel truck as unpressurized rover for short-range operations.

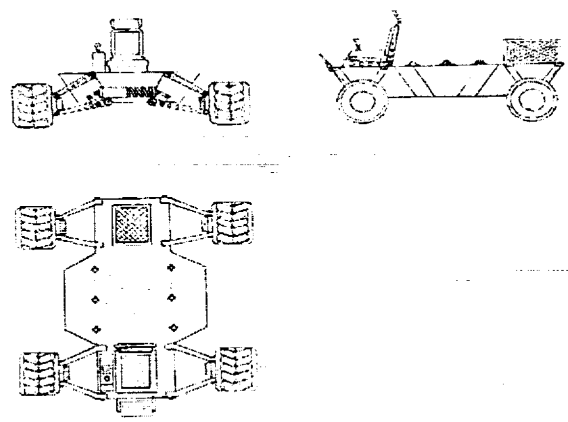

Figure 5.11. Wheel truck configured for short-range use

6.0 POWER

6.1 VEHICLE POWER

Each wheel truck will contain a hydrogen/oxygen regenerative fuel cell (RFC). Use of RFC's will allow the vehicle to travel further distances or longer periods without the need to return to the mission base for refueling. A beneficial byproduct of the energy generation process is potable water.

The RFC system components are the fuel cell stack, electrolyzer, reactants, tankage for the O_2, H_2, and H_2O, radiator, and power management and distribution (PMAD). A gallium arsenide on germanium tracking array would also be required to power the electrolyzer . Figure 6.1 shows a block diagram of a regenerative fuel cell.

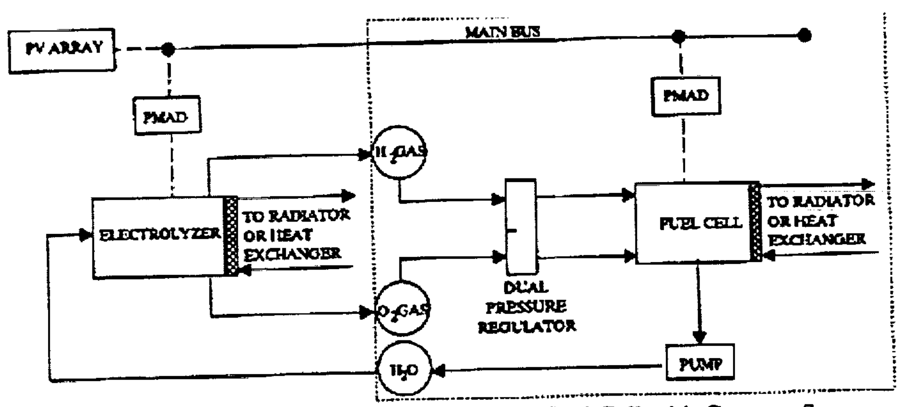

Fig. 6.1 Block Diagram of Regenerative Fuel Cell with Gaseous Storage

6.1.1 REGENERATIVE FUEL CELL OPTIONS

Four RFC options will be considered for vehicle power:

1. Low pressure gas storage.
2. High pressure gas storage.
3. Low pressure gas storage with photovoltaic arrays.
4. High pressure gas storage with photovoltaic arrays.

Each option has advantages and disadvantages. These must be weighed before selecting a power system. High-pressure gas storage RFC's have much smaller tanks than low pressure RFC's, but there is a greater safety concern. Although photovoltaic arrays will add mass to the transportation system, the advantage of being able to use solar power to convert the H_2O byproduct back into H_2 and O_2 make the arrays highly desireable. Options 3 or 4 will most likely end up being the power source for the vehicle. Safety will be the deciding factor in determining which type of storage to use.

6.2 HABITAT POWER

The inflatable habitat module will be powered by its own H_2/O_2 regenerative fuel cells. The placement of the RFC's within the module will be determined once the payload and components of the habitat are known. The additional power produced by the wheel trucks can be redirected to the habitat when the trucks are not in motion. In the same fashion, power produced by the habitat's power systems can be used to supplement the wheel truck power during travel periods.

During the stationary periods, photovoltaic arrays will be deployed so that fuel cells that are not being used can be recharged. The arrays will also be a backup power source in the event of fuel cell failures.

7.0 CONTROLS

7.1 AUTONOMOUS OPERATION

A flexible control package will be incorporated into the truck system design. The system will allow for semi-autonomous and autonomous operation depending on the truck configuration that is required. In autonomous operation the truck will navigate without user interface to a desired position with the aid of an installed digital gyro-compass, inertial measurement units (IMU), and the Mars equivalent of a Global Positioning System (GPS). These units will provide the position quickly and accurately with respect to a Mars centered, Mars fixed (MCF) coordinate system.

GPS requires a constellation of closely monitored satellites in orbit around Mars. A minimum of four satellites is necessary to provide a position solution, longitude and latitude, of the truck. In the event the required numbers of satellites are not within the truck's GPS receiver line-of-sight, the system will revert to inertial navigation mode using the IMU. The IMU will initialize its position based on the last GPS navigation message and operate using the IMU's rate gyros and accelerometers. The gyros are installed in three mutual perpendicular directions to measure the attitude of the vehicle. The accelerometers are installed in a similar manner. They will provide information about the vehicle's acceleration about three coordinate axes.

The system utilizes a close-loop feedback control structure to provide accurate responses to desired outputs. A simplified control structure is described in Fig. 7.1. A program schedule of the truck's travel itinerary is fed into the vehicle's navigation computer. These signals are amplified and sent to the truck's actuators such as the drive motors and steering mechanisms to provide the desired mobility to satisfy the vehicle's next position. Continuous GPS or IMU updates will provide the truck's position in real time. The error between the desired and actual positions will be used as control inputs for the navigation computer to process the necessary course corrections.

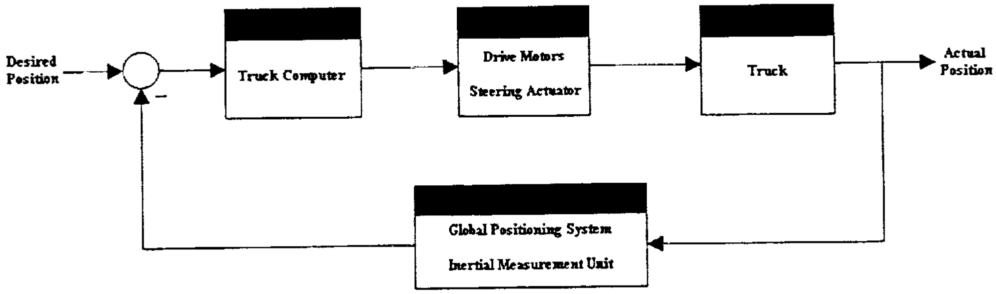

Fig. 7.1 Control architecture for a single truck unit.

7.2 SEMI-AUTONOMOUS OPERATION

The truck's control system allows for user interface via remote control or cable fed inputs to provide a desired response of the vehicle. A typical example of remote operation would be day-to-day work duties such as moving surface components on board the truck from one site to another. The feedback in the control algorithm of the vehicle would simply be "dead reckoning" by the astronaut as to the final position of the truck.

Multiple truck units require a central computer to manage the overall output of the system. Such a configuration is illustrated in Fig. 7.2 where four trucks are used as drive units for a mobile habitat. A pre-programmed travel plan can be installed in the habitat's navigation computer where the system of trucks can deliver the crew autonomously (autopilot) to a desired position. In the event of unforeseen difficulties the crew can remove computer control and operate the system manually using conventional control devices such as wheel and throttle mechanisms. The manual control system is strictly speaking "fly-by-wire" where mechanical inputs are converted into electrical signals for computer processing and routed to the necessary truck unit. Hydraulic actuators incorporated in the mechanical control devices will provide simulated terrain feedback for the pilots.

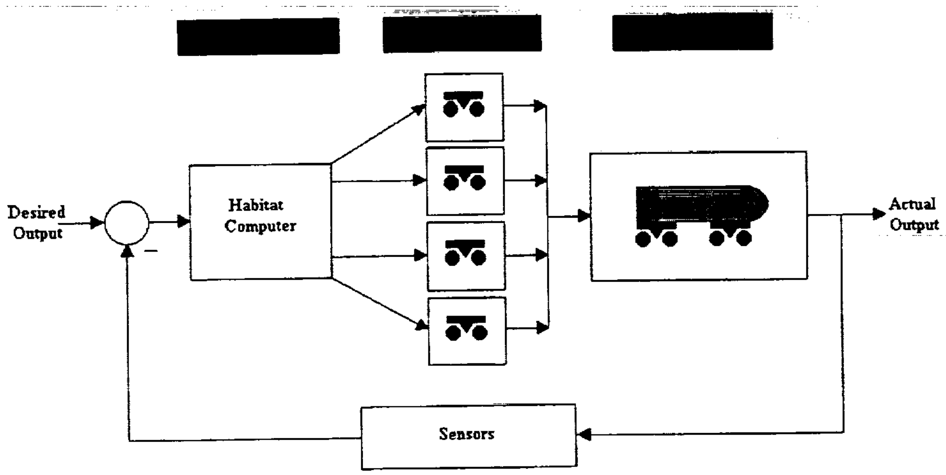

Fig. 7.2 Control architecture for a mobile habitat.

7.3 SYSTEM HARDWARE

The planned hardware systems incorporated into each truck unit would include stereo cameras, laser ranging capabilities, wheel optical encoders, GPS receiver and an IMU. It should be noted that the truck design has a programmable control system structure that can be configured to what is required for a specific operation.

Stereo cameras would provide a remote user who operates the vehicle with a real time image of the terrain around the vehicle. Distances to obstacles are measured using lasers mounted at various positions around the truck's chassis. The orientation of the truck is measured using the IMU's rate gyros. The optical wheel encoders monitor the position of the drive wheels by measuring the steering angle and the shaft position of the wheel unit.

The navigation hardware would include a GPS receiver and an IMU unit. The navigation control systems are managed by the truck's central computer, in which diagnostic subroutines are written to provide reports of system status.

8.0 HUMAN CONSIDERATIONS

A detailed discussion of the effects of space travel on human physiology and psychology is beyond the scope of this report. The conditions of weightlessness and reduced gravity, sunlight deprivation, radiation exposure and many other aspects of spaceflight have been shown to have significant and deleterious effects on human beings. Bone demineralization, immune dysfunction, cardiac and muscle deconditioning and increased cancer risk are but a few of the damaging outcomes of space travel on human physiology. This section will concentrate on two aspects that impact most heavily on the design of mission components – radiation and life-support systems.

8.1 RADIATION PROTECTION

8.1.1 RADIATION ENVIRONMENT

Consideration of the radiation environment on the Martian surface is of tremendous importance in the design of a Mars surface mission (Wilson, 1993, 1998). Furthermore, the radiation exposure and the resultant health risks for any particular component of such a mission should be examined in light of the exposure and health risks of the complete planetary expedition.

Radiation can be subdivided into non-ionizing and ionizing radiation. Non-ionizing radiation such as ultraviolet and X-rays, while of significance in the design of sun exposed materials, will not be discussed in depth. This section will instead concentrate on ionizing radiation. Ionizing radiation consists of high-energy particles that exert their deleterious effects by stripping electrons from matter through which they pass. These particles possess energies on the order of tens to hundreds of MeV. Ionizing radiation poses significant dangers to a manned mission to the surface of Mars, and an understanding of these dangers is essential in the design of habitable structures. This section will review what is currently known about the radiation risks of the Mars surface and what the implications will be to personnel on the surface. Shielding considerations will be reviewed, and recommendations made for both design considerations and for further research into the problem. An assessment is made as to the feasibility of shielding against the various component radiations and what the mass and design implications are of attempting such shielding.

Note: Measurement of radiation in SI is generally expressed in terms of grays (Gy) for absorbed dose and sieverts (Sv) for dose equivalents. Dose equivalents are calculated by adjusting radiation dosages to better compare for effects such as cancer.

Absorbed dose:
1 gray = 1Gy = 1 joule/kilogram = 100 rads = 10000 ergs/gram

Dose equivalent: 1 sievert = 1 Sv = 1 joule/kilogram = 100 rems = 10000 ergs/kilogram

8.1.1.1 COSMIC RADIAITION

Cosmic radiation, otherwise known as Galactic Cosmic Rays (GCR), is radiation of galactic origin. Comprised of ionized atomic particles ranging from hydrogen to heavier particles such as carbon (C) and iron (Fe), these particles are extremely energetic on the order of tens to millions of electron volts. The penetration of these highly energized particles into the inner solar system is limited somewhat by the solar magnetosphere. Penetration of these particles to the surface of the Earth is further limited by both the Earth's magnetosphere and by the Earth's atmosphere, resulting in the observation that cosmic rays of little serious concern to living organisms on this planet. On Mars, however, the reduced strength of that planet's magnetic fields and the relative absence of an atmosphere result in GCR flux that poses a risk to living systems (Wilson, 1993). These risks are discussed below.

8.1.1.2 SOLAR RADIATION

Solar radiation, otherwise known as Solar Energetic Particles (SEP) is of solar origin, as the name suggests. SEP consist largely of ionized hydrogen nuclei (protons), and are of lower energy than GCR (lETAW, 1997). As is the case for GCR, SEPs are shield from Earth's surface by the Earth's magnetosphere and pose little risk to living organisms on Earth. Under normal circumstances, solar radiation poses little risk to astronauts either in space or (presumably) on the Martian surface. Large bursts of SEP occur periodically with sunspots and solar flares, and these bursts do pose a significant threat to living organisms. These bursts can be enormous and do pose tremendous threat to all living organisms. On Earth, the Van Allen Radiation Belts and the Earth's atmosphere afford adequate protection from these events. At the present time, observed sunspot activity affords only several hours of advance warning of such events. The danger posed to astronauts by these infrequent SEP events is severe enough to necessitate consideration of shielding and mission timing in the planning of any piloted mission to Mars (Simonsen, 1993). Unless a crew has access to a safe haven, or is able to return to their shielded home base in several hours, they would have to rely upon adequate shielding capability in the design of their mobile craft. The specific risks posed by these high intensity radiation bursts are discussed below.

8.1.1 HEALTH IMPLICATIONS

These two types of radiation, GCR and SEP will pose significant, though different, risks to human life on a mission to Mars. Mission design requires an appreciation of these risks, and solutions to this problem require consideration of shielding materials, mission timing, and an assessment of acceptable risk. A complete discussion of radiation concerns for deep space missions is beyond the scope of this report. Thorough discussions of this problem can be found in the reference section of this. What follows is a brief overview of the problem faced by the design team.

The health implications from radiation exposure are divided into stochastic effects and deterministic effects. The main stochastic effect of importance is cancer, and exposure is calculated in terms of total dose and in terms of lifetime risk for developing cancer. Deterministic effects include prodromal response (radiation sickness), temporary sterility and optic lens opacity (Letaw, 1997).

Galactic cosmic rays are of primary concern in stochastic effects. A two or three year mission to Mars will involve a significant cumulative exposure to cosmic rays, and implications for long term risk of cancer are must be considered. Current recommendations for safe exposure to heavy particle radiation are largely derived from the nuclear industry. Such recommendations are based on the somewhat arbitrary level of an acceptable risk increase of 3 percent of a fatal cancer over the lifetime of an individual (Curtis, 1998).

Solar radiation, arriving as it does in large bursts, is of concern primarily for deterministic effects. The most severe deterministic effect of radiation is death from acute radiation sickness. Health effects of acute radiation exposure are probabilistic in nature and vary somewhat from individual to individual. In high doses, radiation affects primarily rapidly dividing cells and symptoms result accordingly. In human beings, these cells are found largely in the bone marrow (the source of new blood and immune cells), the intestinal lining, the skin, and ocular lens. Symptoms resulting from radiation exposure therefore include immune suppression (marrow cells), diarrhea, nausea, vomiting, skin edema, and lens opacities. High exposure may well result in death. Estimations for dosing effects are taken largely derived from the therapeutic radiation exposure of cancer patients. These limits are adjusted in an effort to apply them to a healthy astronaut population. A graph demonstrating the probability of death resulting from acute exposure is reproduced below.

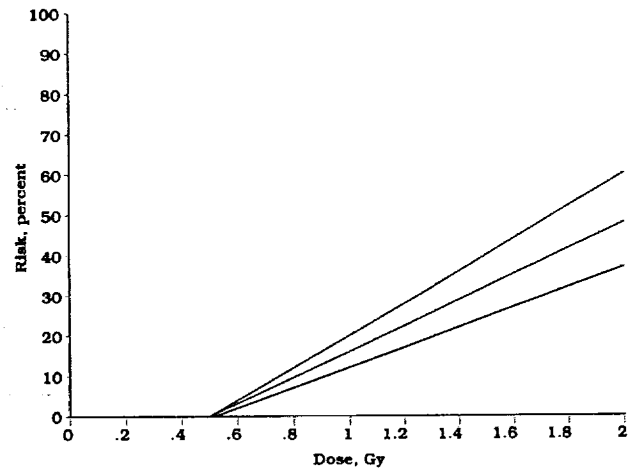

Fig. 8.1 Risk Of Death at 60 Days from Radiation Exposure. Lower line represents average, with 1 standard deviation in the middle, and 2 standard deviations above.

For comparative purposes, the frequently cited event of August, 1972 produced exposures on the order of 1 to 5 Gy. (Letaw, 1997)

8.1.3 SOLAR CYCLE AND RADIATION

Both GCR and SEP vary over the course of the 11 year solar cycle. During the period of solar maximum, GCR penetration to the inner solar system is minimal due to the improved protective influence of the of the solar magnetosphere. Solar maximum occurs at the midpoint of the solar cycle, approximately between years 4 and 9. Unfortunately, the occurrence of increased sunspot activity and therefore of SEP activity is also increased at this time. This relationship is shown in figure 8.2.

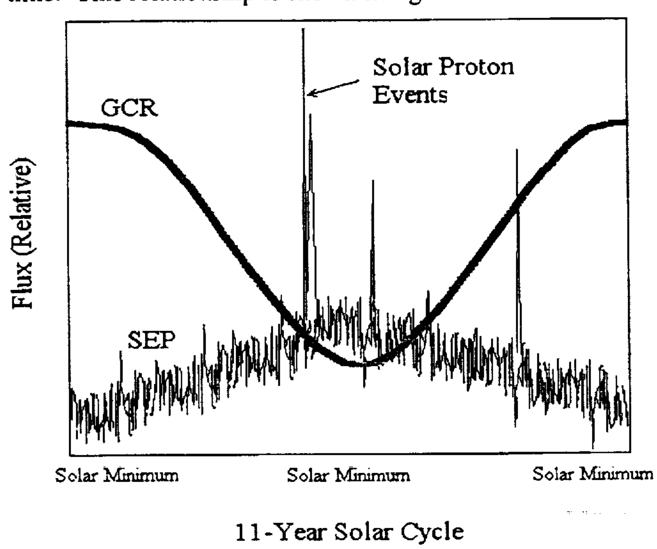

Fig. 8.2 Timing of GCR and SEP with respect to solar cycle - note, fluxes are not comparable in magnitude.

Ideally, the design team should have knowledge of when in the solar cycle a mission to the planet surface would be launched. In the absence of such information, the design must allow for the worst case scenarios for both GCR and SEP exposure.

8.1.4 SHIELDING

Shielding presents a difficult problem, particularly because of the two types of radiation involved. In general, GCR is too energetic to shield against, whereas shielding against SEP events will be required periodically. Further complicating the issue is the effect of nuclear spallation of GCR. Spallation is the process by which the heavy and highly charged cosmic rays impact the shielding materials, causing a cascade of less energetic but equally damaging atomic particles. Calculations by Letaw (1997) and others demonstrate that with materials such as aluminum, even 30 cm of solid shield does little to lessen the damaging effect of GCR. Furthermore, some theorists have demonstrated that shielding with heavy elements such as metals actually produces an increase in radiation dosing. This would suggest that when not protecting against the less energetic SEP radiation, astronauts would be best served by being minimally shielded from the background radiation of the Mars surface. An exception to this problem occurs with hydrogen shielding, as the atoms of hydrogen are unable to be broken down into smaller particles.

8.1.5 DESIGN CONSIDERATIONS

Although hydrogen is a safe shielding material to use against GCR, the sheer amount of hydrogen required precludes an effective shield on a mobile habitat. Such is not the case for SEP, and the protection provided by stored consumables and the laboratory equipment should provide sufficient cover to allow construction of a shielded chamber for the astronauts located on the bottom of the habitat module.

(include shielding calculations for maximum SEP dosing)

8.2 LIFE SUPPORT SYSTEM

Since the design assumptions for the MSTS include a fully functioning home base, the generation and recycling of life supportive materials is not a design requirement of this system. The IHM must include the capacity to store oxygen, water, and food for the two crew members for periods up to four weeks, along with a reserve in event of loss or unexpected delay in return to home base. CO2 scrubbing and environmental control should be easily accomplished with systems similar to those devised for Skylab, Soyuz, Mir, Shuttle, and the International Space Station (ISS).

9.0 WORK REMAINING

The following is a list of areas that represent incomplete analysis by the design team at the time of this report.

1) Sizing and mass estimates.
2) Stress analysis of inflatable materials.
3) Power requirements.
4) Solutions for dust accumulation problems.
5) Suspension system design.
6) Traction calculations in reduced gravity.
7) Explore Transhab work for conversion to this project

10.0 RECOMMENDATIONS

1) Highly effective radiation shielding at a home base increases the safety margin for a poorly shielded excursion on the planet's surface. Consideration should be given to construction of a maximally shielded home base (regolith protection, etc.) for two reasons: 1) A complete picture of the radiation risk does not yet exist on the Martian surface, and 2) Increased protection at a home base will compensate somewhat for unexpected dosing either in transit to and from Mars or while on the Martian surface.

2) Consideration should be giving to launching a prolonged surface mission to the moon prior to a Mars mission. The general consensus from many sources is that the uncertainty of both the deep space radiation environment and the effect of that environment on living organisms poses an unacceptable risk to an astronaut crew. A lack of firm data on this subject also leads to extreme difficulty in the design of safe and adequately shielded mission components.

3) Consideration should be given to landing and expeditionary mission within a Martian canyon. Advantages would include a lowered dose of continual background radiation as a result of natural shielding, and the possibility of using the canyon walls as a safe haven in the event of a solar event.

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M.A.G.I.C.

Mars Advanced Greenhouse Integrated Complex

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ABSTRACT

Human Exploration and Development of Space (HEDS) is a strategic enterprise of the National Aeronautics and Space Administration (NASA). One of the many goals of this initiative is the exploration and colonization of the planet Mars. One approach to this ambitious undertaking is to transport a minimum of resources and utilize as many Martian resources as possible, reducing the overall cost of the mission.

A long duration mission, which utilizes in-situ plant growth-facilities, reduces the dependence on consumable supplies from earth. The reduced number of cargo launches required lowers the cost of the project. Additional equipment may then be shipped in place of consumables. Data obtained from growing food on Mars can be used in planning for permanent habitation of the planet.

A team of undergraduate students and professors at the University of Texas at San Antonio (UTSA) has developed the Mars Advanced Greenhouse Integrated Complex (MAGIC). The project is designed to meet the requirements of the NASA reference mission. A two-phase approach is used. Phase I utilizes resources previously expended by NASA. Phase II is a conceptual design for large-scale growth of food on Mars. [1a, 1b]

1. INTRODUCTION

The project was divided into six teams; Systems Integration; Crop Requirements and Mission Plan; Greenhouse Layout and Structure; Atmosphere Supply and Control; Hydroponic Fluid Supply and Control; and Data Acquisition and Control. A sub-team developed a conceptual design of a robotic harvester. The teams were comprised of a mixture of biology students and civil, electrical, and mechanical engineering students.

A crop list was generated using a variety of parameters. Among these parameters are human nutritional requirements, menu versatility, harvest methods, gas exchange characteristics, Hydroponic nutrient requirements and dimensional restrictions. The technical details supporting the content of this paper are available in our reference report [1a].

The crop size and weight specifications were then established to help choose a greenhouse configuration. Analyses were performed comparing structural configurations (horizontal vs. vertical), and structural designs (rigid vs. inflatable). The vertical configuration provides the most crop space. The inflatable structure provides greater volume for plant growth. To stay within reference mission guidelines, a rigid structure was chosen as the baseline. Four vertical rigid structures provide redundancy, adequate crop space and harvesting area. One structure is modular for plant growth height. [1a, 1b]

Maintaining an atmosphere conducive to productive crop growth requires monitoring and controlling gas concentration, pressure, temperature, and humidity. The systems required to perform these operations involve the use or adaptation of existing atmospheric controls systems. For operations that could not be performed by existing or adapted equipment, new equipment was defined for future development.

The Hydroponic fluid supply and control system involves the design and synthesis of several subsystems. These include a nutrient production system, solution circulation system, water purification system, condensation system, and a sensing system.

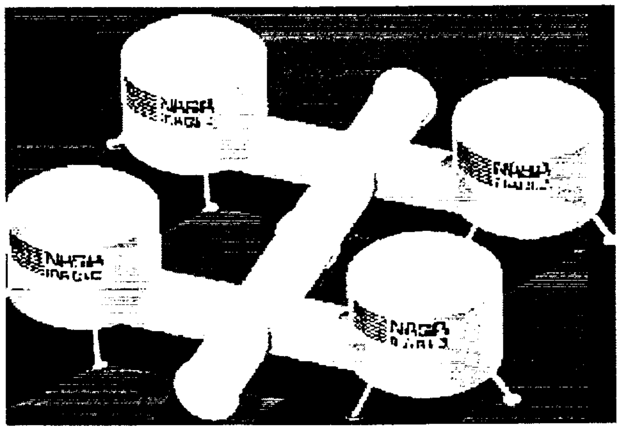
Control systems, power, and data acquisition systems were developed. Computer stations, fiber optics, electrical cable, video cameras, intercom stations, controllers, sensors, communication systems, voice recognition systems, airlock controls, and lighting systems were parameterized and discussed.

Finally, the need for an autonomous robotic harvester was identified with specific tasks for future development. Implementation of a robotic farmer would enable astronauts to utilize their time more productively. Basic requirements for the robot and future technological challenges were addressed.

PHASE I

Phase I of MAGIC was designed to meet the criteria described in NASA's Mars Reference Mission [1b]. The design makes use of four rigid cylinders and an interconnecting tunnel system. Sections two through six describe the design parameters and the atmosphere, hydroponic, and control subsystems. A robotic harvesting concept was then proposed.

Figure 1-1: Phase I Greenhouse Concept



2. LIFE SUPPORT REQUIREMENT

***Human Consumption Requirements*:** The daily needs for a human, based on an average metabolic rate of 4898 calories per person per day are: oxygen, 0.84 kg; food solids, 0.62 kg; and water, 57.28 kg. The effluents per person per day are carbon dioxide, 1.00 kg; water, 29.487 kg; and 0.109 kg. [1a]

***Human Nutrient Requirements*:** Human Nutrient requirements will be met by a combination of plant growth on Mars, and dietary supplements. The crops were chosen to meet U.S. RDA (Recommended Daily Allowance) and NASA Space Requirements for nutrition. The long-term effects on plant and human physiology have not been analyzed under Martian gravitational conditions. Dietary supplements allow NASA doctors to respond to potential physiological changes in the crews due to the diet. [2]

***Human Atmosphere Requirements*:** Life exists in a narrow range of atmospheric oxygen and carbon dioxide pressure. At sea level the partial pressure of oxygen is 21.21 kPa, and carbon dioxide partial pressure is 0.0318 kPa. The minimum partial pressure of oxygen (ppO_2) which a human can tolerate for extended periods is 19 kPa. Lower partial pressure of oxygen (ppO_2) can be tolerated for a short duration. However, there are side effects to lower pressure. Altitude sickness occurs after 8 to 10 hours of ppO_2 at 13.75 kPa. The maximum ppO_2 humans can withstand is 32.4 kPa, however lung irritation occurs after 12 to 72 hours at this level. Humans can tolerate $ppCO_2$ levels as high as 1.01 kPa for short periods (several days), and $ppCO_2$ levels of 1.59 kPa for very short periods under emergency conditions. A $ppCO_2$ of 0.40 kPa can be tolerated for long periods. The atmosphere control systems were designed to maintain the greenhouse within acceptable oxygen and carbon dioxide ranges for human and plant life. [3]

***Plant Productivity*:** The crops chosen and their required daily harvest volume are listed in Table 2-1. These volumes are designed to meet the needs of a six-person crew for two-year period (avg. 4898 cal/person/day). The starter solution will be given to the plants at the beginning of their growth. Plants will initially take up nutrients and store them within their tissues. The nutrient concentration will drop significantly at this time. It is not necessary to add nutrients until the vegetative growth stage. At that point, vegetative growth solution will be added to provide the plants with the nutrients needed at this stage. Nutrient content will be monitored and maintained by the hydroponics subsystems.

TABLE 2-1: Plant Growth Facility Crops

Average harvest requirement, per day, for each greenhouse crop is provided

Crop	**Soybean**	**Wheat**	**White Potato**	**Carrot**	**Spinach**	**Cabbage**	**Lettuce**
kg/day	0.60	1.89	0.77	0.28	0.35	0.08	0.27
Crop	**Tomato**	**Peanut**	**Dry Bean**	**Sweet Potato**	**Celery**	**Green Onion**	**Strawberry**
kg/day	1.52	0.22	0.08	0.67	0.08	0.27	0.26
Crop	**Peppers**	**Rice**	**Pea**	**Snap Bean**	**Beat**	**Radish**	**Broccoli**
kg/day	0.32	0.17	0.17	0.04	0.30	0.17	0.14

Each plant species has unique atmospheric temperature and humidity requirements. Two separate growing environments are required. One environment will be maintained in a temperature range of 16-20 •C and a relative humidity range of 65%. This environment will grow the following crops: wheat, white potato, dry bean, celery, peas, lettuce, spinach, broccoli, green onion, cabbage, strawberry, sugar beet, carrot, and radish. The second environment will be maintained in a temperature range of 22-26 •C and a relative humidity range of 65%. This environment will grow the following crops: rice, soybean, sweet potato, peanut, tomato, peppers, and snap bean. Warm temperature crops can be grown in cooler temperatures with a loss of yield. However, growing cool temperature crops under warm temperatures will lead to little or no edible biomass. The photoperiod for both environments will be 12 hours of light and 12 hours of dark. *Wheat yield is higher when grown with a 24-hour photoperiod. This extended photoperiod adversely affects other crops in the same environment. The lower yield in wheat will be offset by increased yields for other crops.* It is possible, and recommended to provide a greenhouse dedicated to growing wheat. This would allow for the 24-hour photoperiod and increase the yield. Due to flexibility of design as further research is completed, or at the astronauts request additional plants may be added to the menu.

The minimum growth area required to meet mission requirements is 500 m^2. The area needed per crop per maturation period is listed in reference [1a]. However, this does not allow for emergency contingencies, such as a crop failure or a systems failure. Therefore, a minimum growth area of 600 m^2 should be constructed. Additional space provides for crop research, and a safety margin. It is possible to decrease the 600m^2 recommendation if wheat were grown in a separate greenhouse containing at least 200 m^2, and rice and soybeans are grown in a separate greenhouse containing at least 200 m^2. This reduced diet would provide the astronauts with sufficient caloric intake, however many necessary nutrients would be omitted from the diet. Dietary supplements would be required.

***Air and Water Revitalization via Bioregenerative Process*:** The requirement for O_2 per day is 0.84 kg/person. A safety margin should be incorporated. Research has shown that an active growing area of 25 m^2 will provide air revitalization for one moderately active person. This area can be in any configuration desired. In one study, potatoes were used to provide O_2. They were grown in a continuous production mode (the periodic harvesting and planting of crops at short intervals to maintain a steady state of life support). This implies various maturity rates of each crop. It is not known exactly how much O_2 each individual plant produced. Assuming 25 m^2 growing area of any crop produces 0.63 kg per day per person of O_2, at least 150 m^2 of plant growth is required. [5]

***Crop Mission Plan*:** Plants increase photosynthesis at elevated CO_2 levels. At sea level, the CO_2 partial pressure is 0.0318 kPa. Plants cannot survive in a partial pressure CO_2 level greater than 0.2 kPa [6]. Plants will tolerate much lower partial pressure O_2 levels than humans. O_2 levels must be high enough for germination and respiratory metabolism during dark hours, however this amount is quite small. The exact partial pressure of oxygen required by plants is specific to the individual species. The required germination oxygen level is lower than the necessary partial pressure of oxygen needed by humans. [7]

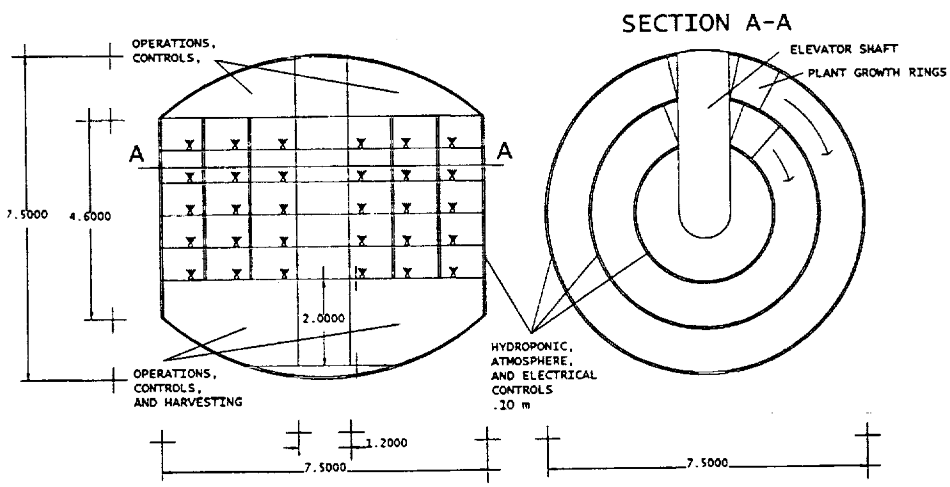
Plants grown at pressures as low as 14 kPa have only slightly lower germination percentages and stem lengths. Plants grown at 33 kPa do not show any significant changes in germination percentages and stem lengths as those grown at 101 kPa. [8]

Phase I of the plant growth facility will have an atmospheric pressure of 101 kPa, with a partial pressure of CO_2 of between 0.10% and 0.15% and a partial pressure of O_2 of between 15% to 18%. The airflow will be 1 meter per second at the top of each plant canopy. In Phase II, minimum atmospheric pressure will be 25 kPa. This will ensure maintaining partial pressures of CO_2 and O_2 within the required levels.

3. GREENHOUSE ARCHITECTURAL LAYOUT

General Approach: The primary function of the Mars greenhouse structure is to provide an adequate environment to grow and process food. The structure should be pressurized, be easily constructed, and be easily maintained. The challenges that need to be addressed are maximizing the use of available space, providing a simple, modular construction scheme, and providing access for automated systems. The structural layout is provided in figure 3-1. A three dimensional cross-section is shown in figure 3-3.

Figure 3-1: Architectural Layout



Space Requirements: A study of plant growth area in horizontal versus vertical configurations revealed the superiority in the vertical configuration's use of space. Vertical configuration utilizes available volume better than the horizontal configuration. The horizontal configuration presented hindrances to a modular shelving system because of the dimensional layout of the structure. The floor area of each level in the vertical configuration is identical therefore simplifying the use of a modular shelving system. In order to meet a crop requirement of 600 square meters [1a] five horizontal configurations would be needed whereas the vertical configuration would fulfill this requirement with four structures. The need to maximize plant growth area within a limited space makes the vertical configuration the logical choice. As indicated in Table 3-1, the better solution would be to choose the vertical inflatable cylinder. A single vertical inflatable cylinder provides $544m^2$ of growth area. This would meet mission requirements. However, for the purposes of this study we used a baseline vertical rigid cylinder per the reference mission.

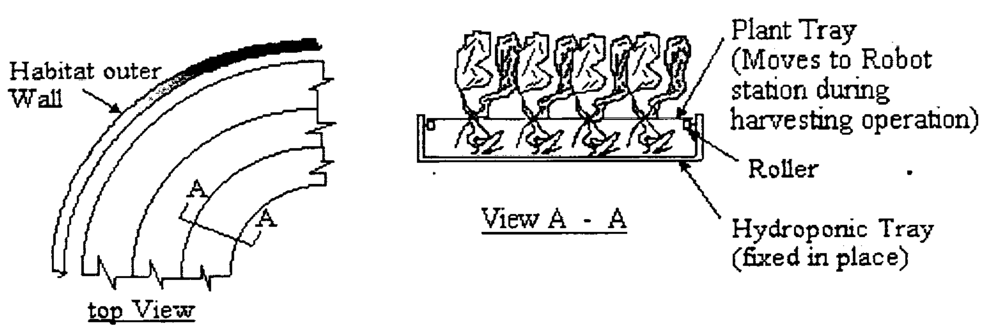
Table 3-1 Greenhouse Dimensional Analysis

Orientation	Vertical (Rigid)	Horizontal	Vertical (Inflatable)
Overall Dimensions (m)	7.5 ϕ x 7.5 L	7.5 ϕ x 7.5 L	9.5 ϕ x 9.5 L
Length Of Cylinder (m)	4.6	4.6	7.5
Plant Growth Capacity (m^2) 0.5m / 0.75 m shelf spacing	259 / 185	205 / 143	715 / 455

Two evolutionary phases are envisioned for the Mars greenhouse. Phase I utilizes structures currently identified by the Mars Reference Mission [1b]. Figure 3-1 shows a single unit from the selected Phase I layout based on the use of rigid cylinders.

***Modular Shelf System*:** The vertical configuration uses a modular shelving system. The shelving systems consist of lighting, air circulation, hydroponics and plant trays. As indicated in Figure 3-1, the modular shelving system is arranged into four concentric circular segments with plant trays that can be moved into the elevator opening for harvesting, cleaning and reseeding.

Figure 3-2: Modular Shelf System



***Space Allocation*:** Four vertical structures are required to obtain 600 m^2 of growth area. [1a] Each crop growth level will accommodate 37 m^2 of crops. Two structures will contain five levels with 0.75 m of vertical spacing. The remaining two structures will contain six levels with 0.50 m of vertical spacing. Each structure uses approximately 65% of the total 250 m^3 of volume for crop growth. The remaining 35% (87 m^3) is allocated for the following:

- Hydroponic fluid storage
- Automated controls
- Useable plant material
- Harvesting Equipment
- Atmospheric Controls

The structure provides compact and efficient use of space while maintaining sufficient space for both automated and human operations.

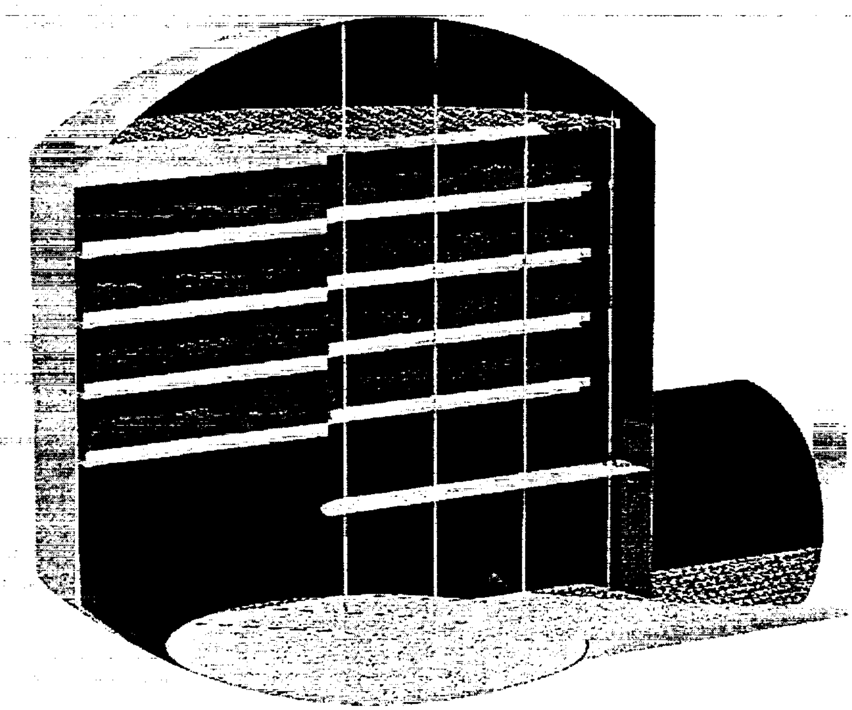
***Support Overview*:** The exterior of the structure is an expended Mars cargo vessel. The framing will use a graphite-reinforced epoxy-material [11]. The supports for the shelving can be fixed to the sides of the structure before or after arriving on the Mars surface. Connections can be bolted pinned or welded as necessary. A rigid support frame for the structures should be used. The support frame would be assembled on the Martian surface. These frames will serve multiple purposes. First, the frame minimizes greenhouse heat loss by limiting direct contact with the Martian surface. Framing material will have low or non-conducting thermal properties to minimize heat transfer from the structure to the surface. A second purpose for the frames will be to maintain the greenhouse level. An auto-leveling system should be built into the frame. This is easily accomplished with electronic sensing devices and adjustable supports built into the frame itself.

The individual greenhouses will be connected together by tubular tunnels with airlocks at each entrance. These tunnels should be between two and three meters in diameter, allowing for equipment accessibility. These connecting tunnels will also be used to house control systems, and harvesting devices for the greenhouses.

Inflatable Technology: Inflatable technology is currently under research for use on future NASA missions. An inflatable structure will provide significantly more space with only a minimal increase in dimensions. A 9.5-meter diameter structure, similar to the current TransHab module, would yield 570 cubic meters of volume [13]. This is a huge benefit in payload size and weight, which leads to an overall cost reduction.

The inflatable structure would use a similar support technology as the rigid structures. The frame could be stored in the "central structural core" removed, and setup prior to inflating the module [13]. This design requires longer setup times than its rigid counter part. However, the significant increase in growth space provides a more economical system than the rigid cylinders. Further research in this area should reveal this the better option for plant growth.

Fig. 3-3: Structural Cross-Section



4. ATMOSPHERE SUPPLY AND CONTROL SYSTEM

For ideal crop growth, the ranges shown in Table 4-1 must be maintained. Heat transfer analysis was conducted for the greenhouse structure. Detailed calculations and methods are available in the reference report [1a]. Air circulation requires two 125 W and two 150 W blowers. Polyurethane Foam insulation will be used on the greenhouse walls. The walls of the greenhouse will serve as heat exchangers. The blower capacity is based on 246 m² of plant shelving. The air-handling system provides from 3 to 4 air exchanges per minute, with air velocities ranging from .1 to 1.0 m/s. Chilled water coils at each of the blower's exits provide heat rejection and humidity control. Condensate, which forms on the coils, will be collected and measured in order to monitor evaporation rates. Atomized streams of water injected directly in the air stream provide supplemental humidification. [14]

Table 4-1: Atmosphere Supply and Control Requirements

Subsystem	Range Of Operation	
Air Revitalization System Oxygen Carbon Dioxide	 18.5 -23.45 300 -5000	 % µL / L
Chamber Pressure	101	kPa
Ventilation and Thermal Control Air Temperature Relative Humidity Air Velocity	 15 - 35 70 - 85 .1 - 1.0	 °C % m / s
Leak Detection and Control Leakage Rate	1	% of the chamber volume/day

The pathogen filtering system will consist of two parts, coarse filters and electrostatic precipitators. The coarse filters will remove large particulate to prevent fouling of the air ducts. The electrostatic precipitators will remove the smaller particulate. A parts list can be found in Table 4-2.

Table 4-2: Parts list

Component	Characteristic
4 Blowers	(2) 125 W and (2) 250 W return blowers
Duct , Fibrous glass liner	Rectangular H=4" W= 5.5" Length = 1139.6'
Pathogen Filters, electrostatic precipitators	4 Area =0.5 m^2
Insulation	Polyurethane Foam 0.006m thick total area 48.5 m^2
2 Condensers	
2 Heat pump	100 W

Atmosphere Controls Analysis: Adequate supplies of oxygen are required for unsuited human entry into the greenhouse. Controlling levels of oxygen, carbon dioxide and other gases is a major concern for proper plant production and processing of plant waste. Maintaining a suitable atmosphere requires the regulation of oxygen, nitrogen, carbon dioxide, and other trace gasses. A system that can generate oxygen on demand, filter out carbon dioxide and replace or remove nitrogen is required. Oxygen and nitrogen separators are commercially available and are easy to integrate into an atmospheric control system.

Oxygen and the other atmospheric gasses will be lost to the Martian atmosphere through inevitable leakage at an assumed rate of 1% of the chamber volume each day. A self-sustained system minimizes the necessity for transport from earth. This loss comes to 0.013 kg of oxygen, 0.011 kg of carbon dioxide and 0.011 kg of nitrogen each day. Replacement oxygen can be provided from two sources. First, oxygen is a byproduct of photosynthesis. Previously, we have determined that 0.63 kg/day of oxygen can be obtained from 25 m^2 of plant area. If there are 300 m^2 in production at any given time, then 7.56 kg of oxygen will be produced each day. If the attendants collectively use 5.04 kg/day the greenhouse will experience a net oxygen production of 2.51 kg/day. Excess gas can then be separated using a commercial separator and stored for future use. The second method of obtaining oxygen involves separating elemental oxygen from bearing gasses in the Martian atmosphere. The Mars Surveyor 2001 lander, scheduled for launch on April 10, 2001, will demonstrate the viability of an oxygen generation system. The Space Technology Laboratory (STL) of Arizona State University has developed an Oxygen Generator System (OGS).

Table 4-3: OGS specifications

Parameter	Value
Total system mass:	1000g
Start up power:	15W
Steady state power:	9.5W
Oxygen flow rate:	>0.5sccm
CO2 supply:	>2.5sccm
cell operating temp.:	750C
envelope:	8" x 6" x 5" (h,l,w)

If the OGS test is a success, then oxygen will be available for life support usage. Carbon dioxide is readily available from the Martian atmosphere and may need only some filtration to remove harmful elements. Nitrogen or some other carrier gas must be transported from earth or generated from some other means. Filtration of unwanted greenhouse gases is an easy task. Separation of oxygen, nitrogen, carbon dioxide, and volatile trace gasses can be accomplished using commercially available separators such as those available from On Site Gas Systems. These separators are available in many configurations and can satisfy almost any specification.

***Temperature*:** Temperature will be measured by Resistance Temperature Detector (RTD). The range of this type of sensor is -40°C to 150°C. A sensor will be installed on every third crop tray to monitor ambient temperatures. This will ensure adequate plant growth requirements.

***Pressure*:** Pressure will be measured by a Sputtered Thin Film pressure sensor. This sensor remains stable in extreme operating conditions. This high performance transducer incorporates a thin film sensor reducing the need for routine maintenance. These sensors will be mounted on the ceiling of the greenhouse. The sensor will be used to determine if a filter requires cleaning.

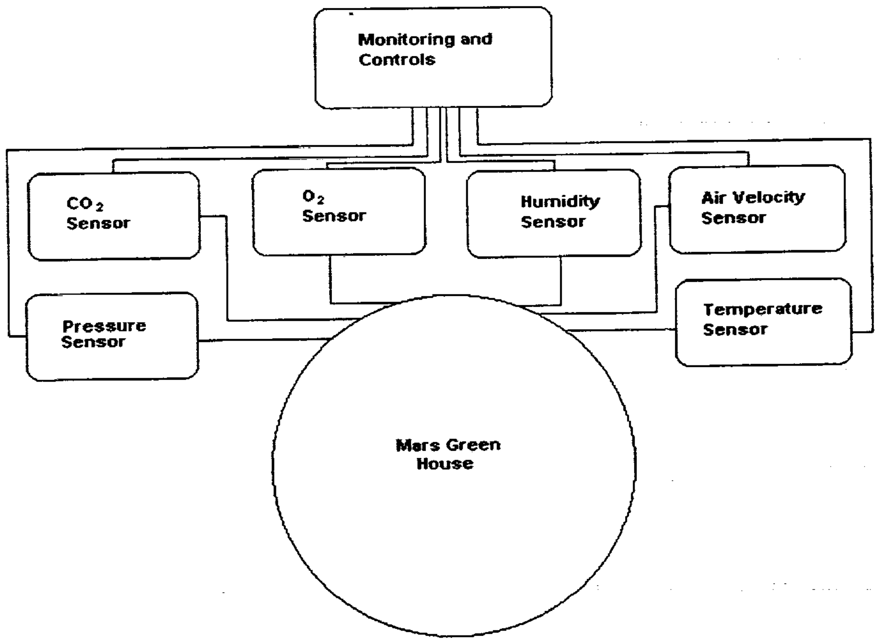
***Humidity*:** Humidity will be monitored by a Relative Humidity sensor, which is configured with integrated circuitry to provide on-chip signal conditioning. These sensors contain a capacitive sensing die set in thermoset polymers that interacts with platinum electrodes. The laser trimmed sensors have an interchangeability of + 5%RH, with stable, low drift performance. The sensor will be placed in the greenhouse ceiling. This type of sensor can be operated in temperatures that range from $^-$40°C to $^+$85°C.

***Sensors*:** Oxygen sensors will be installed at various locations within the greenhouse. The sensor external materials are entirely inert (Teflon and ceramic). The sensor can be used in biological applications or in harsh chemical environments. The sensor can also be operated in either liquids or gases, from vacuum to high pressure. The sensor can resist temperatures from -85°C to 135°C. The dual-chamber oxygen cell design requires biannual calibration. Air circulation will be monitored by a Gas Ultrasonic Flowmeter. This sensor has a wide operating range without pressure drop and does not require routine maintenance. Analog and digital outputs in velocity and actual volumetric flow rate are standard. The meter has a velocity range of 0.1 to 150 ft/s and it has no moving parts. The meter can measure gas flow in pipe or a duct ranging from ½-inch tubing to flue stacks over 25 feet in diameter with appropriate transducers. The meter can resist temperatures from $^-$20°C to $^+$140°C. The metering device will be mounted either in the ducts or next to the ducts. CO_2 and N_2 measurements will be made using a sensor that will be placed in the ceiling in the green house.

***Velocity*:** In order to dissipate the heat generated by the lighting system an adequate air velocity had to be achieved. The surface temperature of the bulb were assumed to be 400 K and temperature of the air stream to be 293 K. Nussels number was evaluated for the velocities ranging from 0.10 m/s to 5.00 m/s. Once Nussels equation was evaluated corresponding average coefficients of convection were calculated. A direct correlation between average coefficient of convection and velocity is now known. Knowing the total amount of heat generated by the bulbs an average convection coefficient can be calculated. This is then cross-referenced with the range of convection coefficients tabulated, yielding an approximate velocity of 3.3 m/s

Figure 4-1 illustrates an overview of the atmosphere supply and control system.

Fig. 4-1: Atmosphere Supply and Control System Layout



5. HYDROPONIC FLUID SUPPLY AND CONTROL SYSTEM

***Overall System*:** The hydroponic fluid supply and control system will produce the hydroponic solution that will be used by the crops. The hydroponic solution will contain the nutrients that the crops need in order to grow in the Mars greenhouse. This control system is broken into five subsystems.

- Nutrient Production System
- Solution Circulation System
- Water Purification System
- Condensation System
- Sensing System

Individual subsystem diagrams are available in the reference report [1a].

***Nutrient Production System*:** The hydroponic solution will consist of a mixture of water and nutrients. The nutrients will be made up of decomposed plants and minerals. A storage tank is provided for the nutrient supply. The system mixes the nutrients with the water that will be flowing to the growing area trays through a system of pipes. A nutrient controller will control the amount of nutrients that are mixed with the water, and a pH controller will control the pH of the hydroponic solution. Once the fluid is produced, it will then go to the fluid circulation system.

An aerobic bioreactor produces the nutrients. Plant biomass will be finely ground and fed into the bioreactor (120-liter volume) at a rate of 0.2 kg per day. The bioreactor contains water at a pH of 6.5, a temperature of 35° C, and dissolved oxygen that is supplied by airflow through the bioreactor. The mixture will remain inside the bioreactor for 21 days. The reactor contents will be removed in batches of 40 liters every following week after the starting period of 21 days. The contents will then be filtered to remove solids. The extracted solution will then be analyzed to determine the type, and amounts of nutrients and chemicals present and add any if necessary.

***Solution Circulation System*:** This system takes in the processed hydroponic solution and distributes it into the growing area trays. The fluid is pumped through a pipe that leads into a row of trays. All of the trays are interconnected by pipes. As fluid begins to fill the first tray, it then will flow to the next tray, until all of the other trays are full of hydroponic solution. A sensor at the beginning and end of the last tray will measure the nutrient concentration of the solution and direct the flow accordingly. If the solution has enough nutrients, the solution will be directed back to the growing area trays through feedback valves and a pump. Otherwise, the solution will be directed to the water purification system. It is assumed that each of the growing area trays require the same amount of solution.

Water Purification System: The water purification system is comprised of a water recovery system, and a condensation system. The water recovery system is based on a diluted plant solution. This source ensures that water can be recovered, and filtered to be reused again. It will supplement the main storage tank that the habitat uses. This recovery system will allow maximized use of the available resources.

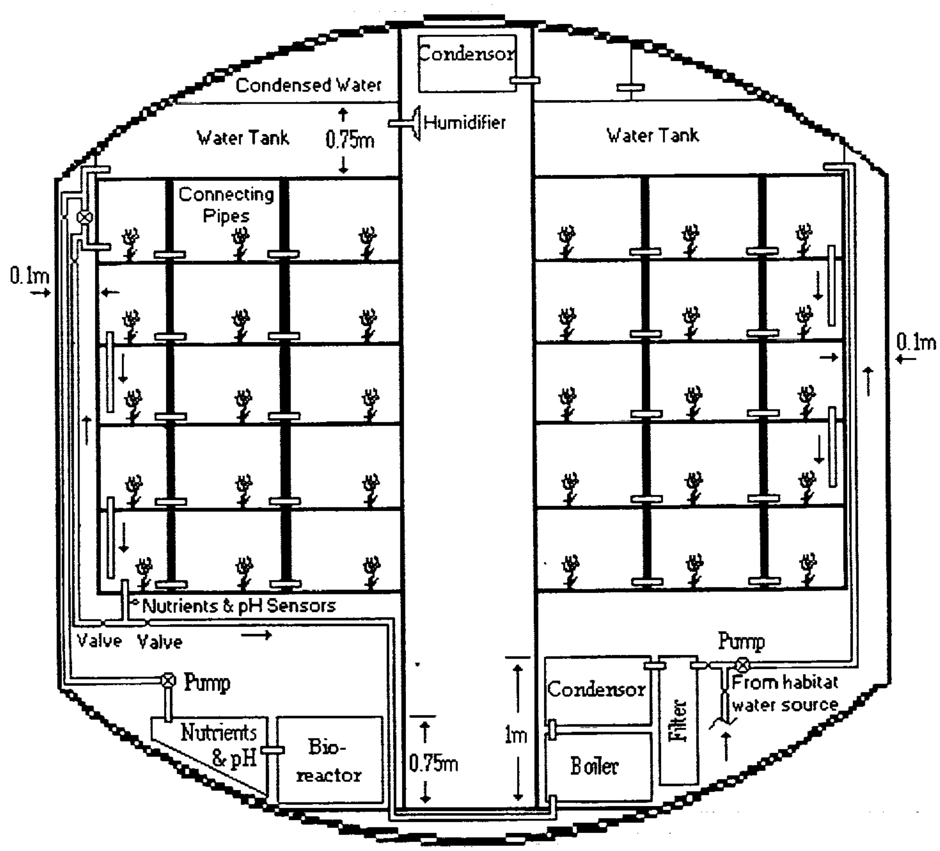
Water will come from any unused solution. This diluted solution must be purified before it returns to the main water tank. The process begins with nutrient sensors indicating to a microcontroller that the concentration of the nutrients in the solution is either within specified parameters, or not. When the concentration falls below acceptable values, the old solution would be removed; while at the same time fresh solution will be provided from the nutrient production system. The removed solution would then go through the purification process. It would start with a boiler that would heat the solution. The heated solution would then be collected in a condenser. Finally, it would be filtered to the proper safety levels and sent to the storage tank. The choice to utilize a boiler-condenser system was based on its ability to disinfect the water as well as purify it.

***Condensation System*:** Condensers will be used to collect any extra humidity inside the greenhouse. The condensers will convert the humidity to water. The water will then go to a storage tank. This system would ensure a maximum use of resources. The storage tank would be tied into the drinking water supply of the habitat. Since plants need a specific percentage of humidity in the air, a humidifier will be used to add humidity to the greenhouse in case the humidity falls to a low percentage. Both of the condenser and humidifier are controlled by a sensor to prevent them from working at the same time, which would defeat the purpose of this system.

***Sensing System*:** The sensing system is responsible for making all of the other systems work properly. This system utilizes a series of sensors that control the functions of each component in each of the subsystems. Most of the nutrient sensors must be custom designed because they are detecting specific chemical compounds, which are not commonly used. All of the sensors will feedback an electrical output to a microprocessor, which will regulate the functions of the hydroponic system according to the requirements.

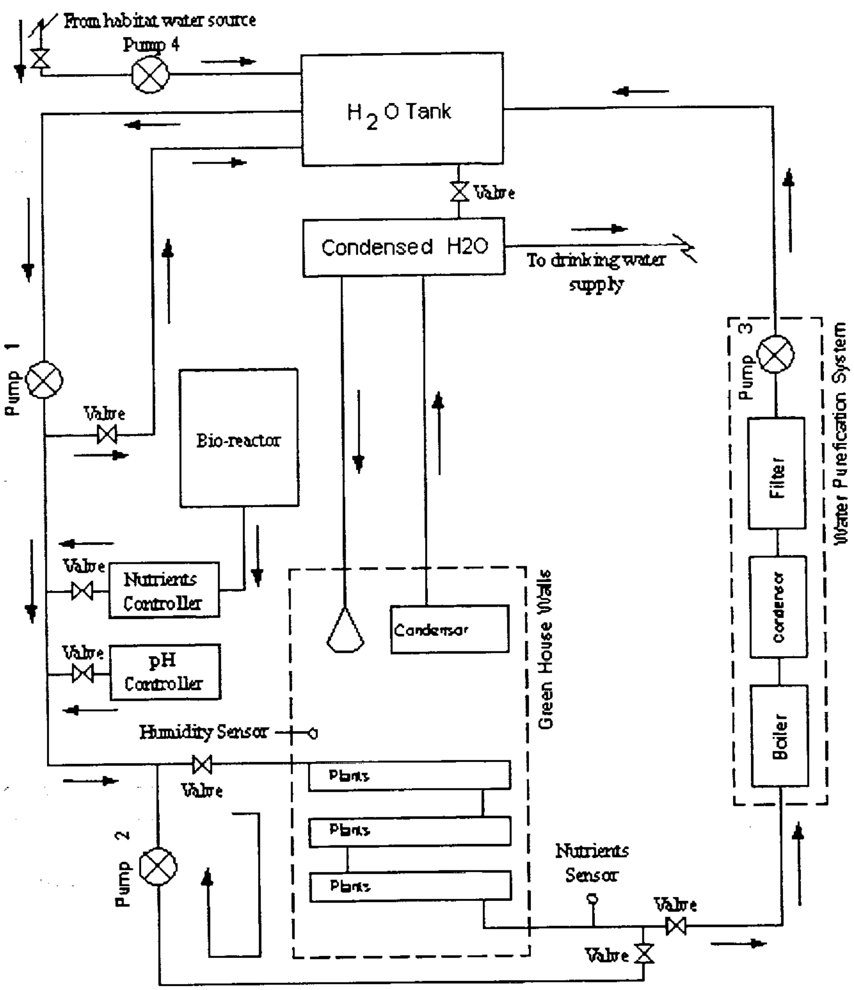
***Phase I* - *Vertical Rigid Structure*:** It will take the system 50 minutes to fill the trays with hydroponic solution. Each tray will be filled with solution up to 5 cm high [1a]. A view of hydroponic system arrangement within the greenhouse structure is provided in figure 5-1.

Fig. 5-1: Phase I Design Requirements



***Safety Measures*:** There are some safety devices installed throughout the system. The first device is a pressure safety valve. This valve is located on the main water pipe. It will redirect the water flow back to the main tank in case of an unexpected pressure increase. The second safety measure is a flow control valve between the main water tank and the condensed water tank. This valve will open to supply the main water tank with water in case of any shortage. Another safety measure is a flow rate gage located at the water exit. This gage will provide a flow rate reading, which would help in regulating the flow. In addition, fluid level sensors are located in each tank to provide fluid volume measurements. Finally, a valve located on the main pipe feeding water to the hydroponics system will enable the operator to shut down the system incase of emergencies. Figure 5-2 is an overview of the complete configuration of the hydroponic system.

Fig. 5-2: The Hydroponic Fluid System



6. DATA ACQUISITION AND CONTROL SYSTEM

Master Control Center **(MCC):** The master control center will be the main point of control for the facility. An overall schematic is provided in figure 6-1. The data line layout is illustrated in figure 6-2. Electrical elements in the greenhouse areas will send signals to, or receive signals from control systems maintained by the central computer system in the MCC. Additionally, the controllers will be able to interface with laptops or portable handheld computers in each area. The local interface with the master control center will be made using fiber optic cable. From the master control center, the astronauts will be able to control and monitor atmospheric sensors, heating systems, atmospheric gas supply system, hydroponic systems, potable water supply system, communications, radar, power control systems, airlock operation and robotics

Computer Stations**:** The elements in each of the greenhouse areas will either send a voltage to a controller in that particular section or it will receive an operating voltage from a master controller. The controller is capable of handling 0-125 V input. It will then feed its output into a laptop that will be stationed in each area. The laptop computer can used to input data or observe data in that section or in any area throughout the complex. The laptop will be linked to the network via a fiber optic cable. Three servers in the master control center will provide redundancy. The monitors will be flat screens to conserve space. The capabilities of each PC are as such, 50 GB hard drive memory (master control center), 3 GB hard drive (area laptops), 1GHz speed and 128 MB RAM. The PCs in each area may be laptops.

***Fiber Optics*:** The computer network will be linked to the areas in the greenhouse with fiber optic cable. It will take approximately 10 meters of fiber optic cable. We are assuming an operational fiber optic PC. Digital to fiber optic converters will be needed for each PC output/input.

***Electrical Cable*:** If nuclear power option is used, the unit power supply will have to be located a safe distance from the greenhouse. A transmission line will be used to bring electricity into the structure. The line should be protected from the adverse temperature. There will be several electrical distribution points in the greenhouse. Not all the motors, pumps and sensors will use the same D.C. input voltage, so several different regulated power supplies will also be required. Some effort should be made to standardize DC inputs and regulation so less power shifting is required. [25]

***Video Cameras*:** Small digital video cameras will be positioned throughout the greenhouse. They will send microwave transmission to a video network that will be setup in the master control room. The number of cameras needed is estimated at 30. Ten of these will be used as spares. Video compression will be used. Using Image Compression places an extra burden of computation time upon the processing of compressed images. This is because the images must first be decompressed before being processed. When the images are large, as in the case of many photogrammetric images, which are typically over 100 Mbytes for gray scale digitized aerial photographs, the time taken to decompress the image can add significantly to the processing time. Algorithms could be designed so that video compression would not be required.

***Intercom Stations*:** Each area in the greenhouse will contain intercom stations. They will be small in size and transmit their audio via microwaves. The number of intercom stations needed is estimated at 15. Seven of these will be used as spares. Station personnel will also be required to wear on person wireless communication modules at all times. The total number of modules required is 15.

***Controllers*:** Electrical elements, such as sensors will be fed into a controller. The sensor-input voltage will be from 0-10 volts. In response to commands from the PC stations, the controller will output the necessary control voltages to the pumps, filters, motors and blowers in the greenhouse. The controller will use two VMIC2700 data acquisition boards. The total number of inputs needed per controller will be 30.

***Sensors*:** The sensors that will be used will provide a 0-10 Volt signal to the controller in each area. [26]

***Communications Systems*:** The stations will posses several communication systems. One will be a microwave-based radio. The very long length of communication distance between Mars and Earth presents some problems. One is that, even at the speed of light, it will take 10 minutes for signals to reach the Earth from Mars and vice versa. This means that the mode of communication that will be used is on a one person sending only basis. They will have to wait 20 minutes minimum for a reply. [27]

***Voice Recognition Systems*:** To make the astronauts work involving computer stations easier and less time consuming, a voice recognition system should be used. Speech Recognition (SR) software is software that has the ability to audibly detect human speech and parse that speech in order to generate a string of words, sounds or phonemes to represent what the person said. Natural Language Processing (NLP) software has the ability to process the output from Speech Recognition software and understand what the user meant. The NLP software could then translate what it believes to be the user's command into an actual machine command announces it, acquire "OK" and execute it. A current problem with voice recognition software is that it is only good for one person. Current technology permits only one individual voice to be recognized per computer station. This is a major problem and does not appear solvable in the next 5 to 10 years. Voice recognition software requires greater than 100 MB of memory for use. Perhaps, a system will be developed where any user can interface audibly with any computer station one individual at a time.

***Airlock Controls*:** There will be one airlock connected to the greenhouse structure. An air pump will infuse O2 into the chamber until atmospheric sensors detect at least a 95/5 ratio of O2 to CO2. A pressure, O2 and CO2 sensors will be required. Motors will open the inside and outside hatches. The process will be able to be controlled from the master control center and set in motion in an automatic cycle. Alarms will actuate when the airlock is in use. The whole processed will be monitored at the master control center, with manual override commands available.

Lighting Systems: The lighting system for the plant growth area will be provided by 400 W high pressure sodium lamps (6 per plant area) which yield an average photosynthetic photon flux of 1500 u mol m S/msqd s when operating at full power. The lamps are powered by dimming ballast (Zone Mate, Widelite Corp, San Marcos TX), which will allow variable light levels for each crop area, which are controlled by the chambers data acquisition recording and control system. The crop will be separated from the lamp bank by means of a polycarbonate plastic sheet barrier. To obtain better environmental conditions for the plants, porthole windows have been designed to let in sunlight. Because the sodium lamps are pressurized, there is a problem with transporting them in a cargo hold exposed to vacuum. Light emitting diode (LED) banks are unpressurized and can produce the necessary light for plant growth. Further study on using LED banks in necessary.

Fig. 6-1: Controls

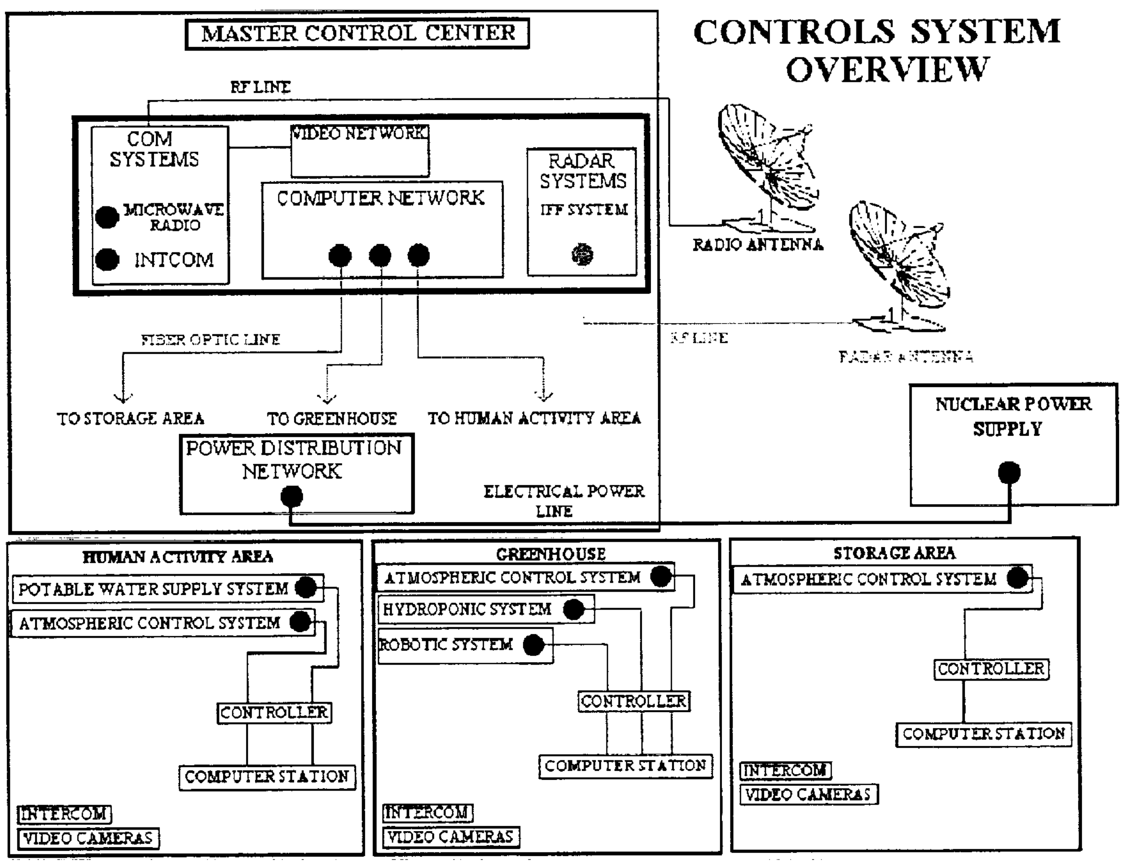
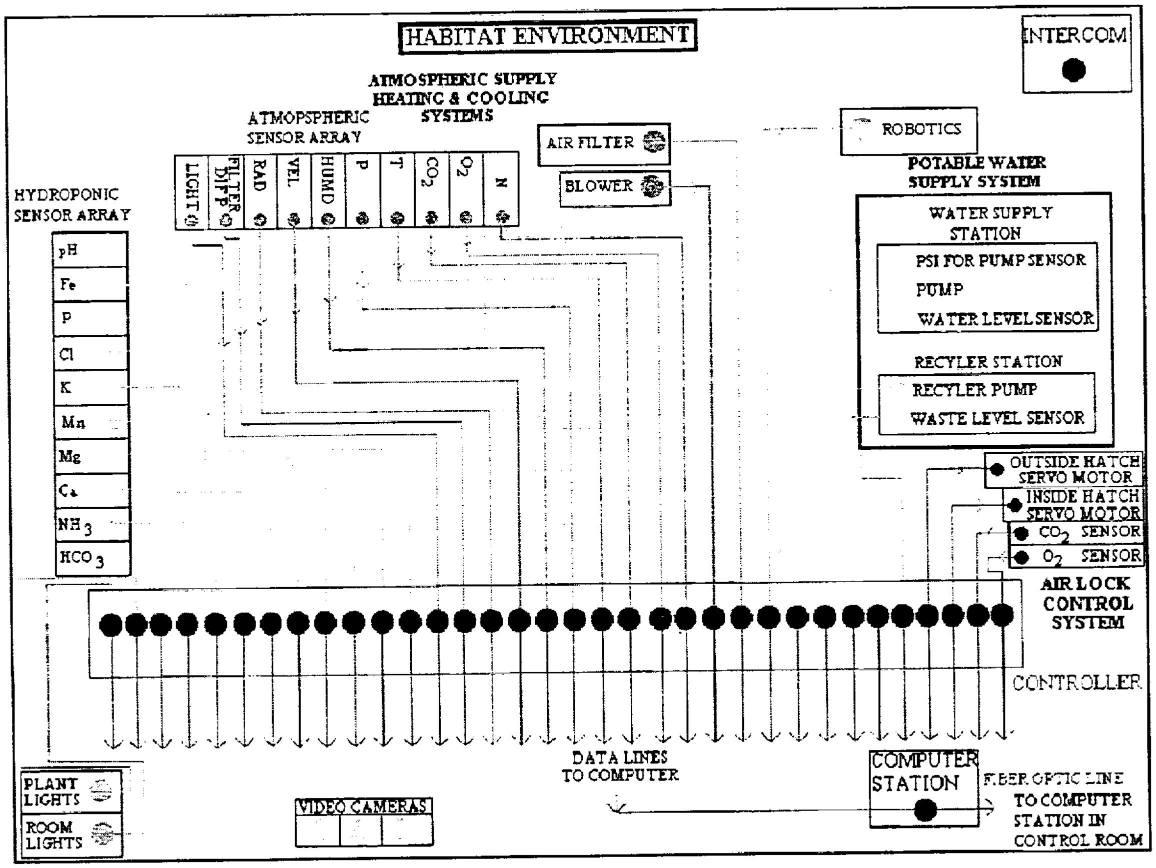


Fig.6-2: Data Line Layout



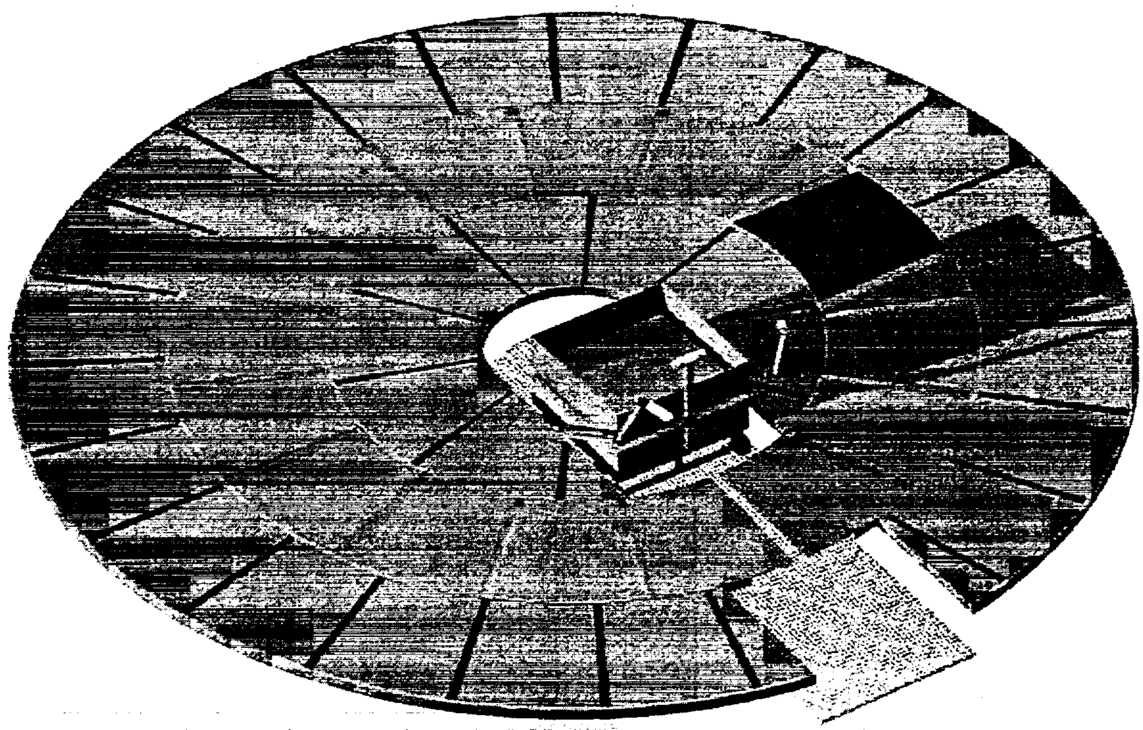
7. AUTONOMOUS ROBOTIC HARVESTER

The autonomous robotic harvester will perform farming duties in the MAGIC greenhouse. The goal is to have the robot monitor the crops and perform the harvesting with minimal human assistance. The robot will have a variety of programs depending on the task required. There will be a manual override, which can be remotely operated, both from Mars and on Earth.

The design of the robot was based on the vertical, rigid cylinder (Phase I) design. The robot will have the capability of elevating approximately 1.5 m. It should have four hydraulic actuators that will elevate and level the robot at all times. This allows the robot to travel in various terrain. The robot should elevate to the different levels of the greenhouse. This elevator can be operated by the robot. The aisles of the elevator will have a maximum width of 1.5 m. The robot will need this space for mobility and versatility. The crop area and the storage/processing area are separate, with storage and processing being done at the ground level. If necessary, the robot should be able to pick the trays up and transport them to the harvest area.

The robot will perform most of the operations at the crop level. Operations such as cutting, sorting, placement of crops, planting of new seeds and cleaning of the tray will be performed at the crop level. These operations will be accomplished by the use of two versatile mobile arms. The arm will have interchangeable tooling. The robot will be programmed to interchange the tools by itself. It will have the capability to remove the trays from its location if necessary. This task will be accomplished by using a lift mechanism, which will pick up the trays and then transport them to the harvest area. The robot should have a lift mechanism that will allow it to carry a load, for example a basket, so that it can place crops after it performs the desired operation.

Fig. 7-1: Rotating Trays and The Robotic 'Farmer'



PHASE II

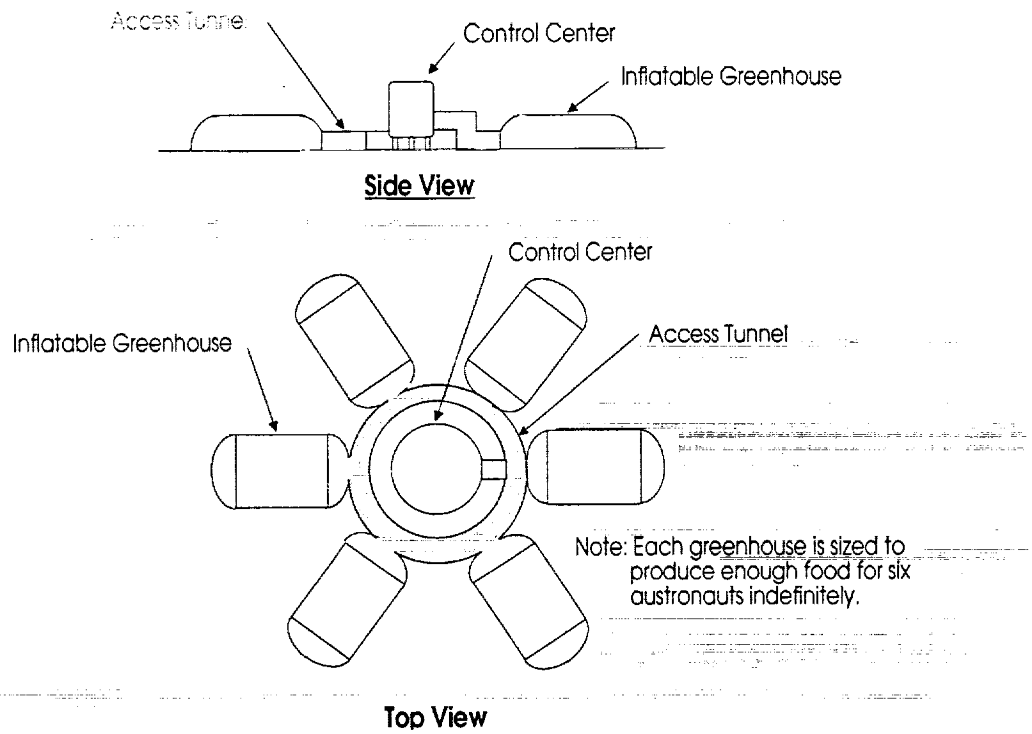
Phase II of MAGIC is a conceptual design for a future extension of the food production facility to support a growing population, beyond the constraints of the reference mission. Using a single rigid structure as a central control and harvesting center, large, tent-like greenhouses could be arranged in a spoked pattern, as seen figure 8-1.

8. PHASE II AND THE MARTIAN REGOLITH

Some compounds found in Martian regolith are also found in Earth's continental crust at comparable amounts. Martian regolith contains some compounds that are not found in Earth's continental crust. Some of these compounds include Na_2O, SO_3, and Cl. Reference [1a] outlines the composition of oxides in weight percent of several Martian sites. A comparison of the Earth's continental crust is also listed. Use of Martian regolith in Phase II will require chemical fertilization.

Phase II of the plan incorporates one expended cargo vessel as the control center serving, and as many as six inflatable low-pressure greenhouses. Each of the inflatable units will have the capacity to serve six astronauts. Thirty-six crewmembers could be continuously served. Figure 8-1 is provided as an illustration of the Phase II concept.

Fig. 8-1: Phase II Concept



9. FUTURE STUDIES

***Mars In-situ Manufacture Of Structural Materials*:** Over a long-term presence on Mars, it is expected that there will eventually exist many more plant waste parts (stems, leaves, and roots) than are needed for conversion to plant nutrients. There is a company in Amarillo, Texas that is currently producing wood-like structural materials out of the waste parts of wheat plants. The processing of plant waste into composite materials could have advantages to a Martian colony and should be investigated.

***Inflatable Structures*:** Inflatable structures provide a greater volume for plant growth than the rigid cylinders selected for Phase I of MAGIC. The inflatable habitat concept (TransHab) currently under consideration for the International Space Station should be investigated further for use as a possible replacement for the rigid cylinders.

***Use of Indigenous Martian CO_2 for Plant and Human Life*:** The concentration of CO_2 in the Martian atmosphere is too high to support plant life. Mars has a preponderance of CO_2 and practically none of the carbon and oxygen necessary for life. A process or mechanism devoted to converting CO_2 into the basic elements of carbon and oxygen needs to be developed. [1a]

***Greenhouse Thermal Protection*:** Thermal analysis should be done on the effects of the extremely low temperatures of Mars on the greenhouse facilities. Much can be done to optimize the maintenance of plants in a healthy thermal environment. Simple, reliable, and effective methods of achieving this should be pursued.

***The Robotic Farmer*:** The autonomous robotic harvester has a mission to replace human activities for tending the greenhouse and processing plant food. The robot should ideally be able to seed and tend plants prior to the arrival of humans and have the facility in full production when humans arrive. It then should continue the harvesting and food processing duties during human presence. Trade-off analysis should be performed to weigh the cost of developing and maintaining the robot against the need for the inhabitants to devote their time to other duties. [1a]

Martian Crop Development: New breeds of plants need to be developed that have high productive capacities and minimal foliage. This development would be relevant for missions of long duration on Mars. Improvements in the productive capacity of wheat, with reduced foliage compared to typical wheat stocks, have been achieved on Earth. Such improvements may be attainable with other plants as well.

Further Development of MAGIC, Phase II: Phase II of our study calls for the use of the Martian soil in lieu of hydroponics, and a crop arrangement not much different than that used on Earth. This quasi-terraforming concept is a challenge in, among others, the issues of materials, heat transfer, gas leakage, and crop choice. Further information on the nature of the Martian soil will be instrumental in the development of this phase. [1a]

10. LESSONS

For many of the students participating in this project, MAGIC was the first opportunity to experience the design of a large scale, complex system. Working in small teams, as part of a larger team, with common goals requires an effective, reliable system of communication among the members and the individual teams. Ideally, this system should be in place before even the first design parameter is established. This reduces the number of redundant tasks being performed and ensures that everyone understands what is required of them. A constantly updated interactive web site can assist in this communication.

11. OUTREACH

Several actions were taken to ensure exposure of MAGIC to the community at large, so that others may learn from the challenges encountered during the project. Following the conference at the Lunar and Planetary Institute, a newsletter circulated at UTSA as well as the student newspaper will publish articles about MAGIC. In addition, a local television station may cover the project. The posters that were developed for the conference will be on display for high school students and parents during "Engineering Week", hosted by engineering students at the university. Finally, the World Wide Web site, which was developed for intercommunication among the MAGIC design teams, will become a permanent part of the UTSA division of engineering web site. This will allow material to be accessed and referenced by those working on further aspects of the design, or on similar projects.

12. CONCLUSION

MAGIC addressed the prospect of developing a food production facility on Mars by implementing a two-phase, expanding approach. Each individual design challenge involved varying levels of trade off analyses, projections, and assumptions. As further information is gained on the nature of the Martian atmosphere, soil, and climate - designs can be refined and finalized. Furthermore, advances in gas production and water production technology, as well as advances in robotic autonomy, will improve the design of each individual subsystem. Like the original pioneers of the American frontier, future astronauts will need to learn how to "live off the land" as they venture further away from home. MAGIC is the first bold step in that direction.

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MARVIN:
MARtian Vehicular INvestigator
A Proposal for a Long-range Pressurized Rover

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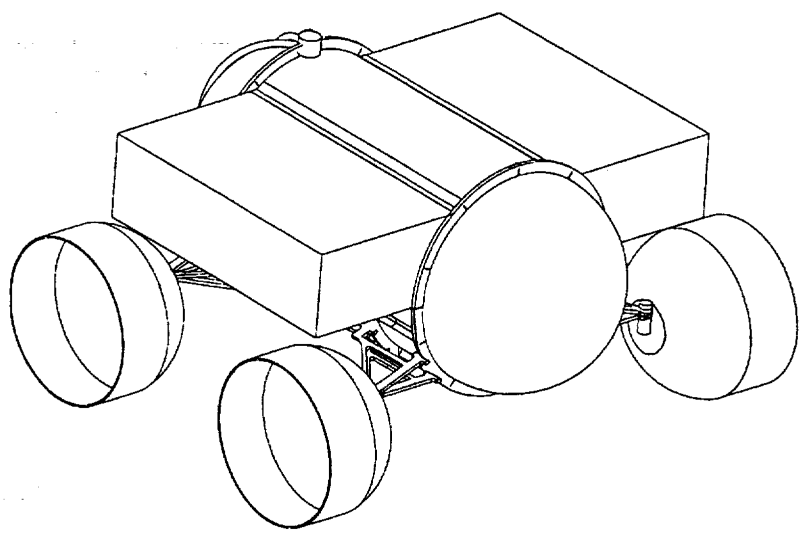


Front Quartering View of MARVIN

Abstract
NASA is planning manned missions to Mars in the near future. In order to fully exploit the available time on the surface for exploration, a roving vehicle is necessary. A nine-member student design team from the Wichita State University Department of Aerospace Engineering developed the MARtian Vehicular INvestigator (MARVIN) a manned, pressurized, long distance rover. In order to meet the unique requirements for successful operation in the harsh Martian environment a four wheeled, rover was designed with a composite pressure vessel six meters long and 2.5 meters in diameter. The rover is powered by twin proton exchange membrane fuel cells which provide electricity to the drive motors and onboard systems The MARVIN concept is expected to have a 1500 km range with a maximum speed of 25 km/hr and a 14-day endurance.

1 Introduction

NASA's reference mission calls for means of exploring Mars's surface. A pressurized rover seems fitted for the task. Therefore, our team of nine students undertook the conceptual design of such a rover, as a requirement for a two-semester design class at Wichita State University. As additional challenges, the team decided to participate in the HEDSUP competition and build a $1/8^{th}$-scale model of the rover to use in an outreach program.

1.1 Mission

The MARtian Vehicular INvestigator's (MARVIN's) mission will be twofold. First, as an exploratory vehicle, it will be used to collect, photograph and analyze samples from the Martian surface. Secondly, it will provide the unique opportunity to study human behavior in an unfamiliar and inhospitable environment. This paper deals only with the exploratory aspect of MARVIN's mission. MARVIN will carry a crew of two to three people over 1500 km, during a two-week mission. At an average speed of 15 to 20 km/h, the crew will be driving approximately 6 hours per day.

The design of MARVIN was based on several assumptions. First, fuel was assumed to be readily available on Mars upon arrival of a manned mission. This would be done by producing large amounts of oxygen, hydrogen, water, and other fuels such as methane in situ. The in situ fuel production process will have started several years before the arrival of man on Mars. It was also assumed that the infrastructures necessary to partially assemble the rover would be available on Mars. In addition, no provisions were made for transport to Mars except to assume that the rover could be delivered to the Martian surface in a nearly assembled state.

The MARVIN rover Project was undertaken strictly as a conceptual project. No detailed design or analysis, nor a cost analysis was performed.

1.2 Design Criteria

Most of the design criteria are dictated by the hostile nature of the Martian environment and by the autonomy required in a place millions of kilometers away from home. Hence a Martian rover has to be rugged, lightweight, and have redundant systems. Mars presents several unique design challenges. First, Mars's atmosphere is composed almost entirely of carbon dioxide, which is toxic to humans and renders all surface operations all the more difficult. Also, very fine (≈1micron) surface dust is carried about by surface winds and has the potential to foul electronics, windows, filters, and moving parts. Temperatures on the surface vary widely between day and night and also vary widely over short vertical distances above the surface. These variations have the potential to cause material failures and necessitate careful material selection during design.

Transporting the rover from Earth to Mars also imposes its own constraints on the design. The rover must be both compact and lightweight due to launch cost and volume limitations. To meet the volume constraint, the rover was designed to be partially assembled on Mars. Advanced materials such as composites and titanium were used to satisfy the mass requirements.

In the following sections, our solutions to these unique constraints are presented and explained.

2 Pressure Vessel Structure

2.1 General Dimensions

The rover structure consists of a pressure vessel with a cylindrical mid-section and two semi-spherical end-caps. The cylindrical mid-section has an outside diameter of 2.5 meters, the two semi-spherical end caps have an outside radius of 1.25 meters each, and the overall length of the vessel is 6 meters. The thickness of the walls is approximately 3 cm. The rear end cap will be partitioned for use as an air lock, and the front-end cap will contain the cockpit and controls. In addition to a windshield in front, dome windows on the top of the cylindrical section will aid in natural interior lighting.

2.2 Pressure Vessel Materials and Structure

2.2.1 Composite Vessel Materials

Advanced composite materials are well suited for space structures due to their high strength and stiffness, light weight and low coefficient of thermal expansion (CTE). For a given design application, composites can be tailored to achieve the desired material properties. The composite fibers carry the structural loads in the direction of the fiber, and the composite matrix holds the fibers together, aids in transferring loads and provides environmental protection.

The main structural fibers of the pressure vessel will be carbon. Carbon is chosen due to its high specific strength, high specific modulus, low CTE and high fatigue strength. Aramid Kevlar 49® is an organic fiber that will be used for the outer surface of the pressure vessel. It exhibits a high degree of yielding in compression giving it superior damage tolerance and resistance to impact and other dynamic loading. It also has low weight, good tensile strength and is fire retardant. The fatigue endurance limit for aramid and carbon fiber reinforced epoxies may approach 60% of the ultimate tensile strength

2.2.2 *Sandwich Structure*

A tape-laying process will be used to construct the vessel in a sandwich structure with a honeycomb core. Filament winding is commonly used for cylindrical shapes such as pressure vessels. An automated fiber placement machine uses a mandrel similar to filament winding, but will allow for a more detailed lay-up in complex areas such as those around the windows and at the bottom cradle attachment area.

The laminate will be symmetric about the mid-plane to avoid bending and twisting due to in-plane loads. Since Kevlar plies will be used on the outside of the structure, the same Kevlar lay-up will also be used on the inside walls. In addition to creating symmetry, the inside Kevlar plies will separate the carbon plies from metal support brackets, which will prevent galvanic corrosion. Composite plies will exist in pairs of opposite orientation to avoid shear distortions due to normal loads. Since Kevlar is non-conductive, fine aluminum filaments will be woven in the outer fabric to dissipate electrostatic energy generated by high winds. These filaments, along with a metallic clear-coat, will also serve to reflect ultraviolet radiation, which contributes to the decomposition of Kevlar. The basic lay-up configuration of the pressure vessel is shown in Figure 2.1.

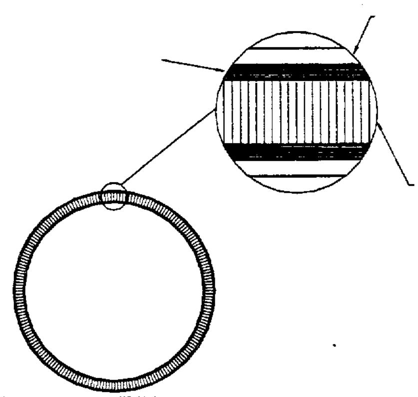


Figure 2.1: Lay-up Configuration

Between the carbon layers, a fiberglass reinforced phenolic honeycomb will be bonded to the carbon with film adhesive. This material has excellent thermal stability and has been used from -423°F up to 400°F. The honeycomb increases the structure's rigidity, its ability to withstand bending and shear loads, and has good strength to weight and stiffness to weight ratios. It provides acoustic absorption, radio frequency shielding, and it is an extremely effective mechanical energy absorber. The faces of a sandwich panel act similarly to the flanges of an I-beam by taking the bending loads. The core corresponds to the web, resisting the shear loads and increasing the stiffness of the structure. Unlike the I-beam's web, it gives continuous support to the facings. The adhesive joins the components and allows them to act as one unit with high torsional rigidity.

The honeycomb at the lower portion of the vessel will be filled with potting compound to distribute localized loads. This type of compound has a typical density of 0.189 g/in^3 and a compressive strength of 17 ksi. Since the average surface temperature on the Martian surface is -65°C, the remainder of the honeycomb will be filled as necessary to aid in internal thermal equilibrium to maintain a "shirt sleeve" environment. Polyethylene is chosen to fill the core since it also provides radiation protection. It also will aid in core stabilization, stiffening and core crush reinforcement.

2.3 *Lay-up Configuration*

The pressure vessel will need to be able to contain the pressure differential and to withstand the stresses due to the internal loads. The ply-stacking configuration will be designed to allow fibers orientated in the axial and hoop directions, as well as off axis layers to provide some circumferential and torsional stiffness and to hold the load carrying layers together.

The critical stress resultant can be calculated using a form of Hooke's Law:

$$N_{xx} = E_{xx} \cdot \varepsilon_{xx} \cdot n \cdot t_p$$

where N_{xx}= the critical stress resultant
E_{xx}= the longitudinal modulus of the material
ε_{xx}= the allowable strain
n = the number of plies
t_p = the thickness of each ply

Using a maximum strain allowable of 0.003 and a carbon unitape with a longitudinal tensile modulus of 18 Msi, 2.5 plies with fibers in the hoop direction are necessary to withstand the hoop stress due to the pressure loads.

As a starting point, a factor of safety of 5 is applied. As a result there are 12 plies with fibers in the hoop direction, 6 plies in the axial direction, and 4 plies are added off-axis to provide torsional stiffness. The plies are arranged symmetrically without grouping the same oriented plies. When plies of the same orientation are grouped together, there are higher interlaminar stresses. Eight Kevlar plies are added for impact resistance. Table 2.1 shows the initial lay-up configuration.

Table 2.1: Composite Layup

	Number of Plies	Thickness per Ply (cm)	Orientation Sequence
Kevlar	4	0.0254	0/+45/-45/90
Carbon	11	0.0127	0/90/0/+45/0/90/0/-45/0/90/0
Honeycomb	1	2.5	1 layer
Carbon	11	0.0127	0/90/0/-45/0/90/0/+45/0/90/0
Kevlar	4	0.0254	90/-45/+45/0

The configuration was put in a classical lamination program with the pressure loads applied. Using the Tsai-Wu failure criteria, all plies have a factor of safety of 7.4 to 11.5.

2.4 *Mass Calculations*

Using the dimensions of the pressure vessel and ply thicknesses of 0.0254 cm for the Kevlar, 0.0127 for carbon unitape, and a honey comb thickness of 2.5 cm, the following mass table was generated.

Table 2.2: Mass Calculations

Material	Outer Radius (cm)	Inner Radius (cm)	Panel Thickness (cm)	Volume (cm^3)	Density (g/cm^3)	Mass (kg)
Kevlar	125.00	124.90	0.102	47850	1.33	63.64
Carbon	124.90	124.76	0.140	65700	1.54	101.20
Honeycomb	124.76	122.26	2.500	1158000	0.048	55.60
Carbon	122.26	122.12	0.140	63750	1.54	98.17
Kevlar	122.12	122.02	0.102	46300	1.33	61.58
Total	--	--	--	--	--	380.19

2.5 *Radiation Shielding for MARVIN*

Studies have shown that in order to achieve necessary radiation shielding for a maximum exposure of 20 mSv/year , 0.15cm of aluminum or 0.3-cm protection of Polyethylene is needed. Polyethylene is used to fill the honeycomb core and therefore provide a good thermal insulator as well as a significant radiation protection.

3 Cradle Structure

3.1 *Material Selection*

The cradle structure will be made of titanium. Specifically, Ti-6Al-4V alloy. This alloy can withstand large temperature gradients, has an low CTE, and 3 times the strength to weight ratio of aluminum. A suitable paint/coating will further enhance the corrosive properties of Ti-6Al-4V and increase the structure's ability to resist the damaging effects of the Martian environment.

3.2 *Results of Stress Analysis*

MARVIN has five (5) longerons and five (5) frames. The longerons were given a T shape to reduce critical stresses. The frames were designed to resist torsion as much as possible, and an "r" shape was used for them., and the whole assembly minus the ears is shown in Figure 3.1.

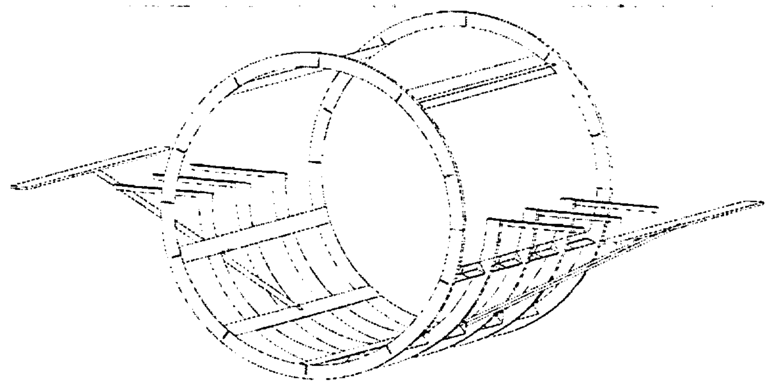


Figure 3.1: Structure with Ear Attachments

3.3 External Storage System

The rover's power production and storage systems are stored outside the pressure vessel in two rectangular containers or "ears". The fuel cell dimensions dominated the overall geometry of the ears. The rectangular structure runs the length of the cylindrical part of the pressure vessel and is 1.5 meters in width and 0.75 meters high. MARVIN has two identical ears, one on each side of the pressure vessel. The ears extend inboard to the contour of the pressure vessel and the extra volume is used for water storage tanks. Figure 3.2 shows the general geometry and inside arrangement of the ears.

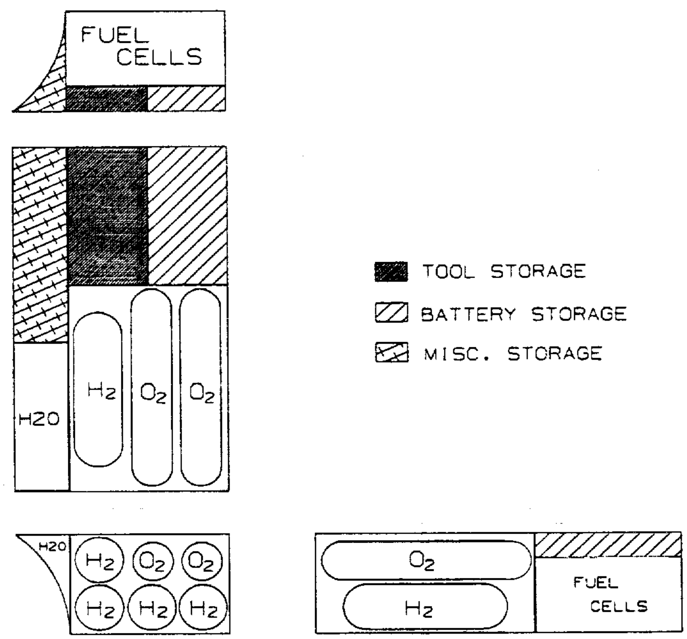


Figure 3.2: Internal Configuration of the Ears

4 Powerplant

4.1 Power Requirements

4.1.1 Mission Definition

The energy requirements for three reference missions were calculated. The first mission represents a 750 km-drive on flat terrain. The second mission has 500 km long climbing and descending portions at an angle of 25 degrees and a 500-km long flat portion. The third reference mission has 750 km long climbing and descending portions at an angle of 10 degrees. Figure 4.1 is an illustration of these three reference missions.

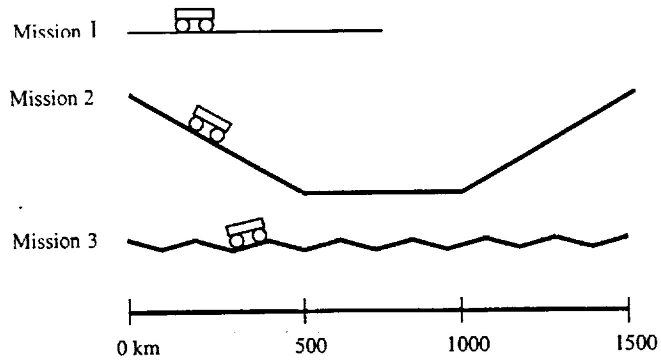


Figure 4.1: Reference Missions Profiles

4.1.2 Energy and Power Requirements

The following simplified equation was used to calculate the energy requirement for each mission:

$$dE = Fdx$$

Where dE is the total energy required to overcome the resistive force F over a distance dx. The resistive force is approximated as follows:

$$F = mg_M(\mu + \sin\theta)$$

Where m is the Rover's mass, g_M is Mars'gravity, μ is the surface's friction coefficient and θ is the terrain's slope. Aerodynamic drag was voluntarily omitted from this equation. Drag's contribution to the resistive force is about 2kN in a 300 km/hr Martian storm, which is negligible. The total mass of the rover is 4000 kg and μ was approximated at 0.5, which gives the following energy requirements:

Table 4.1: Energy and Power Requirements

Mission	Energy Requirement (MJ)	Power Requirement (kW)
1	5592	37
2	11184	37
3	11184	37
Max Slope @ Max Speed		69 (Max power required)

Therefore, the powerplant's power output has to be about 40 kW continuous and 70 kW peak. Note that these power requirements are only for the drivetrain. Hence they do not include the powering of MARVIN's internal systems and external tools and accessories.

4.2 Power Generation

4.2.1 Powerplant Selection

In order to satisfy to mission requirements, a fuel cell was chosen to power our rover. The low-temperature fuel cells (alkaline and Proton Exchange Membrane (PEM)) are favored. The alkaline fuel cell, however, is CO_2-intolerant, which makes it a poor choice for Mars's CO_2 atmosphere. As the following table illustrates, the logical choice is the PEM fuel cell, because it operates at low temperatures, is CO_2-tolerant and has a high energy density.

Table 4.2: Fuel Cell Comparison

Fuel Cell Type	Operating Temp. (°C)	Power Density (kW/liter)	Power Density (kW/kg)	CO_2 tolerant	CO tolerant
Solid Oxide	1000	1-4	1-8	Yes	Good
Molten Carbonate	600	-	-	Yes	Good
Phosphoric Acid	150-205	0.16	0.12	Yes	Fair
Alkaline	65-220	0.1-1.5	0.1-1.5	No	Poor
PEM	25-120	0.1-1.5	0.1-1.5	Yes	Poor

4.2.2 *Operating Principle of a PEM Fuel Cell*

In any kind of fuel cell, the combination of hydrogen and oxygen creates electricity and water. The reaction at the anode is:

$$H_2 \Rightarrow 2H^+ + 2e^-$$

The electrons travel to the cathode through an external circuit where they perform work. The protons travel through the proton exchange membrane to the cathode. There, the protons and the electrons combine with the oxygen to form water, according to the following reaction:

$$\frac{1}{2}O_2 + 2H^+ + 2e^- \Rightarrow H_2O$$

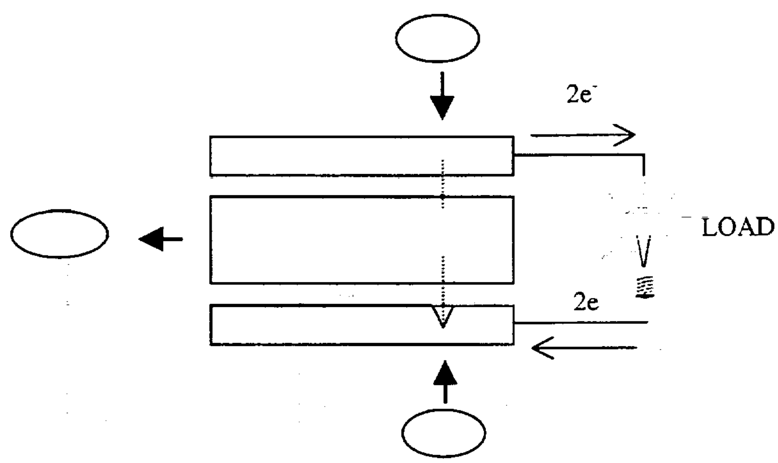


Figure 4.2: Working Principle of a PEM Fuel Cell

4.3 *Specifications of a PEM Fuel Cell Powerplant for MARVIN*

4.3.1 *Weight and Dimensions*

The best laboratory PEM fuel cells currently have power densities of 1.5 kW/kg and 1.5 kW/liter. The observed efficiency for a PEM fuel cell ranges from 50 to 90% . Therefore, assuming a fuel cell efficiency of 80 % and a drivetrain efficiency of 80 %, the system's nominal power has to be: 62.5kW, and a maximum power of 110 kW

Allowing for the rover's internal and external systems, the total power requirement is 83 kW. Therefore, a PEM fuel cell with $P_n = 85$ kW and $P_{\max} = 150$ kW was selected. Such a fuel cell would weigh 150 kg and occupy a volume of 0.15 m^3.

4.3.2 *Fuel Requirement*

The fuel requirement calculations are based on the "Ballard Fuel Cell Powered ZEV Bus". In that design, a 120 kW PEM fuel cells powers a city bus with a range of 160 km on 12 kg of hydrogen and oxygen in 2:1 stoechiometric proportion. The hydrogen needed to achieve a range of 1500 km can be estimated at 120 kg. The volume of H_2 can therefore be calculated to be 1700 liters. In order to satisfy the chemical reaction's proportions, the volume of oxygen is 750 liters with a mass of 960 kg.

4.3.3 *Powerplant Specifications*

The following table summarizes the specifications of MARVIN's PEM fuel cell powerplant:

Table 4.3: Fuel Cell Powerplant Specifications

Stack Weight	150 kg
Stack Volume	0.15 m^3
Hydrogen Mass/Volume	120 kg / 1.7 m^3
Hydrogen Storage Temperature/Pressure	20 K / 0.1 MPa
Hydrogen Delivery Pressure	0.3 MPa
Oxygen Mass/Volume	960 kg / 0.75 m^3
Oxygen Storage Temperature/Pressure	60 K / 0.1 MPa
Oxygen Delivery Pressure	0.3 MPa
Continuous Power Output (kW)	85
Maximum Power Output (kW)	150
Output Voltage (Vdc)	160 – 280
Stack Efficiency	0.8
Drivetrain Efficiency	0.8

4.4 *Powerplant Subsystems*

The following figure represents the power plant and most of its associated subsystems:

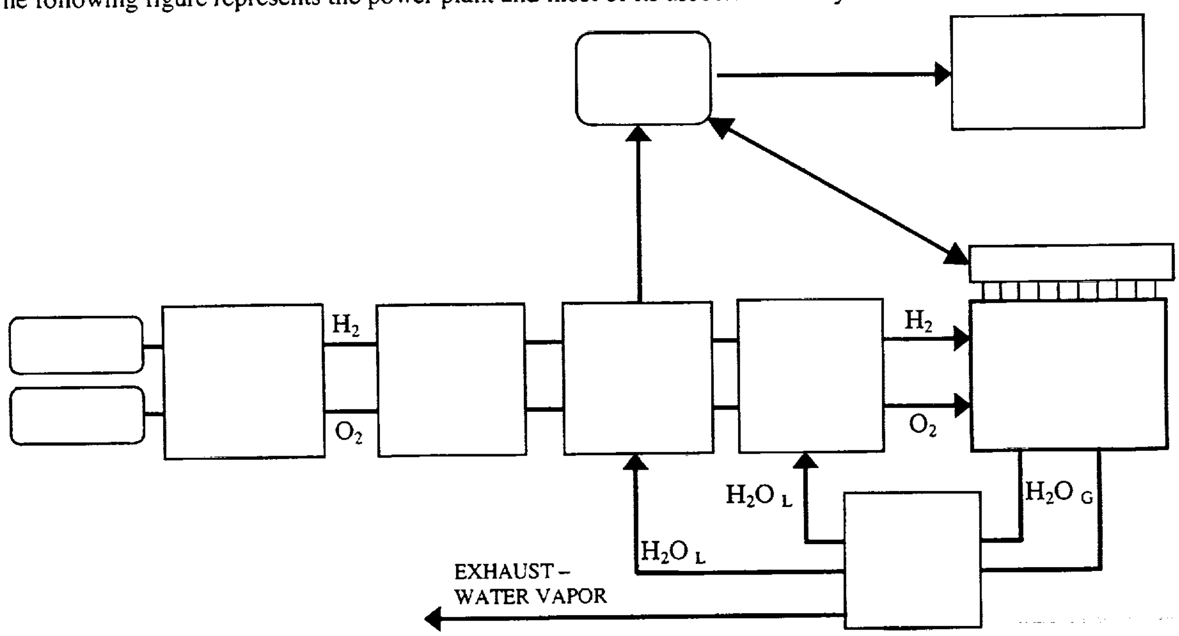


Figure 4.3: PEM Fuel Cell Powerplant and Associated Subsystems

4.4.1 *Electrical Subsystem/Battery*

A 400-kg Al/Air battery would provide the rover with 2kW of continuous power for 48 hours and provide starting power to the fuel cell

4.4.2 *Water Management System*

An 80 % efficient fuel cell produces approximately 7.3 kg of water for each kg of hydrogen it uses. Therefore, during the 14-day mission, the fuel cell will produce around 900 kg of water. About half of it will be used as potable supply. Some of it will be exhausted as water vapor in the Martian atmosphere and the rest, in liquid form, will be used as a coolant. Some of the vapor is released into the atmosphere and the rest goes through a condenser. The resulting warm liquid water is used in a heat exchanger that regulates hydrogen and oxygen temperatures and in the humidifier. The cold liquid water is then distributed between an electrolysis tank, the potable supply and the fuel cell cooling system. Water is produced by the fuel cell only as the rover is running. Therefore, there will need to be an initial amount of water in MARVIN's tank upon departure for a new mission.

5 Suspension and Traction System

5.1 Terrain Features

Photographs from the recent NASA Pathfinder mission clearly show the harsh terrain that surrounds the Pathfinder landing site. The surface is covered by a mixture of fine sandy soil and small pebbles, and strewn with stones ranging from gravel to boulder. The terrain shown in Figure 5.1 is likely the terrain that could be encountered by MARVIN during its missions

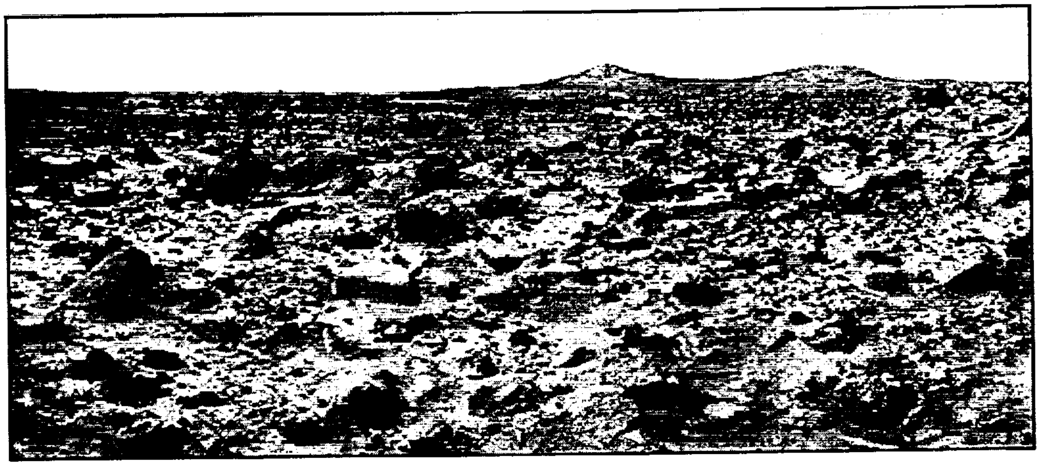


Figure 5.1 Pathfinder Photo of Martian Surface

5.2 Wheels

The wheels need to be reliable and easily maintainable by astronauts in EVA suits. Also, the selected wheel design must be able to provide sufficient traction to successfully move the rover over the terrain. In light of the mission design criteria composite cone type wheels were chosen.

5.2.1 Design and Material Selection

The cone type wheel geometry offers several benefits. The wheels can be closely stacked onto each other and fitted over one end of the pressure vessel during interplanetary transport thus conserving payload volume. The cone type wheel can also be constructed in a very lightweight manner.

Kevlar could be used to construct the entire wheel or just the outer layers for abrasion and impact resistance. Honeycomb and fiberglass are perfect materials for strengthening and stiffening while still keeping mass low.

In order to maintain the wheel shape, stiffening protrusions are added inside the tread surface. The flat, hub portion of the wheel is to be used for motor mounting and as such will require much stiffness in order to make motor mounting feasible. Honeycomb and fiberglass can also be used in this area to maintain bending strength. The proposed wheel cross-section is shown in Figure 5.2.

Wheel traction can be aided by attaching protrusions onto the tread surface that can be tailored exclusively to the type of terrain expected to be encountered during a mission. Similar to specialized snow or mud tires used on Earth-based terrain vehicles.

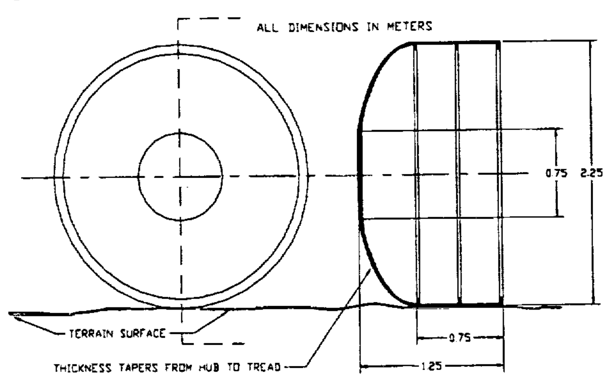


Figure 5.2: Wheel Cross-Section

A limiting factor in the wheel sizing is the physical space the wheel has in which to operate. A minimum ground clearance of one meter is expected and the wheel must not interfere with the ear structures that contain the rover's power generation and fuel storage facilities. When these constraints are balanced with the desire for high suspension travel the acceptable range of wheel diameters becomes clear. MARVIN's wheel diameter of 2.25 meters balances all these needs and falls in the middle of the acceptable values for wheel diameter.

The two types of terrain vehicle clearance failure modes are hang-up failure (HUF) and nose-in failure (NIF). Hang up failure occurs when the terrain contour changes suddenly and the middle of the vehicle between the wheels interferes with the terrain. Nose-in failure occurs when the front (or rear) of the vehicle interferes with the terrain during a sudden contour change

To avoid these failure modes the wheels should be of sufficient diameter and near enough to the ends of the vehicle to prevent terrain vehicle interference. MARVIN's wheelbase and wheel diameter are both sufficient to provide adequate protection against these failure modes during average terrain encounters. Terrain that could cause these types of failure modes in MARVIN is also high risk terrain for causing rollover failure.

5.2.2 *Ground Pressure Analysis*

The soil sinkage analysis is very simplistic and considers the static loading case. The equation in Figure 5.3 describes rover sinkage in terms of ground pressure and two soil constants, the Bernstein's modulus of soil deformation and the exponent of soil deformation. These constants are obtained through testing of the soil. The approximate values used are 0.50 for the soil deformation exponent and 39,000 N/m^3 for the Bernstein's modulus of soil deformation. Based on the curve, the rover is expected to sink between two and seven centimeters into the Martian soil when static. These values are quite acceptable and validate the wheel sizing.

Assumptions about wheel slippage were also made in order to calculate MARVIN's turning radius. Essentially, the wheels were assumed to have no slip in turns. Some slip is expected to occur, thus the turning radius calculated for the rover is a best-case turning radius.

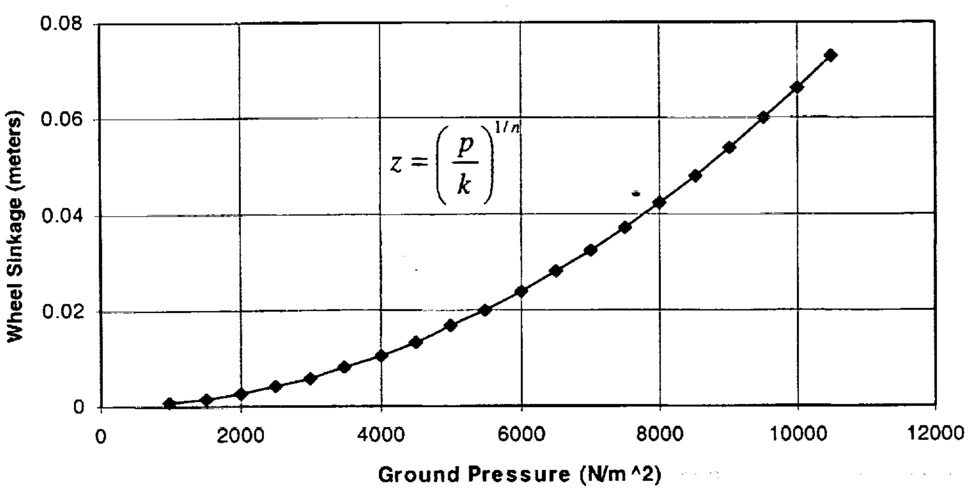


Figure 5.3: Estimated Rover Sinkage versus Ground Pressure

5.3 *Suspension and Steering*

5.3.1 *Four-Wheel Steering*

Ride quality will affect the astronauts both physically and psychologically. Turn radius calculations were simplified by assuming zero wheel slip in turns and by using a simple two-dimensional geometrical model to derive the turn radius equation. Since wheel slippage and lateral wheel separations were not considered, the results obtained can only be considered as 'best case' values.

For the rover's maximum expected steering angle of approximately 35 degrees, the rover has a three-meter turning radius or approximately half the vehicle's length. This is exceptional when compared to most four-wheel Earth based terrain vehicles, which generally have a turn radius larger than the vehicle length. This low turn radius was achieved with four wheel steering.

5.3.2 *Dual A-Arm Suspension*

Dual A-arm suspensions offer many benefits over other terrain vehicle suspension methods. The dual A-arm offers independent motion to each wheel. Thus, each wheel's vertical position over the terrain can be optimized without consideration to the other wheels' positions. In essence, the rover has more freedom to adapt to the varying terrain it is sure to encounter. Also, the 4 wheel independent suspension is very conducive to control by an active damping and wheel actuation system.

One advantage of wheel actuators is the ability to manipulate the pressure vessel's position relative to the ground. When stopping the rover for EVA traverses the vehicle could be positioned so that the rear ladder assembly could easily reach the ground. Also the rover could be leveled for extended parking such as an overnight site stay.

Expected suspension travel is on the order of 0.3meters up or down. The main limitations on suspension travel are the position of the ear assemblies directly above the wheels and the need to keep the pressure vessel relatively close to the ground to minimize potential rollover hazards.

5.4 *Drive Motors*

5.4.1 *Mounting*

The mounting concept we have chosen requires no flexible driveline and maintains the simplicity, ruggedness, and compactness that was required of the drive system. The motor housings are fixed to the wheel hub and rotate with the wheels when driving. The output shaft of the drive system is fixed to the A-arm assembly and serves as the axle for the wheel. This mounting concept is illustrated Figure 5.4.

By placing the motors inside the wheels, a large amount of the rover's weight is placed near the ground and far from the rover's centerline. This increases both the lateral and longitudinal rollover margins by lowering the rover's center of gravity.

5.4.2 *Gearing*

At MARVIN's top speed of 25 km/h, each of the wheels will be turning nearly 60 rpm. Electrical motors typically operate best at much higher rotational speeds. Thus, some type of gearbox is necessary to translate the motor's high rpm rotation into the much lower rotation speed needed for the wheels.

Planetary gears seem to be the most obvious choice, given our unique mounting situation and the need to retain the axially centered shaft position. A planetary gearbox could be placed between the motor and the wheel hub as shown in Figure 5.4. In this configuration, the gearbox also serves as the means of mounting the motor to the wheel hub.

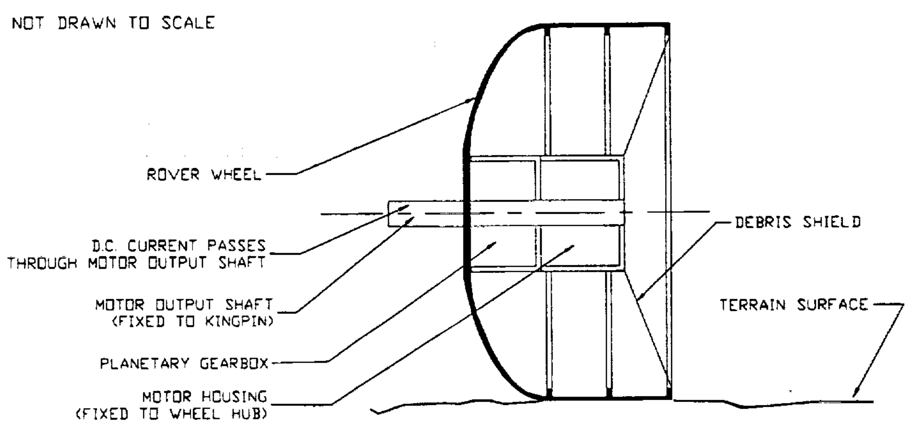


Figure 5.4: Drive Motor and Gearbox Mounting Schematic

6 Cockpit Design

This study was to look at the communication and navigation requirements and determine the basic flight instrument weights. An Integrated Modular Avionics (IMA) system is recognized as providing an answer to the requirements and constraints of modern spacecraft. According to the IMA concept, a system implementation is built up from hardware modules and software components with standardized interfaces. In comparison with the previous generation of federated avionics architectures, the benefits provided by IMA systems will include improved fault-tolerant operation, leading to improved operational and mission performance, as well as a greater openness to growth and innovation, and a reduction of life cycle costs.

The cockpit will be a drive-by-wire system. The design of MARVIN's glass cockpit was based on that of the new space shuttle and Boeing 777. In these cockpits, dual Honeywell aircraft information management system (AIMS) contains the processing equipment required to collect, format and distribute onboard avionics information, including the fight management system (FMS), engine thrust control, digital communications management, operation of cockpit displays and monitoring of MARVIN's condition. Both pilots and ground engineers can assess the condition of all onboard avionics systems.

The Multifunction Electronic Display System (MEDS) is comprised of ten Multifunction Display Units, Integrated Display Processor (IDP), Mass Storage, and Honeywell analog-to-digital (A/D) converters.

Mars is a little more than half the diameter of the Earth, so the horizon is correspondingly closer. If the terrain on Mars were as flat as Kansas, the horizon would only be about 40 kilometers away. So, if the excursion team wants to go anywhere on Mars, they're definitely going over the horizon, which rules out line-of-sight radio transmissions. Communication satellites, on the other hand, cost money, and are subject to failure.

One alternative is ham radio. Mars has an ionosphere, a layer of charged particles in the high upper reaches of its atmosphere, that can be used to reflect radio signals, enabling global surface-to-surface communication in the short-wave radio bands, just as on Earth. According to previous data gathered on Mars's ionosphere, such a radio would operate at about 4 MHz during the day and 700 kHz at night. The latter figure is too low to transmit images or engage in other kinds of high data rate transmission, but it is more than adequate for engineering telemetry or voice communication.

In addition to maintaining communication with the home base, Mars's explorers will also need to navigate. While good maps of Mars are available from orbital imaging, the essential problem for a Mars rover crew will be determining their own location. A radio beacon at the base could help a crew find its way home, but its range would reach at most only to the nearby horizon (just 40 kilometers away). Upon approaching the limits of the base beacon's range, a departing rover crew could station a second beacon on a hilltop, and then another, and another, and another, to mark a return path. Such techniques are, however, quite limiting, and as in the story about the bread-crumb trail being eaten by birds, are subject to catastrophic failure if one of the beacons composing the trail should cease functioning. Inertial navigation and the use of navigation satellites are two other options.

The table shows the total weight for the different instruments and the seats in the cockpit.

Table 6.1: Total Weight for Instruments and Seats

System	Quantity	Total Mass (kg)
SKN-2443 High-Accuracy Inertial Navigation System	2	30
Albus-1553-8 Bus Monitor Data Handling	3	21
Mil-Std-1553B Data Buses	3	21
Multi-Purpose Color Display for F-15C & F-15E	4	48
Wiring and Cooling Fans and Pipes	1	50
Seats	2	100
Total		**270**

The power required for the cockpit is 2kW, plus 4 kW for the exterior lighting and 1 kW for the external cameras, which add up to 7kW.

7 Interior Living Quarters

7.1 MIS Units

Adequate supplies must be carried on MARVIN while leaving storage space for scientific equipment. Since MARVIN's mission objectives will vary, cargo storage with modularity, accessibility and functionality is ideal. MIS (Multipurpose Interior Storage) Units have been developed to accommodate each of MARVIN's missions. The MIS Units are capable of storing all the necessary perishable and expendable items such as water, food, clothing as well as the food preparation equipment. Additionally, there is plenty of storage space available to carry scientific equipment.

The MIS units are located on both sides of the fuselage in the living quarters. (See Figure 7.1) The right MIS unit is 2.97 meters long and runs from the back of the cockpit to the front of the waste management system. The left (not shown) MIS Unit is identical to the right MIS Unit except that it extends the full length of the living quarters, 2.97 meters, from the cockpit to the airlock pressure bulkhead because of the absence of the Waste Management System (See Figure 7.1). Note the person standing inside is 1.83 m, or six feet tall.

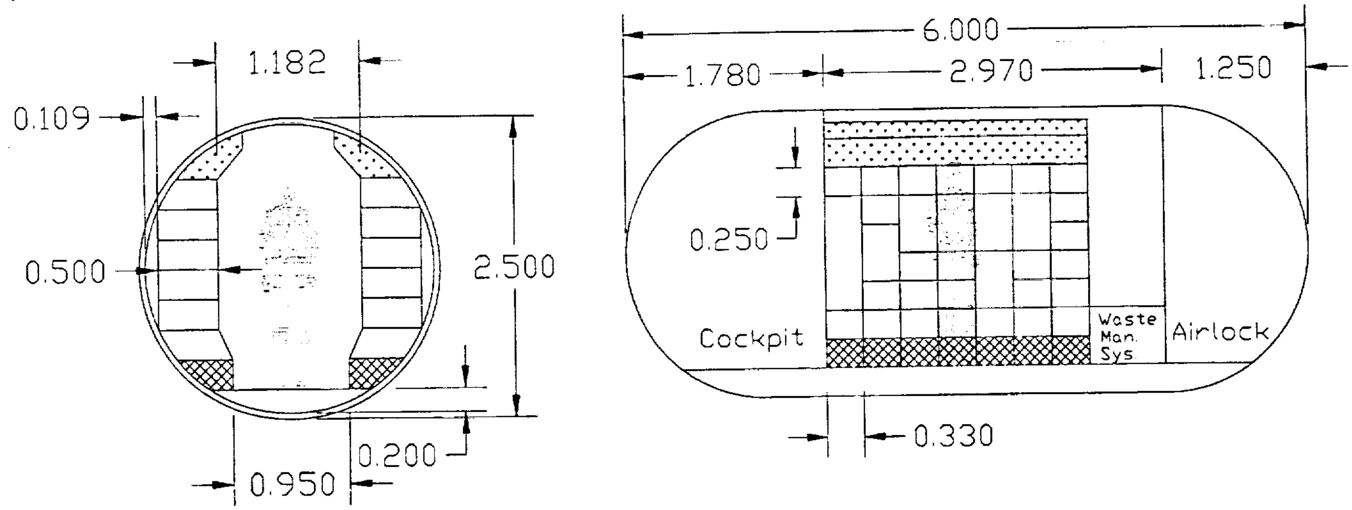


Figure 7.1: Front and Right Side Cross-section View of MARVIN

7.1.1 *Potable and Waste Water Storage*

The fuel cells that power MARVIN will produce approximately 900 kg of water on the two-week mission. The 3-person crew will use approximately 420 kg of this water, based on a 10 kg per person per day. At any one time, enough potable water will be stored in MARVIN to safely bring the crew back to base should a problem arise. Upon return to the base, the majority of the water stored in the MIS Units would be wastewater. This water can then transferred to the base for reclamation of potable water.

7.1.2 *Fire Suppression and Lighting*

The top portion of the MIS Units is where the Fire Extinguishing (FE) and Interior Lighting (IL) Systems are located.. A HALON® or a FE-241(chlorotetrafluoroethane) FE system will be employed on MARVIN.

The IL system will be used to light MARVIN in low light conditions such as a sandstorm or at dusk. Additionally, the IL system will be capable of supporting MARVIN during nighttime missions by illuminating the cabin with red light. It is estimated that this system will require about .25 kW of peak continuous power and .10 kW of power at normal operating conditions.

7.1.3 *Modular Storage Capabilities*

The MIS units are designed to be very modular since MARVIN's mission will be likely different on each excursion. Both the left and the right MIS units are six drawers high, minus the bottom layer for water storage, with the inner four layers being completely interchangeable. Because of MARVIN's geometry, the top and bottom drawer layers are not interchangeable and are used for more permanent equipment such as food preparation equipment, the crew's personal items, etc.... A variety of drawer combinations can be used within these four inner layers. Each inner drawer is .330 meters wide, .250 meters high and .500 meters deep. This gives each inner drawer approximately .041m^3 of storage space. Currently each drawer is fabricated in the same manner as the ones used on the Space Shuttle which employ a Kevlar-Epoxy sandwich structure. The resultant empty mass each drawer on MARVIN is approximately 2 kg.

However, it is conceivable that different missions will require different size equipment and so one(1) double high drawer can replace two(2) single high drawers or one(1) triple high drawer can replace three(3) single high drawers and so on. Electrical connections are provided behind each of the MIS Units for providing power to the scientific, hygiene and food preparation equipment.

The overall storage capacity, minus water storage, of the left MIS Unit is ~2.16 m^3 while the right MIS Unit is ~1.68 m^3 and is slightly smaller because of the presence of the Waste Management System. This gives MARVIN a total internal storage capacity of ~3.84 m^3.

7.2 *Life Support and Environmental Equipment*

The Cabin Air Conditioning (CAC) system and the Pressure Regulation (PR) system are necessary to keep MARVIN's cabin air clean, cool and semi-humid as well as pressurized. Below the main floor is where the CAC and PR systems are located. Raising panels located in the floor can access these systems.

The CAC controls the environment inside MARVIN to approximately 20 °C with 25 to 50% relative humidity. Lithium Hydroxide filters in the CAC clean the air of deadly carbon dioxide and other airborne contaniments while fans and ducting circulate the air through MARVIN.

7.3 Sleeping Arrangements

The shape of MARVIN was predetermined before serious work and research began on the interior of MARVIN. Sleeping arrangements had to be ultimately developed without compromising the MIS Units and the limited space left inside MARVIN. This was accomplished by using a conventional bed system that could be broken down and stored during the daytime when not in use. Figure 7.2 shows a cross section of the Sleeping Bunks (SB) in place for a three-person crew.

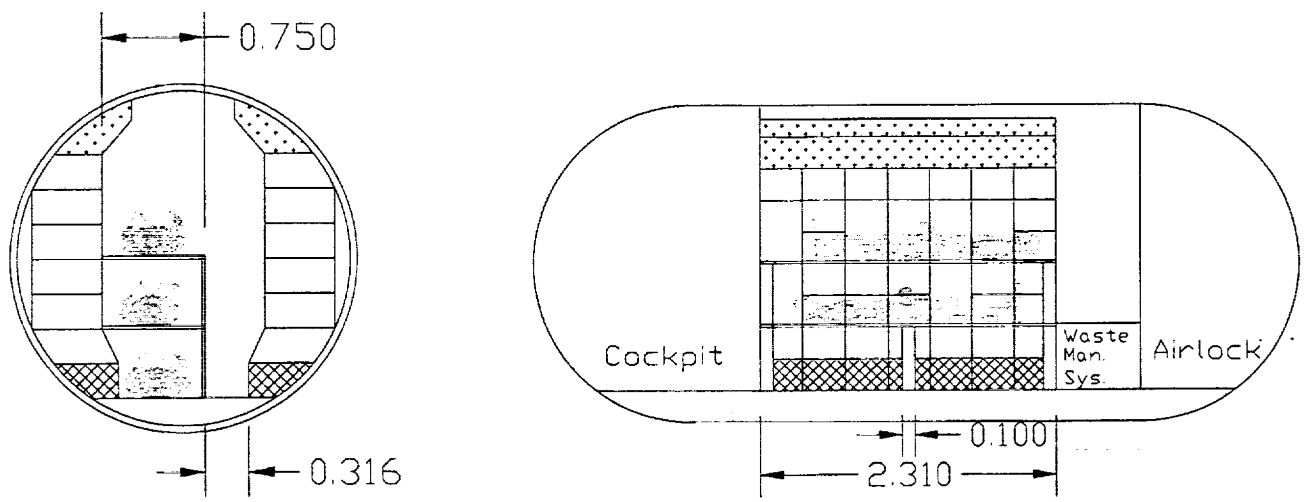


Figure 7.2: Front and Right-side View of the Sleeping Bunks for a Three-person Crew.

Cushioning and bedding are stored in the MIS Units during the daytime. This arrangement allows for approximately 0.316 meters of clearance on the side so the crew can still walk by without too much restriction. It also gives the top two crewmembers a sleeping area of ~1.75 m^2 and the bottom crew member ~1.41 m^2 of sleeping area. Figure 7.2 shows the side view of the SB in place along the right side of MARVIN.

In the daytime the SB can be broken down and stored on the floor. The bottom space between the bottom of the MIS Units is 0.95 meters wide. Each of the supports for the SB is 0.10 meters wide (See Figure 7.2) and each bed itself are 0.75 meters wide. This allows for the SB to be stacked on the floor between the MIS Units.

Access to the Life Support and Environmental Systems is accomplished by removing the bed to gain access to the floor panels. An estimated weight for this bedding arrangement is approximately 25 kg.

7.4 Interior Systems Summary

Table 7.1 Summary of the Interior Systems Located in the Living Quarters.

Equipment	Mass(kg)	Volume(m^3)	Power Required (kW)
MIS Units(2 total)	200	3.84	N/A
Water Storage	425	0.43	N/A
Fire Extinguishing(FE)	~25	0.11	N/A
Interior Lighting	~25	0.11	0.25 peak /0.10 Continuous
Scientific Equipment	650	1.5	2.8
Crew's Personal Gear and Hygiene Equipment	50	0.50	0.10 peak
Food and Preparation Equipment	100	0.50	0.10
Misc. Equipment	300	1.0	1.5
Cabin Air Conditioning(CAC)	~100	0.10	1.5
Pressure Regulation (PR)	~100	0.10	1.5
Beds and Bedding	~25	0.01	N/A
Total	**2000**	**8.20**	**7.75**

8 Airlock

8.1 Airlock General

The airlock's main purpose is to allow its occupants the ability to enter and exit the pressure vessel without depressurizing the entire rover. The airlock is located inside the last 1.25 meters of the pressure vessel. It has two semi-spherical-like, 1.25 m diameter interior and exterior hatch opening doors. The interior door will separate the living quarters from the airlock and the exterior hatch door will allow the occupants to leave the pressure vessel. The airlock hatch doors have dual pressure seals to maintain pressure integrity. The airlock is also used to store two spacesuits and a retractable ladder system that will be use to exit onto Martian soil. The spacesuits are kept at approximately 4-6 psi of pressure. This pressure is the same as that inside the entire rover, which eliminates pre-breathing exercises.

8.2 Airlock Design

The airlock is modeled after the space shuttle airlock. The airlock has five major components: an exterior hatch door, the airlock section of the pressure vessel, steps to reach the exterior hatch door, a pressure bulkhead and the interior hatch door. A layout of each component and a fully assembled configuration, excluding the spacesuits, can be found in Figures 8.1. The bulkhead functions like the end cap of the rover when the airlock is depressurized. Due to the major dust problem in the Mars atmosphere, a dust removal system needs to be designed to remove excess dust from space suits.

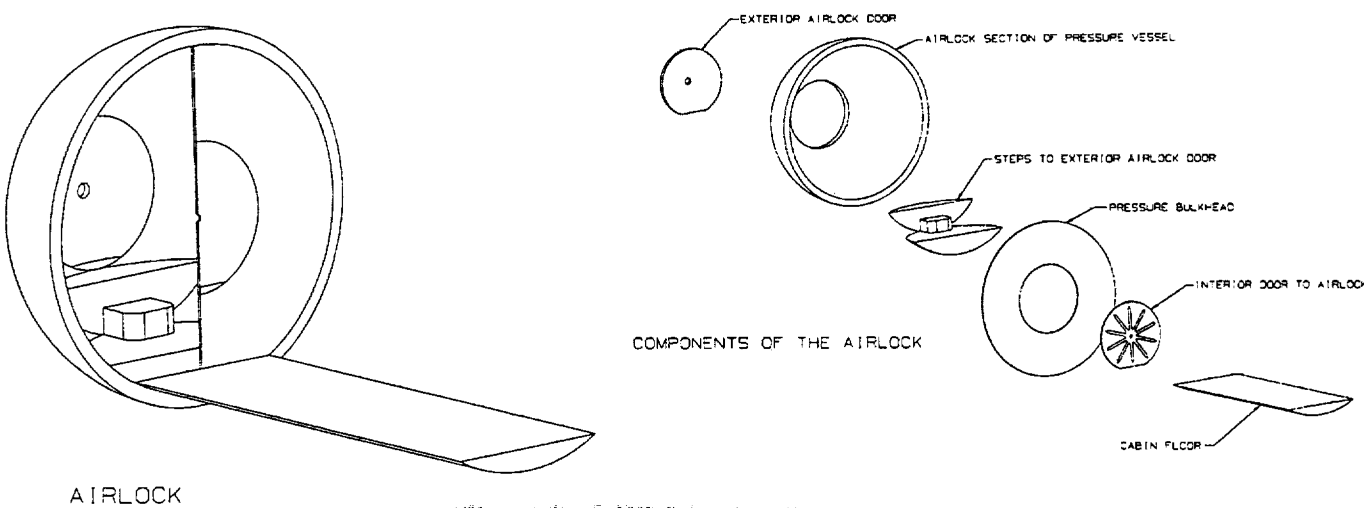


Figure 8.1: Layout of All Structural Components of the Airlock

8.3 Airlock Structure and Dimensions

The airlock section has a thickness of .10 m, an outer diameter of 2.5 m and inner diameter of 2.3 m. The two spherical hatch doors have a diameter of 1.25 m. The doors are constructed out of titanium. Titanium was chosen because of its high strength to weight ratio and its lightweight. Each airlock hatch door has a dual pressure seal to prevent depressurizing the entire pressure vessel. One seal is attached to the airlock hatch door and the other on the airlock structure. A leak check quick disconnect verifies pressure integrity. The interior hatch door has a thickness of .02 m and is located on the pressure bulkhead. The exterior hatch door is .10 m thick and is attached to the end cap of the pressure vessel. There is a circular view window with a diameter of 0.2032 m located in the center of the exterior hatch door.

The pressure bulkhead is located between the living quarters and the airlock structure. The circular bulkhead has a 2.3m diameter and is made from the same composite structure as the entire pressure vessel. The bulkhead is .20 m thick. The bulkhead serves as a substitute end cap for the pressure vessel when the rover is depressurized to allow the occupants to leave for exploration. The interior hatch door is located in the center of the bulkhead.

8.4 Extra Vehicular Activity

Attached to the backside of the spacesuits is an Extra-Vehicular Mobility Unit (EMU). The life-support system which is located inside the EMU help control and maintain temperature and air pressure of the suit while supplying sufficient amount of oxygen to the occupants. Temperatures are maintained by circulating water from a cooling system throughout the interior of the suits.

These suits are made to eliminate carbon dioxide with a hydroxide lithium cartidrige. These are also equipped with special back-up oxygen supply unit in case the main sysytem fails. Electrical power is also supplied from the EMU to run cameras and flashlights.

In order to go on an EVA mission, the astronauts must go through numerous steps to prepare for it. Among these, they must: go through pre-breathing exercises. Pre-breathing exercises are meant to reduce the chances of the astronauts becoming ill in low-pressure atmosphere. Moving rapidly from a high-pressure area to a low-pressure area can cause nitrogen bubbles to form in the bloodstream. Therefore, astronauts would inhale 100 percent oxygen to remove all traces of nitrogen from the body, which can take up to six hours. Keeping MARVIN pressurized at approximately 4-6 psi will eliminate the pre-breathing exercises.

8.5 Rotating Ladder System

Once embarked on a mission, the crew will need to get in and out of the rover and reach the storage compartments on the ears. A rotating ladder was designed to perform these two tasks. The rotating ladder system consists of a platform and a ladder that rotate around the vertical axis of the spherical endcap, as shown in Figure 8.2.

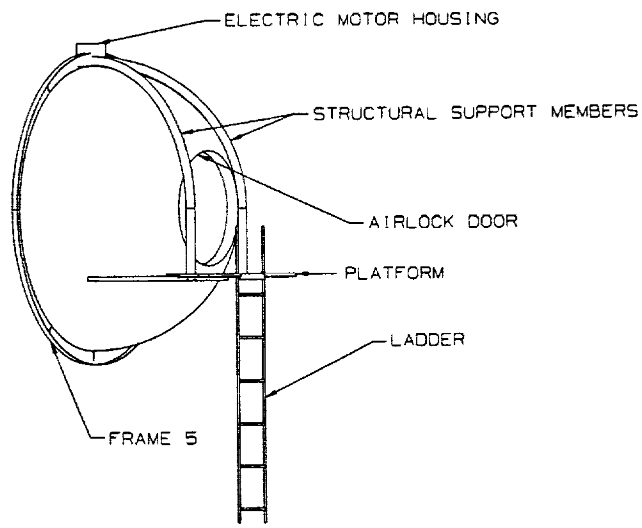
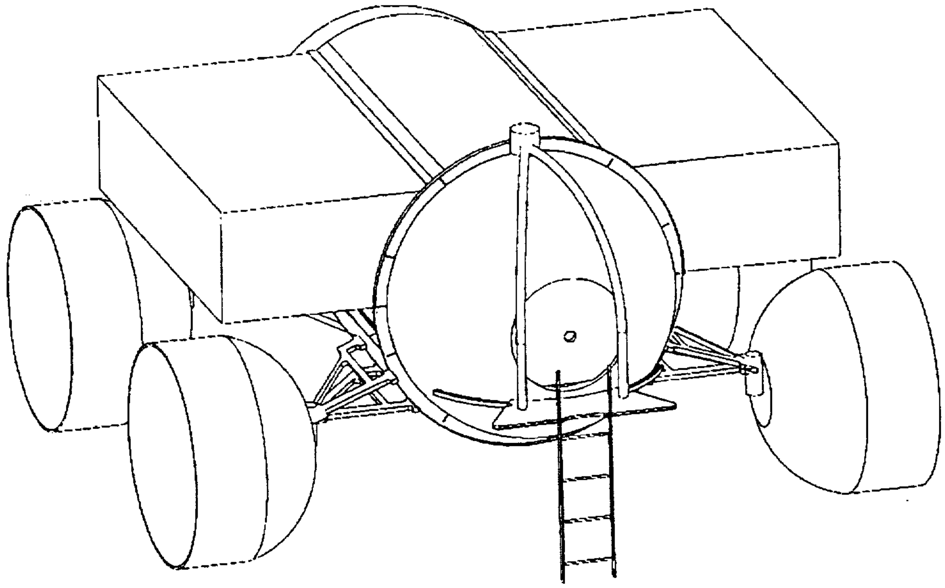


Figure 8.2: Ladder System Layout

9 Conclusions/Recommendations

As stated earlier, MARVIN is a purely conceptual study. Therefore, it lacks the in-depth analysis required for a direct application to a manned mission to Mars. However, the team did its best to present innovative design solutions and gained valuable experience and knowledge from this endeavor. Hopefully, some of these ideas may prove useful or stimulating for future mission plans to Mars.

In an effort to inform young potential engineers about the real possibility of someone landing on Mars in their lifetime, the MARVIN team voluntarily spoke to numerous schools in all grade levels. This was an opportunity to help inform them of the issues an engineer deals with when designing a rover for Mars and answer questions they might have. The main goal of the outreach program was to get the students to think about Mars and hopefully generate new and interesting ideas and concepts. We spoke to groups at three high schools, roughly 290 students, and one elementary school, which had approximately 90 students attend our presentation. To reach a more broad audience, we displayed our project at the University Open House as well as Engineering Open House at WSU. Approximately 500 people from the Wichita community saw our project during these events.



Quartering Rear View of MARVIN

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